Module 13

FOR B2 CERTIFICATION

AIRCRAFT STRUCTURES AND SYSTEMS

Aviation Maintenance Technician Certification Series







- Theory of Flight
- Structures—General Concepts
- Autoflight
- Communication/Navigation
- Electrical Power
- Equipment and Furnishings
- Flight Controls
- Instruments
- Lights
- On Board Maintenance Systems
- Air Conditioning and Cabin
- Pressurization
- Fire Protection
- Fuel Systems
- Hydraulic Power
- Ice and Rain Protection
- Landing Gear
- Oxygen
- Pneumatic/Vacuum
- Water/Waste
- Integrated Modular Avionics
- Cabin Systems
- Information Systems

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MODULE 13

FOR B2 CERTIFICATION

AIRCRAFT AERODYNAMIC STRUCTURES AND SYSTEMS

Aviation Maintenance Technician Certification Series

COMPLIANT WITH EASA
European Aviation Safety Agency
EASA PART 66/147

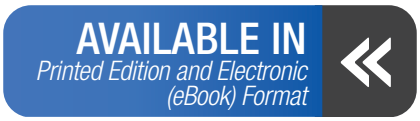


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AVIATION MAINTENANCE TECHNICIAN CERTIFICATION SERIES

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WELCOME

The publishers of this Aviation Maintenance Technician Certification Series welcome you to the world of aviation maintenance. As you move towards EASA certification, you are required to gain suitable knowledge and experience in your chosen area. Qualification on basic subjects for each aircraft maintenance license category or subcategory is accomplished in accordance with the following matrix. Where applicable, subjects are indicated by an "X" in the column below the license heading.

For other educational tools created to prepare candidates for licensure, contact Aircraft Technical Book Company.

We wish you good luck and success in your studies and in your aviation career!

REVISION LOG

VERSION	EFFECTIVE DATE	DESCRIPTION OF CHANGE
001	2016 01	Module Creation and Release
002	2017 02	Format Update
003	2017 08	Module 03 Revisions. Typo corrections in Sub-Modules 03, 04 and 08.
004	2020 02	Updated and enhanced content

Version 4 - Enhanced or modified content within the following Sub-Modules:

Sub-Module 03 Added Autopilot Modes of Operation.

Sub-Module 04 Reorganized based on Part-66 2018 changes; added content: ADS-B and Datalink; corrected Figures 4-78 and 4-79.

Sub-Module 07 Added Fly-By-Wire Failsafe and Fly-By-Wire operation.

FORWARD

PART-66 and the Acceptable Means of Compliance (AMC) and Guidance Material (GM) of the European Aviation Safety Agency (EASA) Regulation (EC) No. 1321/2014, Appendix 1 to the Implementing Rules establishes the Basic Knowledge Requirements for those seeking an aircraft maintenance license. The information in this Module of the Aviation Maintenance Technical Certification Series published by the Aircraft Technical Book Company meets or exceeds the breadth and depth of knowledge subject matter referenced in Appendix 1 of the Implementing Rules. However, the order of the material presented is at the discretion of the editor in an effort to convey the required knowledge in the most sequential and comprehensible manner. Knowledge levels required for Category A1, B1, B2, and B3 aircraft maintenance licenses remain unchanged from those listed in Appendix 1 Basic Knowledge Requirements. Tables from Appendix 1 Basic Knowledge Requirements are reproduced at the beginning of each module in the series and again at the beginning of each Sub-Module.

How numbers are written in this book:
This book uses the International Civil Aviation Organization (ICAO) standard of writing numbers. This method displays large numbers by adding a space between each group of 3 digits. This is opposed to the American method which uses commas and the European method which uses periods. For example, the number one million is expressed as so:

ICAO Standard	1 000 000
European Standard	1.000.000
American Standard	1,000,000

SI Units:
The International System of Units (SI) developed and maintained by the General Conference of Weights and Measures (CGPM) shall be used as the standard system of units of measurement for all aspects of international civil aviation air and ground operations.

Prefixes:
The prefixes and symbols listed in the table below shall be used to form names and symbols of the decimal multiples and submultiples of International System of Units (SI) units.

MULTIPLICATION FACTOR		PREFIX	SYMBOL
1 000 000 000 000 000 000	$= 10^{18}$	exa	E
1 000 000 000 000 000	$= 10^{15}$	peta	P
1 000 000 000 000	$= 10^{12}$	tera	T
1 000 000 000	$= 10^{9}$	giga	G
1 000 000	$= 10^{6}$	mega	M
1 000	$= 10^{3}$	kilo	k
100	$= 10^{2}$	hecto	h
10	$= 10^{1}$	deca	da
0.1	$= 10^{-1}$	deci	d
0.01	$= 10^{-2}$	centi	c
0.001	$= 10^{-3}$	milli	m
0.000 001	$= 10^{-6}$	micro	μ
0.000 000 001	$= 10^{-9}$	nano	n
0.000 000 000 001	$= 10^{-12}$	pico	p
0.000 000 000 000 001	$= 10^{-15}$	femto	f
0.000 000 000 000 000 001	$= 10^{-18}$	atto	a

International System of Units (SI) Prefixes

EASA LICENSE CATEGORY CHART

Module Number and Title		A1 Airplane Turbine	B1.1 Airplane Turbine	B1.2 Airplane Piston	B1.3 Helicopter Turbine	B1.4 Helicopter Piston	B2 Avionics
1	Mathematics	X	X	X	X	X	X
2	Physics	X	X	X	X	X	X
3	Electrical Fundamentals	X	X	X	X	X	X
4	Electronic Fundamentals		X	X	X	X	X
5	Digital Techniques / Electronic Instrument Systems	X	X	X	X	X	X
6	Materials and Hardware	X	X	X	X	X	X
7A	Maintenance Practices	X	X	X	X	X	X
8	Basic Aerodynamics	X	X	X	X	X	X
9A	Human Factors	X	X	X	X	X	X
10	Aviation Legislation	X	X	X	X	X	X
11A	Turbine Aeroplane Aerodynamics, Structures and Systems	X	X				
11B	Piston Aeroplane Aerodynamics, Structures and Systems			X			
12	Helicopter Aerodynamics, Structures and Systems				X	X	
13	Aircraft Aerodynamics, Structures and Systems						X
14	Propulsion						X
15	Gas Turbine Engine	X	X		X		
16	Piston Engine			X		X	
17A	Propeller	X	X	X			

GENERAL KNOWLEDGE REQUIREMENTS

MODULE 13 SYLLABUS AS OUTLINED IN PART-66, APPENDIX 1

Level 1
A familiarization with the principal elements of the subject.
Objectives:
a. The applicant should be familiar with the basic elements of the subject.
b. The applicant should be able to give a simple description of the whole subject, using common words and examples.
c. The applicant should be able to use typical terms.

Level 2
A general knowledge of the theoretical and practical aspects of the subject and an ability to apply that knowledge.
Objectives:
a. The applicant should be able to understand the theoretical fundamentals of the subject.
b. The applicant should be able to give a general description of the subject using, as appropriate, typical examples.
c. The applicant should be able to use mathematical formula in conjunction with physical laws describing the subject.
d. The applicant should be able to read and understand sketches, drawings and schematics describing the subject.
e. The applicant should be able to apply his knowledge in a practical manner using detailed procedures.

Level 3
A detailed knowledge of the theoretical and practical aspects of the subject and a capacity to combine and apply the separate elements of knowledge in a logical and comprehensive manner.
Objectives:
a. The applicant should know the theory of the subject and interrelationships with other subjects.
b. The applicant should be able to give a detailed description of the subject using theoretical fundamentals and specific examples.
c. The applicant should understand and be able to use mathematical formula related to the subject.
d. The applicant should be able to read, understand and prepare sketches, simple drawings and schematics describing the subject.
e. The applicant should be able to apply his knowledge in a practical manner using manufacturer's instructions.
f. The applicant should be able to interpret results from various sources and measurements and apply corrective action where appropriate.

	LEVELS B2
Sub-Module 01 - Theory of Flight	
(a) *Airplane Aerodynamics and Flight Controls* Operation and effect of: Roll control: ailerons and spoilers, pitch control: elevators, stabilators, variable incidence stabilizers and canards, yaw control, rudder limiters; Control using elevons, ruddervators; High lift devices: slots, slats, flaps, drag inducing devices: spoilers, lift dumpers, speed brakes; Operation and effect of trim tabs, servo tabs, control surface bias;	1
(b) *High Speed Flight* Speed of sound, subsonic flight, transonic flight, supersonic flight; Mach number, critical Mach number;	1
(c) *Rotary Wing Aerodynamics* Terminology; Operation and effect of cyclic, collective and anti-torque controls.	1
Sub-Module 02 - Structures — General Concepts	
(a) Fundamentals of structural systems.	1
(b) Zonal and station identification systems; Electrical bonding; Lightning strike protection provision.	2
Sub-Module 03 - Autoflight (ATA 22)	
(a) Fundamentals of automatic flight control including working principles and current terminology; Command signal processing; Modes of operation: roll, pitch and yaw channels; Yaw dampers; Stability Augmentation System in helicopters; Automatic trim control; Autopilot navigation aids interface.	3
(b) Auto-throttle systems; Automatic landing systems: principles and categories, modes of operation, approach, glideslope, land, go-around, system monitors and failure conditions.	3
Sub-Module 04 - Communication / Navigation (ATA 23/34)	
(a) Fundamentals of radio wave propagation, antennas, transmission lines, communication, receiver and transmitter; Working principles of following systems: - Very High Frequency (VHF) communication; - High Frequency (HF) communication; - Audio; - Emergency Locator Transmitters (ELTs); - Cockpit Voice Recorder (CVR); - Very High Frequency Omnidirectional Range (VOR); - Automatic Direction Finding (ADF); - Instrument Landing System (ILS);	3

- Flight Director Systems (FDSs), Distance Measuring Equipment (DME);
- Area navigation, RNAV systems;
- Flight Management Systems (FMSs);
- Global Positioning System (GPS), Global Navigation Satellite Systems (GNSSs);
- Data Link.

(b) Air Traffic Control transponder, secondary surveillance radar; — 3
Traffic Alert and Collision Avoidance System (TCAS);
Weather avoidance radar;
Radio altimeter;
Automatic Dependent Surveillance Broadcast (ADS-B).

(c) Microwave Landing System (MLS); — 3
Very Low Frequency and hyperbolic navigation (VLF/Omega);
Doppler navigation;
Inertial Navigation System (INS);
ARINC (Aircraft Radio Incorporated) communication and reporting.

Sub-Module 05 - Electrical Power (ATA 24)

Batteries Installation and Operation; — 3
DC power generation;
AC power generation;
Emergency power generation;
Voltage regulation;
Power distribution;
Inverters, transformers, rectifiers, circuit protection;
External/Ground power.

Sub-Module 06 - Equipment and Furnishings (ATA 25)

Electronic emergency equipment requirements. — 3
Cabin entertainment equipment.

Sub-Module 07 - Flight Controls (ATA 27)

(a) Primary controls: aileron, elevator, rudder, spoiler; — 2
Trim control;
Active load control;
High lift devices;
Lift dump, speed brakes;
System operation: manual, hydraulic, pneumatic;
Artificial feel, Yaw damper, Mach trim, rudder limiter, gust locks; Stall protection systems.

(b) System operation: electrical, fly-by-wire. — 3

Sub-Module 08 - Instruments (ATA 31) — 3

Classification;
Atmosphere;
Terminology;
Pressure measuring devices and systems;
Pitot-static systems;
Altimeters;
Vertical speed indicators;
Airspeed indicators;
Machmeters;
Altitude reporting/alerting systems;
Air data computers;
Instrument pneumatic systems;
Direct reading pressure and temperature gauges;
Temperature indicating systems;
Fuel quantity indicating systems;
Gyroscopic principles;
Artificial horizons;
Slip indicators;
Directional gyros;
Ground Proximity Warning Systems;
Compass systems;
Flight Data Recording systems;
Electronic Flight Instrument Systems;
Instrument warning systems including master warning systems and centralized warning panels;
Stall warning systems and angle of attack indicating systems;
Vibration measurement and indication;
Glass cockpit.

Sub-Module 09 - Lights (ATA 33) — 3

External: navigation, landing, taxiing, ice;
Internal: cabin, cockpit, cargo;
Emergency.

Sub-Module 10 - On Board Maintenance Systems (ATA 45) — 3

Central maintenance computers;
Data loading system;
Electronic library system;
Printing;
Structure monitoring (damage tolerance monitoring).

	LEVELS B2
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Air conditioning systems;	2
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11.3 - Pressurization	
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(b) Portable fire extinguisher.	1
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Indications and warnings;	3
Refuelling and defuelling;	2
Longitudinal balance fuel systems.	3
Sub-Module 14 - Hydraulic Power (ATA 29)	
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Hydraulic fluids;	1
Hydraulic reservoirs and accumulators;	1
Pressure generation: electrical, mechanical, pneumatic;	3
Emergency pressure generation;	3
Filters;	1
Pressure control;	3
Power distribution;	1
Indication and warning systems;	3
Interface with other systems.	3

	LEVELS B2
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Ice formation, classification and detection;	2
Anti-icing systems: electrical, hot air and chemical;	2
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Rain repellent;	1
Probe and drain heating;	3
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Sub-Module 16 - Landing Gear (ATA 32)	
Construction, shock absorbing;	1
Extension and retraction systems: normal and emergency;	3
Indications and warnings;	3
Wheels, brakes, antiskid and autobraking;	3
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Supply regulation;	3
Indications and warnings;	3
Sub-Module 18 - Pneumatic/Vacuum (ATA 36)	
System lay-out;	2
Sources: engine/APU, compressors, reservoirs, ground supply;	2
Pressure control;	3
Distribution;	1
Indications and warnings;	3
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Sub-Module 19 - Water/Waste (ATA 38)	
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Sub-Module 20 - Integrated Modular Avionics (ATA 42)	
Functions that may be typically integrated in the Integrated Modular Avionic (IMA) modules are, among others: Bleed Management, Air Pressure Control, Air Ventilation and Control, Avionics and Cockpit Ventilation Control, Temperature Control, Air Traffic Communication, Avionics Communication Router, Electrical Load Management, Circuit Breaker Monitoring, Electrical System BITE, Fuel Management, Braking Control, Steering Control, Landing Gear Extension and Retraction, Tyre Pressure Indication, Oleo Pressure Indication, Brake Temperature Monitoring, etc.; Core System; Network Components.	3

Sub-Module 21 - Cabin Systems (ATA 44) 3

The Cabin Network Service may host functions such as:
Access to predeparture/departure reports,
E-mail/intranet/internet access,
Passenger database;
Cabin Core System;
In-flight Entertainment System;
External Communication System;
Cabin Mass Memory System;
Cabin Monitoring System;
Miscellaneous Cabin System.

Sub-Module 22 - Information Systems (ATA 46) 3

The units and components which furnish a means of storing, updating and retrieving digital information traditionally provided on paper, microfilm or microfiche. Includes units that are dedicated to the information storage and retrieval function such as the electronic library mass storage and controller. Does not include units or components installed for other uses and shared with other systems, such as flight deck printer or general use display. Typical examples include Air Traffic and Information Management Systems and Network Server Systems.
Aircraft General Information System;
Flight Deck Information System;
Maintenance Information System;
Passenger Cabin Information System;
Miscellaneous Information System.

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PART-66 SYLLABUS **LEVELS**

CERTIFICATION CATEGORY → **B2**

Sub-Module 01

THEORY OF FLIGHT

Knowledge Requirements

13.1 - Theory of Flight

(a) *Airplane Aerodynamics and Flight Controls* — **1**

Operation and effect of:
- roll control: ailerons and spoilers,
- pitch control: elevators, stabilators, variable incidence stabilizers and canards,
- yaw control, rudder limiters;

Control using elevons, ruddervators;
High lift devices: slots, slats, flaps, drag inducing devices: spoilers, lift dumpers, speed brakes;
Operation and effect of trim tabs, servo tabs, control surface bias;

(b) *High Speed Flight* — **1**

Speed of sound, subsonic flight, transonic flight, supersonic flight;
Mach number, critical Mach number;

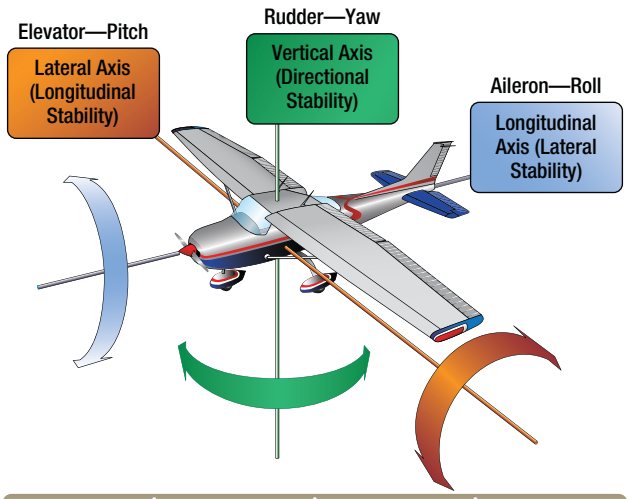
(c) *Rotary Wing Aerodynamics* — **1**

Terminology;
Operation and effect of cyclic, collective and anti-torque controls.

The directional control of a fixed-wing aircraft takes place around the lateral, longitudinal, and vertical axes by means of flight control surfaces designed to create movement about these axes. These control devices are hinged or movable surfaces through which the attitude of an aircraft is controlled during takeoff, flight, and landing. They are usually divided into two major groups: 1) primary or main flight control surfaces and 2) secondary or auxiliary control surfaces.

PRIMARY FLIGHT CONTROL SURFACES

The primary flight control surfaces on a fixed-wing aircraft include: ailerons, elevators, and the rudder. The ailerons are attached to the trailing edge of both wings and when moved, rotate the aircraft around the longitudinal axis. The elevator is attached to the trailing edge of the horizontal stabilizer. When it is moved, it alters aircraft pitch, which is the attitude about the horizontal or lateral axis. The rudder is hinged to the trailing edge of the vertical stabilizer. When the rudder changes position, the aircraft rotates about the vertical axis (yaw). ***Figure 1-1*** shows the primary flight controls of a light aircraft and the movement they create relative to the three axes of flight.



Primary Control Surface	Airplane Movement	Axes of Rotation	Type of Stability
Aileron	Roll	Longitudinal	Lateral
Elevator/ Stabilator	Pitch	Lateral	Longitudinal
Rudder	Yaw	Vertical	Directional

Figure 1-1. Flight control surfaces move the aircraft around the three axes of flight.

Primary control surfaces are usually similar in construction to one another and vary only in size, shape, and methods of attachment. On aluminum light aircraft, their structure is often similar to an all-metal wing. This is appropriate because the primary control surfaces are simply smaller aerodynamic devices. They are typically made from an aluminum alloy structure built around a single spar member or torque tube to which ribs are fitted and a skin is attached. The lightweight ribs are, in many cases, stamped out from flat aluminum sheet stock. Holes in the ribs lighten the assembly. An aluminum skin is attached with rivets. ***Figure 1-2*** illustrates this type of structure, which can be found on the primary control surfaces of light aircraft as well as on medium and heavy aircraft.

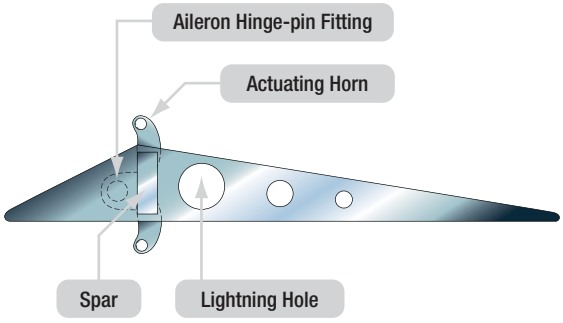


Figure 1-2. Typical structure of an aluminum flight control surface.

Primary control surfaces constructed from composite materials are also commonly used. These are found on many heavy and high-performance aircraft, as well as gliders, home-built, and light-sport aircraft.

The weight and strength advantages over traditional construction can be significant. A wide variety of materials and construction techniques are employed. ***Figure 1-3*** shows examples of aircraft that use composite technology on primary flight control surfaces. Note that the control surfaces of fabric-covered aircraft often have fabric covered surfaces just as aluminum-skinned (light) aircraft typically have all-aluminum control surfaces.

OPERATION AND EFFECT OF ROLL CONTROL DEVICES

AILERONS

Ailerons are the primary flight control surfaces that move the aircraft about the longitudinal axis. In other

words, movement of the ailerons in flight causes the aircraft to roll. Ailerons are usually located on the outboard trailing edge of each of the wings. They are built into the wing and are calculated as part of the wing's surface area. ***Figure 1-4*** shows aileron locations on various wing tip designs.

Ailerons are controlled by a side-to-side motion of the control stick in the cockpit or a rotation of the control yoke. When the aileron on one wing deflects down, the aileron on the opposite wing deflects upward. This amplifies the movement of the aircraft around the longitudinal axis. On the wing on which the aileron trailing edge moves downward, camber is increased and lift is increased. Conversely, on the other wing, the raised aileron decreases lift. (***Figure 1-5***)

The result is a sensitive response to the control input to roll the aircraft. The pilot's request for aileron movement and roll are transmitted from the cockpit to the actual control surface in a variety of ways depending on the aircraft. A system of control cables and pulleys, push-pull tubes, hydraulics, electric, or a combination of these can be employed. (***Figure 1-6***)

Simple, light aircraft usually do not have hydraulic or electric fly-by-wire aileron control. These are found on heavy and high-performance aircraft. Large aircraft and some high performance aircraft may also have a second set of ailerons located inboard on the trailing edge of the wings. These are part of a complex system of primary and secondary control surfaces used to provide lateral control and stability in flight. At low speeds, the ailerons may be augmented by the use of flaps and spoilers. At high speeds, only inboard aileron deflection is required to roll the aircraft while the other control surfaces are locked out or remain stationary.



Figure 1-3. Composite control surfaces and some of the many aircraft that utilize them.

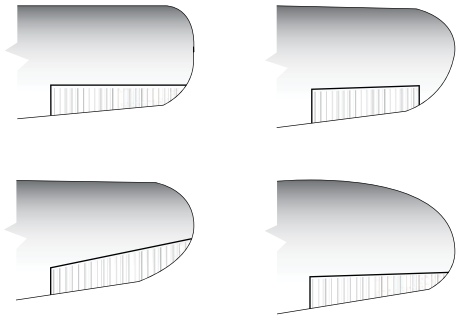


Figure 1-4. Aileron location on various wings.

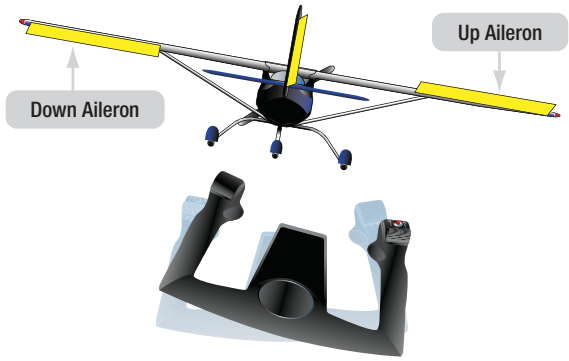


Figure 1-5. Differential aileron control movement. When one aileron is moved down, the aileron on the opposite wing is deflected upward.

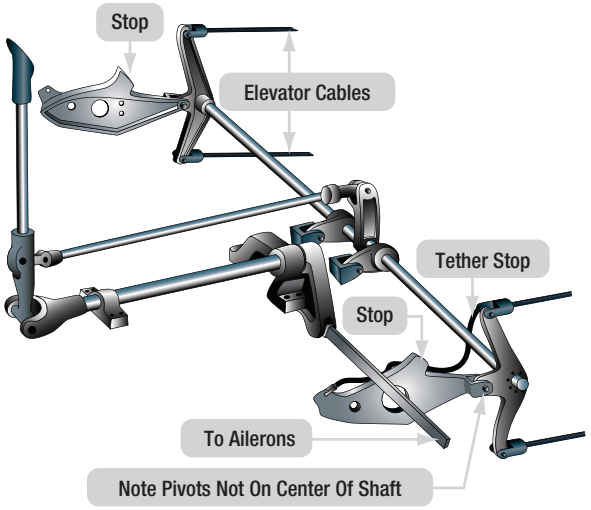


Figure 1-6. Transferring control surface inputs from the cockpit.

Figure 1-7 illustrates the location of the typical flight control surfaces found on a transport category aircraft.

SPOILERS

A spoiler is a device found on the upper surface of many heavy and high-performance aircraft. It is stowed flush to the wing's upper surface. When deployed, it raises up into the airstream and disrupts the laminar airflow of the wing, thus reducing lift. Spoilers are made with similar construction materials and techniques as the other flight control surfaces on the aircraft. At low speeds, spoilers are rigged to operate when the ailerons operate to assist with the lateral movement and stability of the aircraft. On the wing where the aileron is moved up, the spoilers also raise thus amplifying the reduction of lift on that wing. (***Figure 1-8***) On the wing with downward aileron deflection, the spoilers remain stowed. As the speed of the aircraft increases, the ailerons become more effective and the spoiler interconnect disengages. Note that spoilers are also used in as drag inducing devices.

OPERATION AND EFFECT OF PITCH CONTROL DEVICES

ELEVATORS

The elevator is the primary flight control surface that moves the aircraft around the horizontal or lateral axis. This causes the nose of the aircraft to pitch up or down. The elevator is hinged to the trailing edge of the horizontal stabilizer and typically spans most or all of its width. It is controlled in the cockpit by pushing or pulling the control yoke forward or aft. Light aircraft use a system of control cables and pulleys or push pull tubes to transfer cockpit inputs to the movement of the elevator. High performance and large aircraft typically employ more complex systems. Hydraulic power is commonly used to move the elevator on these aircraft.

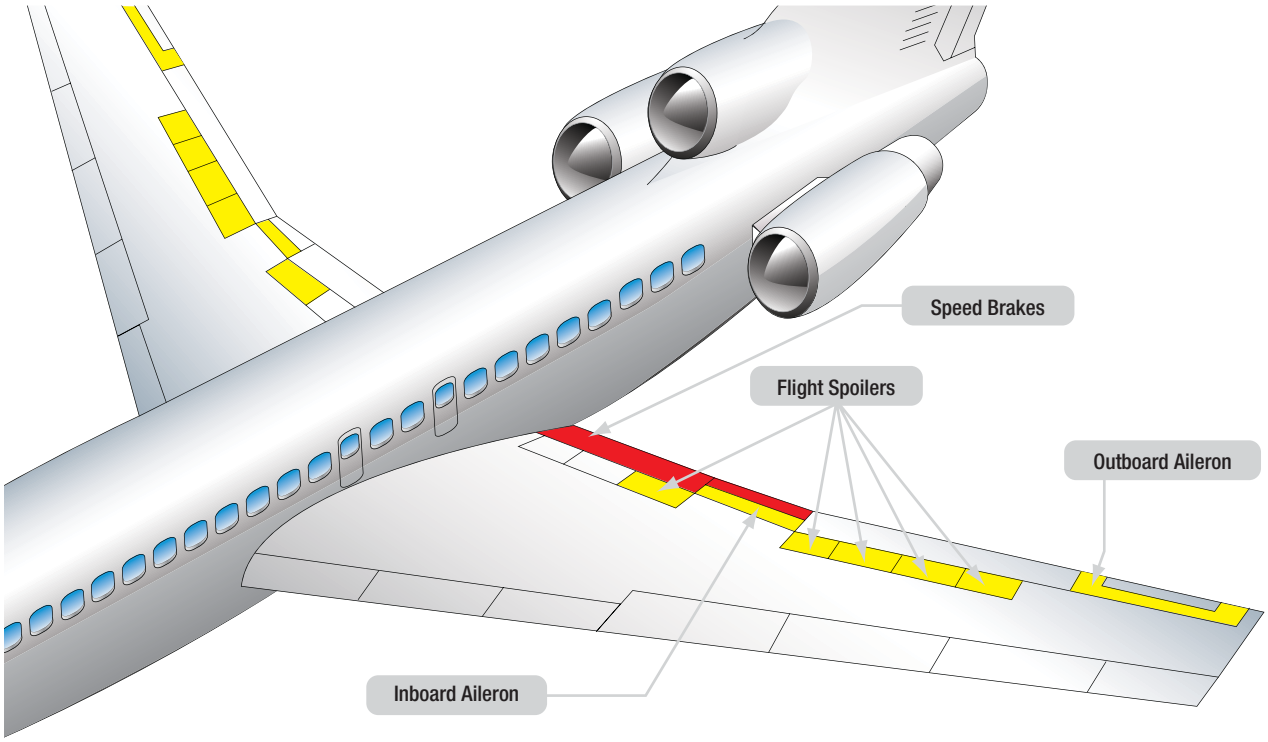


Figure 1-7. Typical flight control surfaces on a transport category aircraft.



Figure 1-8. Spoilers deployed upon landing a transport category aircraft.

On aircraft equipped with fly-by-wire controls, a combination of electrical and hydraulic power is used.

STABILATORS

A movable horizontal tail section, called a stabilator, is a control surface that combines the action of both the horizontal stabilizer and the elevator. (***Figure 1-9***) Basically, a stabilator is a horizontal stabilizer that can also be rotated about the horizontal axis to affect the pitch of the aircraft.

VARIABLE INCIDENCE STABILIZERS

A variable incidence stabilizer refers to any horizontal stabilizer in which the angle of incidence of the horizontal stabilizer is adjustable. Thus, a stabilator is a variable incidence horizontal stabilizer. Various mechanisms and operating rigging are available. Most large aircraft use a motorized jackscrew to alter the position of the stabilizer often energized by the trim tab switch on the control yoke. The reason for a stabilator or any horizontal stabilizer variable incidence device is to minimize drag when trimming the aircraft in flight. Deflection of the elevator via the use of a trim tab causes drag and requires a relatively large elevator on large aircraft to achieve all desired trim settings. By varying the angle of the horizontal stabilizer to adjust pitch, less drag is created and elevator size and deflection may be reduced. (***Figure 1-10***)



Figure 1-9. A stabilizer and index marks on a transport category aircraft.

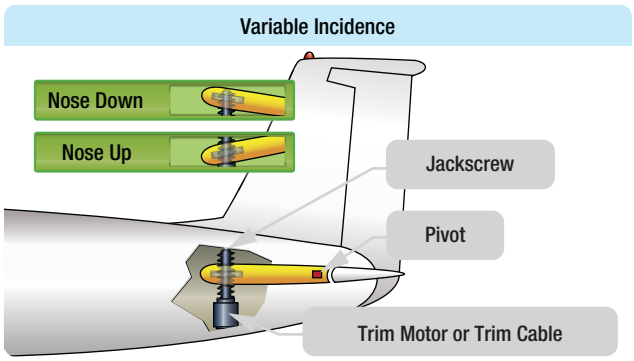


Figure 1-10. Some airplanes, including most jet transports, use an variable stabilizer to provide the required pitch trim forces.

CANARDS

A canard utilizes the concept of two lifting surfaces. It functions as a horizontal stabilizer located in front of the main wings. In effect, the canard is an airfoil similar to the horizontal surface on a conventional aft-tail design. The difference is that the canard actually creates lift and holds the nose up, as opposed to the aft-tail design which exerts downward force on the tail to prevent the nose from rotating downward. (***Figure 1-11***)

The canard design dates back to the pioneer days of aviation, most notably used on the Wright Flyer. Recently, the canard configuration has regained popularity and is appearing on newer aircraft. Canard designs include two types–one with a horizontal surface of about the same size as a normal aft-tail design, and the other with a surface of the same approximate size and airfoil shape of the aft-mounted wing known as a tandem wing configuration. Theoretically, the canard is considered more efficient because using the horizontal surface to help lift the weight of the aircraft should result in less drag for a given amount of lift.



Figure 1-11. The Piaggio P180 includes a variable-sweep canard design, which provides longitudinal stability about the lateral axis.

OPERATION AND EFFECT OF YAW CONTROL DEVICES

RUDDERS

The rudder is the primary control surface that causes an aircraft to yaw or move about the vertical axis. This provides directional control and thus points the nose of the aircraft in the direction desired. Most aircraft have a single rudder hinged to the trailing edge of the vertical stabilizer. It is controlled by a pair of foot-operated rudder pedals in the cockpit. When the right pedal is pushed forward, it deflects the rudder to the right which moves the nose of the aircraft to the right. The left pedal is rigged to simultaneously move aft. When the left pedal is pushed forward, the nose of the aircraft moves to the left.

As with the other primary flight controls, the transfer of the movement of the cockpit controls to the rudder varies with the complexity of the aircraft. Many aircraft incorporate the directional movement of the nose or tail wheel into the rudder control system for ground operation. This allows the operator to steer the aircraft with the rudder pedals during taxi when the airspeed is not high enough for the control surfaces to be effective. Some large aircraft have a split rudder arrangement. This is actually two rudders, one above the other. At low speeds, both rudders deflect in the same direction when the pedals are pushed. At higher speeds, one of the rudders becomes inoperative as the deflection of a single rudder is aerodynamically sufficient to maneuver the aircraft.

RUDDER LIMITERS

In flight, most large aircraft oscillate slightly from side to side. Yaw dampener units automatically detect this movement and send signals to the hydraulic power control unit (PCU) that moves the rudder so that it can correct for these yaw oscillations. Similarly, rudders are known to deflect without being commanded to do so by the flight crew. Again, the yaw dampener is designed to correct the fluctuations by signaling the PCU. However, too large of an involuntary deflection to a rudder can cause a loss of control of the aircraft. A rudder limiter is fitted to many aircraft to prevent any more than a few degrees of involuntary motion of the rudder. Essentially, it limits the movement unless it is commanded from the flight deck.

SECONDARY OR AUXILIARY CONTROL SURFACES

There are several secondary or auxiliary flight control surfaces. Their names, locations, and functions of those for most large aircraft are listed in ***Figure 1-12***.

OPERATION AND EFFECT OF TABS

Trim Tabs

The force of the air against a control surface during the high speed of flight can make it difficult to move and hold that control surface in the deflected position. A control surface might also be too sensitive for similar reasons. Several different tabs are used to aid with these types of problems. The table in ***Figure 1-13*** summarizes the various tabs and their uses. While in flight, it is desirable for the pilot to be able to take his or her hands and feet off of the controls and have the aircraft maintain its flight condition.

Trims tabs are designed to allow this. Most trim tabs are small movable surfaces located on the trailing edge of a primary flight control surface. A small movement of the tab in the direction opposite of the direction the flight control surface is deflected, causing air to strike the tab, in turn producing a force that aids in maintaining the flight control surface in the desired position. Through linkage set from the cockpit, the tab can be positioned so that it is actually holding the control surface in position rather than the pilot. Therefore, elevator tabs are used to maintain the speed of the aircraft since they assist in maintaining the selected pitch. Rudder tabs can be set to hold yaw in check and maintain heading. Aileron tabs can help keep the wings level.

Secondary/Auxiliary Flight Control Surfaces		
Name	**Location**	**Function**
Flaps	Inboard trailing edge of wings	Extends the camber of the wing for greater lift and slower flight. Allows control at low speeds for short field takeoffs and landings.
Trim Tabs	Trailing edge of primary flight control surfaces	Reduces the force needed to move a primary control surface.
Balance Tabs	Trailing edge of primary flight control surfaces	Reduces the force needed to move a primary control surface.
Anti-balance Tabs	Trailing edge of primary flight control surfaces	Increases feel and effectiveness of primary control surface.
Servo Tabs	Trailing edge of primary flight control surfaces	Assists or provides the force for moving a primary flight control.
Spoilers	Upper and/or trailing edge of wing	Decreases (spoils) lift. Can augment aileron function.
Slats	Mid to outboard leading edge of wing	Extends the camber of the wing for greater lift and slower flight. Allows control at low speeds for short field takeoffs and landings.
Slots	Outer leading edge of wing forward of ailerons	Directs air over upper surface of wing during high angle of attack. Lowers stall speed and provides control during slow flight.
Leading Edge Flap	Inboard leading edge of wing	Extends the camber of the wing for greater lift and slower flight. Allows control at low speeds for short field takeoffs and landings.

NOTE: An aircraft may possess none, one, or a combination of the above control surfaces.

Figure 1-12. Secondary or auxiliary control surfaces and respective locations for larger aircraft.

Flight Control Tabs			
Type	**Direction of Motion (in relation to control surface)**	**Activation**	**Effect**
Trim	Opposite	Set by pilot from cockpit. Uses independent linkage.	Statically balances the aircraft in flight. Allows "hands off" maintenance of flight condition.
Balance	Opposite	Moves when pilot moves control surface. Coupled to control surface linkage.	Aids pilot in overcoming the force needed to move the control surface.
Servo	Opposite	Directly linked to flight control input device. Can be primary or back-up means of control.	Aerodynamically positions control surfaces that require too much force to move manually.
Anti-balance or Anti-servo	Same	Directly linked to flight control input device.	Increases force needed by pilot to change flight control position. De-sensitizes flight controls.
Spring	Opposite	Located in line of direct linkage to servo tab. Spring assists when control forces become too high in high-speed flight.	Enables moving control surface when forces are high. Inactive during slow flight.

Figure 1-13. Various tabs and their uses.

Occasionally, a simple light aircraft may have a stationary metal plate attached to the trailing edge of a primary flight control, usually the rudder. This is also a trim tab as shown in ***Figure 1-14***. It can be bent slightly on the ground to trim the aircraft in flight to a hands off condition when flying straight and level. The correct amount of bend can be determined only by flying the aircraft after an adjustment. Note that a small amount of bending is usually sufficient.

Balance Tabs

The aerodynamic phenomenon of moving a trim tab in one direction to cause the control surface to experience a force moving in the opposite direction is exactly what occurs with the use of balance tabs. (***Figure 1-15***) Often, it is difficult to move a primary control surface due to its surface area and the speed of the air rushing over it.

Deflecting a balance tab hinged at the trailing edge of the control surface in the opposite direction of the desired control surface movement causes a force to position the surface in the proper direction with reduced force to do so. Balance tabs are usually linked directly to the control surface linkage so that they move automatically when there is an input for control surface movement. They also can double as trim tabs, if adjustable on the flight deck.

Servo Tabs

A servo tab is similar to a balance tab in location and effect, but it is designed to operate the primary flight control surface, not just reduce the force needed to do so. It is usually used as a means to back up the primary control of the flight control surfaces. (***Figure 1-16***)

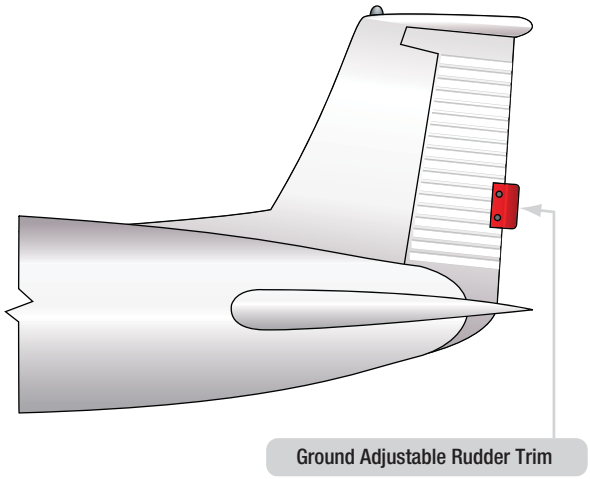


Figure 1-14. Example of a trim tab.

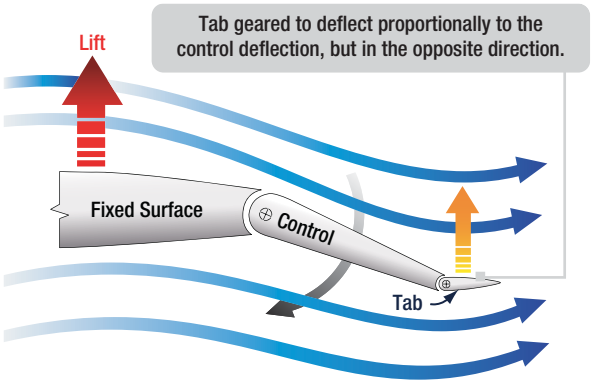


Figure 1-15. Balance tabs assist with forces needed to position control surfaces.

On heavy aircraft, large control surfaces require too much force to be moved manually and are usually deflected out of the neutral position by hydraulic actuators. These power control units are signaled via a system of hydraulic valves connected to the yoke and rudder pedals. On fly-by-wire aircraft, the hydraulic actuators that move the flight control surfaces are signaled by electric input. In the case of hydraulic system failure(s), manual linkage to a servo tab can be used to deflect it. This, in turn, provides an aerodynamic force that moves the primary control surface.

Anti-Servo/Anti-Balance Tabs

Anti-servo tabs, as the name suggests, are like servo tabs but move in the same direction as the primary control surface. On some aircraft, especially those with a movable horizontal stabilizer, the input to the control surface can be too sensitive. An Anti-servo tab tied through the control linkage creates an aerodynamic force that increases the effort needed to move the control surface. This makes flying the aircraft more stable for the pilot.

Figure 1-17 shows an Anti-servo tab in the near neutral position. Deflected in the same direction as the desired stabilator movement, it increases the required control surface input. Anti-servo tabs are also known as anti-balance tabs.

Control Surface Bias

When a control surface is in the neutral position, is faired with the wing rudder or horizontal stabilizer and no effect on the aircrafts aerodynamic surfaces. Some aircraft are designed with control surface bias.

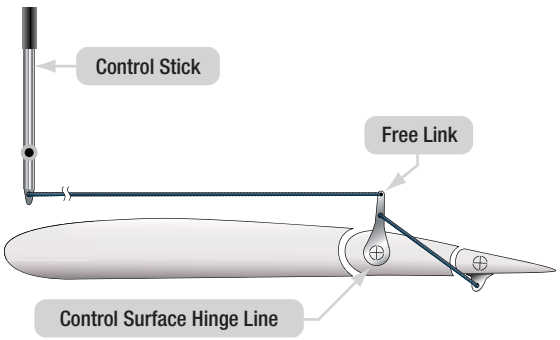


Figure 1-16. Servo tabs can be used to position flight control surfaces in case of hydraulic failure.

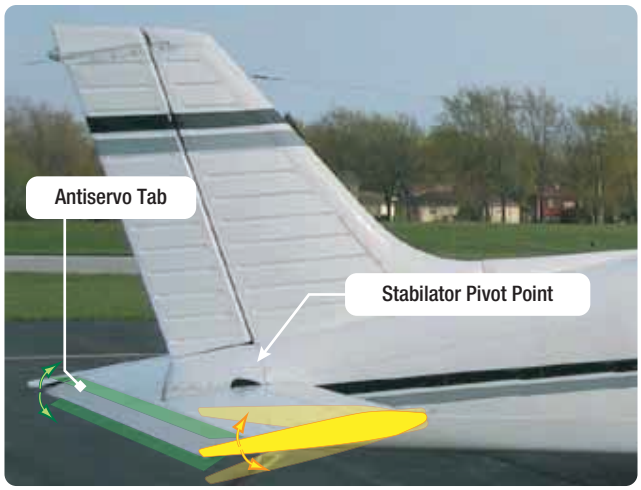


Figure 1-17. An Anti-servo tab moves in the same direction as the control tab. Shown here on a stabilator, it desensitizes the pitch control.

This means that a control surface is not naturally in the neutral position. It is designed to impart a force on the airfoil at all times. The force is generally used to counter balance a design imbalance and alter the aircraft's aerodynamics for easy hands-off flight. This means that when the aircraft is flying straight and level, the control surface bias has effect but all trim position gauges on the flight deck indicate zero trim.

HIGH LIFT DEVICES

Aircraft wings contain devices that are designed to increase the lift produced by the wing with the devices deployed during certain phases of flight.

FLAPS

Flaps are one such high lift device found on most aircraft. They are usually inboard on the wings' trailing edges adjacent to the fuselage. Leading edge flaps are also common. They extend forward and down from the inboard wing leading edge. The flaps are lowered to increase the camber of the wings and provide greater lift and control at slow speeds. They enable landing at slower speeds and shorten the amount of runway required for takeoff and landing. The amount that the flaps extend and the angle they form with the wing can be selected from the cockpit. Typically, flaps can extend up to 45-50˚. ***Figure 1-18*** shows various aircraft with flaps in the extended position.

Flaps are usually constructed of materials and with techniques used on the other airfoils and control surfaces of a particular aircraft. Aluminum skin and structure flaps are the norm on light aircraft. Heavy and high performance aircraft flaps may also be aluminum, but the use of composite structures is also common.

There are various kinds of flaps. Plain flaps form the trailing edge of the wing when the flap is in the retracted position. (***Figure 1-19A***) The airflow over the wing continues over the upper and lower surfaces of the flap, making the trailing edge of the flap essentially the trailing edge of the wing. The plain flap is hinged so that the trailing edge can be lowered. This increases wing camber and provides greater lift.

A split flap is normally housed under the trailing edge of the wing. (***Figure 1-19B***) It is usually just a braced flat metal plate hinged at several places along its leading edge. The upper surface of the wing extends to the trailing edge of the flap. When deployed, the split flap trailing edge lowers away from the trailing edge of the wing. Airflow over the top of the wing remains the same. Airflow under the wing now follows the camber created by the lowered split flap, increasing lift.



Figure 1-18. An aileron balance panel and linkage uses varying air pressure to assist in control surface positioning.

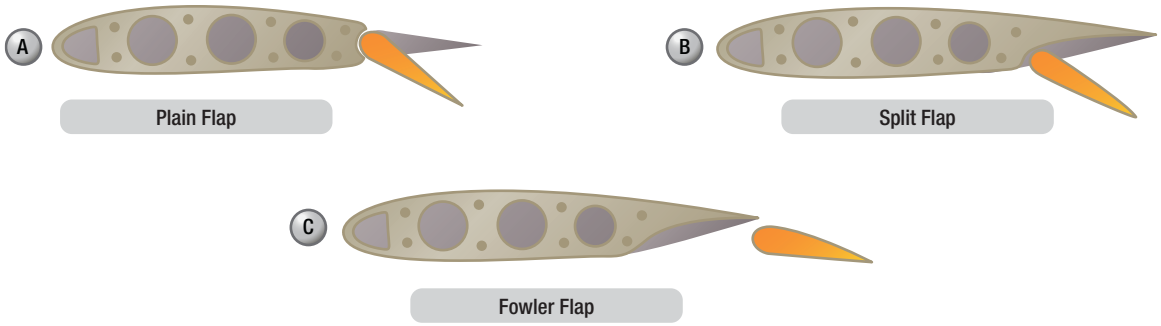


Figure 1-19. Various types of flaps.

Fowler flaps not only lower the trailing edge of the wing when deployed but also slide aft, effectively increasing the area of the wing. (***Figure 1-19C***) This creates more lift via the increased surface area, as well as the wing camber. When stowed, the fowler flap typically retracts up under the wing trailing edge similar to a split flap. The sliding motion of a fowler flap can be accomplished with a worm drive and flap tracks.

An enhanced version of the fowler flap is a set of flaps that actually contains more than one aerodynamic surface. ***Figure 1-20*** shows a triple-slotted flap. In this configuration, the flap consists of a fore flap, a mid flap, and an aft flap.

When deployed, each flap section slides aft on tracks as it lowers. The flap sections also separate leaving an open slot between the wing and the fore flap, as well as between each of the flap sections. Air from the underside of the wing flows through these slots. The result is that the laminar flow on the upper surfaces is enhanced. The greater camber and effective wing area increase overall lift.

Heavy aircraft often have leading edge flaps that are used in conjunction with the trailing edge flaps. (***Figure 1-21***) They can be made of machined magnesium or can have an aluminum or composite structure.

While they are not installed or operate independently, their use with trailing edge flaps can greatly increase wing camber and lift. When stowed, leading edge flaps retract into the leading edge of the wing. The differing designs of leading edge flaps essentially provide the same effect. Activation of the trailing edge flaps automatically deploys the leading edge flaps, which are driven out of the leading edge and downward, extending the camber of the wing. ***Figure 1-22*** shows a Krueger flap, recognizable by its flat mid-section.

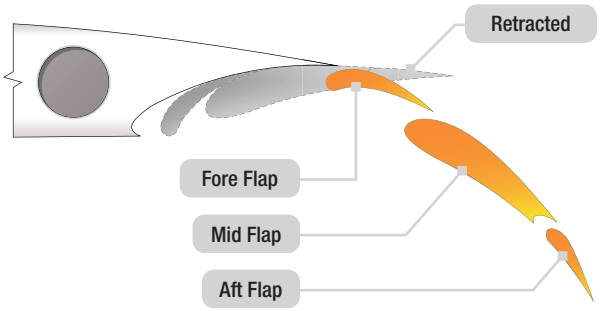


Figure 1-20. Triple slotted flap.

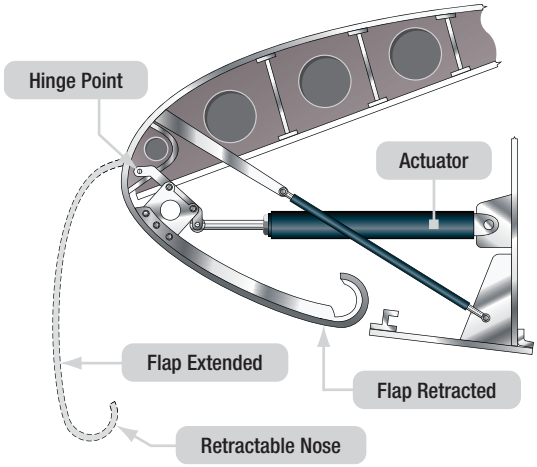


Figure 1-21. Leading edge flaps.

FLAPERONS

Some aircraft are equipped with flaperons. (***Figure 1-23***) Flaperons are ailerons which can also act as flaps. Flaperons combine both aspects of flaps and ailerons. In addition to controlling the bank angle of an aircraft like conventional ailerons, flaperons can be lowered together to function much the same as a dedicated set of flaps. The pilot retains separate controls for ailerons and flaps. A mixer is used to combine the separate pilot inputs into this single set of control surfaces called flaperons. Many designs that incorporate flaperons mount the control surfaces away from the wing to provide undisturbed airflow at high angles of attack and/or low airspeeds.



Figure 1-22. Side view (left) and front view (right) of a Krueger flap on a Boeing 737.

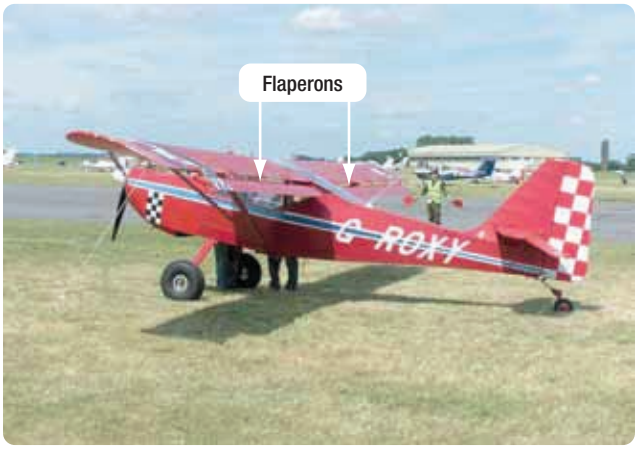


Figure 1-23. Flaperons on a Skystar Kitfox MK 7.

SLATS

Another leading-edge device which extends wing camber is a slat. Slats can be operated independently of the flaps with their own switch in the cockpit. Slats not only extend out of the leading edge of the wing increasing camber and lift, but most often, when fully deployed leave a slot between their trailing edges and the leading edge of the wing. (***Figure 1-24***) This increases the angle of attack at which the wing will maintain its laminar airflow, resulting in the ability to fly the aircraft slower and still maintain control.

SLOTS

A fixed device mounted to extend the leading edge of the wing forward and downward is known as a slot or cuff. (***Figure 1-25***) It essentially increases the camber of the wing and allows the aircraft to fly at slower speeds and higher angles of attack. Moreover, slots reduce the stall speed of the aircraft by mixing high speed air flow exiting the slot with boundary layer air. The result is a delay in boundary layer separation. However, slots

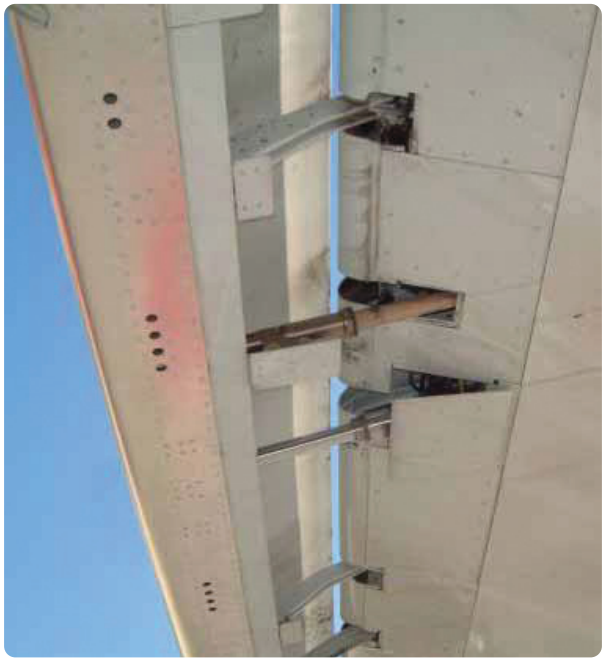


Figure 1-24. Air passing through the slot aft of the slat promotes boundary layer airflow on the upper surface at high angles of attack.



Figure 1-25. A leading edge slot on a STOL aircraft.

increase drag. The benefits of good low-speed handling characteristics when weighed against the increased drag that a slot causes at higher speeds limits the use of slots.

Full-span slots span the full wing from root to tip. They are commonly used on STOL (short takeoff and landing) aircraft. Partial-span slots are positioned on the outboard section of the wing leading edge. This increases the angle of attack at which the outboard wing stalls and ensures that the wing root stalls first. When the wing root stalls first, stall characteristics are docile. Recovery is easier because the partial-span slots maintain air flow over the ailerons during the stall.

ELEVONS AND RUDDERVATORS

Elevons perform the combined functions of the ailerons and the elevator. (***Figure 1-26***) They are typically used on aircraft that have no true separate empennage such as a delta wing or flying wing aircraft.

They are installed on the trailing edge of the wing. When moved in the same direction, the elevons cause a pitch adjustment. When moved in opposite directions, the aircraft rolls. Elevons may also move differentially in the same direction causing adjustments to roll and pitch. The control yoke or stick activated elevon movement through a mechanical or electronic mixing device. A ruddervator combines the action of the rudder and elevator. (***Figure 1-27***)

This is possible on aircraft with V–tail empennages where the traditional horizontal and vertical stabilizers do not exist. Instead, two stabilizers angle upward and outward from the aft fuselage in a "V" configuration. Each contains a movable ruddervator built into the trailing edge. Movement of the ruddervators can alter the movement of the aircraft around the horizontal and/or vertical axis.

DRAG INDUCING DEVICES

SPOILERS

Spoilers are unique in that they may be fully deployed on both wings to act as speed brakes. The reduced lift and increased drag can quickly reduce the speed of the aircraft in flight. Spoilers are sometimes called lift dumpers.

SPEED BRAKES

Dedicated speed brake panels similar to flight spoilers in construction can be found on the upper surface of the wing trailing edge of heavy and high-performance aircraft. They are designed specifically to increase drag and reduce the speed of the aircraft when deployed. These speed brake panels do not operate differentially with the ailerons at low speed like the spoilers.

A speed brake control lever in the cockpit can deploy all spoiler and speed brake surfaces fully when operated. Often, speed brakes surfaces are rigged to deploy on the ground automatically when engine thrust reversers are activated. The location of speed brake panels is visible in ***Figure 1-7***.

HIGH SPEED FLIGHT

SPEED OF SOUND

Sound, in reference to aeroplanes and their movement through the air, is nothing more than pressure disturbances in the air. It is like dropping a rock in the water and watching the waves flow out from the center. As an aeroplane flies through the air, every point on the aeroplane that causes a disturbance creates sound energy in the form of pressure waves. These pressure waves flow away from the aeroplane at the speed of sound, which at standard day temperature of 59 °F, is 761 mph.

The speed of sound in air changes with temperature, increasing as temperature increases. ***Figure 1-28*** shows how the speed of sound changes with altitude.

MACH NUMBER, SUBSONIC, TRANSONIC AND SUPERSONIC FLIGHT

In high-speed flight and/or high-altitude flight, the measurement of speed is expressed in terms of a "Mach number" - the ratio of the true airspeed of the aircraft to the speed of sound in the same atmospheric conditions. An aircraft traveling at the speed of sound is traveling at Mach 1.0.

Aircraft speed regimes are defined approximately as follows:

- Subsonic - Mach numbers below 0.75
- Transonic - Mach numbers from 0.75 to 1.20
- Supersonic - Mach numbers from 1.20 to 5.00
- Hypersonic - Mach numbers above 5.00



Figure 1-26. Elevons.



Figure 1-27. Ruddervator.

When an aeroplane is flying at subsonic speed, all of the air flowing around the aeroplane is at a velocity of less than the speed of sound (known as Mach 1). Keep in mind that the air accelerates when it flows over certain parts of the aeroplane, like the top of the wing, so an aeroplane flying at 500 mph could have air over the top of the wing reach a speed of 600 mph. How fast an aeroplane can fly and still be considered in subsonic flight varies with the design of the wing, but as a Mach number, it will typically be just over Mach 0.8. When an aeroplane is flying at transonic speed, part of the aeroplane is experiencing subsonic airflow and part is experiencing supersonic airflow.

Over the top of the wing the velocity of the air will reach Mach 1 and a shock wave will form. The shock wave forms 90 degrees to the airflow approximately halfway between the leading and trailing edge of the wing. It is known as a normal shock wave. Stability problems can be encountered during transonic flight, because the shock wave can cause the airflow to separate from the wing. The shock wave also causes the center of lift to shift aft, causing the nose to pitch down. The speed at which the shock wave forms is known as the critical Mach number.

Altitude in Feet	Temperature (°F)	Speed of Sound (mph)
0	59.00	761
1 000	55.43	758
2 000	51.87	756
3 000	48.30	753
4 000	44.74	750
5 000	41.17	748
6 000	37.60	745
7 000	34.04	742
8 000	30.47	740
9 000	26.90	737
10 000	23.34	734
15 000	5.51	721
20 000	–12.32	707
25 000	–30.15	692
30 000	–47.98	678
35 000	–65.82	663
*36 089	–69.70	660
40 000	–69.70	660
45 000	–69.70	660
50 000	–69.70	660
55 000	–69.70	660
60 000	–69.70	660
65 000	–69.70	660
70 000	–69.70	660
75 000	–69.70	660
80 000	–69.70	660
85 000	–64.80	664
90 000	–56.57	671
95 000	–48.34	678
100 000	–40.11	684

*Altitude at which temperature stops decreasing

Figure 1-28. Altitude and temperature versus speed of sound.

When an aeroplane is flying at supersonic speed, the entire aeroplane is experiencing supersonic airflow. At this speed, the shock wave which formed on top of the wing during transonic flight has moved all the way aft and has attached itself to the wing trailing edge. Supersonic speed is from Mach 1.20 to 5.0. If an aeroplane flies faster than Mach 5, it is said to be in hypersonic flight.

SHOCK WAVE

Sound coming from an aeroplane is the result of the air being disturbed as the aeroplane moves through it, and the resulting pressure waves that radiate out from the

source of the disturbance. For a slow moving aeroplane, the pressure waves travel out ahead of the aeroplane, traveling at the speed of sound. When the speed of the aeroplane reaches the speed of sound, however, the pressure waves (sound energy) cannot get away from the aeroplane. At this point the sound energy starts to pile up, initially on the top of the wing, and eventually attaching itself to the wing leading and trailing edges. This piling up of sound energy is called a shock wave. If the shock waves reach the ground, and cross the path of a person, they will be heard as a sonic boom. ***Figure 1-29A*** shows a wing in slow speed flight, with many disturbances on the wing generating sound pressure waves that are radiating outward. ***Figure 1-29B*** is the wing of an aeroplane in supersonic flight, with the sound pressure waves piling up toward the wing leading edge.

CRITICAL MACH NUMBER

While flights in the transonic and supersonic ranges are common occurrences for military aircraft, civilian jet aircraft normally operate in a cruise speed range of Mach 0.7 to Mach 0.90.

The speed of an aircraft in which airflow over any part of the aircraft or structure under consideration first reaches (but does not exceed) Mach 1.0 is termed "critical Mach number" or "Mach Crit." Thus, critical Mach number is the boundary between subsonic and transonic flight and is largely dependent on the wing and airfoil design. Critical Mach number is an important point in transonic flight. When shock waves form on the aircraft, airflow separation followed by buffet and aircraft control difficulties can occur. Shock waves, buffet (airflow becomes unsmooth), and airflow separation take place above critical Mach number. A jet aircraft typically is most efficient when cruising at or near its critical Mach number.

At speeds 5–10 percent above the critical Mach number, compressibility effects begin. Drag begins to rise sharply. Associated with the "drag rise" are buffet, trim and stability changes, and a decrease in control surface effectiveness. This is the point of "drag divergence." (***Figure 1-30***)

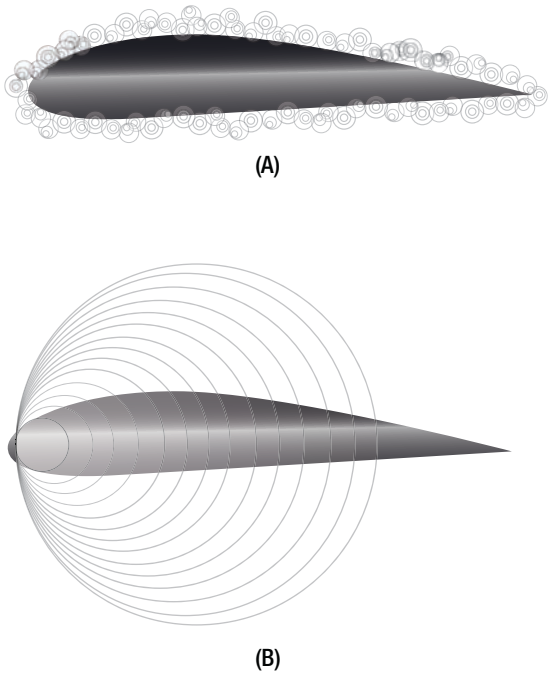


Figure 1-29. Sound energy in subsonic and supersonic flight.

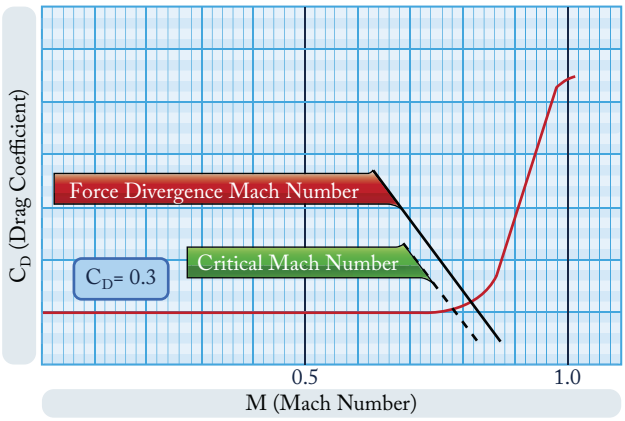


Figure 1-30. Critical Mach.

ROTARY WING AERODYNAMICS

TERMINOLOGY

ROTARY WING AIRCRAFT CONFIGURATION

The structures of the rotary wing aircraft are designed to give rotary wing aircraft its unique flight characteristics. One of the differences between a rotary wing and a fixed-wing aircraft is the main source of lift. The fixed-wing aircraft derives its lift from a fixed airfoil surface while the rotary wing aircraft derives lift from a rotating airfoil called the rotor. Changing the

angle of attack of the rotating airfoils (blades) increases or decreases lift, respectively raising or lowering the aircraft. Tilting the rotor plane of rotation causes the aircraft to move horizontally.

The typical configurations of rotary wing aircraft are:

- Autogyro - an aircraft with a free-spinning horizontal rotor that turns due to passage of air upward through the rotor. This air motion is created from forward motion of the aircraft resulting from either a tractor or pusher configured engine/propeller design. (***Figure 1-31***)
- Single rotor helicopter - An aircraft with a single horizontal main rotor that provides both lift and direction of travel is a single rotor helicopter. A secondary rotor mounted vertically on the tail counteracts the rotational force (torque) of the main rotor to correct yaw of the fuselage. (***Figure 1-32***)
- Dual rotor helicopter - An aircraft with two horizontal rotors that provide both the lift and directional control is a dual rotor helicopter. The rotors are counterrotating to balance the aerodynamic torque and eliminate the need for a separate antitorque system. (***Figure 1-33***)



Figure 1-31. An autogyro.

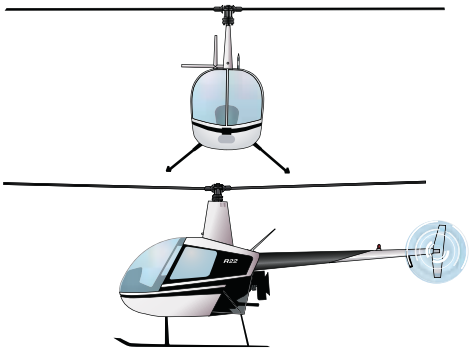


Figure 1-32. Single rotor helicopter.

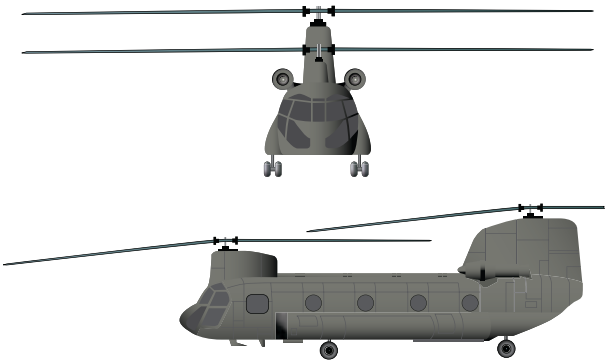


Figure 1-33. Dual rotor helicopter.

FLIGHT CONDITIONS

Hovering Flight

During hovering flight, a helicopter maintains a constant position over a selected point, usually a few feet above the ground. For a helicopter to hover, the lift and thrust produced by the rotor system act straight up and must equal the weight and drag, which act straight down. (***Figure 1-34***) While hovering, the amount of main rotor thrust can be changed to maintain the desired hovering altitude. This is done by changing the angle of incidence of the rotor blades and hence the angle of attack of the main rotor blades. Changing the angle of attack changes the drag on the rotor blades, and the power delivered by the engine must change as well to keep the rotor speed constant.



Figure 1-34. To maintain a hover at a constant altitude, enough lift and thrust must be generated to equal the weight of the helicopter and the drag produced by the rotor blades.

The weight that must be supported is the total weight of the helicopter and its occupants. If the amount of lift is greater than the actual weight, the helicopter accelerates upwards until the lift force equals the weight gain altitude; if thrust is less than weight, the helicopter accelerates downward. When operating near the ground, the effect of the closeness to the ground

changes this response. An important consequence of producing thrust is torque. Newton's Third Law states that for every action there is an equal and opposite reaction. Therefore, as the engine turns the main rotor system in a counterclockwise direction, the helicopter fuselage tends to turn clockwise. This tendency for the fuselage to rotate is called torque. The amount of torque is directly related to the amount of engine power being used to turn the main rotor system. The greater the engine power, the greater the torque effect. The force that compensates for torque and provides for directional control can be produced by various means. The defining factor is dictated by the design of the helicopter, some of which do not have a torque issue. Single main rotor designs typically have an auxiliary rotor located on the end of the tail boom (***Figure 1-32***).

This auxiliary rotor, generally referred to as a tail rotor, produces thrust in the direction opposite the torque reaction developed by the main rotor. A pilot can vary the amount of thrust produced by the tail rotor in relation to the amount of torque produced by the engine. As the engine supplies more power to the main rotor, the tail rotor must produce more thrust to overcome the increased torque effect. Other methods of compensating for torque and providing directional control include the Fenestron® tail rotor system, an SUD Aviation design that employs a ducted fan enclosed by a shroud. Another design, called NOTAR®, a McDonald Douglas design with no tail rotor, employs air directed through a series of slots in the tail boom, with the balance exiting through a 90° duct located at the rear of the tail boom. (***Figure 1-35***)

Basically, these units do the same things, regardless of the type of helicopter on which they are installed; however, the operation of the control system varies greatly by helicopter model.

Vertical Flight

Hovering is actually an element of vertical flight. Increasing the angle of attack of the rotor blades (pitch) while keeping their rotation speed constant generates additional lift and the helicopter ascends. Decreasing the pitch causes the helicopter to descend. In a no wind condition, when lift and thrust are less than weight and drag, the helicopter descends vertically. If lift and thrust are greater than weight and drag, the helicopter ascends vertically. (***Figure 1-36***)

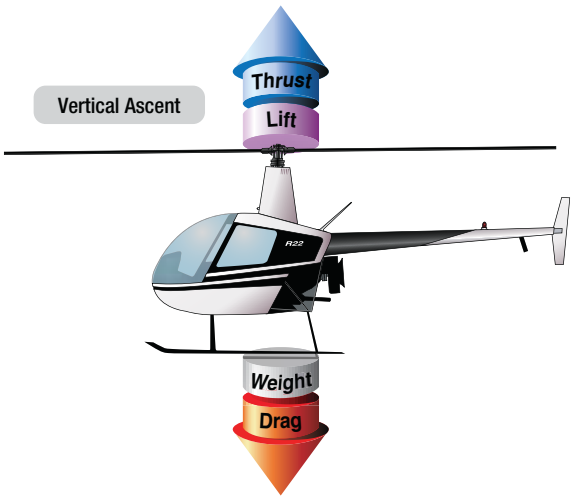


Figure 1-36. To ascend vertically, more lift and thrust must be generated to overcome the forces of weight and drag.



Figure 1-35. Aerospatiale Fenestron tail rotor system (left) and the McDonnell Douglas NOTAR® System (right).

Forward Flight

In steady forward flight with no change in airspeed or vertical speed, the four forces of lift, thrust, drag, and weight must be in balance. Once the tip-path plane is tilted forward, the total lift-thrust force is also tilted forward. This resultant lift-thrust force can be resolved into two components - lift acting vertically upward and thrust acting horizontally in the direction of flight. In addition to lift and thrust, there is weight (the downward acting force) and drag (the force opposing the motion of an airfoil through the air). (***Figure 1-37***)

In straight-and-level (constant heading and at a constant altitude), unaccelerated forward flight, lift equals weight and thrust equals drag. If lift exceeds weight, the helicopter accelerates vertically until the forces are in balance; if thrust is less than drag, the helicopter slows until the forces are in balance. As the helicopter moves forward, it begins to lose altitude because lift is lost as thrust is diverted forward. However, as the helicopter begins to accelerate, the rotor system becomes more efficient due to the increased airflow.

The result is excess power over that which is required to hover. Continued acceleration causes an even larger increase in airflow through the rotor disk and more excess power. In order to maintain unaccelerated flight, the pilot must not make any changes in power or in cyclic movement. Any such changes would cause the helicopter to climb or descend. Once straight-and-level flight is obtained, the pilot should make note of the power (torque setting) required and not make major adjustments to the flight controls.

Autorotation

Autorotation is the state of flight in which the main rotor system of a helicopter is being turned by the action of air moving up through the rotor rather than engine power driving the rotor. In normal, powered flight, air is drawn into the main rotor system from above and exhausted downward, but during autorotation, air moves up into the rotor system from below as the helicopter descends. Autorotation is permitted mechanically by a freewheeling unit, which is a special clutch mechanism that allows the main rotor to continue turning even if the engine is not running. If the engine fails, the freewheeling unit automatically disengages the engine from the main rotor allowing the main rotor to rotate freely. It is the means by which a helicopter can be landed safely in the event of an engine failure; consequently, all helicopters must demonstrate this capability in order to be certificated. (***Figure 1-38***)

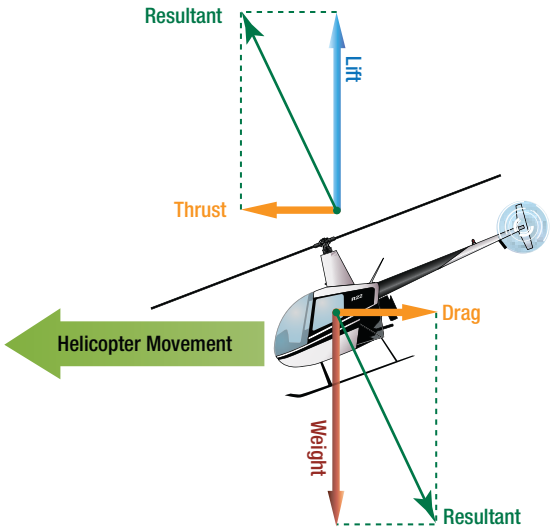


Figure 1-37. The power required to maintain a straight-and-level flight and a stabilized airspeed.

OPERATION AND EFFECT OF ROTORCRAFT CONTROLS

FLIGHTS CONTROLS CONFIGURATION

The flight controls of a helicopter differ slightly from those found in an aircraft. The control units located in the flight deck of all helicopters are very nearly the same. There are three major controls in a helicopter that the pilot must use during flight. They are the collective pitch control, cyclic pitch control, and antitorque pedals or tail rotor control. In addition to these major controls, the pilot must also use the throttle control, which is mounted directly to the collective pitch control in order to fly the helicopter. (***Figure 1-39***)

Swash Plate Assembly

The purpose of the swash plate is to transmit control inputs from the collective and cyclic controls to the main rotor blades. It consists of two main parts: the stationary swash plate and the rotating swash plate. (***Figure 1-40***)

The stationary swash plate is mounted around the main rotor mast and connected to the cyclic and collective controls by a series of pushrods. It is restrained from rotating by an antidrive link but is able to tilt in all directions and move vertically. The rotating swash plate is mounted to the stationary swash plate by a uniball sleeve. It is connected to the mast by drive

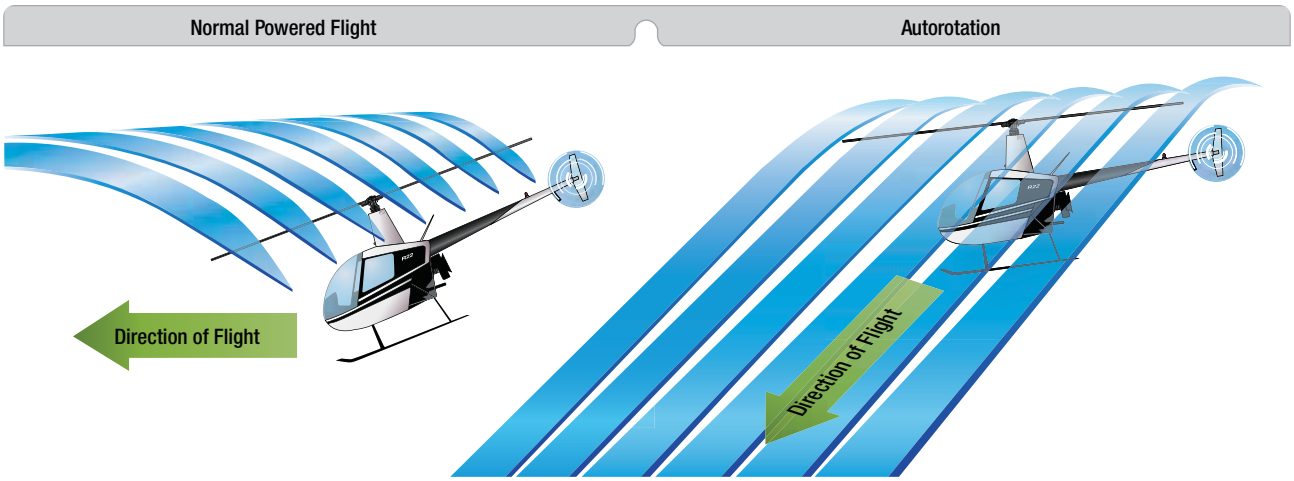


Figure 1-38. During an autorotation, the upward flow of relative wind permits the main rotor blades to rotate at their normal speed. Ineffect, the blades are "gliding" in their rotational plane.

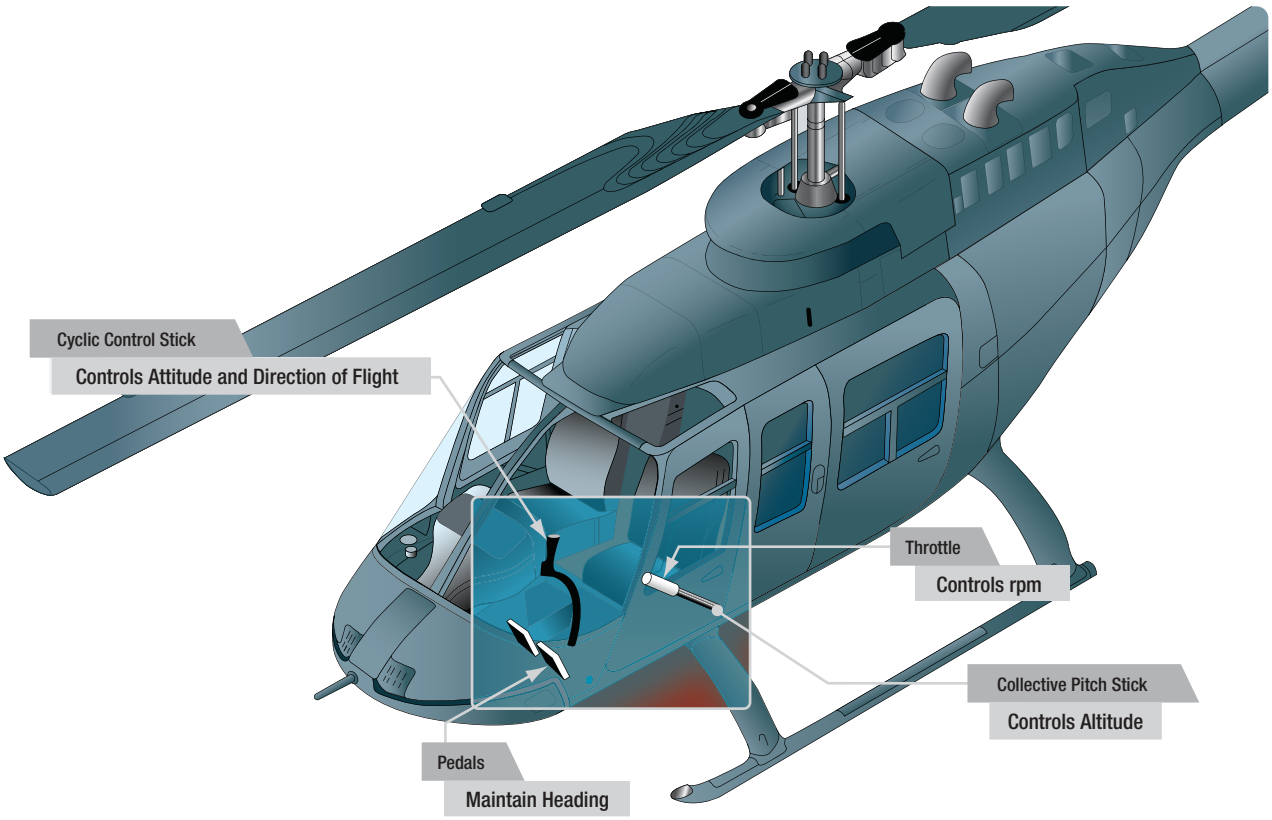


Figure 1-39. Controls of a helicopter and the principal function of each.

links and is allowed to rotate with the main rotor mast. Both swash plates tilt and slide up and down as one unit. The rotating swash plate is connected to the pitch horns by the pitch links.

Collective Pitch Control

The collective pitch control is located on the left side of the pilot's seat and is operated with the left hand. The collective is used to make changes to the pitch angle of all the main rotor blades simultaneously, or collectively, as the name implies. As the collective pitch control is raised, there is a simultaneous and equal increase in pitch angle of all main rotor blades; as it is lowered, there is a simultaneous and equal decrease in pitch angle. This is done through a series of mechanical linkages, and the amount of movement in the collective lever determines the amount of blade pitch change. (***Figure 1-41***) An adjustable friction control helps prevent inadvertent collective pitch movement.

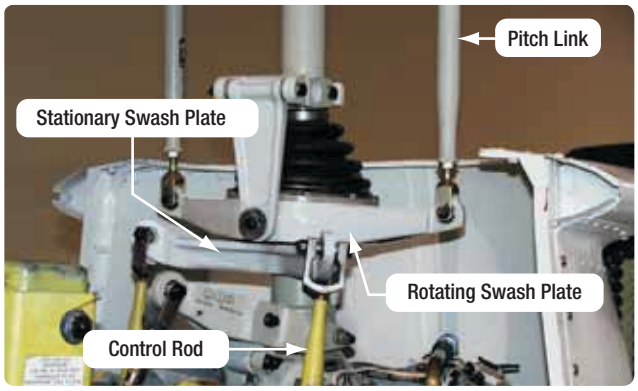


Figure 1-40. Stationary and rotating swash plate.

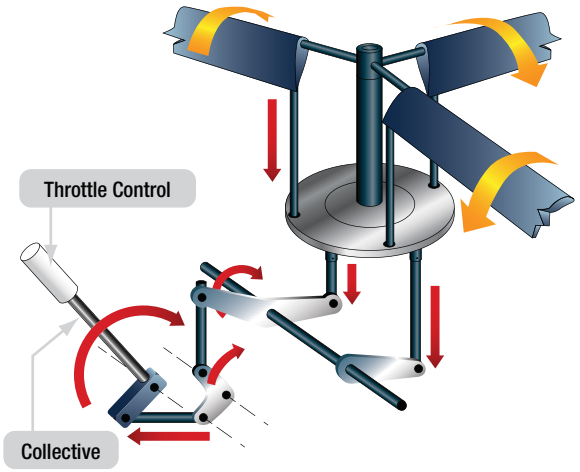


Figure 1-41. The collective changes the pitch of all of the rotor blades simultaneously and by the same amount, thereby increasing or decreasing lift.

Throttle Control

The function of the throttle is to regulate engine rpm. If the correlator or governor system does not maintain the desired rpm when the collective is raised or lowered, or if those systems are not installed, the throttle must be moved manually with the twist grip to maintain rpm. The throttle control is much like a motorcycle throttle, and works almost the same way; twisting the throttle to the left increases rpm, twisting the throttle to the right decreases rpm. (***Figure 1-42***)

Governor/Correlator

A governor is a sensing device that senses rotor and engine rpm and makes the necessary adjustments in order to keep rotor rpm constant. Once the rotor rpm is set in normal operations, the governor keeps the rpm constant, and there is no need to make any throttle adjustments. A correlator is a mechanical connection between the collective lever and the engine throttle. When the collective lever is raised, power is automatically increased



Figure 1-42. A twist grip throttle is usually mounted on the end of the collective lever. The throttles on some turbine helicopters are mounted on the overhead panel or on the floor in the cockpit.

and when lowered, power is decreased. This system maintains rpm close to the desired value, but still requires adjustment of the throttle for fine tuning. Governors are common on all turbine helicopters (as it is a function of the fuel control system of the turbine engine), and used on some piston-powered helicopters. Some helicopters do not have correlators or governors and require coordination of all collective and throttle movements.

Cyclic Pitch Control

The cyclic pitch control is mounted vertically from the cockpit floor, between the pilot's legs or, in some models, between the two pilot seats. (***Figure 1-43***) This primary flight control allows the pilot to fly the helicopter in any horizontal direction; fore, aft, and sideways (***Figure 1-44***). The total lift force is always perpendicular to the tip-path place of the main rotor. The purpose of the cyclic pitch control is to tilt the tip-path plane in the direction of the desired horizontal direction. The cyclic control changes the direction of this force and controls the attitude and airspeed of the helicopter.

The rotor disk tilts in the same direction the cyclic pitch control is moved. If the cyclic is moved forward, the rotor disk tilts forward; if the cyclic is moved aft, the disk tilts aft, and so on. Because the rotor disk acts like a gyro, the mechanical linkages for the cyclic control rods are rigged in such a way that they decrease the pitch angle of the rotor blade approximately 90° before it reaches the direction of cyclic displacement, and increase the pitch angle of the rotor blade approximately 90° after it passes the direction of displacement. An increase in pitch angle increases angle of attack; a decrease in pitch angle decreases angle of attack.



Figure 1-43. The cyclic pitch control may be mounted vertically between the pilot's knees or on a teetering bar from a single cyclic located in the center of the helicopter. The cyclic can pivot in all directions.

For example, if the cyclic is moved forward, the angle of attack decreases as the rotor blade passes the right side of the helicopter and increases on the left side. This results in maximum downward deflection of the rotor blade in front of the helicopter and maximum upward deflection behind it, causing the rotor disk to tilt forward.

Antitorque Pedals

The antitorque pedals are located on the cabin floor by the pilot's feet. They control the pitch and, therefore, the thrust of the tail rotor blades. (***Figure 1-45***)

Newton's Third Law applies to the helicopter fuselage and how it rotates in the opposite direction of the main rotor blades unless counteracted and controlled. To make flight possible and to compensate for this torque, most helicopter designs incorporate an antitorque rotor or tail rotor. The antitorque pedals allow the pilot to control the pitch angle of the tail rotor blades which in forward flight puts the helicopter in longitudinal trim and while at a hover, enables the pilot to turn the helicopter 360°. The antitorque pedals are connected to the pitch change mechanism on the tail rotor gearbox and allow the pitch angle on the tail rotor blades to be increased or decreased.

Helicopters that are designed with tandem rotors do not have an antitorque rotor (***Figure 1-33***). These helicopters are designed with both rotor systems rotating in opposite directions to counteract the torque, rather than using a tail rotor. Directional antitorque pedals are used for directional control of the aircraft while in flight, as well as while taxiing with the forward gear off the ground. With the right pedal displaced forward, the forward rotor disk tilts to the right, while the aft rotor disk tilts to the left. The opposite occurs when the left pedal is pushed forward; the forward rotor disk inclines to the left, and the aft rotor disk tilts to the right. Differing combinations of pedal and cyclic application can allow the tandem rotor helicopter to pivot about the aft or forward vertical axis, as well as pivoting about the center of mass.

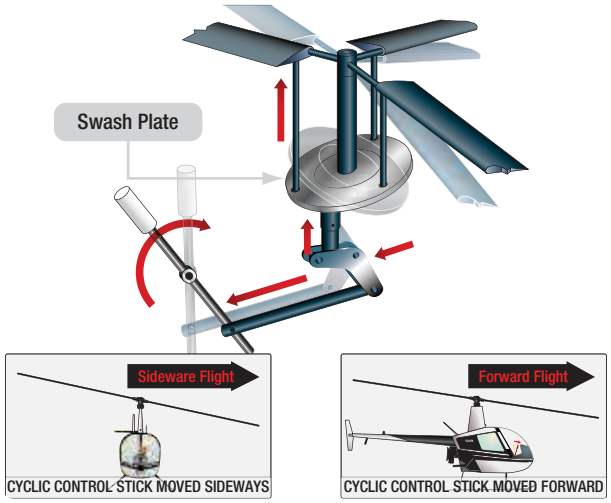


Figure 1-44. The cyclic changes the angle of the swash plate which changes the plane of rotation of the rotor blades. This moves the aircraft horizontally in any direction depending on the positioning of the cyclic.



Figure 1-45. Antitorque pedals compensate for changes in torque and control heading in a hover.

Question: 1-1
Around what three axis do the primary flight controls move an aeroplane?

Question: 1-2
Movement of the _____ in flight causes the aircraft to roll.

Question: 1-3
The _____ is the primary flight control that moves the aircraft around the horizontal or lateral axis.

Question: 1-4
An _____ tab is used to maintain the speed of an aircraft since it assists in maintaining the selected pitch.

Question: 1-5
_____ and _____ are lowered to increase the camber of the wings and provide greater lift and control at slow speeds.

Question: 1-6
Elevons perform the combined functions of the _____ and the _____.

Question: 1-7
At which speed a shock wave is generated during transonic flight?

Question: 1-8
Which are the three main flight controls of a helicopter?

ANSWERS

Answer: 1-1
Lateral or Horizontal.
Longitudal.
Vertical.

Answer: 1-2
ailerons.

Answer: 1-3
elevator.

Answer: 1-4
Elevator.

Answer: 1-5
Flaps, slats.

Answer: 1-6
Ailerons, Elevator.

Answer: 1-7
Critical Mach Number

Answer: 1-8
Collective pitch control, cyclic pitch control, and antitorque pedals or tail rotor control



PART-66 SYLLABUS **LEVELS**

CERTIFICATION CATEGORY → **B2**

Sub-Module 02

STRUCTURES — GENERAL CONCEPTS

Knowledge Requirements

13.2 - Structures — General Concepts

(a) Fundamentals of structural systems. **1**

(b) Zonal and station identification systems; Electrical bonding; Lightning strike protection provision. **2**

FUNDAMENTALS OF STRUCTURAL SYSTEMS

STRUCTURAL CLASSIFICATION

Aircraft structure is divided into three categories for the purposes of assessing damage and the application of repair protocol that are suitable for the structure under consideration. Manufacturer manuals designate which category a structure falls under and the technician is required to repair and maintain that structure in accordance with rules specified for the category under which it falls. The three categories for structure are: primary, secondary and tertiary.

PRIMARY STRUCTURE

Primary structure is any portion of the aircraft structure that, if it fails, on the ground or in flight, would likely cause any of the following:

- A loss of control of the aircraft
- Catastrophic structural collapse
- Injury to occupants
- Power unit failure
- Unintentional operation/inability to operate a service

Some examples of primary structure are wings spars, engine mounts, fuselage frames, and main floor structural members. Within the primary structure are elements called principle structural elements (PSE's). These elements are those which carry flight, ground and pressurization loads. Primary structure may also be represented as a structurally significant item or SSI. These elements are specified in a supplemental structural inspection document. Due to their structural importance, they may require special inspection and have specific repair limitations.

SECONDARY STRUCTURE

Secondary structure is all non-primary structure portions of the aircraft which have integral structural importance and strength exceeding design requirements. These structures weakening without risk of failure such as those described for primary structure. Prominent examples of secondary structure are wing ribs, fuselage stringers and specified sections of the aircraft skin.

TERTIARY STRUCTURE

Tertiary structure is the remaining structure. Tertiary structures are lightly stressed structures that are fitted to the aircraft for various reasons. Fairings, fillets, various support brackets, etc. are examples of tertiary structure.

STRUCTURAL STRESSES

Aircraft structural members are designed to carry a load or to resist stress. In designing an aircraft, every square inch of wing and fuselage, every rib, spar, and even each metal fitting must be considered in relation to the physical characteristics of the material of which it is made. Every part of the aircraft must be planned to carry the load to be imposed upon it. The determination of such loads is called stress analysis. Although planning the design is not the function of the aircraft technician, it is, nevertheless, important that the technician understand and appreciate the stresses involved in order to avoid changes in the original design through improper repairs. Stress is a material's internal resistance, or counterforce, that opposes deformation. The degree of deformation of a material is strain. When a material is subjected to a load or force, that material is deformed, regardless of how strong the material is or how light the load is.

There are five major stresses (***Figure 2-1***) to which all aircraft are subjected:

- Tension
- Compression
- Torsion
- Shear
- Bending

Strength or resistance to the external loads imposed during operation may be the principal requirement in certain structures. However, there are numerous other characteristics in addition to designing to control the five major stresses that engineers must consider. For example, cowling, fairings, and similar parts may not be subject to significant loads requiring a high degree of strength. However, these parts must have streamlined shapes to meet aerodynamic requirements, such as reducing drag or directing airflow.

A critical failure mechanism in aircraft structural elements is metal fatigue. Metal fatigue is experienced by a component or structural member when a load is repeatedly applied and released or applied and reversed. This cycling weakens the material over time even though the load applied may be well below that which causes damage in a single application. Over time, these load changes cause fatigue in the form of minute cracks in the metal structure. Each tiny, seemly inconsequential crack exposes new material to the elements. Aircraft structure is tested at the manufacturer to determine a limit not to be exceeded for an aircraft in service.

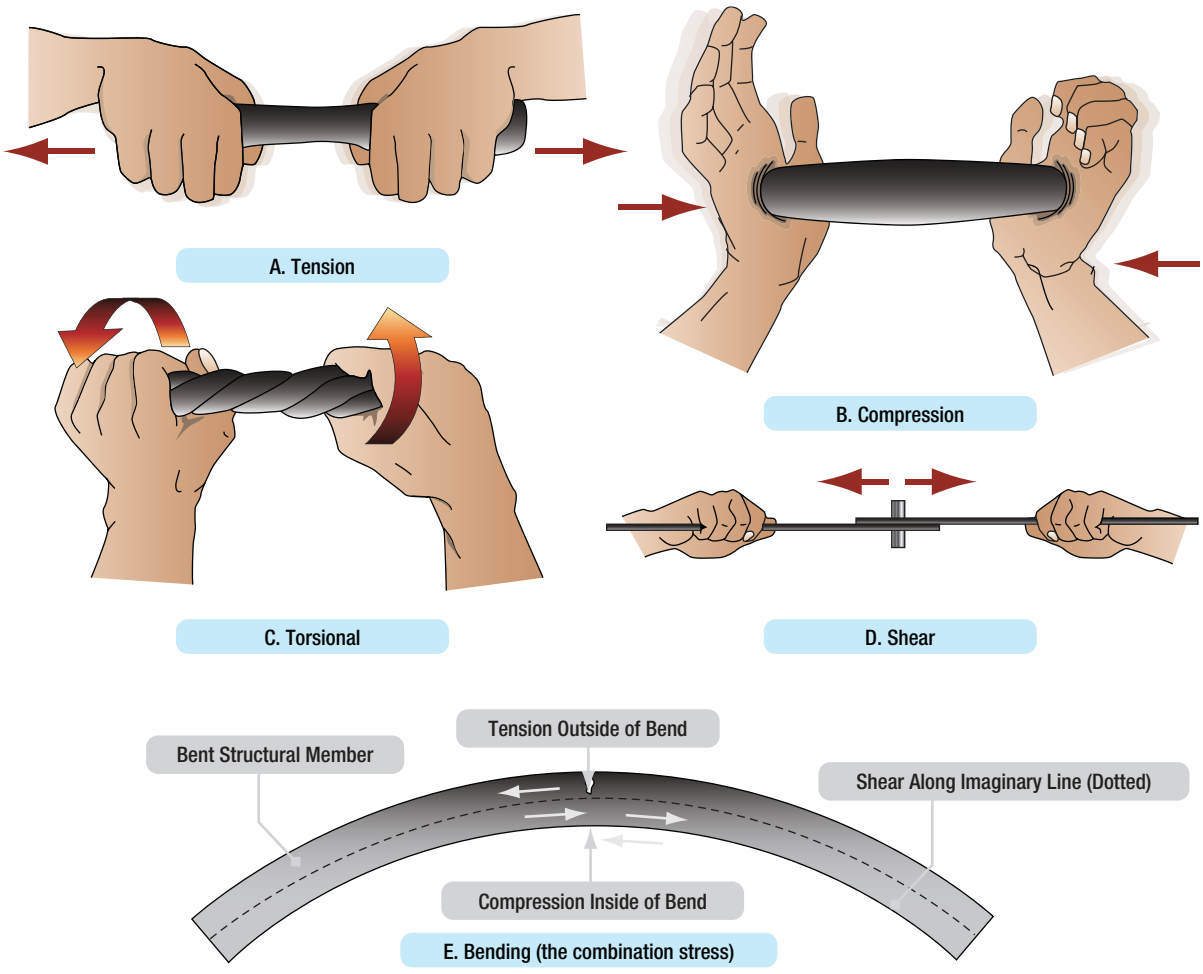


Figure 2-1. The five stresses that may act on an aircraft and its parts.

CONSTRUCTION METHODS

FUSELAGE

The fuselage is the main structure or body of the fixed-wing aircraft. It provides space for cargo, controls, accessories, passengers, and other equipment. In single-engine aircraft, the fuselage houses the powerplant. In multi-engine aircraft, the engines may be either in the fuselage, attached to the fuselage, or suspended from the wing structure.

There are two general types of fuselage construction:

- Truss: A truss is a rigid framework made up of members, such as beams, struts, and bars to resist deformation by applied loads. The truss-framed fuselage is generally covered with fabric. The truss-type fuselage frame is usually constructed of steel tubing welded together in such a manner that all members of the truss can carry both tension and compression loads. (***Figure 2-2***)
- Monocoque: The monocoque (single shell) fuselage (***Figure 2-3***) relies largely on the strength of the skin or covering to carry the primary loads. The design is called stressed skin and may be divided into monocoque and semimonocoque. Different portions of the same fuselage may belong to either of the two classes, but most modern aircraft are considered to be of semimonocoque type construction. (***Figure 2-4***)

Attached to the outside of the aircraft structure is the aircraft skin, be it stressed or not. Simple, light aircraft generally have skin made from sheet aluminum which is formed to fit, wrapped and riveted to the structural members. Larger, more complex and heavier aircraft used heavier material to form the aircraft skin. This is to transfer and carry the greater loads experience during high performance flight.

A simpler way to reinforce an area of skin on the aircraft which receives greater loads than can easily be carried by a single sheet of material is to create a doubler for that area. A doubler is simply a second, reinforcing layer of skin material used to strengthening the load carrying capacity of the skin. In addition to the structural members already mentioned, additional beams, floor structural members and various other reinforcement members are also used to construct an aircraft.

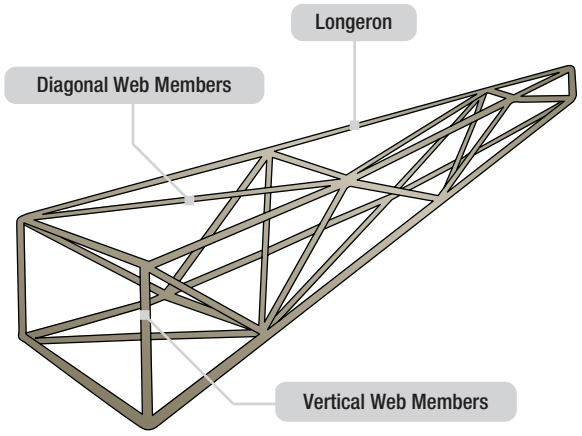


Figure 2-2. A truss-type fuselage. A Warren truss uses mostly diagonal bracing.

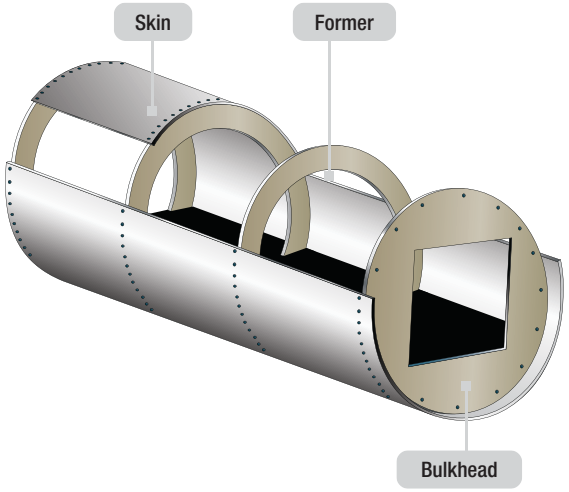


Figure 2-3. An airframe using monocoque construction.

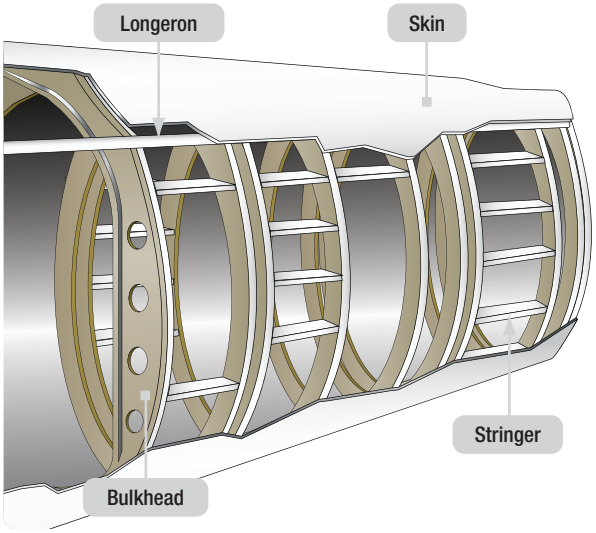


Figure 2-4. The most common airframe construction is semimonocoque.

A beam may be installed laterally or longitudinally. Beams typically support the floor of the flight deck and the passenger compartment. Struts and ties are also used in aircraft structure. A strut is a bar or rod shaped reinforcement designed to resist compression loads. A tie is a rod or beam designed to take a tensile load. Both are used as needed to reinforce the aircraft structure throughout the fuselage to carry the loads experienced.

WING, EMPENNAGE AND ENGINE ATTACHMENT

The wings, empennage and engines must be attached to the fuselage. The type of attachment varies with the aircraft design. Typically, special pins or bolts are used. Wings and empennage structure is often constructed with load carrying main members called spars. Attach lugs securely fitted to these spars mate with lugs that are fitted to strengthened sections of the fuselage and mounting pins or bolts are passed through both lugs and secured. ***Figure 2-5*** shows the internal fuselage structure of what is considered the center section of the horizontal stabilizer on a Boeing 737.

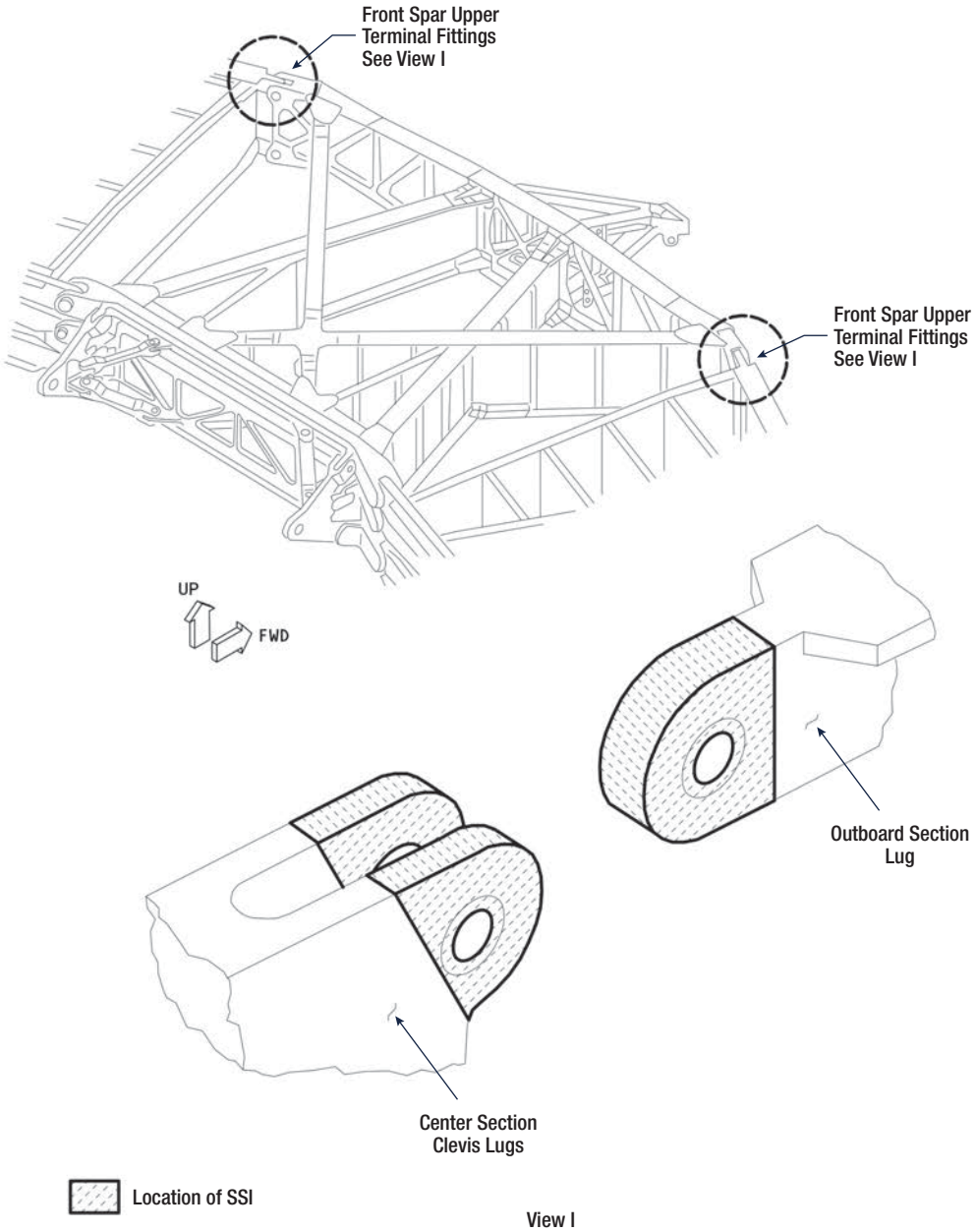


Figure 2-5. Horizontal stabilizer center section outboard front spar terminal fitting lugs.

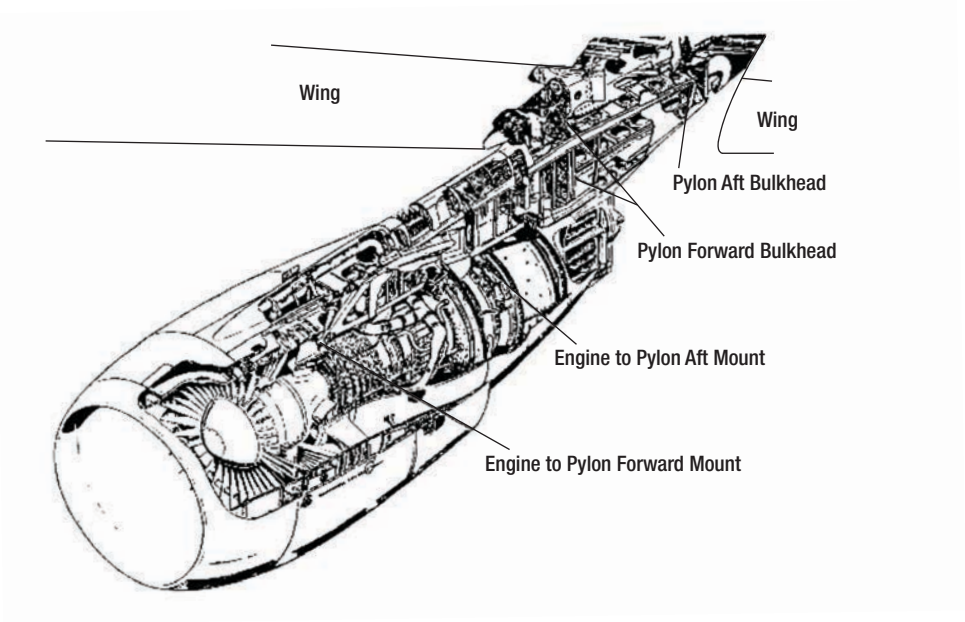


Figure 2-6. Turbofan engine pylon.

Engine attachments vary widely on aircraft depending on where the engines are located and the size and design of the aircraft and engine. A typical arrangement found on transport aircraft is to extend support structure forward and down from the wing spars. The structure is called a pylon.

Figure 2-6 is a rough cutaway drawing of a turbofan engine pylon. It is built to be very strong to support the engine. Attached to the pylon structure are engine mounts to which the engine is bolted or visa versa.

AIRCRAFT CONSTRUCTION

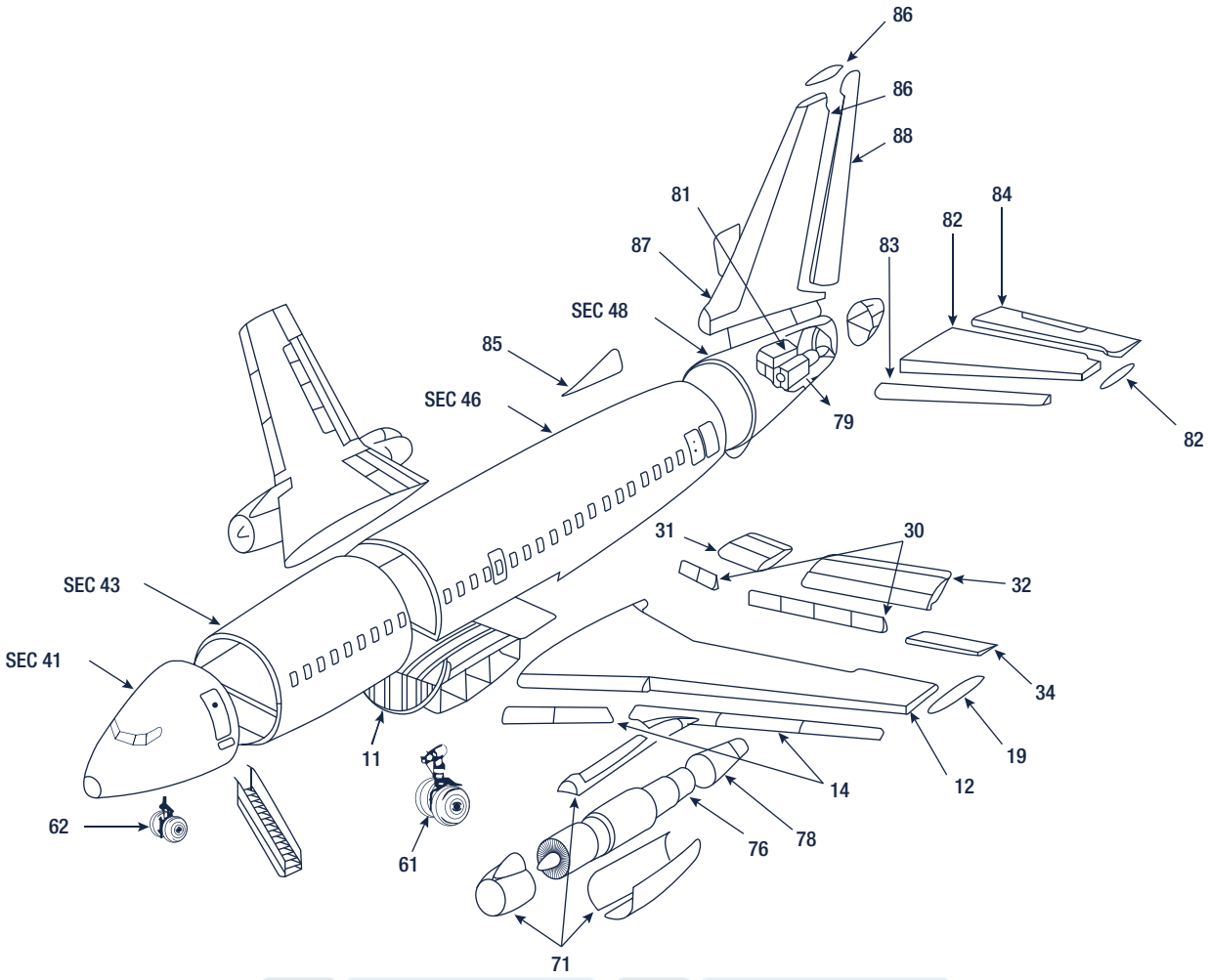
Large aircraft maintenance manuals contain specific instruction for the bonding of all materials and sections of the aircraft. ATA section 51 gives a descriptive overview of the aircraft structure and general rules followed in construction of airframe components and sections. A large aircraft fuselage is manufactured in sections that are then mated and fastened together. The structural sections of a Boeing 737 are shown in ***Figure 2-7***.

Sections 41, 43, and 48 comprised the pressurized portion of the fuselage. Section 48 is not pressurized but does supply the support structure for the vertical and horizontal stabilizer. It also contains a bay for installation of the auxiliary power unit. A rear pressure bulkhead separates body section 46 from body section 48.

SYSTEM INSTALLATION PROVISIONS

In addition to designing functioning support systems for operation of the aircraft, design engineers must also make the system components fit into the aircraft. Depending on the system and components, provisions for access and servicing must also be addressed. Items that receive regular maintenance such as filters, fluid level checks, bearing lubrication, etc. must be located so that technicians can easily access them. Line replaceable units (LRUs) must be able to be quickly uninstalled and installed. Aircraft maintenance is a significant expense for the operator. Anything that can be done to locate system components for easy access for maintenance saves time and lowers the cost of operating the aircraft.

Modern airliner designers often group the components of a various systems in a single bay for easy access. Air conditioning, for example, may have its several key components mounted next to each other in an air conditioning bay. The hydraulic reservoir, pumps and filters may all be located in a different bay or in the wheel well area. Avionics and electronics are frequently mounted in an avionics bay. Not only are the 'black boxes' easily accessible but environmental conditions can be better controlled than if the units were spread throughout the aircraft.



Struct	Title
10	Wing
11	Wing Stub
12	Wing, Outboard
14	Slats and Flaps, L.E.
19	Wing Tip
30	Spoilers
31	Flap, Inboard
32	Flap, Outboard
34	Aileron
35	Flap, Center
40	Body
41	Section 41
43	Section 43
46	Section 46
48	Section 48

Struct	Title
60	Landing Gear
61	Main Gear
62	Nose Gear
70	Powerplant
71	Cowling
78	Thrust Reverser, Tail Pipe
79	Auxiliary Power Unit
80	Empennage
81	Stabilizer Center Section
82	Stabilizer
83	Stabilizer L.E.
84	Stabilizer Elevator
85	Dorsal Fin
86	Fin
86	Fin Tip
87	Fin L.E.
88	Rudder

Figure 2-7. Aeroplane section numbers.

STATION NUMBERING

Even on small, light aircraft, a method of precisely locating each structural component is required. Various numbering systems are used to facilitate the location of specific wing frames, fuselage bulkheads, or any other structural members on an aircraft.

Most manufacturers use some system of station marking. For example, the nose of the aircraft may be designated "zero station" and all other stations are located at measured distances in inches behind the zero station. Thus, when a blueprint reads "fuselage frame station 137" that particular frame station can be located 137 inches behind the nose of the aircraft.

To locate structures to the right or left of the center line of an aircraft, a similar method is employed. Many manufacturers consider the center line of the aircraft to be a zero station from which measurements can be taken to the right or left to locate an airframe member. This is often used on the horizontal stabilizer and wings. The applicable manufacturer's numbering system and abbreviated designations or symbols should always be reviewed before attempting to locate a structural member. They are not always the same.

The following list includes location designations typical of those used by many manufacturers.

- Fuselage stations (Fus.Sta. or FS) are numbered in inches from a reference or zero point known as the reference datum. (***Figure 2-8***) The reference datum is an imaginary vertical plane at or near the nose of the aircraft from which all fore and aft distances are measured. The distance to a given point is measured in inches parallel to a center line extending through the aircraft from the nose through the center of the tail cone. Some manufacturers may call the fuselage station a body station, abbreviated BS.
- Buttock line or butt line (BL) is a vertical reference plane down the center of the aircraft from which measurements left or right can be made (***Figure 2-9***).
- Water line (WL) is the measurement of height in inches perpendicular from a horizontal plane usually located at the ground, cabin floor, or some other easily referenced location. (***Figure 2-10***)
- Aileron station (AS) is measured perpendicular from, and parallel to, the inboard edge of the aileron, perpendicular to the rear beam of the wing.
- Flap station (KS) is measured perpendicular to the rear beam of the wing and parallel to, and outboard from, the inboard edge of the flap.
- Nacelle station (NC or Nac. Sta.) is measured either forward of or behind the front spar of the wing and perpendicular to a designated water line.

In addition to the location stations listed above, other measurements are used, especially on large aircraft. Thus, there may be horizontal stabilizer stations (HSS), vertical stabilizer stations (VSS) or powerplant stations (PPS). (***Figure 2-11***)

In every case, the manufacturer;s terminology and station location system should be consulted before locating a point on a particular aircraft.

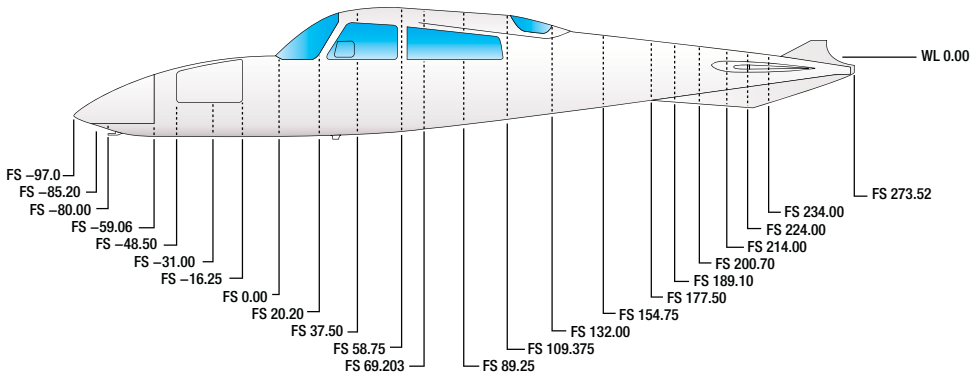


Figure 2-8. The various body stations relative to a single point of origin illustrated in inches or SI equivalent.

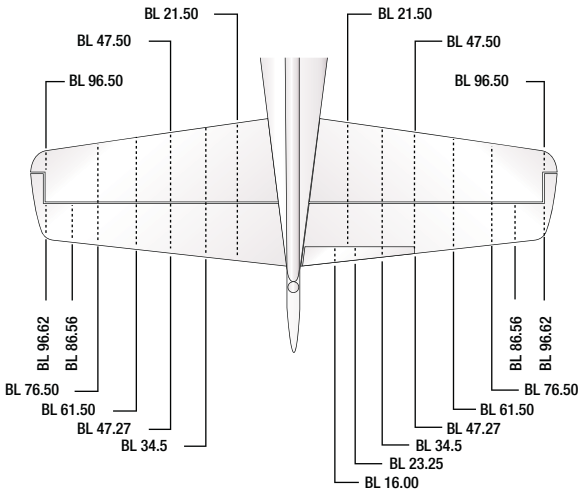


Figure 2-9. Butt line diagram of a horizontal stabilizer.

ZONAL IDENTIFICATION SYSTEM

Another method is used to facilitate the location of aircraft components on air transport aircraft. This involves dividing the aircraft into zones. Large areas or major zones are further divided into sequentially numbered zones and sub-zones. The digits of the zone number are reserved and indexed to indicate the location and type of system of which the component is a part. ***Figure 2-12*** illustrates these zones and sub-zones on a transport category aircraft.

ACCESS AND INSPECTION PANELS

Knowing where a particular structure or component is located on an aircraft needs to be combined with gaining access to that area to perform the required inspections or maintenance. To facilitate this, access and inspection

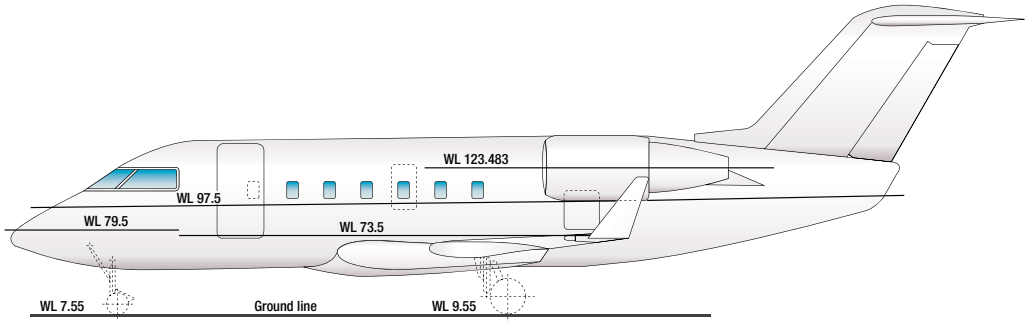


Figure 2-10. Water line diagram.

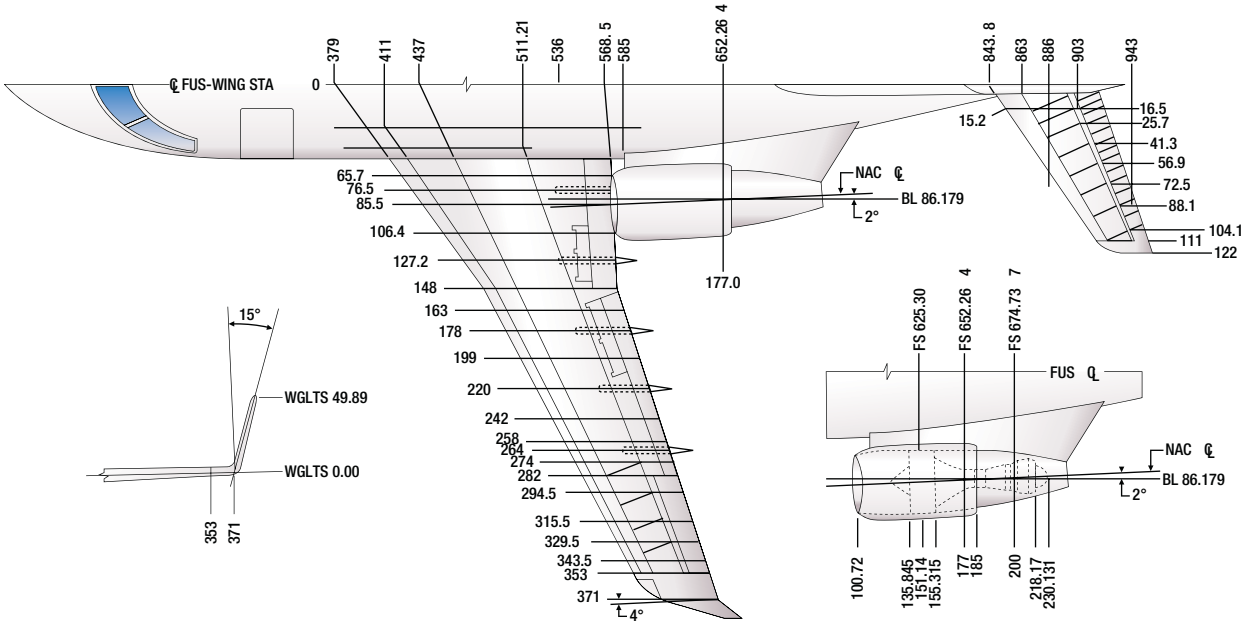


Figure 2-11. Wing stations are often referenced off the butt line, which bisects the center of the fuselage longitudinally. Horizontal stabilizer stations referenced to the butt line and engine nacelle stations are also shown.

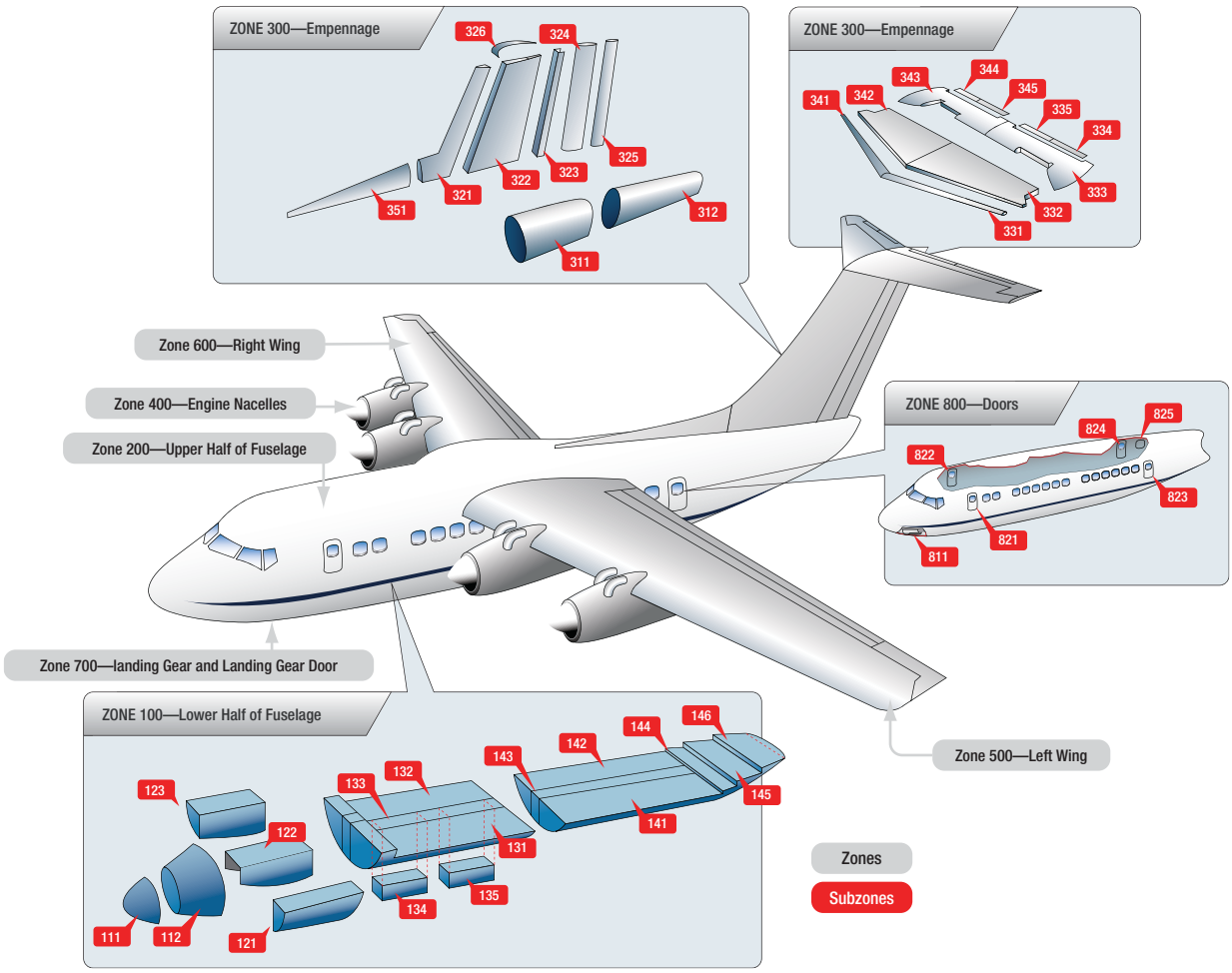


Figure 2-12. Large aircraft are divided into zones and sub-zones for identifying the location of various components.

panels are located on most surfaces of the aircraft. Small panels that are hinged or removable allow inspection and servicing. Large panels and doors allow components to be removed and installed, as well as human entry for maintenance purposes.

The underside of a wing, for example, sometimes contains dozens of small panels through which control cable components can be monitored and fittings greased. Various drains and jack points may also be on the underside of the wing. The upper surface of the wings typically have fewer access panels because a smooth surface promotes better laminar airflow, which causes lift. On large aircraft, walkways are sometimes designated on the wing upper surface to permit safe navigation by mechanics and inspectors to critical structures and components located along the wing's leading and trailing edges. Wheel wells and special component bays are places where numerous components and accessories are grouped together for easy maintenance access.

Panels and doors on aircraft are numbered for positive identification. On large aircraft, panels are usually numbered sequentially containing zone and sub-zone information in the panel number. Designation for a left or right side location on the aircraft is often indicated in the panel number. This could be with an 'L' or 'R' or panels on one side of the aircraft could be odd numbered and the other side even numbered.

The manufacturer's maintenance manual explains the panel numbering system and often has numerous diagrams and tables showing the location of various components and under which panel they may be found. Each manufacturer is entitled to develop its own panel numbering system.

LIGHTENING STRIKE PROTECTION AND BONDING

Precautions are taken to ensure safe and continuous operation of an aircraft should it happen to be struck by lightening. A single lightening strike may contain 100 000 amperes of current. It must not be allowed to build up or arc from one point on the structure to another. Aircraft use the predominantly aluminum structure as a ground path for operation of electrical devices. Most components are therefore mounted to structure or attached to the structure with bonding straps.

This ensures that all components are at the same potential level electrically and that equal, low resistance paths for current flow exist. (***Figure 2-13***) Not only are electrical components bonded to aircraft structure but different parts of the aircraft structure are bonded together as well. Hinged flight controls, for example, have a bond strap between the movable control surface and the main airframe structure.

As an aircraft flies throughout the air, its surface can become highly charged with static electricity. Static dischargers, or wicks, are installed on aircraft to reduce radio receiver interference. This interference is caused by corona discharge emitted from the aircraft as a result of precipitation static. Corona occurs in short pulses which produce noise at the radio frequency spectrum. Static dischargers are normally mounted on the trailing edges of the control surfaces, wing tips and the vertical stabilizer. They discharge precipitation static at points a critical distance away from avionics antennas where there is little or no coupling of the static to cause interference or noise.

Flexible and semi-flexible dischargers are attached to the aircraft structure by metal screws, rivets, or epoxy. The connections should be checked periodically for security. A resistance measurement from the mount to the airframe should not exceed 0.1 ohm. Inspect the condition of all static dischargers in accordance with manufacturer's instructions. ***Figure 2-14*** illustrates examples of static dischargers.

When lightning strikes an aircraft with all bonding devices in tact and working, there is no difference in potential from one part of the aircraft to another. The electrical energy dissipates over the entire surface of the aircraft and returns to the atmosphere through the static wicks. Note that all bonding straps should be inspected periodically to ensure that no potential is allowed to build so that lightning is dissipated in this manner.

Composite materials used to construct modern aircraft are not naturally conductive. To achieve the same static and lightening protection as an aluminum aircraft, conductive wires or layering of conductive material into composite components during layup ensures even distribution of electrical charges when all bonding procedures are followed.

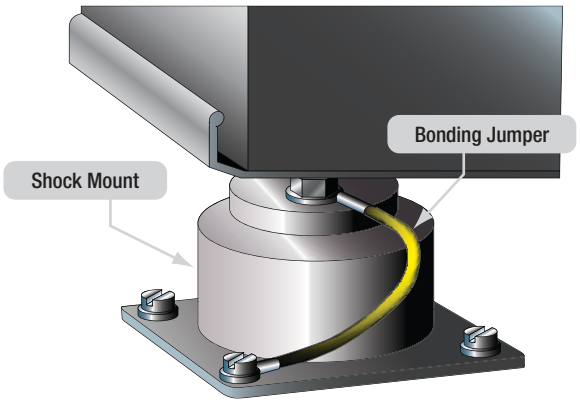


Figure 2-13. A bonding jumper is used to ground an equipment rack and avionics chassis around the non-conductive shock mount material.



Figure 2-14. Static discharger's or wicks dissipate built up static energy in flight at points a safe distance from avionics antennas to prevent radio frequency interference.

Question: 2-1
If the failure of a structure causes a loss of control of the aircraft or leads to a catastrophic structural collapse, the structure is classified as _____ structure.

Question: 2-2
What are the 5 major stress to which all aircraft are subject?

Question: 2-3
To maintain an airframe wherein all components are at the same potential level electrically, _____ is used.

Question: 2-4
Longerons and stringers are used in _____ fuselage construction.

Question: 2-5
Aileron station (AS) is measured _____ from, and parallel to the _____ edge of the aileron.

Question: 2-6
Metal fatigue is caused by _____ repetitive loads in the structure.

Question: 2-7
Access and _____ panels are used to facilitate maintenance on aircraft.

Question: 2-8
Fuselage stations (Fus.Sta. or FS) are numbered from a zero point known as the _____.

ANSWERS

Answer: 2-1
primary.

Answer: 2-2
Tension.
Compression.
Torsion.
Shear.
Bending.

Answer: 2-3
bonding.

Answer: 2-4
semimonocoque.

Answer: 2-5
outboard.
inboard.

Answer: 2-6
small.

Answer: 2-7
inspection.

Answer: 2-8
reference datum.



PART-66 SYLLABUS **LEVELS**

CERTIFICATION CATEGORY → **B2**

Sub-Module 03

AUTOFLIGHT (ATA 22)

Knowledge Requirements

13.3 - Autoflight (ATA 22)

(a) Fundamentals of automatic flight control including working principles and current terminology; — **3**
Command signal processing;
Modes of operation: roll, pitch and yaw channels;
Yaw dampers;
Stability Augmentation System in helicopters;
Automatic trim control; Autopilot navigation aids interface;

(b) Autothrottle systems; — **3**
Automatic landing systems: principles and categories, modes of operation, approach, glideslope, land, go-around, system monitors and failure conditions.

SECTION A

FUNDAMENTALS OF AUTOMATIC FLIGHT CONTROL

INTRODUCTION

Aircraft fly under many conditions. External conditions can alter the desired flight characteristics of the aircraft. To maintain the desired characteristics of the aircraft, the pilot moves the control surfaces either manually or automatically. Multiple instruments in the cockpit feed vital information to the pilot, who then reacts to get the desired performance or orientate the aircraft in a desired attitude. In most fighter aircraft capable of supersonic flight, aircraft speed is so great that the pilot's normal response time is far too slow. For example, by the time the pilot reacts to an indicator to position a control surface, the aircraft may already be out of control.

In the early days, the system that was used to control the aircraft was called the flight control system (FCS), which was primarily mechanical in nature. By means of cables and pulleys, the control surfaces of the aircraft were given the necessary deflections to control the aircraft. However, new technologies brought with it the fly-by-wire (FBW) FCS. In this system, electrical signals are sent to the control surfaces. The signals are sent by a device usually called the flight (control) computer (FC/FCC) (***Figure 3-1***).

But what is the advantage of automatic flight control? Why would we use an FC instead of a pilot? There are several reasons for this. First of all, a computer has a much higher reaction time than a pilot. Also, it isn't subject to concentration losses and fatigue. Finally, a computer can more accurately know the state the aircraft is in. (Computers can handle huge amounts of data better and also don't need to read a small indicator to know, for example, the velocity or the height of the aircraft.) However, there also is a downside to FCs. They are only designed for a certain flight envelope. When the aircraft is outside of the flight envelope, the system can't really operate the aircraft anymore. For these situations, we still need pilots.

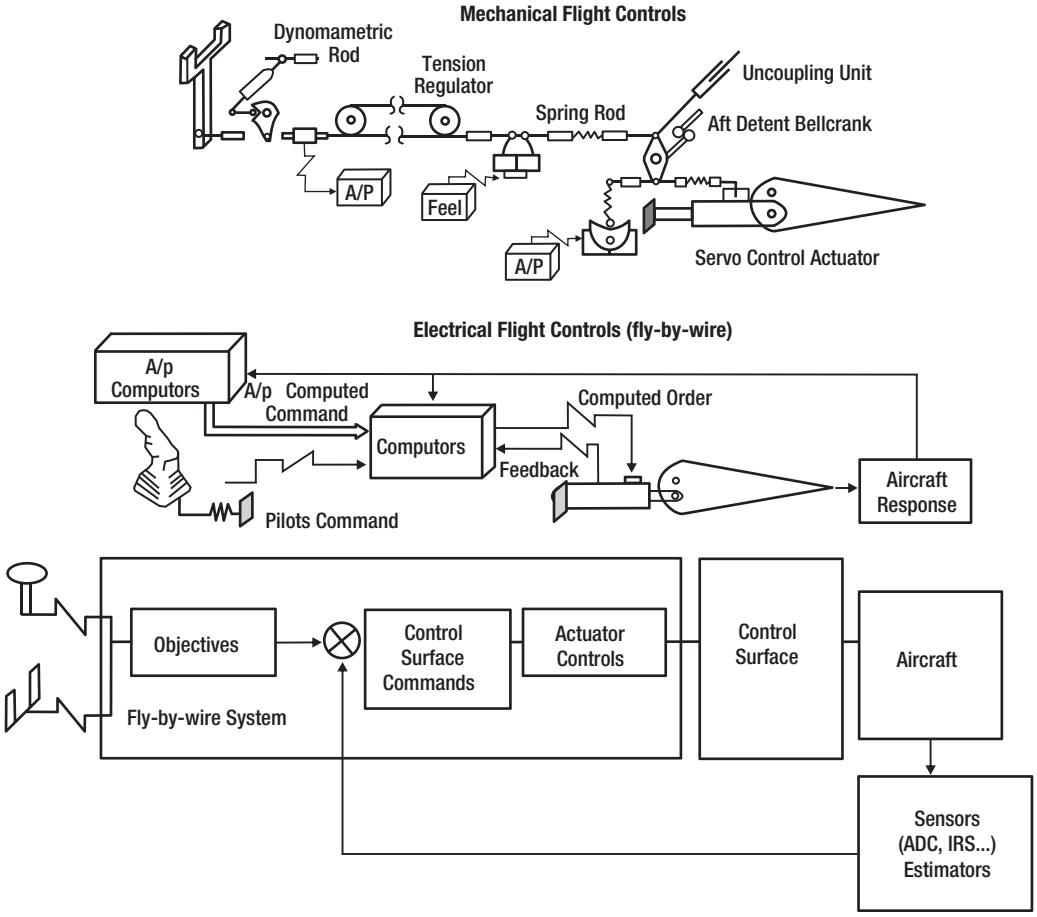


Figure 3-1. Old Mechanical Flight Control System in comparison to the Fly-By-Wire (FBW) concept of the Automatic Flight Control System.

An Automatic flight control and stabilization system, significantly eases the pilot's workload and provides increased aircraft stability at all speeds. Some automatic flight control systems are capable of flying the aircraft by radio navigation aids, correcting for wind, and making pilot-unaided landings. Nowadays, the term Automatic Flight Control System (AFCS) is more commonly used instead of the older term, automatic pilot, or the shortened version, autopilot.

Airbus and Boeing commercial airplanes differ in their approaches in using fly-by-wire systems. In Airbus airliners, the flight-envelope control system always retains ultimate flight control when flying under normal law, and it will not permit the pilots to fly outside these performance limits unless flying under alternate law. However, in the event of multiple failures of redundant computers, the A320 does have a mechanical back-up system for its pitch trim and its rudder. The A340-600 has a purely electrical (not electronic) back-up rudder control system, and beginning with the A380, all flight-control systems have back-up systems that are purely electrical through the use of a so called three-axis Backup Control Module.

With the Boeing 727 model airliners, the two pilots can completely override the computerized flight-control system to permit the aircraft to be flown beyond its usual flight-control envelope during emergencies. Airbus's strategy, which began with the A320, has been continued on subsequent Airbus airliners.

BASIC ARCHITECTURE OF AFSC

The AFCS consist of three independent systems: Digital Flight Control System (DFCS), Yaw Damper system, and Auto-throttle system. These systems provide automatic airplane stabilization about the pitch, roll, and yaw axes and control the airplane with selective guidance from radio, heading, flight management computer, and air data computer inputs. The DFCS is a two-axis (pitch and roll) system which operates the elevators and ailerons to automatically maintain altitude, airspeed and/or guide the airplane to designated locations and make automatic landings. Control functions are also translated into flight director commands for display on the pilots' attitude director indicators (ADI's); thereby providing the pilots' flight attitude commands during manual operation or allowing the pilots to monitor autopilot operation. Automatic stabilizer trimming relieves sustained elevator loads which might be incurred due to fuel burn-off. The yaw damper systems operate the rudders to correct any periodic yaw oscillations (dutch roll). An auto-throttle system automatically maintains selected airspeeds or Mach during cruise conditions and maintains selected engine thrust settings when making flight director controlled take-offs or autopilot/flight director controlled landing approaches by adjusting engine thrust levers.

ROBUSTNESS OF AFCS

AFCS maintains a commanded flight path (direction, altitude and velocity of flight) against perturbations in atmospheric conditions. This is accomplished through a feedback control mechanism, in which deviations from the defined flight path are automatically corrected. AFCS is generally composed of three modules with the same functions, thereby creating redundancy, although each is designed differently (heterogeneity) to avoid a common mode failure. Three computers are made that are modular, so that failure in one module does not affect the functions of other parts of the system. This type of mechanism is implemented using digital technologies that decouple low-level voltages from digital signals (ON/OFF of pulses), thereby preventing noise from influencing its functions. Although this is a simplified explanation of the actual system, the concept applies to details of the basic system as much as it does to the more complex systems (***Figure 3-2***).

COMMAND SIGNAL PROCESSING

BASIC TYPES OF AFSC

The AFSC is designed to ease the pilot workload in such a way that his physical as well as psychological stress can be reduced. The AFCS can be generally categorized into four different types:

- AFCS as the trimmed flight holding system.
- AFCS as the stability augmentation system.
- AFCS as the command augmentation system.
- AFCS as the stability maker and command optimization.

TRIMMED FLIGHT HOLDING

Such an AFCS encompasses a flight condition and configuration holding system, also known as "Hold Systems" such as speed hold, altitude hold, attitude hold and directional hold. This autopilot system does not work continuously but only at a certain period of time. The

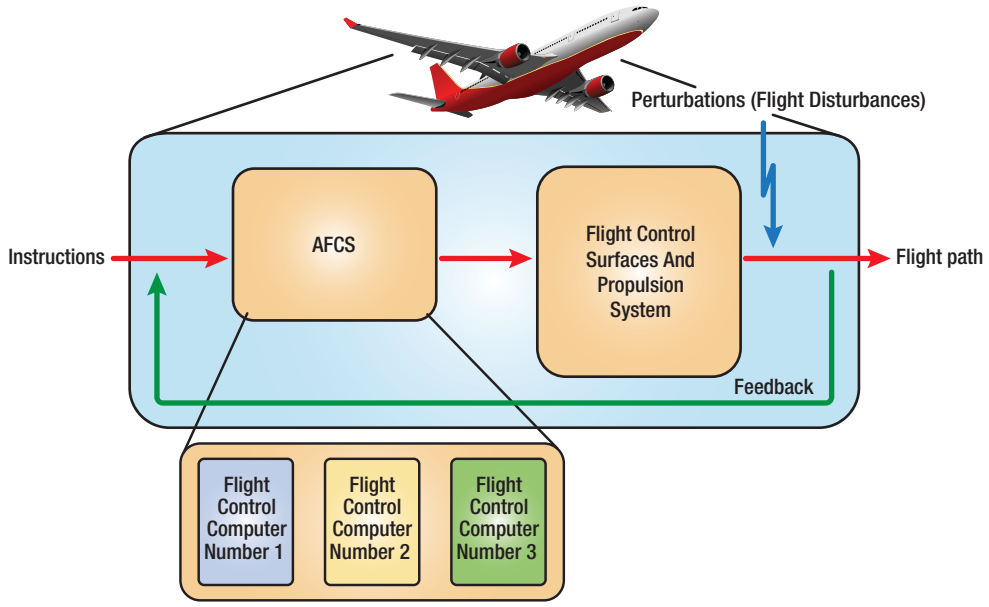


Figure 3-2. AFSC block diagram and redundancy elements.

AP system can be activated by turning the AP switch and deactivated by over-riding the system through the movement of the control manipulator. Therefore, the characteristic of the AP system is of limited authority.

Note that by determining the desired attitude, the pilot can then press the autopilot switch in order that the reference attitude can be obtained and maintained (***Figure 3-3***).

During the period in which the AP is working the pilot does not need to grasp the controller stick. To deactivate the autopilot, the pilot just needs to slightly move the control stick to cut the AP control circuit. The AP system can work well if the aircraft has a good stability characteristic.

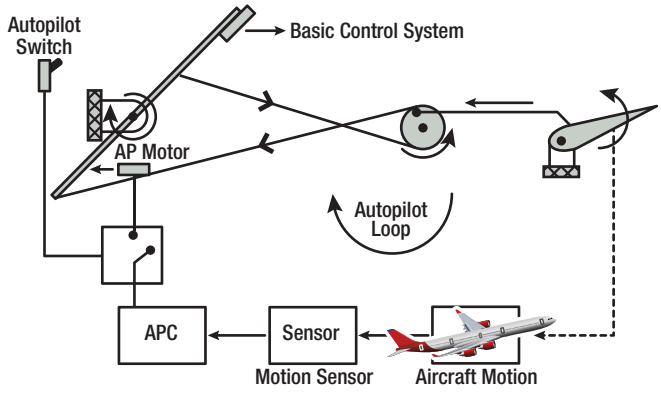


Figure 3-3. Functional diagram of the autopilot system used in the pitch longitudinal channel of an aircraft.

STABILITY AUGMENTATION SYSTEM (SAS)

The type of automatic flight control system that adds stability to the aircraft is usually called the Stability Augmentation System or SAS. This type of automatic flight control system improves the stability of an aircraft at certain flight configurations and conditions within the flight envelope. For conventional aircraft, the stability augmentation will be needed during flight at low speed and low altitude for instance during landing or approach. The control optimization of typical aircraft is conducted only at certain flight configurations such as cruise configuration. This makes the aircraft stability at other flight configurations such as approach, and landing slightly deteriorated. The stability augmentation is therefore necessary for those configurations. The stability augmentation can be achieved by increasing the damping of the existing aerodynamic damping ratio through the application of a feedback control system.

The types of SAS, for example, are:

- Damping ratio augmentation system such as pitch damper, yaw damper and roll damper
- Dynamic compensation supplier system such as wing leveler and turn coordinator

Figure 3-4 shows the example of pitch damper SAS implemented for the aircraft pitch longitudinal channel. Note that the SAS signal comes out of the flight control computer which processes the logic of stability augmentation. This signal directly enters the ECU and is combined with the command signal from the pilot

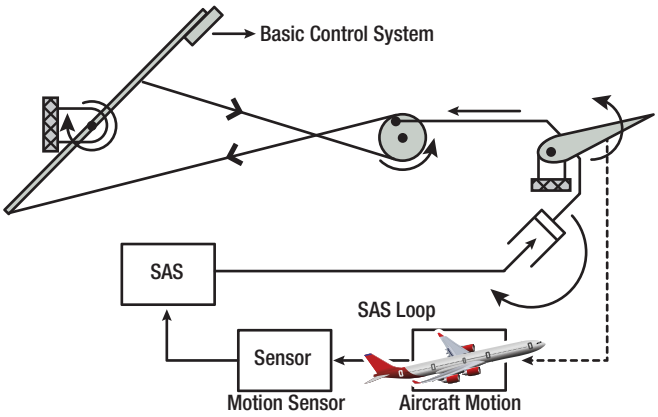


Figure 3-4. Stability Augmentation System (Pitch) Diagram.

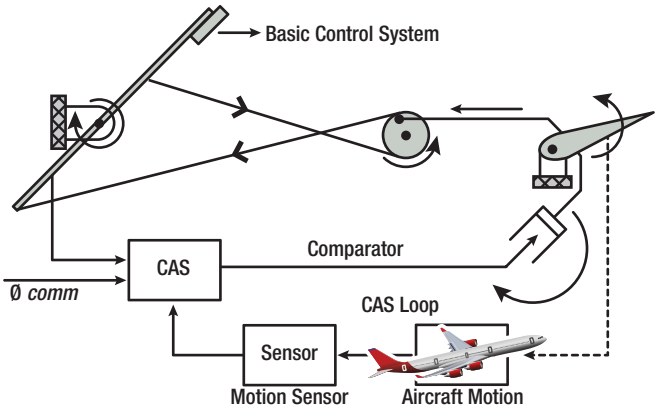


Figure 3-5. Command Augmentation System (Pitch) Diagram.

to move the elevator. The SAS signal coupled with the aircraft dynamics will improve the pitch damping ratio such that the aircraft dynamics is more stable.

It can be observed that the SAS is different from AP in some ways. In the AP system, the output from the AP computer is used to move the control stick in lieu of the pilot input. In the SAS system, the output from the SAS computer is entered into the ECU which forms a closed loop in order to increase the stability of the aircraft. Thus, the SAS system will keep working even though there is an input command from the pilot. Whereas in the AP system the AP loop will be automatically turned off once the pilot moves the controller stick. The SAS therefore has a higher level of authority compared to the AP system. The SAS system is called the flight control system with partial authority. To deactivate the SAS, the pilot can turn the SAS switch.

COMMAND AUGMENTATION SYSTEM (CAS)

Command Augmentation Systems (CAS), adds the power of input command of the pilot by processing the input command and the generated aircraft motion to optimize the input command to the aerodynamic control surface. The working principle of this system is analogous to that of the power steering of the ground vehicle. The command augmentation system works to facilitate Pitch, Roll and Yaw movements. ***Figure 3-5*** shows the example of CAS for the pitch oriented column steering. From the diagram, it is evident that input command from the controller stick is processed to follow the pilot's desired pitch angle. The command signal is then corrected by the actual pitch angle and is processed and sent through the ECU to a electronically controlled hydraulically powered actuator. It can also be inferred from the diagram that the pilot's desired pitch angle can effectively be achieved by an appropriate control stick input. In summary, the differentiating features of the CAS, SAS and AP are:

- CAS — reacts due to control stick input and results in the desired orientation. If the pilot does not move the control stick, CAS is not operating
- SAS — reacts continuously regardless of the motion of the controller stick. When `the SAS is operating, the stability of the aircraft is increased.
- AP — is operating in the condition that the control stick is not moved. When AP is working, the aircraft will maintain its trimmed condition as desired by the pilot.

From the perspective of control circuit, the following feature distinguishes CAS, SAS and AP:

- CAS — the circuit is closed through the Flight Control Computer at the junction point of controller stick and output from the aircraft motion sensor
- SAS — the circuit is closed through the Flight Control Computer directly to the actuator.
- AP — the circuit is closed by the motion of the AP electromotor at the controller stick.

From the above comparison, it is clear that the CAS system has a higher authority than the SAS does because it always reacts to follow the desired attitude set by the pilot.

SUPER AUGMENTATION

This kind of automatic flight control is commonly called Super Augmentation Flight Control System or the Stability Provider and Control Optimizer. This control system is typically used to create an artificial stability for the class of aircraft which are

statically unstable. The same system is simultaneously used to optimize the control power through the application of control laws provided by the Flight Control Computer. The domain of this type of control system is electronic and hydraulic. The super augmented control system is often called electro (opto) — hydraulic flight control system or Fly-by-Wire (Light) flight control system which is abbreviated as FbW or FbL.

Figure 3-6 depicts a Super Augmentation AFSC which principally combines three main activities:

- Super augmentation: providing an artificial stability and optimizing the control power of the aircraft. This subsystem works continuously and cannot be overridden by the pilot.
- Autopilot: taking over some parts of pilot's routine tasks. If this system is in operation, the pilot does not need to hold the control stick. This subsystem can be overruled by the pilot by moving the controller stick.
- Control Law: governing the optimization of the aircraft motion output following the desired mission. Using the control law, the aircraft motion is optimized in such a way that it will not always be the same as the motion due solely to the input command from the pilot. The control law is also used for protection or limits the variables of the aircraft at certain flight configurations.

The artificial stability provided by the super augmentation system is the longitudinal and/or lateral directional static stability. This static stability is created through the continuous feedback process in such a way that the trimmed condition of the aircraft is maintained. It can be observed that this type of flight control system is a control system with a full authority. Without the availability of this type of system, the aircraft that are statically unstable will not be able to fly. Thus, the characteristic of this control system is flight critical.

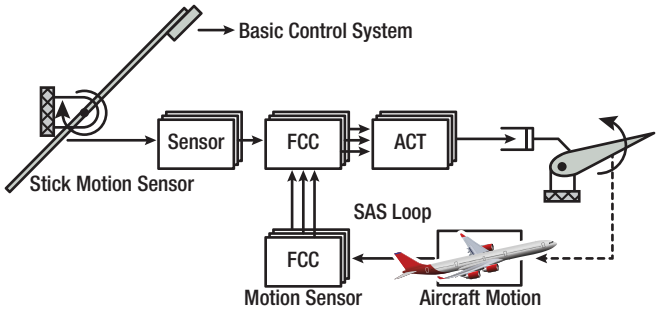


Figure 3-6. Super Augmentation System Diagram.

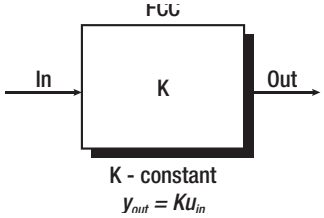
AFCS INTERFACE AND SIGNAL PROCESSING

In the back-end AFCS interface, the control command generator is the part of the system through which pilot command is received from the controller manipulators (stick, steering wheel or pedal) and propulsion controller manipulator (power lever and condition lever). In the study of control engineering, the ability to move the control manipulator is called the controllability.

Another very critical element of the controller is the information processor system. In the manual controller (human controller), this system is represented by the pilot's brain supported by the basic information processing computers to speed up the decision-making process. In the automatic controller, the information processing element is represented by a FCC. The FCC works continuously in real time depending on the authority level of the implemented automatic control system. The software inside the FCC that manipulates the input of the FCC to be converted to the desired control signal by the control system, is called the control law. The control law can be in the form of simple instructions which are typically used by the autopilot.

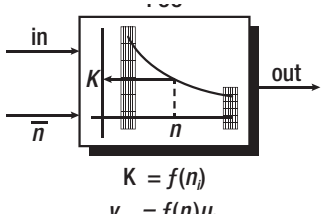
Some of the examples of control law are:

- Constant Gain. The FCC represents a multiplier or an amplifier only.



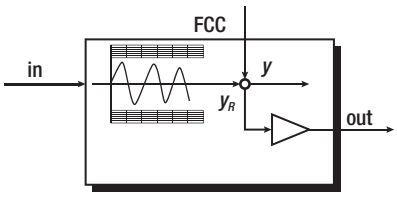
Constant Gain.

- Variable Gain (gain scheduling). The FCC works as a modulated transformer. The gain can be regulated as a function of one or more parameters.



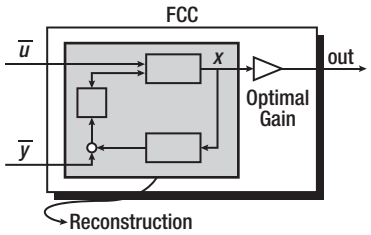
Variable Gain (gain scheduling).

- Robust Gain. The FCC gives the value of the gain K in the admissible control region. The robust property means that control law will still work when there exist some level of uncertainty or parameter changes in the plant.



Robust Gain.

- Optimal Gain. The FCC calculates the optimal gain based on a certain predetermined optimization criterion such as minimum control power, minimum time and minimum fuel. FCC u out x y optimal gain Reconstruction.



Optimal Gain.

- Adaptive Gain. The FCC determines the varying gain that adjusts to the most suitable model of a certain configuration.

Other than the above control laws, there are many other approaches that are getting more applications in the automatic flight control design, namely: neural networks, fuzzy logic, H2 and H∞ control, and passivity based control.

AUTOPILOT MODES OF OPERATION

PITCH MODE (ALTITUDE HOLD) - BASIC MODE

The elevator deflection command, sent to the power loop is a function of the difference between the commanded attitude and the instantaneous attitude of the pitch rate and the flight condition changes. The gains are based on the aerodynamic characteristics of the aircraft, primarily the changing values based on speed.

During turns, if the TURN selector switch is turned "ON", a signal proportional to the bank angle is applied to the pitch channel which compensates for the load factor increase. Depending on the type of aircraft, automatic attitude corrections are also introduced when flaps are extended to anticipate and changes in attitude.

A "command attitude" function makes it possible to modify the reference attitude using the controller embedded in control wheel steering (CWS). Any action on the pitch control wheel disengages the selected superior mode without disengaging the autopilot.

In CWS, one of the control rods in the flight control linkages provides a signal based on the direction and range of the attitude commanded with the control column. To avoid any sudden control movements at the moment of engagement, the autopilot is synchronized with the human pilot, thereby ensuring the deflection from the control unit is zero at the moment of engagement.

Synchronization can also be carried out via the power loop by a synchronization loop equipped with an integrator whose output signal acts directly on the power amplifier to continuously override the deflection deviation.

VERTICAL SPEED MODE (V/S) – BASIC MODE

On modern systems, vertical speed hold mode (VS) corresponds to the basic mode of the longitudinal autopilot system. In this case, the commanded attitude depends on the difference between the vertical speed at the moment of engagement or the difference between a new selected vertical speed and the actual vertical speed of the aircraft.

When flight level change modes are available on a system, the V/S mode is more suitable for smaller level changes. When selecting the mode, the displayed VZ, if the logic control authorizes it, results in the mode becoming active. If the ATS is armed, it engages in the speed hold mode.

ALTITUDE HOLD MODE (ATL HLD) - SUPERIOR MODE

This mode keeps the altitude of the aircraft stabilized at the moment of autopilot engagement, provided the vertical speed is less than the programmed threshold.

The mode also engages automatically at the moment of intercept of a selected altitude. When the autopilot is in ALT mode with the ATS engaged, it results in the activation of the SPEED/MACH mode. Activation of any of the following modes including V/S, LVL CH, PROFILE or VNAV, SRS, or LAND would automatically disengage the SPEED/MACH mode.

Any changes in altitude are sensed and corrected by deflection of the appropriate control surfaces to maintain the desired altitude. The commanded attitude to maintain the desired altitude at mode engagement varies with speed, weight, and thrust of a particular aircraft.

A change in one of these parameters, not including the natural load shifts of the aircraft in flight, would result in attitude and altitude changes while maintaining the same command deflection angle. The problem can be overcome using an override fade or a pitch limiter.

ALTITUDE ACQUISITION MODE (ALT/S OR ALT ACQ) - SUPERIOR MODE

The altitude deviation is calculated relevant to the altitude selected by the pilot or in relation to the vertical flight path. This mode is armed from the moment it is selected and becomes active once the reference altitude is acquired and the vertical speed is within prescribed limits. During the capture phase, if the selected altitude is modified and the capture condition is lost, the V/S or ATT mode engages automatically and the ALT ACQ mode returns to the armed phase. When altitude deviation becomes lower than the set threshold (for example 20 feet), the ATT mode is disengaged and the altitude hold mode is automatically engaged once again. The altitude acquisition mode disengages during the manual selection of any of the following modes, namely PROFILE, ALT HLD or in case of GLIDE SLOPE beam intercept (G/S).

YAW DAMPER

A yaw damper is a device used on many aircraft (usually jets and turboprops) to damp (reduce) the rolling and yawing oscillations known as the Dutch roll mode (***Figure 3-7***). It requires yaw rate sensors and a processor that provides a signal to an actuator connected to the rudder. The use of a yaw damper helps to provide a better ride for passengers. On some aircraft, the yaw damper must be operational at all times during flight above a specified altitude. The Yaw Damper System is engaged by a switch on the flight control module through the AFC accessory unit. When the system is engaged, a rate gyro in the coupler senses yaw rate of the airplane. The yaw rate signal is processed to generate a command to the rudder power control unit which moves the rudder. The signal is translated from electrical to mechanical in the rudder power control unit by a transfer valve. Position feedback is accomplished by a Linear Variable Differential Transformer (LVDT). A position indicator in the flight compartment indicates rudder movement due to yaw damper inputs. There is no feedback to the rudder pedals since a series hydraulic actuator is used in the power control unit.

A typical Yaw Damper System consists of a yaw damper engage switch and warning annunciator, a coupler, an engage solenoid, transfer valve and position transducer on the rudder power control unit, and a rudder position indicator.

YAW DAMPER COUPLER

The coupler (***Figure 3-8***) provides rudder deflection commands to compensate for movement about the yaw axis. The Yaw Damper coupler is a rack assembly containing solid state sub-assemblies for yaw computation, BITE, and power supply and a rate sensor with yaw BITE panel subassembly.

YAW DAMPER ENGAGE SWITCH

The engage switch is used to place the Yaw Damper system in the operating mode. (***Figure 3-9***) The engage switch is a 2-position solenoid hold switch. The solenoid is controlled by the Yaw Damper engage logic and holds the switch in the ON position.

WARNING ANNUNCIATOR

The Yaw Damper warning annunciator indicates that the system is not engaged in the operating mode. The warning annunciator is generally an amber light that illuminates whenever the Yaw Damper is not engaged (***Figure 3-9***). The master caution lights and the flight control master caution annunciator lights, located on the glare shield, also illuminate when the warning annunciator is on.

YAW DAMPER RUDDER POSITION INDICATOR

The rudder position indicator (***Figure 3-10***) displays rudder movement induced by Yaw Damper coupler

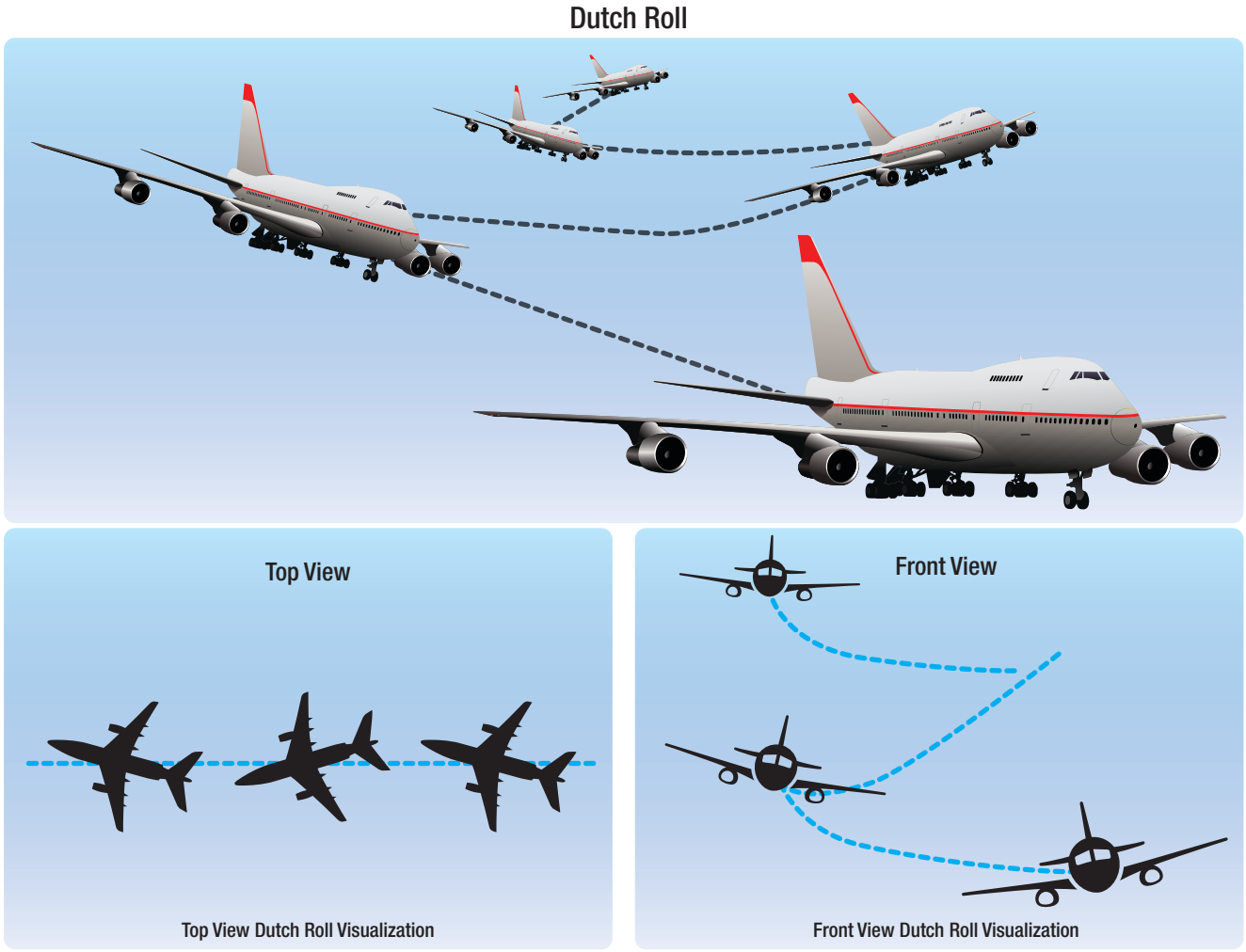


Figure 3-7. Dutch Roll Visualization.



Figure 3-8. Yaw Damper Coupler.

commands. Whenever the Yaw Damper system is engaged, the rudder position indicator reflects the movements of the rudder. The position transducer on the rudder power control unit senses rudder displacement from neutral and provides a signal to the indicator.



Figure 3-9. Yaw Damper Engage Switch & Warning Annunciator.

RUDDER POSITION TRANSDUCERS

A linear position transducer mounted on the rudder power control unit supplies a rudder position feedback signal to the Yaw Damper coupler. The transducer's



Figure 3-10. Yaw Damper Position Indicator.

magnitude of the output signal varies directly with the length of the input stroke and the phase of the signal changes with the direction of the stroke from the center null position.

RUDDER TRANSFER AND SOLENOID VALVES

The engage solenoid and the transfer valve control hydraulic flow for Yaw Damper control to the rudder power control unit. The engage solenoid is mounted on the rudder power control unit. It is a solenoid actuated valve which is energized when the Yaw Damper is engaged; it then completes hydraulic power to the transfer valve. The transfer valve, also mounted on the rudder power control unit, converts the Yaw Damper electrical signals into hydraulic flow to move the rudder. The unit consists of a torque motor which moves a jet pipe assembly that regulates the hydraulic flow to the control valve.

YAW DAMPER OPERATION

The Yaw Damper may be turned on by placing the Yaw Damper switch to ON. Hydraulic power must be available, and two seconds after the yaw damper is turned ON, the YAW DAMPER light extinguishes. The engagement is monitored by a power monitor circuit in the AFC accessory unit. If AC power is lost for over 2 seconds, the YAW DAMPER light illuminates and the Yaw Damper switch returns to OFF. The yaw rate gyro senses oscillations in the yaw axis. The signal is filtered, programmed by the ADC for changes in speed, and amplified. It is then applied to the transfer valve. The transfer valve ports hydraulic pressure to move the actuator. The Yaw Damper actuator output is summed with rudder pedal input to move the main actuator which controls the rudder, but there is no mechanical feedback to move the rudder pedals. As the Yaw Damper actuator moves, the LVDT supplies the feedback signal to cancel the signal from the yaw rate gyro. When the oscillation stops, the feedback signal returns the rudder to the original position. The feedback signal is also applied to the position indicator; however, during normal operation, the signals are of insufficient value to be monitored by the position indicator.

STABILITY AUGMENTATION SYSTEM IN HELICOPTERS

Some helicopters incorporate a SAS to help stabilize the helicopter in flight and in a hover. The original purpose and design allowed decreased pilot work load and lessened fatigue. It allowed pilots to place an aircraft at a set attitude to accomplish other tasks or simply stabilize the aircraft for long cross-country flights. Stability augmentation systems reduce pilot workload by improving basic aircraft control harmony and decreasing disturbances. These systems are very useful when the pilot is required to perform other duties, such as sling loading and search-and-rescue operations. Other inputs such as heading, speed, altitude, and navigation information may be supplied to the computer to form a complete autopilot system.

Modern helicopter SAS architecture is generally an attitude-based system that accepts attitude source data using a combination of the installed attitude gyro or digital attitude heading reference system (AHRS) inputs, and motion sensors located in the FCC. The FCC commands the servo actuators to apply small corrections to the cyclic as required to maintain attitude. The SAS is designed to maintain the helicopter at the datum to which it has been trimmed. It uses a simple feedback control in which a rate gyro senses pitch rate, for example, which, on integration, provides a correcting input at the swash plate (if this is the means of rotor control).

SAS COMPONENTS

The SAS along with the associated autopilot system shown in ***Figure 3-11*** consists of the following components:

- HeliSAS Control Panel (HCP) – The HCP interfaces with the FCC. This push button panel

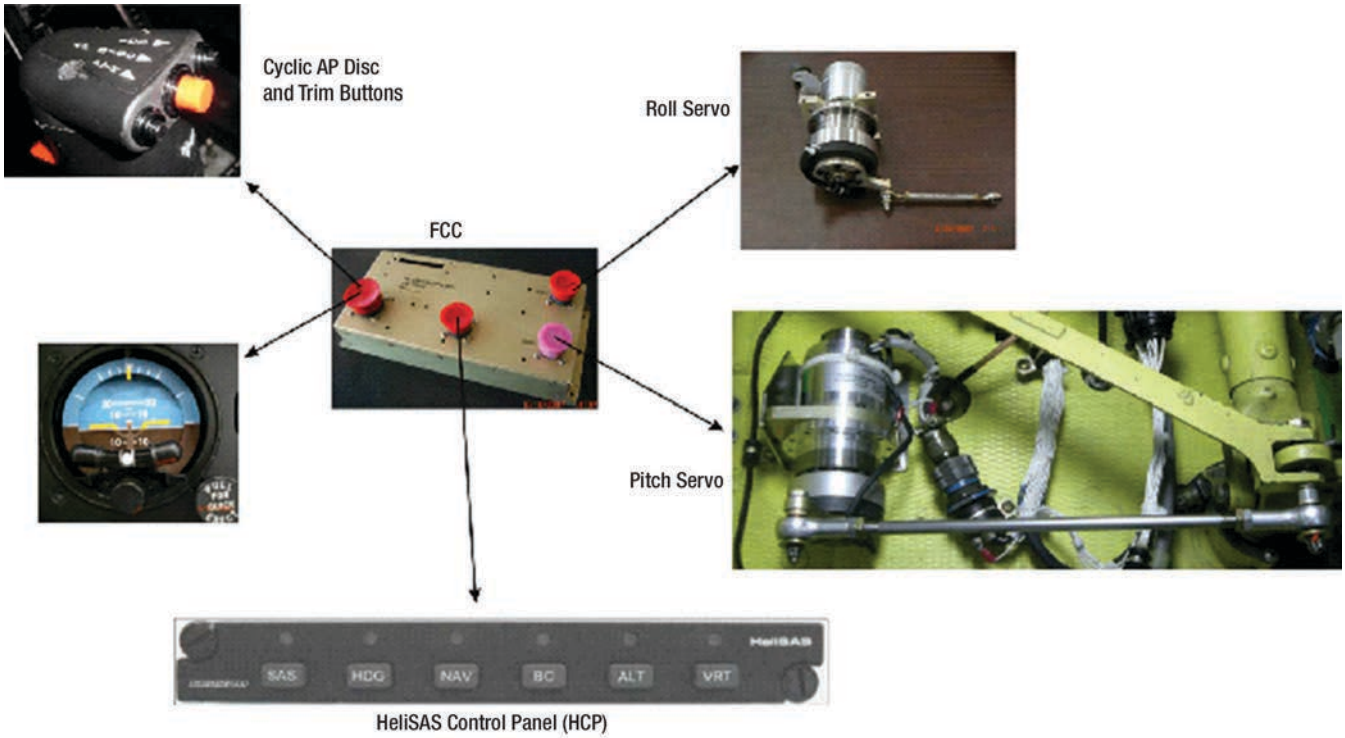


Figure 3-11. Helicopter Stability Augmentation System Main Components.

located in the cockpit within the pilot's reach provides for engagement of the SAS and the desired autopilot mode selections of altitude hold, heading hold, navigation signal tracking, and vertical (Approach) navigation features, which are controlled via the six push buttons on the panel.

- Flight Control Computer (FCC) – The Flight Control Computer receives inputs from the HCP as well as from the on-board NAV, GPS and Attitude systems and commands the pitch and roll servos to perform the selected autopilot function. Internal to the FCC are three body axis rate sensors, a tri-axial accelerometer, two differential pressure sensors, and one absolute pressure sensor. The rate sensors and accelerometer are used to sense the rotational velocities and acceleration of the SAS in three orthogonal axes. The pressure sensors are used to detect and calculate the indicated airspeed and altitude of the airframe.
- Roll Servo – It receives roll error signals from the FCC and provides roll correction to the cyclic controls through electromagnetic clutches located within the servo.
- Pitch Servo – It receives pitch error signals from the FCC and provides pitch correction to the cyclic controls through electromagnetic clutches located within the servo.
- Attitude Gyro – It sends attitude signals to the FCC. The attitude reference for the FCC can also be provided by a digital AHRS (if provisioned).
- AP DISC Switch - The cyclic-mounted AP DISC switch disengages the SAS mode when pressed. If the autopilot is engaged, pushing the AP DISC button causes the autopilot modes to disengage, while the SAS remains engaged. Pushing the button twice causes the autopilot and SAS to disengage. If only the SAS is engaged, pushing the button once disengages the SAS.
- Cyclic Controls - The SAS installation includes a modification to the cyclic controls to add a Momentary Trim button and an AP/SAS DISC button. The Momentary Trim button provides a force-trim-release function to allow the pilot to retrim to a new pitch or roll attitude in ATT mode.
- Servos - The servos are electromechanical servo-actuators consisting of a DC brushless commutating motor, low ratio gearbox, clutch, and servo position feedback resolvers that control the pitch and roll axes of the helicopter. The servo-actuators are connected to the flight control system in parallel with the basic helicopter control rods and have manual servo motor back-drive capability. The clutches consist of an electromagnetic pressure plate design that disconnects the servo-actuators

from the flight control system when the SAS is selected off. Loss of power to the clutches causes them to fail to the open, clutch face separated position. The gear ratio between the cyclic stick and the servo-motor is sufficiently low so that the helicopter can be safely flown with the SAS disengaged and the clutch stuck closed (i.e., the pilot can back drive the gearbox and servo-motor with negligible resistance).

SAS OPERATION

The Helicopter SAS is typically a two-axis attitude hold, attitude command, flight control system. The system has two basic functions: SAS to aid with aircraft stability and autopilot outer loop control modes for altitude hold, and heading select and navigation sensor coupling. The SAS and Autopilot System provide a significant reduction in pilot workload, from takeoff to landing. The SAS mode should be engaged prior to liftoff, and disengaged following touchdown. The various autopilot modes can only be engaged when the SAS mode is already engaged and the airspeed is greater than designated minimum engagement airspeed for the autopilot (for example, 44 KIAS in case of Bell 206 helicopter).

A pair of servos is coupled to the cyclic through electromagnetic clutches as a means to control the helicopter for a given mode of operation. One servo controls the cyclic about the roll axis, and the other servo controls the cyclic about the pitch axis. These servos are driven by error signals received from the Flight Control Computer (FCC), which in turn receives inputs from AHRS, Heading System (HSI or Electronic Flight Instrument System (EFIS), VHF Navigation Receiver and GPS Navigation Receiver.

The SAS and autopilot (ATT) system makes it possible to fly for indefinite periods, with hands off the cyclic when in ATT mode. Due to the unstable nature of helicopters, the pilot must always be prepared to assume immediate manual control of the cyclic in the event of an automatic SAS disengagement resulting from a system failure. In addition, despite an engaged SAS mode and the main rotor speed governor in the helicopter, the pilot is not relieved of their responsibility to closely monitor helicopter attitude and main rotor RPM at all times.

AUTOMATIC TRIM CONTROL

INTRODUCTION

The purpose of trimming is to free the pilot from having to exert a constant pressure on the controls. This is often used to maintain straight and level flight, however trimming can also be used at any phase of flight - for example to maintain a constant rate of climb or descent. In addition to the servo control of primary flight control surfaces, it is also necessary to provide methods of controlling the trim of an aircraft via its secondary flight control system. However, whereas in manually controlled flight, trimming is usually affected about the three axes, under automatically controlled conditions it is generally confined to control about the pitch axis. In most cases it is accomplished by a separate trim servo-actuator coupled to the elevator trim tab system, and operating in parallel with the elevator servo-actuator. On the other hand, for those aircraft in which trimming is effected by means of a variable incidence horizontal stabilizer, a separate trim servo-actuator is coupled to the stabilizer. In cases where the stabilizer incidence is varied by hydraulic motors (e.g. Boeing 747 and 767 aircraft) the required automatic trim signals are used to control the flow of hydraulic fluid to the motors.

WORKING OF AUTOMATIC STABILIZER TRIM SYSTEM

A schematic representation of a control system adapted for trimming by means of a horizontal stabilizer is shown in ***Figure 3-12***.

The trim servo in this application is a three-phase, dual-speed, dual-winding motor which operates in parallel with the elevator power control unit. The 115 volts AC power is supplied to the windings of the motor through the contacts of the aircraft nose up and nose down relays, and also a speed change relay. The motor is coupled to the stabilizer jackscrew through an electromagnetic clutch and a differential gearbox. The 'nose up' and 'nose down' relays are energized by appropriate command signals from the pitch channel of the automatic control system. Energizing and de-energizing of the speed change relay is controlled by a flap position switch. This process is simplified in ***Figure 3-13***.

In the aircraft trimmed condition, all three relays are de-energized and so the trim motor is de-clutched from the stabilizer and has no power supplied to it. The elevators,

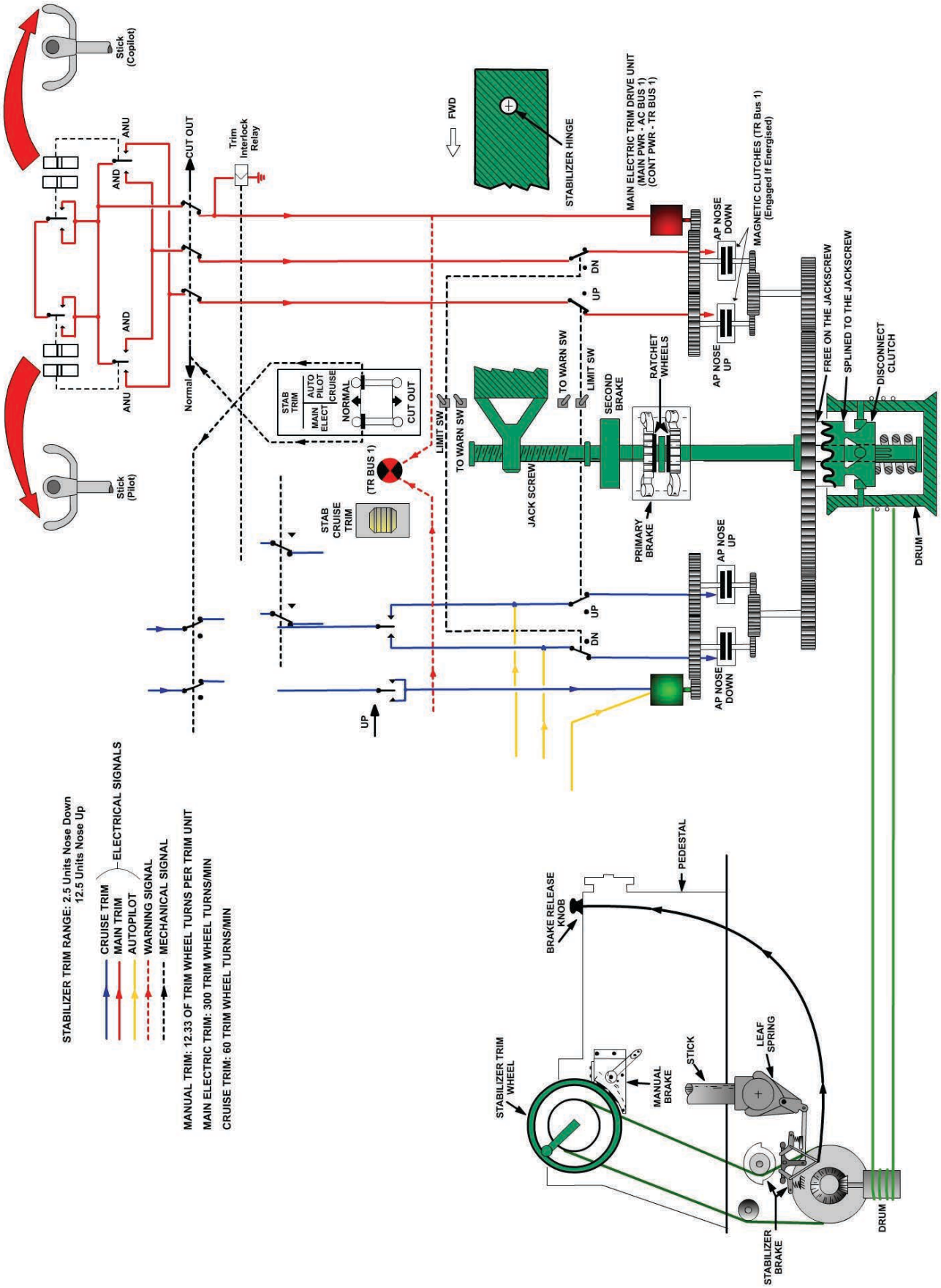


Figure 3-12. Automatic Stabilizer Trim System.

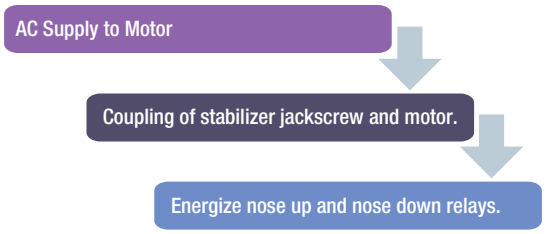


Figure 3-13. Block Diagram of Automatic Stabilizer Trim System.

which provide the primary control of the aircraft's attitude, are locked to their hydraulic power control unit when the automatic control system is engaged.

Assuming that a climb command has been initiated, the command signal will be applied to the transfer valve of the power control unit causing it to apply up elevator. At the same time, the pitch control channel will supply a signal to the coil of the 'nose up' relay which, on being energized, completes an engage signal circuit (across the contacts of the de-energized down relay) to the trim motor clutch, and also completes the AC supply circuit to one set of trim motor windings. The operation of the trim motor is governed as a function of stabilizer position versus elevator displacement from the neutral shift position, and this establishes threshold levels at which the stabilizer will start and/or stop trimming. It is important to know the usage of other component for the proper functioning of aircraft.

Trim Potentiometer

The threshold levels can, in turn, be changed by a trim potentiometer, which is mechanically connected to the stabilizer. If the 'up' elevator displacement already assumed exceeds the threshold level at which stabilizer trim will start, then the trim motor will drive the jackscrew in a direction that moves the stabilizer leading edge down.

Sensor

The movement of the stabilizer also moves the elevator feel and centering unit and a neutral shift sensor. The sensor supplies a feedback signal to the pitch channel as the stabilizer takes up its new trim position. Since the elevators must also take up a compatible neutral position, as the aircraft responds to the required attitude change the pitch channel supplies a signal to the elevator power control unit transfer valve commanding a downward displacement of the elevators.

Feedback Unit

A feedback signal from the elevator position transducer limits elevator displacement and by summing it with that of the neutral shift sensor the new neutral position of the elevators is determined. Signals from the pitch trim potentiometer are also fed back into the pitch channel to change the threshold levels of trim motor operation appropriate to the new trim position of the stabilizer.

Limit Switches

The purpose of the limit switches shown in the trim signal circuits is to disengage the nose up and nose down relays, and thereby the clutch, in the event that the stabilizer is driven beyond its pre-set limits in either the upward or downward directions.

AUTOPILOT NAVIGATION AIDS INTERFACE

Autopilot navigation aids interface is composed of instrument landing system, localizer element, glide path element, indicating element, marker beacons. (***Figure 3-14***)

INSTRUMENT LANDING SYSTEM

An Instrument Landing System (ILS) is a short-range navigational aid, which provides azimuth and vertical guidance during the approach to an airport runway. The system comprises ground-based transmitting elements and also receiving elements carried on board an aircraft. (***Figure 3-15***)

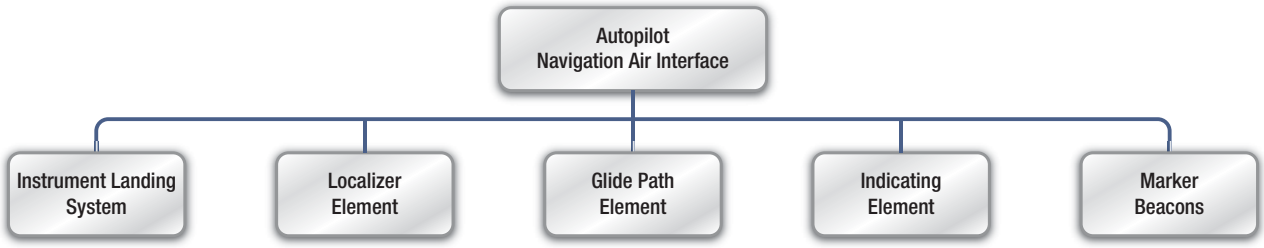


Figure 3-14. Block diagram of Elements of Autopilot Navigation Aids Interface.

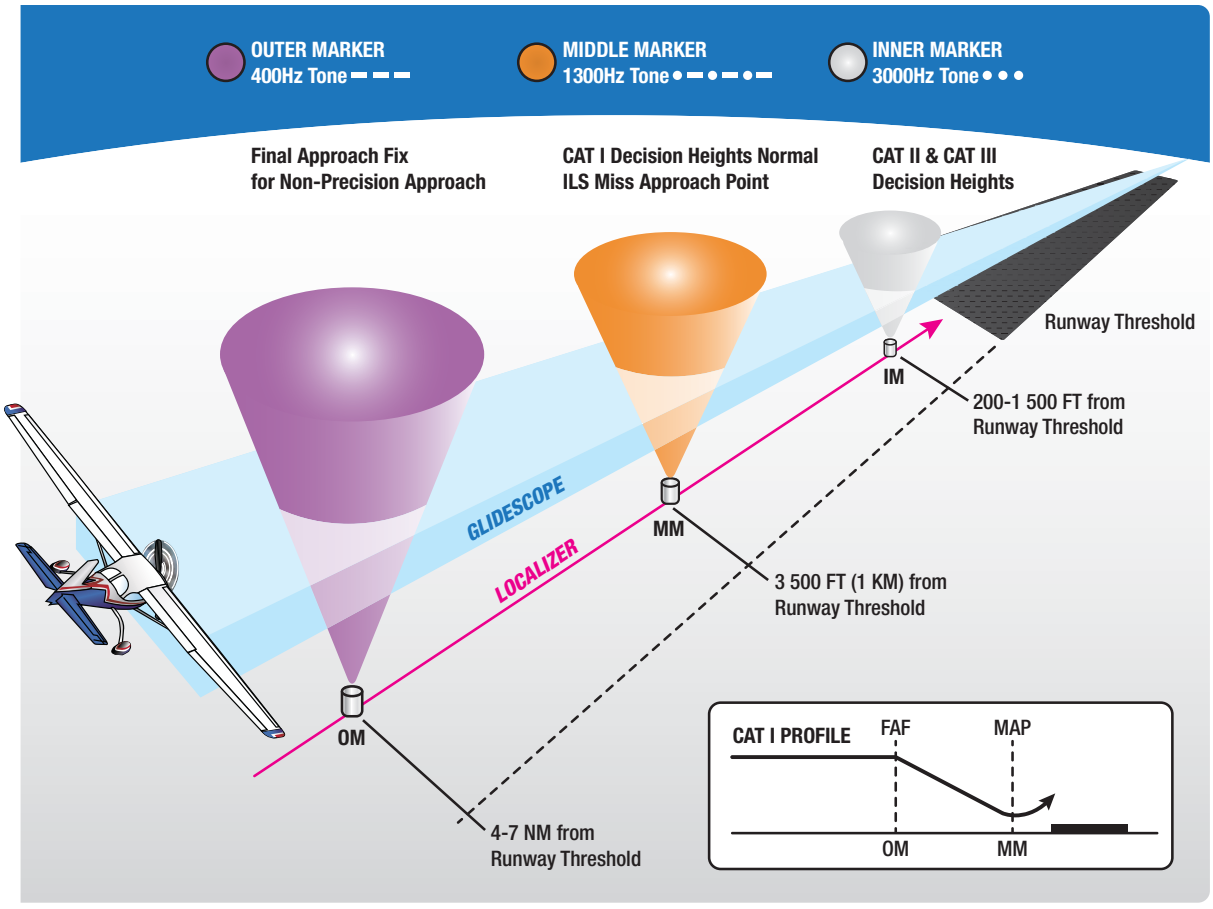


Figure 3-15. ILS ground based elements.

The ground based elements are:

- Localizer-transmits runway azimuth approach information;
- Glide path, or glide slope, transmitter-provides vertical approach information; and
- Marker beacons-transmit information on the distance to the runway threshold.

The airborne elements are:

A localizer signal receiving antenna (usually this is the same antenna as the one used for the VOR installation since both the localizer element and the VOR operate in the same frequency band);

- A glide path signal receiving antenna;
- An ILS receiver unit;
- An indicator which shows whether the aircraft is on the correct approach path; (***Figure 3-16***)
- Marker beacon antenna and receiver;
- Marker lights on the main instrument panel.

LOCALIZER ELEMENT

The transmitter is located at the far end of the runway and it transmits on a given frequency in the band 108.0 and 112.0 MHz to direct an approaching aircraft onto the extended center line of the runway, the transmitter radiates azimuth guidance signals to the left and to the right of the centerline as shown in the ***Figure 3-17***. The transmitted signal is then passed through the comparator which further takes the decision as shown.

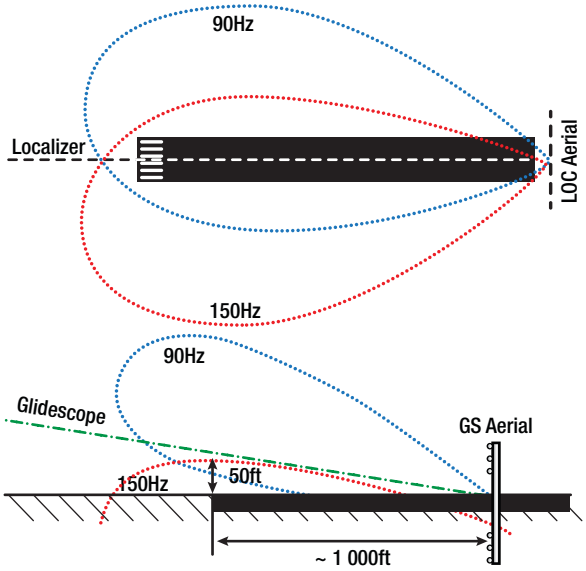


Figure 3-16. ILS guidance signals and commands.

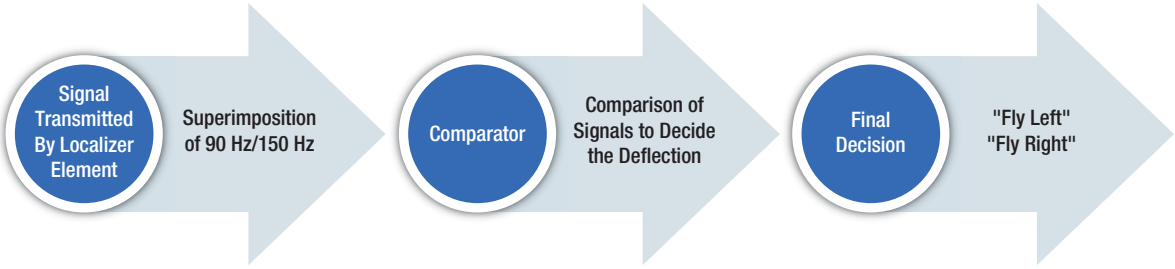


Figure 3-17. Block diagram of Functions of Localizer Element.

The signal transmitted to the left has a 90 Hz signal superimposed on it, while a 150 Hz signal is superimposed on the signal transmitted to the right. The two transmissions overlap along the runway centerline, and when an aircraft is approaching on the centerline extension, the ILS receiver receives both signals at equal strengths. This condition is indicated on the indicating element, usually by a vertical bar, which takes up a central position over the dial of the indicating element. Afterward, there are two possibilities:

- If the aircraft deviates to the left of the centerline, the strength of the 90 Hz signal will be greater than that of the 150 Hz signal. Both signals pass through a comparator circuit within the receiver which then produces an output causing the vertical bar of the indicating element to be deflected to the right, thereby directing the pilot to 'fly right' in order to intercept the centerline again.
- If the aircraft deviates to the right of the centerline, the 150 Hz signal is stronger than the 90 Hz signal, and so after comparison, the receiver output causes the vertical bar to be deflected to the left to direct the pilot to 'fly left' in order to intercept the centerline.

GLIDE PATH ELEMENT

The transmitter is located near the threshold of the runway and it transmits on a given frequency in the band 329.3 to 335.0 MHz The glide path transmitter radiates a signal pattern, which is similar to that of the localizer but, as will be noted from ***Figure 3-18***, the transmissions provide vertical guidance above and below a descent path established at an angle of between 2.5° and 3°.

When the aircraft approaches along this path both the 90 Hz and 150 Hz signals are received at the same strength and this is indicated by a horizontal bar or, as in FD systems, by a pointer which takes up a central position over a scale in the indicating element. If the aircraft deviates below or above the established path,



Figure 3-18. ILS guidance signals and commands.

the pointer will be deflected respectively up or down and this will provide the pilot with the corresponding flight directions.

INDICATING ELEMENT

Different methods of indication are used to meet the requirements. It includes the localizer bar, scales, dots, and warning flags. The localizer bar, or lateral deviation bar as it is more usually referred to, and the glide path pointer are deflected by independent meter movements to which the respective receiver channel signals are supplied.

Scales are provided for reference against the localizer bar and glide path pointer, and thereby indicate by a series of dots in the amount of deviation in degrees from the respective beam centerlines. For example, a two-dot deflection of the localizer bar corresponds to an aircraft deviation of 2.5°, and a full-scale deflection of the glide path pointer corresponds to a deviation of 0.5°. The dots are also indicative of the value (in micro-amps) of the signal currents producing meter deflections. Warning flags are provided and are visible when

power is off, whenever signals are below the required strength, and whenever any malfunction or failure of the system occurs.

MARKER BEACON

Marker beacon transmitters are located at points along the runway approach path and their signals are beamed vertically into the descent path on a frequency of 75 MHz. The signals are coded, and when an aircraft flies over a beacon, they are received and can be heard over the aircraft's intercommunication system; they also illuminate appropriately colored lights on the cockpit instrument panel. There are three types of marker beacons that may be installed as part of their most common application, an Instrument Landing System.

The outer marker is situated approximately four nautical miles from the runway threshold, and for identification purposes, its transmitted signals are coded with continuous dashes. The signal is modulated at a frequency of 400 Hz, made up by a Morse code – a group of two dots per second. On the aircraft, the signal is received by a 75 MHz marker receiver. The pilot hears a tone from the loudspeaker or headphones and a blue indicative bulb lights up. Anywhere an outer marker cannot be placed due to the terrain, a DME unit can be used as a part of the ILS to secure the right fixation on the localizer.

The middle marker is used to mark the point of transition from an approach by instruments to a visual one. The audio signal is made up of two dashes or six dots per second. The frequency of the identification tone is 1 300 Hz. Passing over the middle marker is visually indicated by a bulb of an amber (yellow) color.

The inner marker is sited approximately 3 500 feet from the threshold, its signals being coded with alternate dots and dashes, in frequency of six dots per second. The inner marker emits an AM wave with a modulated frequency of 3 000 Hz. (***Figure 3-19***)

SECTION B

AUTO-THROTTLE SYSTEMS

Auto-throttle is a computer controlled electromechanical device designed to control the thrust of an aircraft's engines within specific engine design parameters. The throttle position of each engine is controlled to maintain a specific value of thrust, in terms of either rotational speed (N1) or engine pressure ratio (EPR) or, a target airspeed, over the full flight regime from take-off to touchdown (***Figure 3-20***). It is designed to operate primarily in conjunction with an AFCS to maintain an aircraft's speed and vertical path, and a flight management computer system (FMCS).

Audio Frequency	Audio Keying	Lamp Actuated
400Hz	▬ ▬ ▬ ▬ ▬ ▬ ▬ ▬	Blue (Outer)
1 300Hz	● ▬ ● ▬ ● ▬ ● ▬	Amber (Middle)
3 000Hz	● ● ● ● ● ● ● ●	White (Airway/Inner)

Figure 3-19. Marker Beacons Audio Keying Signal.

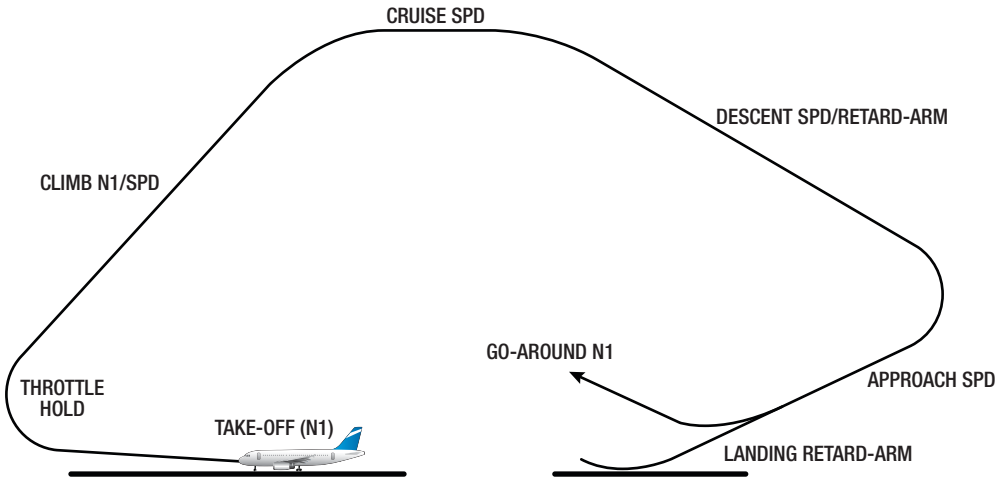


Figure 3-20. Auto-throttle system flight regime.

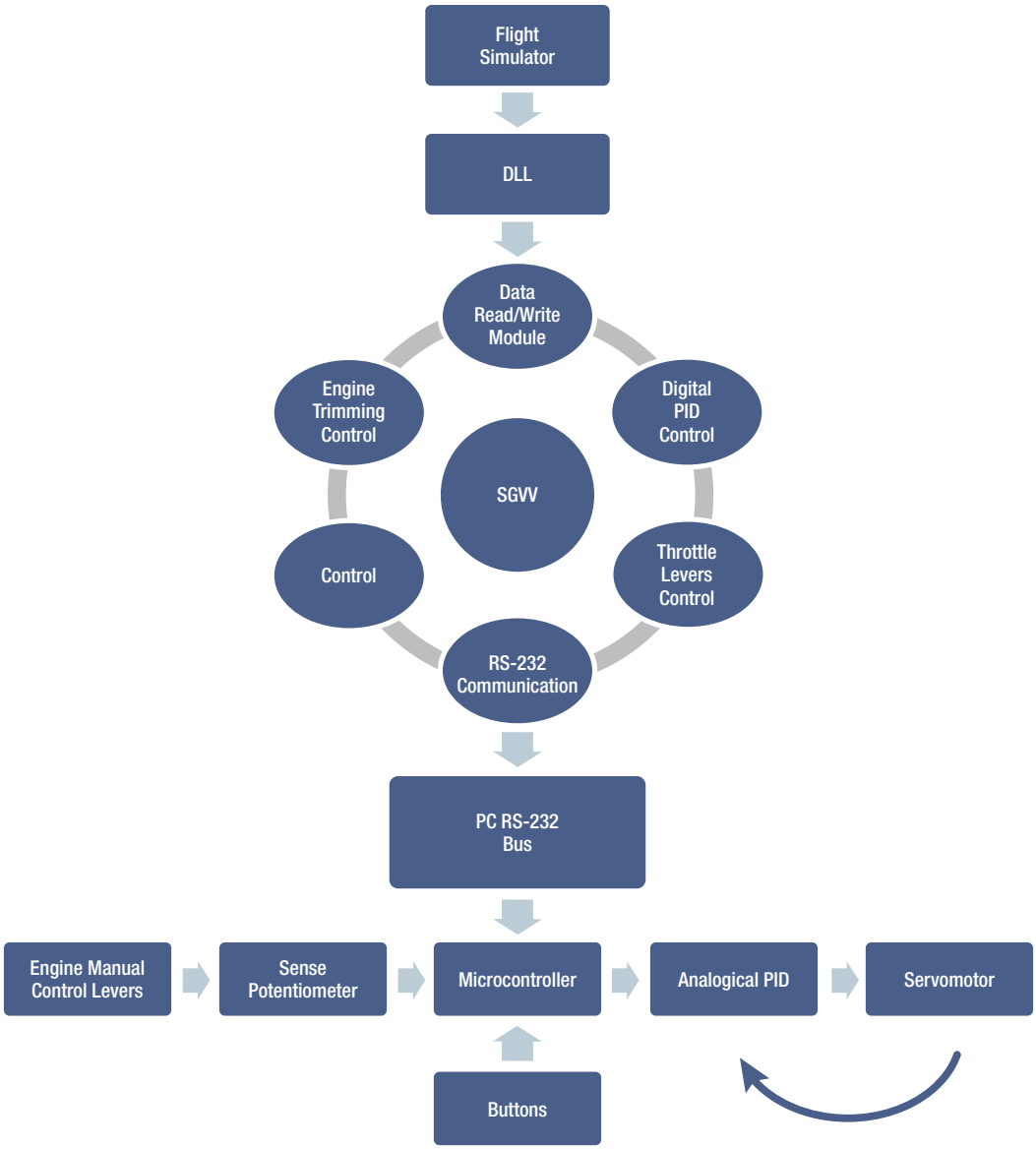


Figure 3-21. Auto-throttle system and signal interfacing.

When an AFCS mode is controlling airspeed, the auto throttle system controls engine thrust to a specific value. When an AFCS mode is controlling the vertical path of an aircraft, the auto throttle maintains airspeed through thrust control. The system computer utilizes data from several other interfacing systems as shown in ***Figure 3-21***.

AUTOMATIC LANDING SYSTEMS

In order to achieve a safe landing, an aircraft has to be so controlled that its wheels make contact with the ground comfortably within the paved surface of the runway, within fairly narrow longitudinal limits along it, and at a sinking speed not greatly in excess of an optimum value of about 1-2 feet per second. The speed at touchdown should have been reduced from the approach margin of about 30% above the stall to about half this value, by progressive reduction of engine power during the landing flare. Finally, the wings should have been leveled prior to the actual landing, and the aircraft yawed to bring its longitudinal axis parallel to the runway center line to remove any drift angle due to crosswind; the latter maneuver being known as decrabbing, or drift 'kick-off'. In summary, it is important to follow the sequence and operations shown in figure. (***Figure 3-22***)

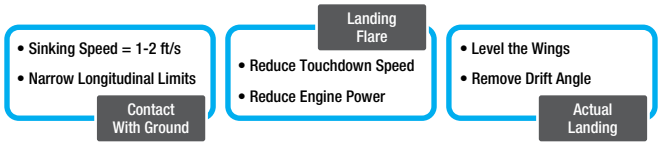


Figure 3-22. Block Diagram of Automatic Landing System.

IMPORTANCE OF CONTROL

Control of the aircraft is needed about all three axes simultaneously, as well as the control of airspeed through engine power changes; it will be appreciated, therefore, that the approach and landing maneuver is the most difficult one demanded of the pilot. Add to this the unfortunate fact that in aircraft operations a large percentage of all accidents can be attributed to the approach and landing phase of a flight. Therefore, a minimum reliability value of 1 in 10^7 is set; in other words, that a system should not cause a fatal accident more often than one in ten million landings.

The control function during the approach and landing maneuver is required on a highly repetitive basis, and although a number of parameters are to be controlled simultaneously, such control is only necessary for a comparatively short period of time, and is therefore most suited to automatic means.

MODIFICATION IN AUTOMATIC LANDING

As a prelude to blind landing, automatic landing has always been the ultimate aim of control systems designers and aircraft operators, throughout the development of automatic flight control systems. In making a broad analysis of available data, it will be found that the many problems which have had to be solved in the development of systems in current use, and having auto land capability, can be grouped in the following three main areas:

- Achieving the highest integrity and reliability of systems bearing in mind that they need to be entrusted with very considerable authority over the controls of an aircraft, including the throttles, and in the presence of the ground.
- The provision of adequate monitoring information on the progress of the approach and landing maneuver, and which will enable the pilot to take over under the most critical conditions of a system malfunction in the presence of the ground.
- The substitution of the pilot's direct vision with an automatic ground guidance system, having an integrity and reliability of the same high order as that demanded of the 'on board' system.

WEATHER MINIMA

In low visibility operations, the weather limits for landing are given in the following terms:

- Runway visual range (RVR) which is an instrumentally derived value that represents the range at which high intensity lights can be seen in the direction of landing along the runway. Its readings are transmitted to the air traffic controller who can inform the pilot of the very latest visibility conditions.
- Decision height is the wheel height above the runway threshold by which a go-round must be initiated by the pilot unless adequate visual reference has been established, and the position and approach path of the aircraft have been visually assessed as satisfactory to safely continue the approach or landing.

Minimum values of these two quantities (known as 'weather minima) are specified by the national licensing authorities for various types of aircraft, and for various airports. When the traffic controller advises that RVR is above the specified minimum the pilot may descend to the specified decision height, and if by then he has sighted a sufficiently large segment of the ground to enable him to be confident of his judgment, he may carry on and land; otherwise he must overshoot, and either enter the holding pattern pending another approach, or divert to an alternative airport. During the approach, the pilot's line of sight is down the glidepath and not along the runway, and this gives rise to another factor, called 'slant visual range', which a pilot must take into account in order to avoid misinterpretation of visual cues.

CATEGORIES

The foregoing terms are related in a system of categorization adopted by ICAO (International Civil Aviation Organization), and which describes low-visibility landing capabilities based on the principle that the probability of having adequate short visual reference, for the range of permitted decision heights, should be as high as possible. The three categories also serve as an indication of the stages through which automatic approach and automatic landing development progresses, and thereby designate the capabilities of individual automatic flight control systems.

WEATHER MINIMA VS ALL WEATHER OPERATIONS

In connection with automatic landing systems, and in describing low weather minima, the term 'all weather operations' is frequently used; a term which can, and sometimes is, taken to mean that there are no weather

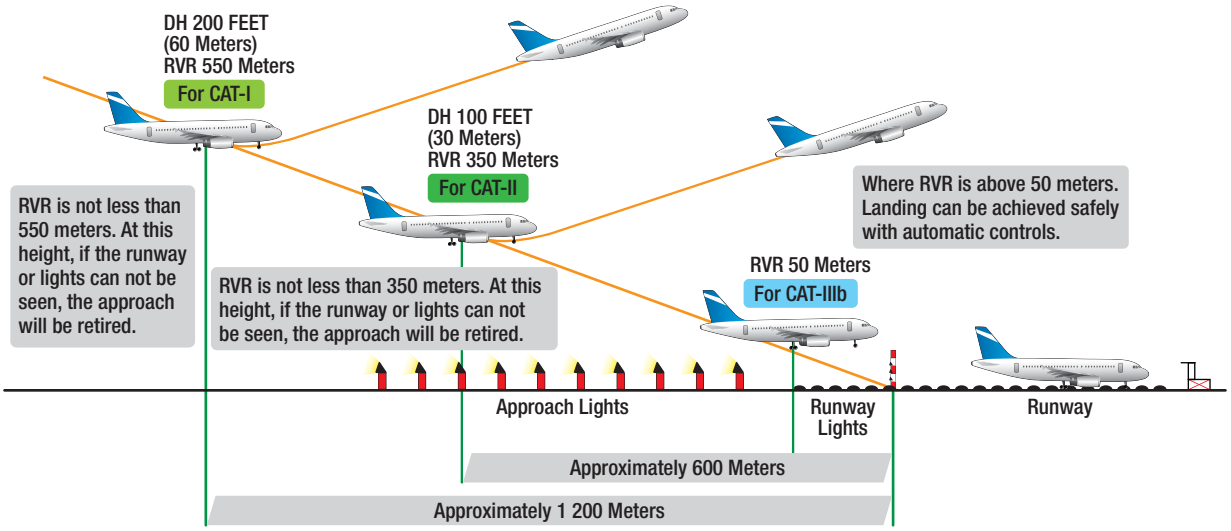


Figure 3-23. Runway Visual Range.

conditions that can prevent an aircraft from taking-off and landing successfully. This is not the case, because no automatic system can, for example, perform the landing task in wind conditions in excess of those for which the aircraft has been certificated, this being primarily governed by the controllability characteristics and strength factors of the aircraft. Similarly, no automatic system can land an aircraft on a runway the surface of which, because of water, slush or ice, is not fit for such an operation. (***Figure 3-23***)

- Category I: Operation down to minima of 200 ft decision height and runway visual range of 800 m with a high probability of approach success.
- Category 2: Operation down to minima below 200 ft decision height and runway visual range of 800 m, and to as low as 100 ft decision height and runway visual range of 400 m with a high probability of approach success.
- Category 3A: Operation down to and along the surface of the runway, with external visual reference during the final phase of the landing down to runway visual range minima of 200 m.
- Category 3B: Operation to and along the surface of the runway and taxiways with visibility sufficient only for visual taxiing comparable to runway visual range value in the order of 50 m.
- Category 3C: Operation to and along the surface of the runway and taxiways without external visual reference.

MODES OF OPERATION

Auto-throttle system operation is controlled primarily through the mode control panel (MCP) of the AFCS (***Figure 3-24***) and the indications of the requisite selections necessary for arming the system, and operating it in the relevant modes, are presented on the MCP and on other annunciator. There are basically two modes of operation: take-off and speed control mode.

TAKE-OFF MODE

This mode is initiated prior to take-off by engaging the FMCS, the computer of which provides the engine rotational speed (Nl) limits for each flight profile, and an N1 target speed. The limits and target speeds are displayed by cursors or 'bugs' on the Nl speed indicators. Arming of the auto-throttle system for take-off is done by moving the engage switch on the MCP to the ARM position. At the same time, the legend ARM is displayed on appropriate annunciator.

Engagement of the system with the servo-actuators controlling the throttle or thrust levers, is done by pressing lever-mounted switches designated as take-off/go-around (TOGA) switches. The servo-actuators then advance the thrust levers at a particular rate to



Figure 3-24. Mode control panel (MCP).

the predicted position to obtain the correct N1 values before a certain take-off speed of the aircraft has been attained. Thus, the auto-throttle system enters a 'hold' state and this is indicated by a 'THR HOLD' (throttle hold) annunciator.

The speed detector circuit is also interlocked with micro switches on the main landing gear shock struts, so that in the event of it failing to perform the 'hold' function, the micro switches will do so as a result of their activation after an aircraft lifts off. The circuits of the micro switches also activate timers, which normally cause the power supplies to the servo-actuator systems to be reinstated after a specified time from lift-off; in this example, 17.5 seconds. During this period the aircraft will have climbed to a specific radio altitude (400 feet in this case). If an engine fails and the initial climb is prolonged such that 400 feet is not reached, an additional interlock circuit prevents reconnection of the throttle servo-actuator systems until this altitude is attained. At this stage, the systems are armed to control N1 speeds during the remainder of the climb to the selected cruise altitude.

SPEED CONTROL MODE

This mode can be selected through the MCP of the AFCS, or it is automatically selected by the AFCS when it is not in a speed mode. The speed target set and displayed on the MCP is referred to as the 'MCP SPD'. If vertical navigation (V NAV) control is selected for AFCS operation, the speed target is provided by the flight management computer and is referred to as FMC SPD.

The auto-throttle system is switched into this speed mode when an aircraft approaches a selected altitude under V NAV control, and will remain in this mode during altitude hold. Airspeed/Mach feedback signals are provided by the air data computers. The auto-throttle system limits the airspeed to minimum and maximum safe values independent of the target airspeed values, and it also limits the angle of attack or alpha angle. Minimum airspeed and maximum alpha angle are computed from signals produced by flap position and alpha angle sensors.

When the aircraft begins to descend under V NAV control, the auto-throttle system retards the thrust levers to idle, and the legend RETARD is annunciated. The retard rate is nominally two degrees/second, and may be terminated by physically stopping the thrust levers, or allowing them to drive against the idle stop position. The system then goes into the ARM mode, which is also annunciated.

When the AFCS captures the glide slope beam, the V NAV mode is disengaged and the auto-throttle system switches to the MCP SPD mode, and the speed displayed corresponds to the last value computed by the FMC system. Selection of a desired speed value is made on the AFCS mode control panel. The 'MCP SPD' is annunciated to indicate to the flight crew that pressing the switch will cause the auto-throttle system to revert to 'ARM'. In the event of speed mode operation with an engine 'out', the throttles advance together to maintain airspeed, and N1 speed equalization is replaced by thrust lever equalization.

Approach gain of the auto-throttle system is determined either by glide slope capture or by radio altitude, and flap position approach gain provides high gain setting for more precise speed control, and reduced throttle motion during changes of flap position. During an approach in turbulent conditions, the gain tends to cause the system to be high on speed. The degree of over speed depends on the magnitude and frequency of the turbulence.

During the landing flare maneuver, the retard rate of thrust reduction is adjusted so that throttle angle is reduced to idle in 6 seconds. Retard occurs at 27 feet of radio altitude during an automatic or manual landing. If it is not initiated by radio altitude, it can also occur 1.5 seconds after an automatic flare. When the aircraft lands and the landing gear shock strut micro switches are activated to their 'ground' position, the thrust levers are moved aft at 8°/s to remove any residual displacement above the idle position. The auto-throttle system is automatically disengaged after 2 seconds.

GO-AROUND MODE

This mode can be engaged when the auto-throttle system is engaged and the radio altitude is less than 2 000 feet. Initial operation of a TOGA switch, on a thrust lever causes the throttles to advance to an angle predicted for a reduced go-around thrust; at the same time, the mode is annunciated by 'GA' on the attitude indicator of the flight director system.

The reduced go-around thrust sets the required throttle angle and also the full go-around thrust rating dependent upon the weight of the aircraft and on density altitude. The TOGA switch is then operated a second time causing the engines to 'spool up' to the full thrust rating. The go-around mode also interfaces with the AFCS so that when it is in command at the time of the mode selection, its pitch channel generates a pitch-up command to establish the aircraft in the requisite climb attitude.

SYSTEM MONITORS AND FAILURE CONDITIONS

The setting of safety devices is dictated by two conflicting requirements:

- They must limit the effect of a 'runaway' such that safe recovery can be effected by the pilot;
- They must allow sufficient authority to the control system so that the required flight path can be followed accurately in the presence of disturbances.

Limited Applications of Safety Devices

Even with a compromise setting of a safety device, there is the possibility of a height loss under 'runaway conditions', which during an automatic landing would be unacceptable. A further factor, which limits the application of safety devices in the manner of conventional control systems, is their inability to protect against passive failures. While not producing flight path changes directly. It follows therefore, to achieve the objective of automatic landing, the operation of an AFCS must be of such a nature that it will:

- Not disturb the flight path as a result of an active malfunction;
- Have adequate authority for sufficiently accurate control along the required flight path;
- Warn of a passive failure;
- Not fail to complete the intended flight maneuver following an active or a passive failure.

Solution-System Redundancy

System redundancy is a concept to utilize multiple systems operating in such a manner that a single failure within a system will have an insignificant effect on the aircraft's performance during the approach and landing operation. In describing failures and the system redundancy concept, it is inevitable that certain terminology must be adopted. It is therefore relevant at this point to review the accepted definitions.

Fail-soft

Is used to describe the ability of a system to withstand a failure without endangering passenger safety, and without producing excessive deviations from the flight path.

Fail-operational

This describes a system in which one failure (sometimes more) can occur, but leaves the overall system still functioning, and without I causing degradation of performance beyond the limits required for automatic landing and roll-out.

Simplex

This term is usually used to define a single automatic control system and its appropriate number of sub channels. Although various elements of the system may be duplicated, a single failure elsewhere will result in complete unserviceability.

Multiplex

This term is applied to a system comprising two or more sensibly independent simplex systems and sub-channels used collectively so that, in the event of a failure of a system or sub-channel, the remaining systems are alone capable of performing the controlling function. The number of systems and sub-channels adopted is qualified by the terms duplex, triplex and quadruplex as appropriate.

Duplex System

Is a system of two complete systems or channels, which are interconnected, and which together provide continuous control. If comparison monitoring is provided, a duplex system can provide fail operational capability. The term should not be confused with the terms Duplicate-Monitored or Duplicate-Redundancy.

Triplex System

Is a fail-operational system of three complete systems or channels which are interconnected and which together provide continuous control. In the event of failure of one of the systems or channels, that system or channel is outvoted by the other two and is automatically disengaged; control is therefore continued in duplex. In the event of a further fault in either of the two remaining systems or channels, they will both disconnect, and the aircraft is returned to the pilot in a trimmed and safe attitude.

Duplicate-monitored

This refers to a system comprising two systems in parallel and with separate power supplies. The components of both are designed to be either self-monitoring or to have their outputs checked by parallel comparator circuits. Only one system is engaged at any particular time, the other system being in a follow-up mode, and thereby serving as an active standby. In the event of a fault being shown up by the self-monitors or comparators of either of the systems, control is automatically changed over to the standby system.

Dual-dual

This term is used by some manufactures to define a twin fail-operational control system having twin passive monitoring systems. It should not be considered synonymous to a duplex system, since the control systems may or may not be active simultaneously. In the event of a monitor detecting a failure in its associated system, the second system with its monitor is switched in.

Monitoring

In its strictest sense and, in particular, when applied to multiplex systems, this term defines the process of making comparisons either between two or more outputs (or inputs) or between an output (or input) and a selected datum. The monitoring process can also assume a limiting function; e.g. when it is set up to cause a system to disconnect whenever an output (or input) exceeds a prescribed limit.

Comparison monitor (or Comparator)

Is one, which operates on data supplied from comparable stages in two or more similar systems.

Equalizer

This is a device, which adjusts the performance of the subsystems in multiplex systems to remove differences between subsystem outputs that may arise other than as a result of fault conditions. Two devices are normally adopted, one called a gain equalizer which adjusts the amplitude of response of sub-systems, and the other called a datum equalizer which adjusts the steady state output of sub-systems.

There are alternate terms associated with the ongoing terms as listed below:

TERMS	ALTERNATE NAMES
Fail-soft	Fail-Passive
Fail-operational	Fail-Active & Fail-Survival
Simplex	Non-Redundant
Duplex system	Dual Active With Passive Monitoring
Triplex system	Triple-Redundant

AUTOMATIC LANDING SEQUENCE

The profile of an automatic approach, flare and landing sequence is illustrated in ***Figure 3-25*** and is based on a system that utilizes triple digital flight control computer channels, allowing for redundancy to operate in the fail operational and fail passive conditions already defined.

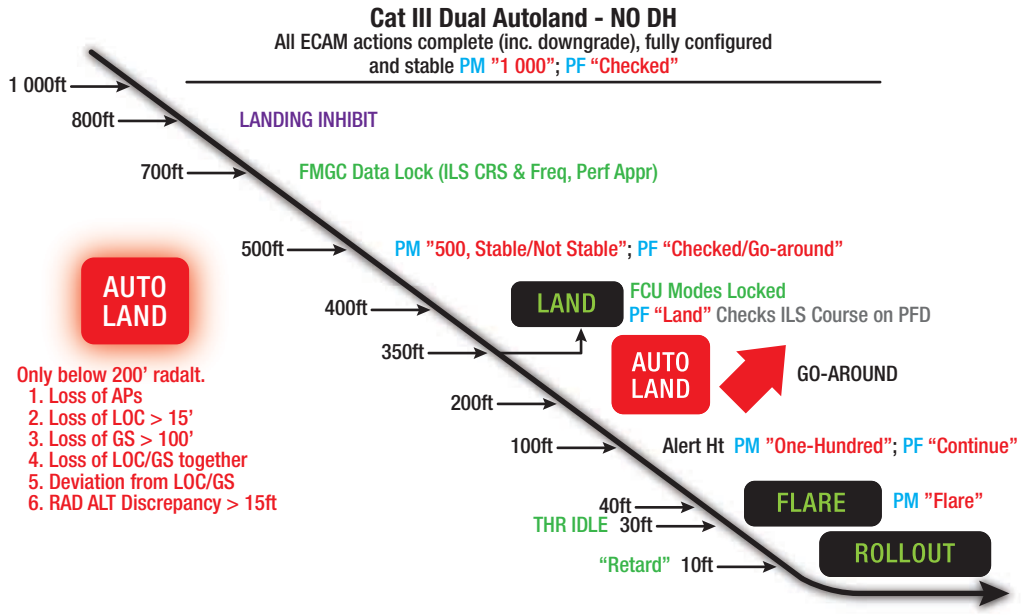


Figure 3-25. Automatic approach, flare and land.

Depending upon the number of channels that are armed and engaged, the system performs what are termed a 'LAND 2' status or 'LAND 3' status autoland.

'LAND 2' signifies there is dual redundancy of engaged flight control computers, sensors and servos (fail passive operation)

'LAND 3' signifies triple redundancy of power sources, engaged flight control computers, sensors and servos (fail operational). Each status is displayed on an autoland status annunciator.

The indicator is of the dual prismatic display type; the upper display indicates the actual autoland capability of the AFCS, while the lower display indicates the degradation from 'LAND 3' capability. Each triangular-shaped prism is actuated by electro-magnetic coils so that the faces of the prisms display the relevant status information as shown. Two test switches are provided; number 1 rotates the upper and lower display faces A into view and number 2 switches rotates the 'B' faces into view. A reset switch (P/RST) can be used to clear the lower display. During cruise and initial stages of approach to land, the control system operates as a single channel system, controlling the aircraft about its pitch and roll axes and providing the appropriate flight director commands.

Since multichannel operation is required for an automatic landing, at a certain stage of the approach, the remaining two channels are armed by pressing an APPR switch on the flight control panel. The operation of this switch also arms the localizer and glide slope modes. Both of the 'off-line' channels are continually supplied with the relevant outer loop control signals and operate on a comparative basis the whole time. Altitude information essential for vertical guidance to touchdown is always provided by signals from a radio altimeter, which becomes effective as soon as the aircraft's altitude is within the altimeter's operating range (typically 2 500 feet).

When the aircraft has descended to 1 500 feet radio altitude, the localizer and glide slope beams are captured, and the armed 'off-line' control channels are then automatically engaged. The localizer and glide slope beam signals control the aircraft about the roll and pitch axes so that any deviations are automatically corrected to maintain alignment with the runway. At the same time, the auto-land status annunciator displays LAND 2 or LAND 3, depending upon the number of channels 'voted into operation' for landing the aircraft, and computerized control of flare is also armed. At a radio altitude of 330 feet, the aircraft's horizontal stabilizer is automatically repositioned to begin trimming the aircraft to a nose-up attitude. The elevators are also deflected to counter the trim and to provide subsequent pitch control in the trimmed attitude.

When an altitude is reached at which the landing gear is 45 feet above the ground (referred to as gear altitude) the flare mode is automatically engaged. The gear altitude calculation, which is preprogrammed into the computer, is based upon radio altitude, pitch attitude, and the known distance between the landing gear, the fuselage and the radio altimeter antenna. The flare mode takes over pitch attitude control from the glide slope, and generates a pitch command to bring the aircraft onto a 2 feet/second descent path.

At the same time, a 'throttle retard' command signal is supplied to the auto-throttle system to reduce engine thrust to the limits compatible with the flare path. Prior to touchdown, and about 5 feet gear altitude, the flare mode is disengaged and there is transition to the touchdown and roll-out mode. At about one foot gear altitude, the pitch attitude of the aircraft is decreased to 2°, and at touchdown, a command signal is supplied to the elevators to lower the aircraft's nose and so bring the nose landing gear wheels in contact with the runway and hold them there during the roll-out. When reverse thrust is applied, the auto-throttle system is automatically disengaged. The AFCS remains in control until disengaged by the flight crew.

Question: 3-1
Auto-pilots systems are divided into three main sections:
_____ _____; and _____.

Question: 3-2
Automatic control systems incorporates closed loop control which incorporates: error actuation, amplification and _____.

Question: 3-3
The acronyms: FADEC and EFIS stand for:
_____ and _____ respectively.

Question: 3-4
Two common types of Servo-Motors are:
_____ and _____.

Question: 3-5
To enable a smooth transition of flight controls from disengaged to engagement of autopilots, a process known as _____ is used.

Question: 3-6
When in CMD mode the force detection threshold for autopilot override is (greater/less) in magnitude than that of CWS mode.

Question: 3-7
ALT, V/S are examples of _____autopilot modes.

Question: 3-8
Side slip oscillations, commonly known as "Dutch Roll", are stabilized by means of a _____ _____ system.

Question: 3-9
Large helicopters may have a stabilization system called SAS, which stands for:
_____, _____, _____. An artificial feel spring effect is built into the controls, known as a: _____ _____.

Question: 3-10
Auto-thrust systems operate in three power zones:
_____, _____ and _____.

ANSWERS

Answer: 3-1
sensors; computers; actuators.

Answer: 3-2
feedback.

Answer: 3-3
full authority digital engine control; electronic flight instrumentation system.

Answer: 3-4
electric; electro-hydraulic.

Answer: 3-5
synchronization.

Answer: 3-6
greater.

Answer: 3-7
pitch.

Answer: 3-8
Yaw Damper.

Answer: 3-9
stability augmentation system; force gradient.

Answer: 3-10
rear, center, front.



PART-66 SYLLABUS **LEVELS**

CERTIFICATION CATEGORY → **B2**

Sub-Module 04

COMMUNICATION/NAVIGATION (ATA 23/34)

Knowledge Requirements

13.4 - Communication/Navigation (ATA 23/34)

(a) Fundamentals of radio wave propagation, antennas, transmission lines, communication, receiver and transmitter; Working principles of following systems: **3**
- Very High Frequency (VHF) communication;
- High Frequency (HF) communication;
- Audio;
- Emergency Locator Transmitters (ELTs);
- Cockpit Voice Recorder (CVR);
- Very High Frequency Omnidirectional Range (VOR);
- Automatic Direction Finding (ADF);
- Instrument Landing System (ILS);
- Flight Director Systems (FDSs), Distance Measuring Equipment (DME);
- Area navigation, RNAV systems;
- Flight Management Systems (FMSs);
- Global Positioning System (GPS), Global Navigation Satellite Systems (GNSSs);
- Data Link.

(b) Air Traffic Control transponder, secondary surveillance radar; **3**
Traffic Alert and Collision Avoidance System (TCAS);
Weather avoidance radar;
Radio altimeter;
Automatic Dependent Surveillance Broadcast (ADS-B).

(c) Microwave Landing System (MLS); **3**
Very Low Frequency and hyperbolic navigation (VLF/Omega);
Doppler navigation;
Inertial Navigation System (INS);
ARINC (Aircraft Radio Incorporated) communication and reporting.

SECTION A

FUNDAMENTALS OF RADIOWAVE PROPAGATION

WAVE THEORY

A wave can be defined as a disturbance (sound, light, radio waves) which moves through a medium (air, water, vacuum). Wave motion can be defined as a recurring disturbance advancing through space with or without the use of a physical medium. Therefore, it is a means of moving or transferring energy from one point to another point. For example, when sound waves strike a microphone, sound energy is converted into electrical energy. When radio waves strike an antenna, they are likewise converted into electrical energy. Thus, sound, light, and radio waves are all forms of energy that are moved by wave motion. There are two types of mechanical waves (***Figure 4-1***), known as transverse waves and longitudinal (compressional) waves.

The transverse and longitudinal waves are also called periodic waves because they consist of cycles or patterns that are produced over and over again by the source.

TRANSVERSE WAVES

The analogy of wave motion in water helps to explain the basic concept of how a radio wave propagates. Dropping a stone into a pool of water result in a disturbance of the water (the medium). From the point of impact, the disturbance is transmitted on the surface of the water as an expanding series of circular waves. ***Figure 4-2*** depicts thin wave motion. View-A pictures the stone an instant before it strikes the water. View-B illustrates the action that occurs when the stone strikes the surface, pushing the water upward and outward. View-C, the stone has sunk deeper into water, which has closed over it, while the leading wave has moved outward. View-D shows the leading wave continuing to move outward, followed by a series of waves gradually diminishing in amplitude. Meanwhile, the disturbance at the original point of impact has gradually subsided.

NOTE: The leading wave has amplitude and wavelength corresponding to one compete cycle. The water is not actually being moved by the outward motion of the wave, but rather by the up and down motion of the water.

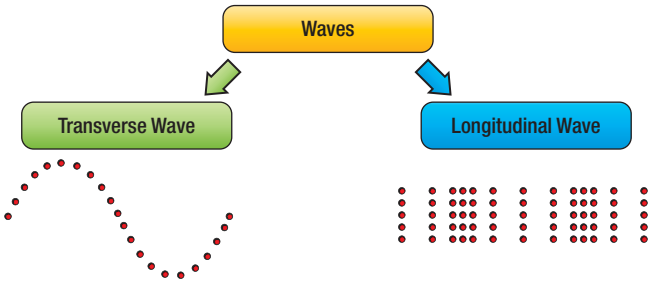


Figure 4-1. The two types of waves, transverse and longitudinal (compressional).

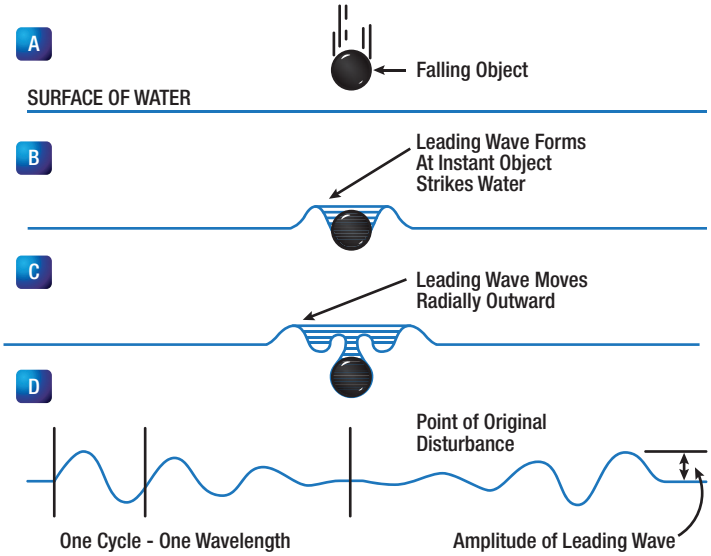




Figure 4-2. How a falling stone imparts wave motion to a water surface. In a transverse wave (above) the water particles displace perpendicularly to the direction of wave propagation. The particles do not move along with the wave; they simply oscillate up and down about their individual equilibrium positions as the wave passes by.

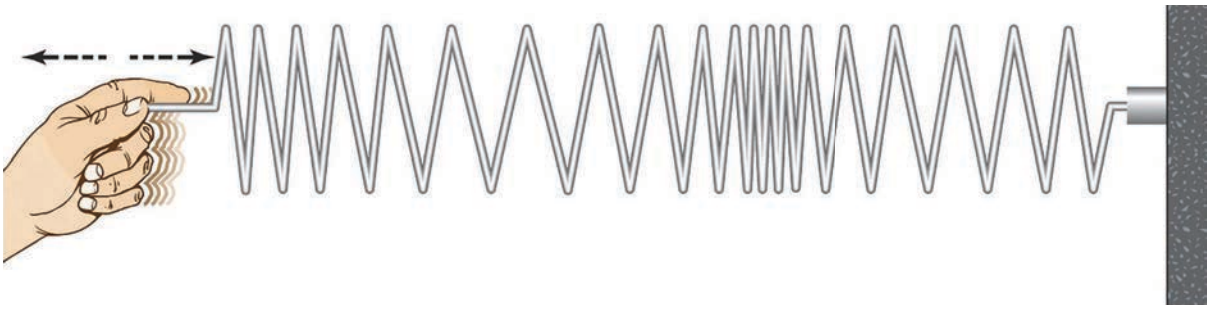


Figure 4-3. A longitudinal (compressional) wave in slinky spring.

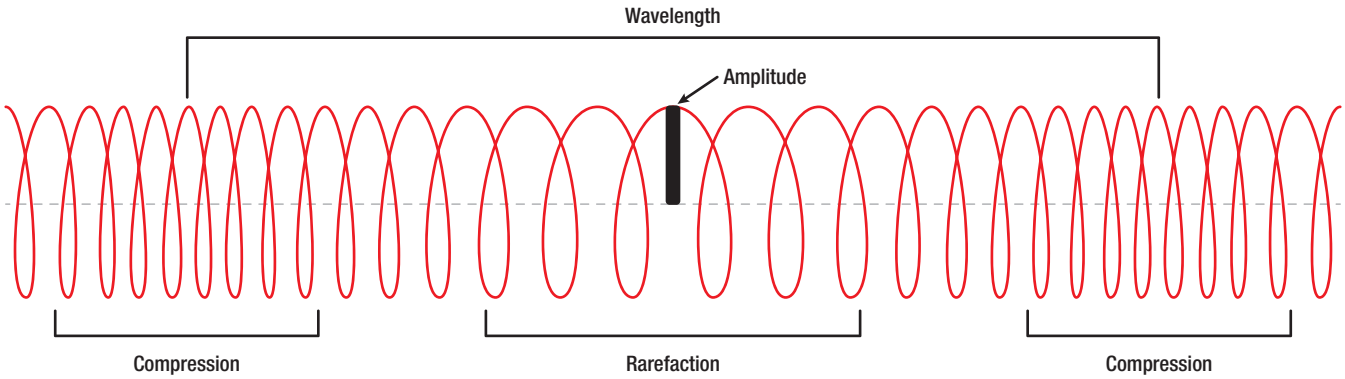


Figure 4-4. Regions of Compression and Rarefaction in a Longitudinal Wave.

The up and down motion is traverse, or at right angles (perpendicular) to the outward motion of the waves. This is called traverse wave motion. Radio waves, light waves, and microwaves are transverse waves. Transverse waves also travel on the strings of instruments such as guitars and banjos.

LONGITUDINAL (COMPRESSIONAL) WAVES

A longitudinal wave is a wave in which particles of the medium move in a direction parallel to the direction that the wave moves. Suppose that a slinky is stretched out in a horizontal direction across the classroom and that a pulse is introduced into the slinky on the left end by vibrating the first coil left and right as shown in ***Figure 4-3***. Energy will begin to be transported through the slinky from left to right. As the energy is transported from left to right, the individual coils of the medium will be displaced leftwards and rightwards. In this case, the particles of the medium move parallel to the direction that the pulse moves. This type of wave is a longitudinal wave, for example, a sound wave.

Longitudinal waves are characterized by regular regions of high and low pressure along the wave. High pressure regions are called compressions and low pressure regions are called rarefactions. (***Figure 4-4***)

BASIC WAVE TERMINOLOGY

Some common terms are used to describe the characteristics of waves. Taking into consideration, the earlier example of ripples created in water due to a falling stone, some basic wave terms can be easily explained as illustrated in ***Figure 4-5***.

Amplitude (A)

The amplitude of a traverse wave is half the distance measured vertically (height) from the crest to the trough.

Wave Length (λ)

A wavelength or one cycle, is the distance from the crest of one wave to the crest of the next, or between any two similar points on adjacent waves. In case of a radio wave, a wavelength would be the distance in space occupied by one cycle of a radio wave for a given period. Wavelengths

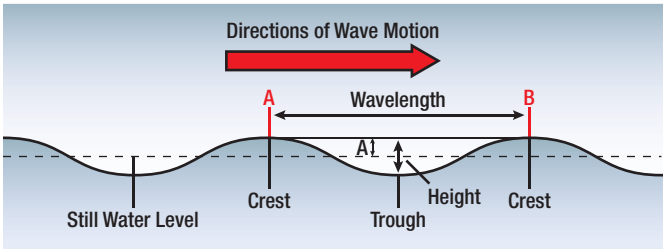


Figure 4-5. Wave terminology explained in a water ripple.

vary from a few hundredths of an inch at extremely high frequencies, to many miles at extremely low frequencies. Wavelengths are, however, expressed in meters.

Frequency (f)

The number of times the wave cycle repeats in a given unit of time is called the frequency of the wave and is measured in hertz (Hz). To honor the German physicist Heinrich Hertz, the term Hz was designated for use in lieu of the term "cycles per second" when referring to the frequency of radio waves. The frequency of household current is 60 Hz. The frequency of airborne radio equipment is generally in the millions-of-Hz range. The graph below (***Figure 4-6***) shows us three different waves with different frequencies. The red wave has the lowest frequency among the three because it has the least number of repeating cycles, and the blue wave has the highest frequency because it has the highest number of repeating cycles.

Frequency (f), can be obtained by dividing the wave's velocity, usually symbolized by the letter v, by its wavelength. Remember, we represent it with the Greek symbol: lambda. We usually measure the wavelength in meters and the velocity in meters per second. The frequency found using these units will be measured in is Hz (hertz), another way of saying cycles per second.

$$Frequency = \frac{Velocity}{Wavelength}$$

$$f = \frac{v}{\lambda}$$

Unit of Frequency: Hz

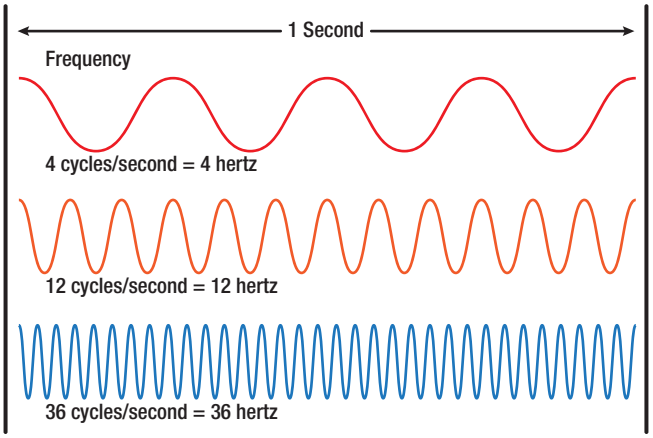


Figure 4-6. High and Low Frequency Waveforms.

Wave Period (T)

The wave period is the measure of time it takes for the wave cycle to complete, measured in seconds. We usually measure the wave period in seconds and represent it with the letter T. The wave period and frequency are inversely proportional to each other, so if the frequency increases, the period decreases, and vice-versa. In other words, if the frequency is large, then the period is short and if the frequency is small, then the period is long.

$$Wave\ Period = \frac{1}{Frequency} \quad or \quad T = \frac{1}{f}$$

$$Wave\ Period = \frac{Wavelength}{velocity} \quad or \quad T = \frac{\lambda}{v}$$

Unit of Wave Period: s

Let's consider an example, while looking at the ocean waves, you observed that the approximate velocity of a wave is 3 m/s, and the distance between the peaks of two waves is approximately 20m. What are the frequency and wave period for these waves? In the given problem, we know the velocity is 3m/s. We also know the wavelength, remember that's the distance between two peaks, so we can call the wavelength 20 meters. We can use these two bits of information to find the frequency. Divide the velocity, 3 m/s, find the wavelength, 20m, and we find that the frequency is 0.15 Hz or 0.15 cycles per second. Then we find the reciprocal of that number; 1 over 0.15 will give us the value of 6.67. That tells us that the wave period is 6.67 seconds.

Given:

velocity (v) = 3m/s
wavelength (λ) = 20m

$$f = \frac{v}{\lambda}$$

$$T = \frac{1}{f}$$

Substitute the Formulas:

$$f = \frac{3m/s}{20m}\ 0.15\ Hz$$

Final Answer:

The frequency, f = 0.15 Hz and the period, T = 6.67s

Wave Front

In the case of a water wave created by a rock dropped into a still pool of water, the wave fronts indicate the crests of water waves (***Figure 4-7***).

Rays

The radial motion of the waves is indicated by the outward-pointing arrows, referred to as rays.

ELECTROMAGNETIC WAVES

An electromagnetic wave consists of two primary components: an electric field and a magnetic field. The electric field results from the force of voltage, and the magnetic field results from the flow of current. Electromagnetic fields that are radiated are commonly considered to be waves and electromagnetic radiation in space can be interpreted as horizontal and vertical lines of force oriented at right angles to each other. These lines of force are made up of an electric field (E) and a magnetic field (H), which when combined make up the electromagnetic field. The electric and magnetic fields radiated from an antenna form just such an electromagnetic field which is responsible for the transmission and reception of electromagnetic energy through free space. (***Figure 4-8***)

An antenna is considered part of the electrical circuit of a transmitter or a receiver and has factors including inductance, capacitance, and resistance. Which means the antenna can be expected to display definite voltage

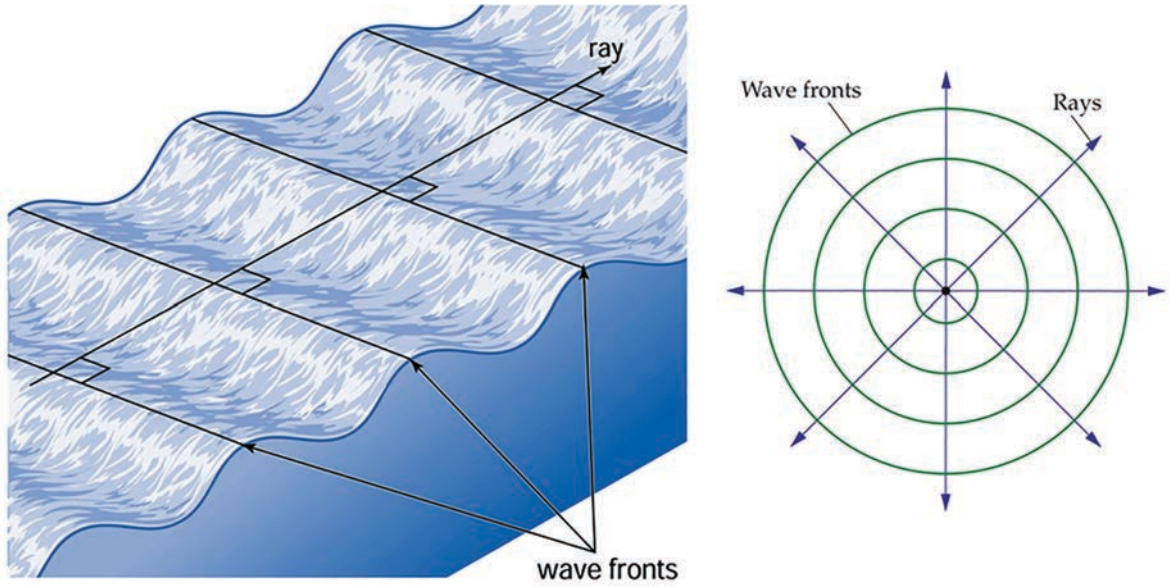


Figure 4-7. Wave fronts and Rays.

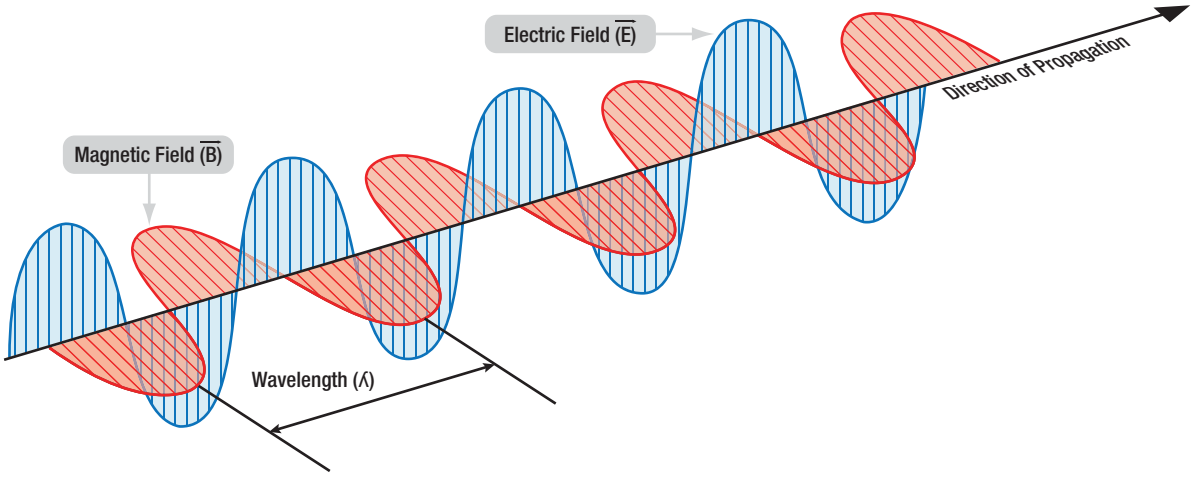


Figure 4-8. Electric and Magnetic Field Components of an Electromagnetic Wave.

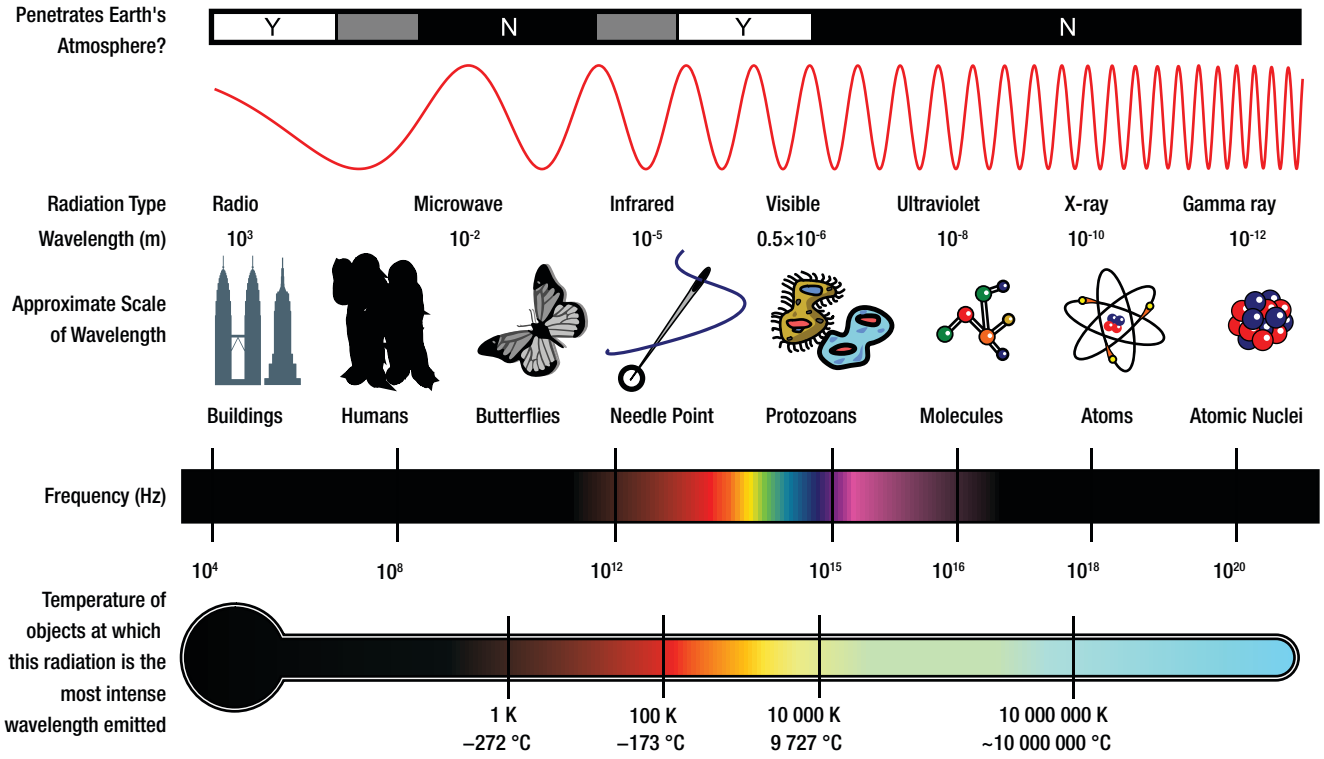


Figure 4-9. The Electromagnetic Spectrum.

and current relationships with respect to a given input. A current through the antenna produces a magnetic field, and a charge on the antenna produces an electric field. These two fields combine to form the inductive field.

ELECTROMAGNETIC SPECTRUM

An electromagnetic radiation is classified by wavelength into radio, microwave, infrared, visible, ultraviolet, X-rays and gamma rays (***Figure 4-9***). The behavior of EM radiation depends on its frequency. Lower frequencies have longer wavelengths, and higher frequencies have shorter wavelengths, and are associated with photons of higher energy. It is pertinent to mention that sound waves are not classified as electromagnetic radiation. At the lower end of the electromagnetic spectrum, about 20 Hz to about 20 kHz, are frequencies that might be considered in the audio range.

RADIO WAVES

Radio waves belong to the electromagnetic radiation family, which includes x-ray, ultraviolet, and visible light — forms of energy we use every day. Much like the gentle waves that form when a stone is tossed into a still lake, radio signals radiate outward, or propagate, from a transmitting antenna and intercepted by a receiver. (***Figure 4-10***) However, unlike water waves, radio waves propagate at the speed of light. We characterize a radio wave in terms of its amplitude, frequency, and wavelength. Radio wave amplitude, or strength, can be visualized as its height — the distance between its peak and its lowest point. Amplitude, which is measured in volts, is usually expressed in terms of an average value called root-mean-square, or RMS. The RMS values of salient waveforms are shown in ***Figure 4-11***.

Since radio waves propagate at the speed of light (300 million meters per second), you can easily determine the wavelength in meters for any frequency by dividing 300 by the frequency in megahertz. So, the wavelength of a 10 MHz wave is 30 meters, determined by dividing 300 by 10.

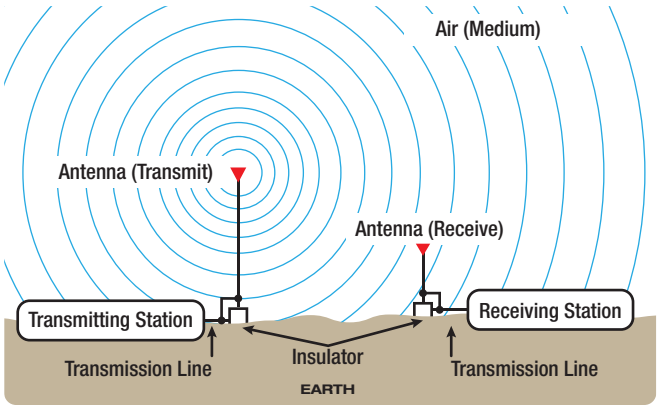


Figure 4-10. Radio wave transmission and reception.

Wave Type	Wave Type	RMS Value	Crest Factor
Sine Wave		$\frac{1}{\sqrt{2}} \approx 0.707$	$\sqrt{2} \approx 1.414$
Half Wave Rectified Sine		$\frac{1}{2} \approx 0.5$	2
Full Wave Rectified Sine		$\frac{1}{\sqrt{2}} \approx 0.707$	$\sqrt{2} \approx 1.414$
Triangle Wave		$\frac{1}{\sqrt{3}} \approx 0.577$	$\sqrt{3} \approx 1.732$
Square Wave		1	2

Figure 4-11. RMS Values of Salient Waveforms.

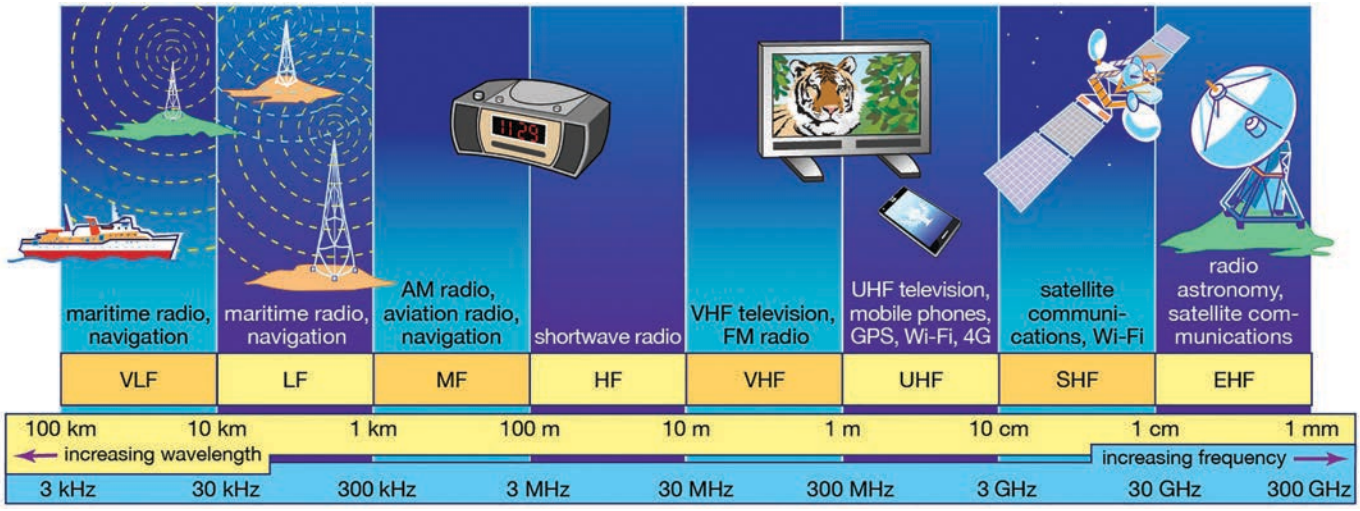


Figure 4-12. Radio frequency spectrum and various usages.

In aviation, a variety of radio waves are used for communication. ***Figure 4-12*** illustrates the radio spectrum that includes the range of common aviation radio frequencies and their applications. A wide range of frequencies are used from low frequency (LF) at 100 kHz (100 000 cycles per second) to super high frequency (SHF) at nearly 10 gHz (10 000 000 000 cycles per second).

RADIO WAVE GENERATION

AC power of a particular frequency has a characteristic length of conductor that is resonant at that frequency. This length is the wavelength of the frequency that can be seen on an oscilloscope. Fractions of the wavelength also resonate, especially half of a wavelength, which is the same as half of the AC sign wave or cycle. The frequency of an AC signal is the number of times the AC cycles every second. AC applied to the center of a radio antenna, a conductor half the wavelength of the AC frequency, travels the length of the antenna, collapses, and travels the length of the antenna in the opposite direction. The number of times it does this every second is known as the radio wave signal frequency or radio frequency. The radio frequency spectrum and wide usage across the aviation industry is shown in ***Figure 4-13***. As the current flows through the antenna, corresponding electromagnetic and electric fields build, collapse, build in the opposite direction, and collapse again. (***Figure 4-14***)

To transmit radio waves, an AC generator is placed at the midpoint of an antenna. As AC current builds and collapses in the antenna, a magnetic field also builds and collapses around it. An electric field also builds and subsides as the voltage shifts from one end of the antenna to the other. Both fields, the magnetic and the

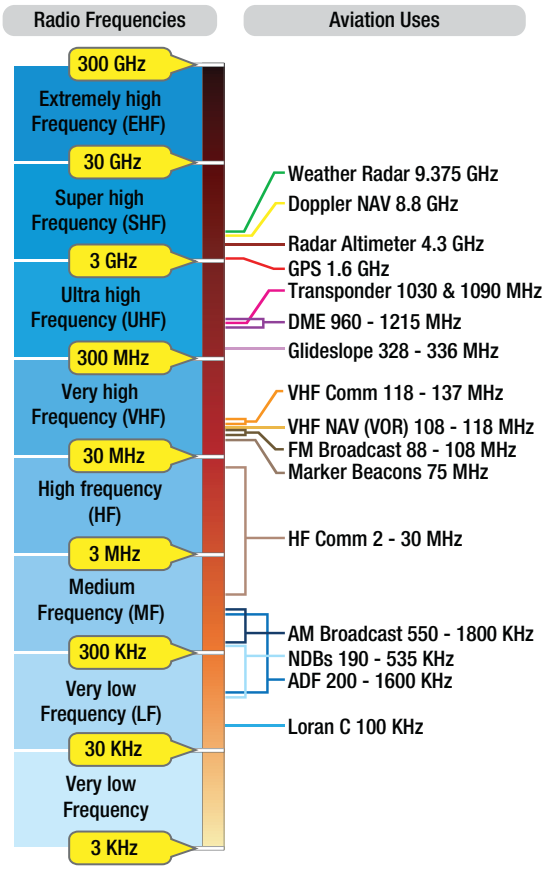


Figure 4-13. Radio frequency spectrum and usage in aviation.

electric, fluctuate around the antenna at the same time. The antenna is half the wavelength of the AC signal received from the generator. At any one point along the antenna, voltage and current vary inversely to each other.

RADIO FREQUENCY SPECTRUM

In the radio frequency spectrum the usable frequency range for radio waves extends from about 20 kHz (just above sound waves) to above 30 000 MHz. (***Figure 4-15***) A wavelength at 20 kHz is 15 kilometers long. At 30 000 MHz or 30 GHz, the wavelength is only 1 centimeter. In aviation, the radio frequency bands of crucial importance are the HF, VHF and UHF bands, which are discussed briefly below.

High Frequency (Hf) Band

The HF band is defined as the frequency range of 3 to 30 MHz. In practice, most HF radios use the spectrum from 1.6 to 30 MHz. Most long-haul communications in this band take place between 4 and 18 MHz. Higher frequencies (18 to 30 MHz) may also be available from time to time, depending on ionospheric conditions and the time of day.

Very High Frequency (VHF) Band

The VHF frequency band is defined as the frequency range from 30 to 300 MHz. From the previous discussion about the relationship between frequency and wavelength, it should be noted that VHF wavelengths vary from 10 meters at the low end to one meter at the high end. This means that the size of antennas and tuning components used in VHF radio are much smaller and lighter than those of HF radios. The higher frequency and shorter wavelengths of VHF radios have a profound effect on radio range, which shall be discussed in detail later.

Ultra High Frequency (UHF) Band

The UHF band goes from 300 MHz to 2450 MHz, although most UHF radios do not utilize frequencies above 512 MHz. The wavelengths associated with 300 to 512 MHz range from one meter to 0.58 meters (58 centimeters). The very small antennas required for these wavelengths make them ideal for use on high-speed aircraft.

MODULATION

By itself, a radio wave conveys no information. It's simply a rhythmic stream of continuous waves (CW). When we modulate radio waves to carry information, we refer to them as carriers. To convey information, a carrier must be varied so that its properties — its amplitude, frequency, or phase (the measurement of a complete wave cycle) — are changed, or modulated, by the information signal. When the Italian inventor Guglielmo Marconi was developing the first practical radio system in 1896, there was no way to modulate the radio wave in order to transmit voice so he used a method of switching the transmitter on and off to transmit Morse code signals. This simplest form of modulation is called CW or radio telegraphy since it borrowed the Morse code from the telegraph industry. The simplest method of modulating a carrier is by turning it on and off by means of a telegraph key. In the early days of radio, On-Off keying, using Morse code, was the only method of conveying wireless messages.

There are many different varieties of modulation but they all fall into three basic categories, namely amplitude modulation, frequency modulation and phase modulation, although frequency and phase modulation are essentially the same. Each type has its own advantages and disadvantages. A review of all three

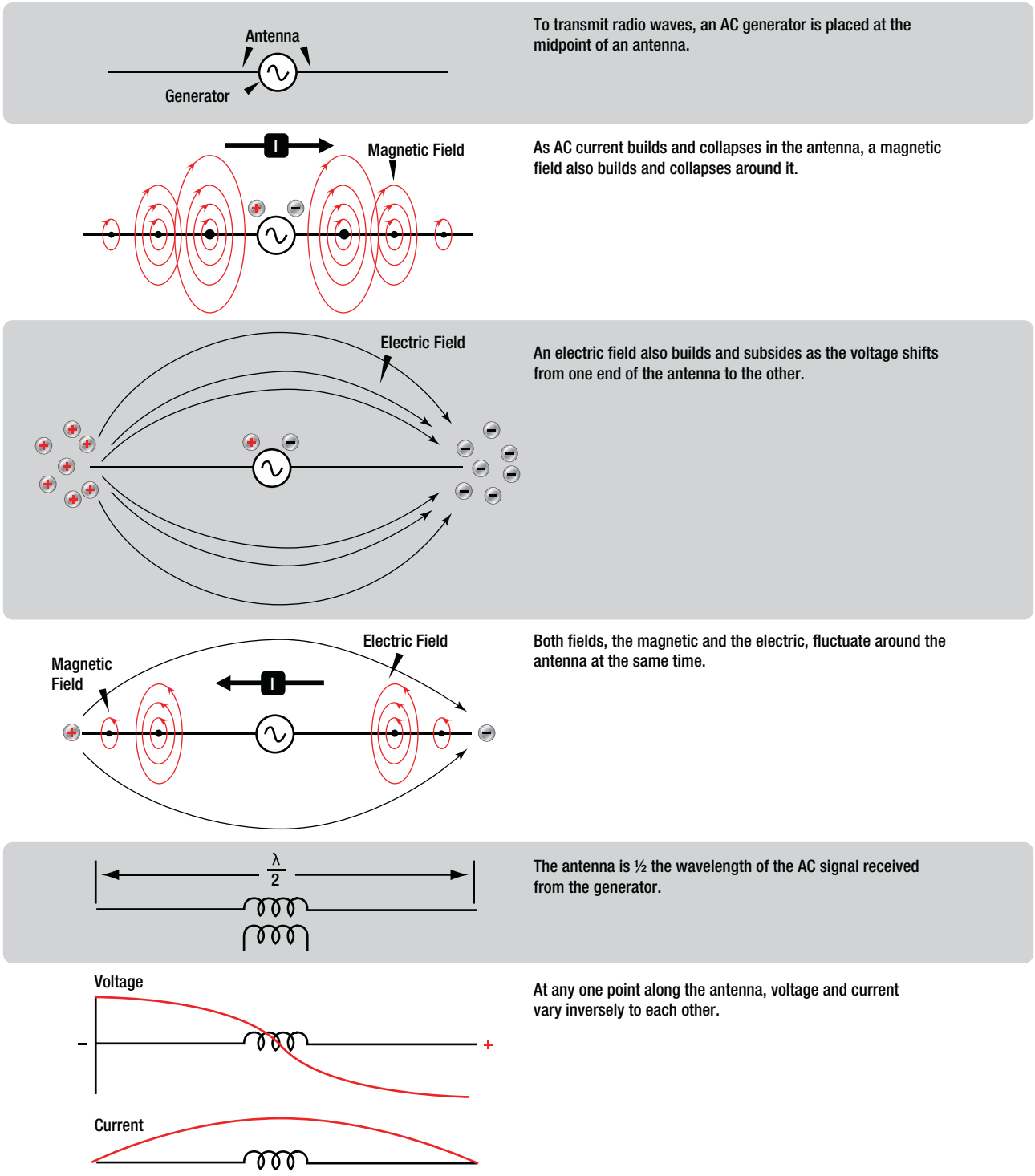


Figure 4-14. Radio waves are produced by applying an AC signal to an antenna. This creates a magnetic and electric field around the antenna. They build and collapse as the AC cycles. The speed at which the AC cycles does not allow the fields to completely collapse before the next fields build. The collapsing fields are then forced out into space as radio waves.

basic types will be undertaken, although a much greater focus will be placed on those types used within phone systems. By reviewing all the techniques, a greater understanding of the advantages and disadvantages can be gained.

Radio Carrier

The basis of any radio signal or transmission is the carrier. This consists of an alternating waveform like that shown in ***Figure 4-16***. This is generated in the transmitter, and if it is radiated in this form it carries no information – it appears at the receiver as a constant signal.

Band	ITU Designation	Frequencies	Wavelength
Very Low Frequency	VLF	3-30 kHz	100 - 10 km
Low Frequency	LF	30-300 kHz	10 - 1 km
Medium Frequency	MF	300 kHz - 3 mHz	1000 - 100 m
High Frequency	HF	3 - 30 mHz	100 - 10 m
Very High Frequency	VHF	30 - 300 mHz	10 - 1 m
Ultra High Frequency	UHF	300 mHz - 3 GHz	1 - 0.1 m
Super High Frequency	SHF	3 - 300 GHz	0.1 m - 1 cm
Extremely High Frequency	EHF	300 GHz - 3 THz	1 cm - 1mm

Figure 4-15. Radio frequency spectrum bands.

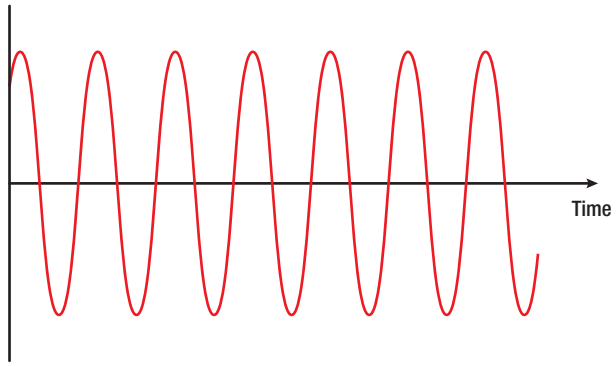


Figure 4-16. An alternating waveform.

Amplitude Modulation

Possibly the most obvious method of modulating a carrier is to change its amplitude in line with the modulating signal. The simplest form of amplitude modulation is to employ a system known as 'on–off keying', where the carrier is simply turned on and off. This is a very elementary form of digital modulation and was the method used to carry Morse transmissions, which were widely used especially in the early days of 'wireless'. Here, the length of the on and off periods defined the different characters.

More generally, the amplitude of the overall signal is varied in line with the incoming audio or other modulating signal, as shown in ***Figure 4-17***. Here, the envelope of the carrier can be seen to change in line with the modulating signal. This is known as Amplitude Modulation (AM).

The demodulation process for AM where the radio frequency signal is converted into an audio frequency signal is very simple. It only requires a simple diode detector circuit like that shown in ***Figure 4-18***. In this circuit the diode rectifies the signal, only allowing the one-half of the alternating radio frequency waveform through. A capacitor is used as a simple low-pass filter to

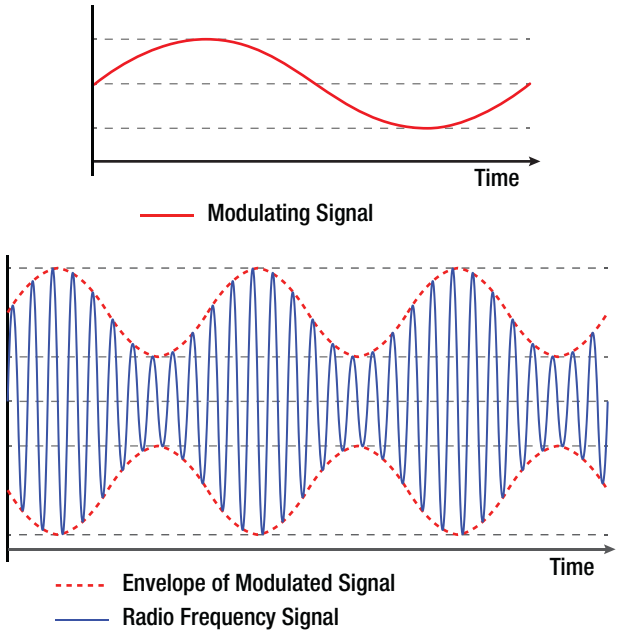


Figure 4-17. An amplitude modulated signal.

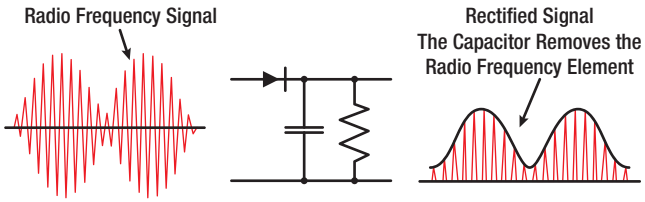


Figure 4-18. A simple diode detector circuit.

remove the radio-frequency parts of the signal, leaving the audio waveform. This can be fed into an amplifier; after which it can be used to drive a loudspeaker. This form of demodulator is very cheap and easy to implement, and is still widely used in many AM receivers today.

The signal may also be demodulated more efficiently using a system known as synchronous detection (***Figure 4-19***). Here, the signal is mixed with a locally generated signal with the same frequency and phase as the carrier. In this way the signal is converted down to the baseband frequency. This system has the advantage of a more linear demodulation characteristic than the diode detector, and it is more resilient to various forms of distortion. There are various methods of generating the mix signal. One of the easiest is to take a feed from the signal being received and pass it through a very high-gain amplifier. This removes any modulation, leaving just the carrier with exactly the required frequency and phase. This can be mixed with the incoming signal and the result filtered to recover the original audio.

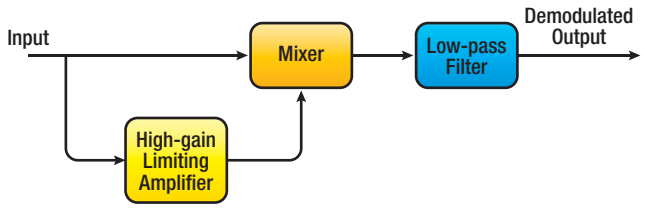


Figure 4-19. Synchronous AM demodulation.

AM has the advantage of simplicity, but it is not the most efficient mode to use – both in terms of the amount of spectrum it takes up and the usage of the power. For this reason, it is rarely used for communications purposes. Its only major communications use is for VHF aircraft communications. However, it is still widely used on the long, medium, and short wave bands for broadcasting because its simplicity enables the cost of radio receivers to be kept to a minimum.

To find out why it is inefficient, it is necessary to look at a little theory behind the operation of AM. When a radio-frequency signal is modulated by an audio signal, the envelope will vary. The level of modulation can be increased to a level where the envelope falls to zero and then rises to twice the unmodulated level. Any increase above this will cause distortion because the envelope cannot fall below zero. As this is the maximum amount of modulation possible, it is called 100 per cent modulation. (***Figure 4-20***)

Even with 100 per cent modulation, the utilization of power is very poor. When the carrier is modulated, sidebands appear at either side of the carrier in its frequency spectrum. Each sideband contains the information about the audio modulation. To look at how the signal is made up and the relative powers, take the simplified case where the 1 kHz tone is modulating the carrier. In this case, two signals will be found: 1 kHz either side of the main carrier, as shown in ***Figure 4-21***.

When the carrier is fully modulated (i.e. 100 per cent), the amplitude of the modulation is equal to half that of the main carrier – that is, the sum of the powers of the sidebands is equal to half that of the carrier. This means that each sideband is just a quarter of the total power. In other words, for a transmitter with a 100 watt carrier, the total sideband power will be 50 W and each individual sideband will be 25 W. During the modulation process the carrier power remains constant. It is only needed as a reference during the demodulation process. This

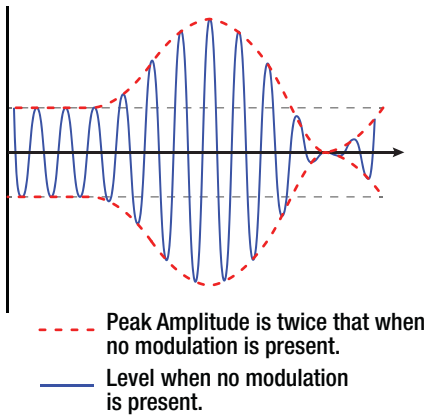


Figure 4-20. Fully modulated signal.

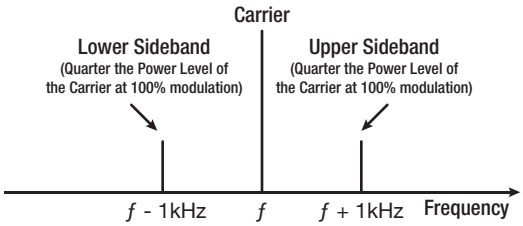


Figure 4-21. Spectrum of a signal modulated with a 1-kHz tone.

means that the sideband power is the useful section of the signal, and this corresponds to (50/150) × 100 per cent, or only 33 per cent of the total power transmitted.

Not only is AM wasteful in terms of power; it is also not very efficient in its use of the spectrum. If the 1 kHz tone is replaced by a typical audio signal made up of a variety of sounds with different frequencies, then each frequency will be present in each sideband. (***Figure 4-22***)

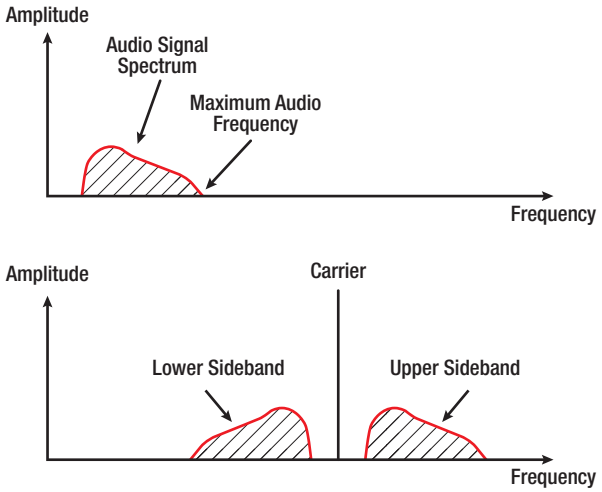


Figure 4-22. Spectrum of a signal modulated with speech or music.

Accordingly, the sidebands spread out either side of the carrier as shown and the total bandwidth used is equal to twice the top frequency that is transmitted. In the crowded conditions found on many of the short wave bands today this is a waste of space, and other modes of transmission that take up less space are often used.

To overcome the disadvantages of amplitude modulation, a derivative known as Single Sideband (SSB) is often used. By removing or reducing the carrier and removing one sideband, the bandwidth can be halved and the efficiency improved. The carrier can be introduced by the receiver for demodulation.

Modulation Index

It is often necessary to define the level of modulation that is applied to a signal. A factor or index known as the modulation index is used for this. When expressed as a percentage, it is the same as the depth of modulation. In other words, it can be expressed as:

$$M = \frac{RMS\ value\ of\ modulating\ signal}{RMS\ value\ of\ unmodulated\ signal}$$

The value of the modulation index must not be allowed to exceed 1 (i.e. 100 per cent in terms of the depth of modulation), otherwise the envelope becomes distorted and the signal will spread out either side of the wanted channel, causing interference to other users.

In summary, Amplitude Modulation (***Figure 4-23***) has the following distinct features:

- A carrier signal is modulated only in amplitude value.
- The modulating signal is the envelope of the carrier.
- The required bandwidth is 2B, where B is the bandwidth of the modulating signal.
- Since on both sides of the carrier freq. fc, the spectrum is identical, we can discard one half, thus requiring a smaller bandwidth for transmission.
- The total bandwidth required for AM can be determined from the bandwidth of the audio signal: BAM = 2B.

Frequency Modulation

While AM is the simplest form of modulation to envisage, it is also possible to vary the frequency of the signal to give frequency modulation (FM). It can be seen from ***Figure 4-24*** that the frequency of the signal varies as the voltage of the modulating signal changes.

The amount by which the signal frequency varies is very important. This is known as the deviation, and is normally quoted in kilohertz. As an example, the signal may have a deviation of ±3 kHz. In this case, the carrier is made to move up and down by 3 kHz.

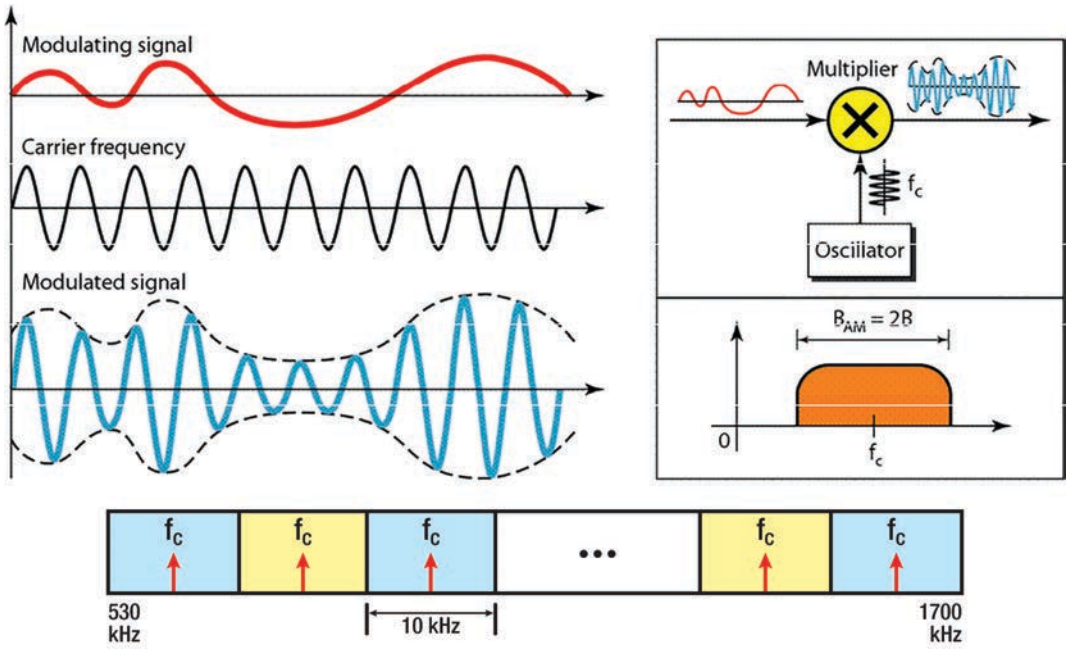


Figure 4-23. Amplitude Modulation and AM Band Allocation.

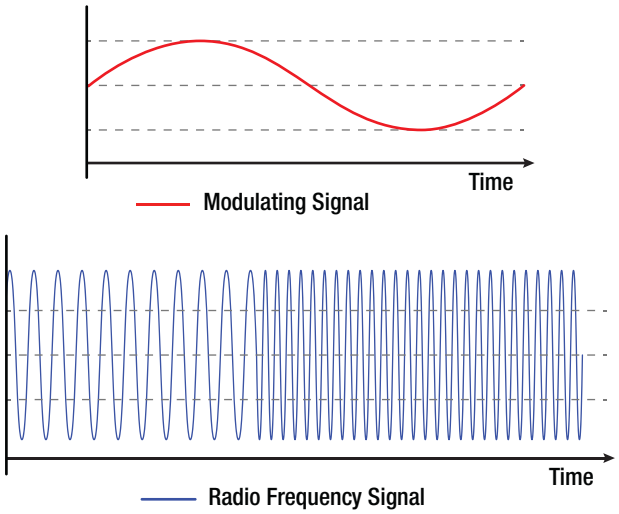


Figure 4-24. A frequency modulated signal.

FM is used for a number of reasons. One particular advantage is its resilience to signal-level variations and general interference. The modulation is carried only as variations in frequency, and this means that any signal-level variations will not affect the audio output provided that the signal is of a sufficient level. As a result, this makes FM ideal for mobile or portable applications where signal levels vary considerably. The other advantage of FM is its resilience to noise and interference when deviations much greater than the highest modulating frequency are used. It is for this reason that FM is used for high-quality broadcast transmissions where deviations of ±75 kHz are typically used to provide a high level of interference rejection. In view of these advantages, FM was chosen for use in the first-generation analogue mobile phone systems.

To demodulate an FM signal, it is necessary to convert the frequency variations into voltage variations. This is slightly more complicated than demodulating AM, but it is still relatively simple to achieve. Rather than just detecting the amplitude level using a diode, a tuned circuit has to be incorporated so that a different output voltage level is given as the signal changes its frequency. There is a variety of methods used to achieve this, but one popular approach is to use a system known as a quadrature detector. It is widely used in integrated circuits, and provides a good level of linearity. It has the advantages that it requires a simple tuned circuit and it is also very easy to implement in a form that is applicable to integrated circuits.

The basic format of the quadrature detector is shown in ***Figure 4-25***. It can be seen that the signal is split into two components. One of these passes through a network that provides a basic 90° phase shift, plus an element of phase shift dependent upon the deviation. The original signal and the phase-shifted signal are then passed into a multiplier or mixer. The mixer output is dependent upon the phase difference between the two signals, i.e. it acts as a phase detector and produces a voltage output that is proportional to the phase difference and hence to the level of deviation of the signal.

In summary, Frequency Modulation (***Figure 4-26***) has the following distinct features:

- The modulating signal changes the frequency of the carrier signal.
- The bandwidth for FM is high.
- It is approximately 10× the signal frequency.
- The total bandwidth required for FM can be determined from the bandwidth of the audio signal: BFM = 2(1 + ß)B. Where b is usually 4.

Modulation Index And Deviation Ratio

In many instances a figure known as the modulation index is of value and is used in other calculations. The modulation index is the ratio of the frequency deviation to the modulating frequency, and will therefore vary according to the frequency that is modulating the transmitted carrier and the amount of deviation:

$$M = \frac{\textit{Frequency deviation}}{\textit{Modulation frequency}}$$

However, when designing a system, it is important to know the maximum permissible values. This is given by the deviation ratio, and is obtained by inserting the maximum values into the formula for the modulation index:

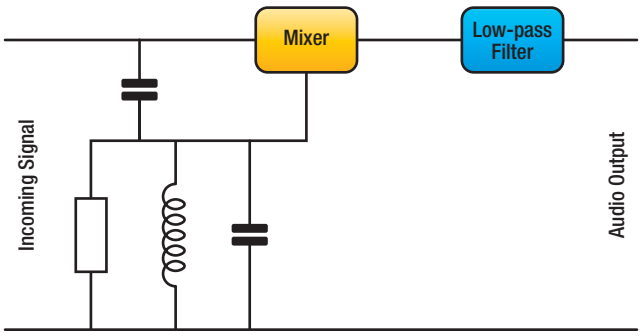


Figure 4-25. Block diagram of an FM quadrature detector.

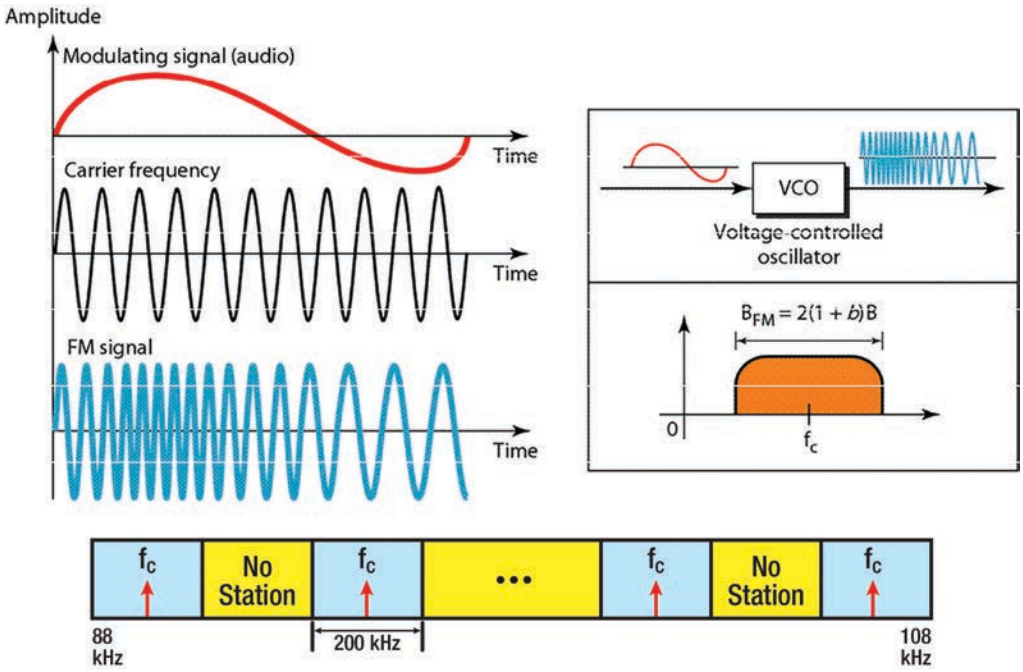


Figure 4-26. Frequency Modulation and FM Band Allocation.

$$D = \frac{Maximum\ frequency\ deviation}{Maximum\ modulation\ frequency}$$

Sidebands

Any signal that is modulated produces sidebands. In the case of an amplitude modulated signal they are easy to determine, but for frequency modulation the situation is not quite as straightforward. They are dependent upon not only the deviation, but also the level of deviation – i.e., the modulation index M. The total spectrum is an infinite series of discrete spectral components, expressed by the complex formula:

$$\begin{aligned} \textit{Spectrum components} = {} & V_c\{J_0(M)\cos \omega_c t \\ & + J_1(M)[\cos(\omega_c + \omega_m)t - \cos(\omega_c + \omega_m)t] \\ & + J_2(M)[\cos(\omega_c + \omega_m)t - \cos(\omega_c + 2\omega_m)t] \\ & + J_3(M)[\cos(\omega_c + \omega_m)t - \cos(\omega_c + 3\omega_m)t] \\ & + \ldots\}. \end{aligned}$$

In this relationship, Jn(M) are Bessel functions of the first kind, ω_c is the angular frequency of the carrier and is equal to $2\pi f$, and ω_m is the angular frequency of the modulating signal. V_c is the voltage of the carrier. It can be seen that the total spectrum consists of the carrier plus an infinite number of sidebands spreading out on either side of the carrier at integral frequencies of the modulating frequency. The relative levels of the sidebands can be read from a table of Bessel functions, or calculated using a suitable computer program.

Figure 4-27 shows the relative levels to give an indication of the way in which the levels of the various sidebands change with different values of modulation index.

It can be gathered that for small levels of deviation (that is, what is termed narrowband FM) the signal consists of the carrier and the two sidebands spaced at the modulation frequency either side of the carrier. The spectrum appears the same as that of an AM signal. The major difference is that the lower sideband is out of phase by 180°.

As the modulation index increases, other sidebands at twice the modulation frequency start to appear (***Figure 4-28***). As the index is increased, further sidebands can also be seen. It is also found that the relative levels of these sidebands change, some rising in level and others falling as the modulation index varies.

Bandwidth

It is clearly not acceptable to have a signal that occupies an infinite bandwidth. Fortunately, for low levels of modulation index all but the first two sidebands may be ignored. However, as the modulation index increases the sidebands further out increase in level, and it is often necessary to apply filtering to the signal. This should not introduce any undue distortion. To achieve this, it is normally necessary to allow a bandwidth equal to twice the maximum frequency of deviation plus the maximum

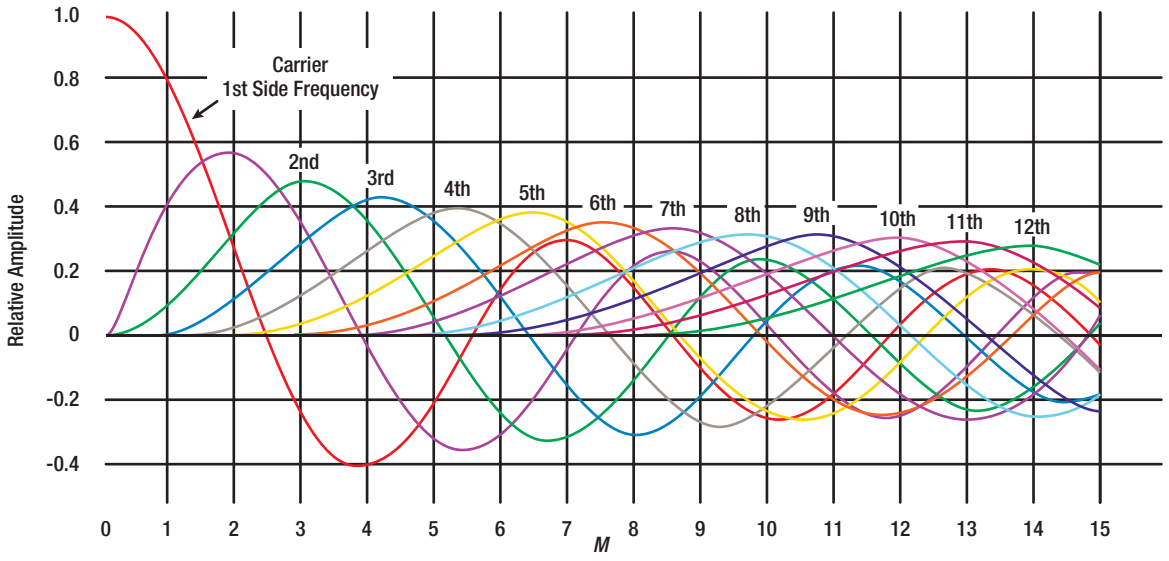


Figure 4-27. The relative amplitudes of the carrier and the first 10 side frequency components of a frequency modulated signal for different values of modulation index.

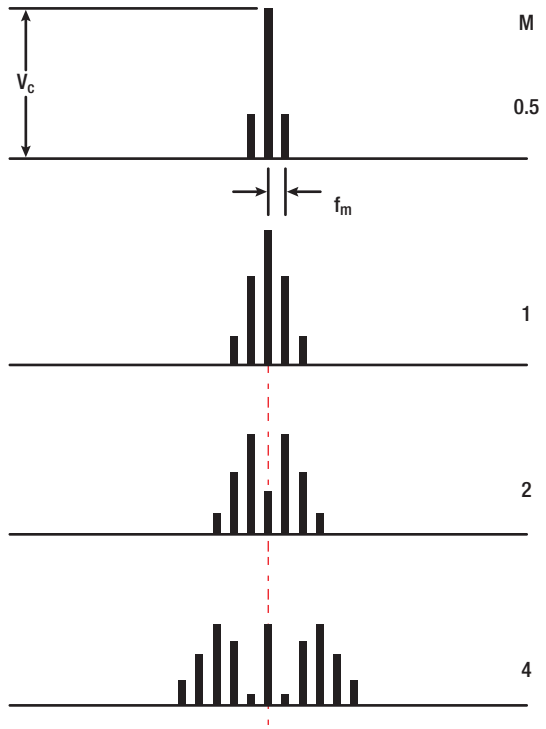


Figure 4-28. Spectra of frequency-modulated signals with various values of modulation index for a constant modulation frequency. It can be seen that for small values of the modulation index M (e.g. M = 0.5), the signal appears to consist of the carrier and two sidebands. As the modulation index increases, the number of sidebands increases and the level of the carrier can be seen to decrease for these values.

modulation frequency. In other words, for a VHF FM broadcast station with a deviation of ±75 kHz and a maximum modulation frequency of 15 kHz, this must be (2 × 75) + 15 kHz, i.e. 175 kHz. In view of this a total of 200 kHz is usually allowed, enabling stations to have a small guard band and their center frequencies on integral numbers of 100 kHz.

Single Side Band (SSB)

When two AC signals are mixed together, such as when a carrier wave is modulated by an information signal, three main frequencies result:

- Original carrier wave frequency;
- Carrier wave frequency plus the modulating frequency;
- Carrier wave frequency minus the modulating frequency.

Due to the fluctuating nature of the information signal, the modulating frequency varies from the carrier wave up or down to the maximum amplitude of the modulating frequency during AM. These additional frequencies on either side of the carrier wave frequency are known as side bands. Each side band contains the unique information signal desired to be conveyed. The entire range of the lower and upper sidebands including the center carrier wave frequency is known as bandwidth. (***Figure 4-29***).

There are a limited number of frequencies within the usable frequency ranges (i.e., LF, HF, and VHF). If different broadcasts are made on frequencies that are too close together, some of the broadcast from one frequency interfere with the adjacent broadcast due to overlapping side bands. The authorities divide the

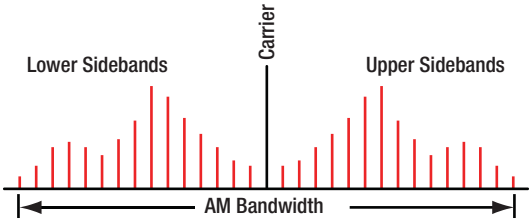


Figure 4-29. The bandwidth of an AM signal contains the carrier wave, the carrier wave plus the information signal frequencies, and the carrier wave minus the information signal frequencies.

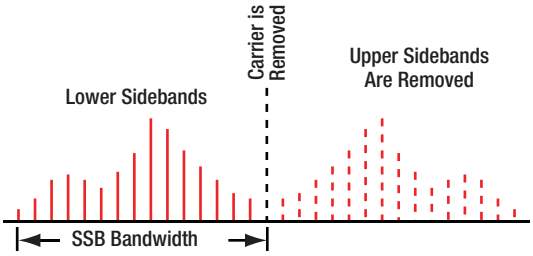


Figure 4-30. The additional frequencies above and below the carrier wave produced during modulation with the information signal are known as sidebands. Each sideband contains the unique information of the information signal and can be transmitted independent of the carrier wave and the other sideband.

various frequency bands and issues rules for their use. Much of this allocation is to prevent interference. The spacing between broadcast frequencies is established so that a carrier wave can expand to include the upper and lower side bands and still not interfere with a signal on an adjacent frequency. As use of the radio frequencies increases, more efficient allocation of bandwidth is imperative. Sending information via radio waves using the narrowest bandwidth possible is the focus of engineering moving forward. At the same time, fully representing all of the desired information or increasing the amount of information conveyed is also desired.

Various methods are employed to keep bandwidth to a minimum, many of which restrict the quality or quantity of information able to be transmitted. In lower frequency ranges, such as those used for ground wave and some sky wave broadcasts, SSB transmissions are a narrow bandwidth solution. Each side band represents the initial information signal in its entirety. Therefore, in an SSB broadcast, the carrier wave and either the upper or lower sidebands are filtered out. Only one sideband with its frequencies is broadcast since it contains all of the needed information. This cuts the bandwidth required in half and allows more efficient use of the radio spectrum. SSB transmissions also use less power to transmit the same amount of information over an equal distance. Many HF long distance aviation communications are SSB. (***Figure 4-30***).

Phase Modulation

Another form of modulation that is widely used, especially for data transmissions, is Phase Modulation (PM). As phase and frequency are inextricably linked (frequency being the rate of change of phase), both forms of modulation are often referred to by the common term 'angle modulation'.

To explain how phase modulation works, it is first necessary to give an explanation of phase. A radio signal consists of an oscillating carrier in the form of a sine wave. The amplitude follows this curve, moving positive and then negative, and returning to the start point after one complete cycle. This can also be represented by the movement of a point around a circle, the phase at any given point being the angle between the start point and the point on the waveform as shown in ***Figure 4-31***.

Modulating the phase of the signal changes the phase from what it would have been if no modulation were applied. In other words, the speed of rotation around the circle is modulated about the mean value. To achieve this, it is necessary to change the frequency of the signal for a short time. In other words, when phase modulation is applied to a signal there are frequency changes and vice versa. Phase and frequency are inseparably linked, as phase is the integral of frequency. Frequency modulation can be changed to phase modulation by simply adding a CR network to the modulating signal that integrates the

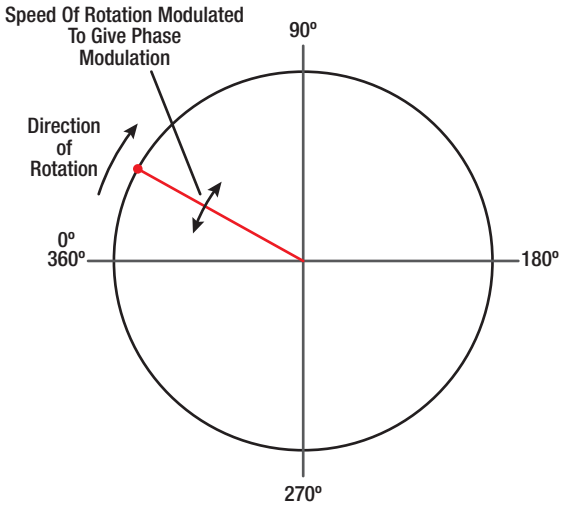


Figure 4-31. Phase modulation.

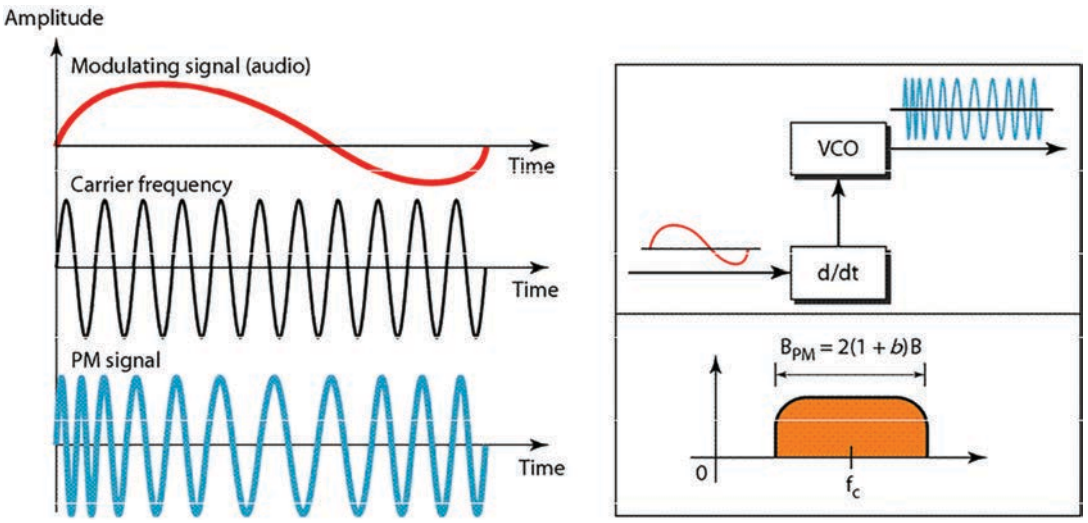


Figure 4-32. Phase modulated signal and bandwidth characteristics.

modulating signal. As such, the information regarding sidebands, bandwidth and the like also holds true for phase modulation as it does for frequency modulation, bearing in mind their relationship. In summary, Phase Modulation (***Figure 4-32***) has the following distinct features:

- The modulating signal only changes the phase of the carrier signal.
- The phase change manifests itself as a frequency change but the instantaneous frequency change is proportional to the derivative of the amplitude.
- The bandwidth is higher than for AM.
- The total bandwidth required for PM can be determined from the bandwidth and maximum amplitude of the modulating signal:
- BPM = 2(1 + β)B.
- Where β = 2 most often.

RADIO WAVE PROPAGATION

The Radio waves of different frequencies have unique characteristics as they propagate through the atmosphere. Very low frequency (VLF), LF, and medium frequency (MF) waves have relatively long wavelengths and utilize correspondingly long antennas. Radio waves produced at these frequencies ranging from 3 kHz to 3 mHz are known as ground waves or surface waves. This is because they follow the curvature of the earth as they travel from the broadcast antenna to the receiving antenna. Ground waves are particularly useful for long distance transmissions. Automatic direction finders (ADF) and LORAN navigational aids use these frequencies. (***Figure 4-33***)

High frequency (HF) radio waves travel in a straight line and do not curve to follow the earth's surface. This would limit transmissions from the broadcast antenna to receiving antennas only in the line-of-sight of the broadcast antenna except for a unique characteristic. HF radio waves bounce off of the ionosphere layer of the atmosphere. This refraction extends the range of HF signals beyond line-of-sight. As a result, transoceanic aircraft often use HF radios for voice communication. The frequency range is between 2-25 MHz. These kinds of radio waves are known as sky waves. (***Figure 4-33***)

Above HF transmissions, radio waves are known as space waves. They are only capable of line-of-sight transmission and do not refract off of the ionosphere (***Figure 4-33***). Most aviation communication and navigational aids operate with space waves. This includes VHF (30-300MHz), UHF (300MHz-3GHz), and super high frequency (SHF) (3Ghz-30Ghz) radio waves. VHF communication radios are the primary communication radios used in aviation. They operate in the frequency range from 118.0 MHz to 136.975MHz. Seven hundred and twenty separate and distinct channels have been designated in this range with 25 kilohertz spacing between each channel. Further division of the bandwidth is possible, such as in Europe where 8.33 kilohertz separate each VHF communication channel. VHF radios are used for communications between aircraft and air traffic control (ATC), as well as air-to-air communication between aircraft. When using VHF, each party transmits and receives on the same channel. Only one party can transmit at any one time.

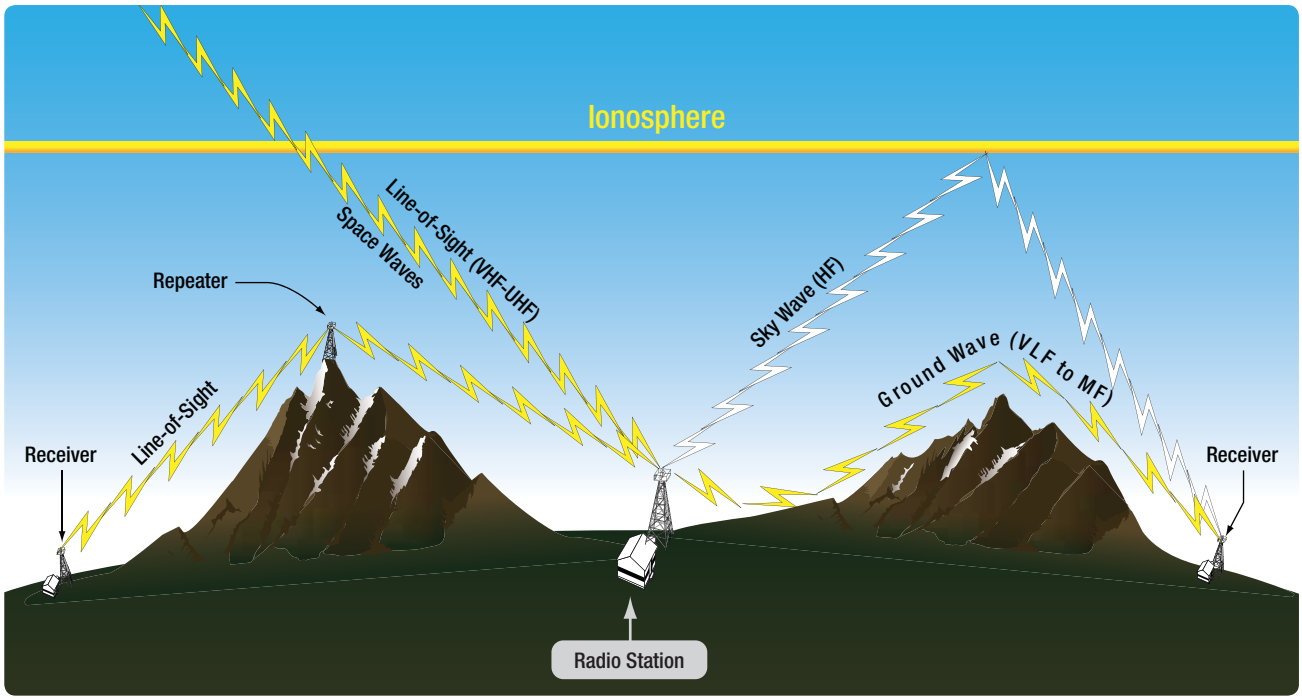


Figure 4-33. Radio waves behave differently in the atmosphere depending in their frequency.

ANTENNAS

As stated, antennas are conductors that are used to transmit and receive radio frequency waves. Although the airframe technician has limited duties in relation to maintaining and repairing avionics, it is the responsibility of the technician to install, inspect, repair, and maintain aircraft radio antennas.

Three characteristics are of major concern when considering antennas:

- Length
- Polarization
- Directivity

The exact shape and material from which an antenna is made can alter its transmitting and receiving characteristics. Also note that some non-metallic aircraft have antennas embedded into the composite material as it is built up.

Antenna Length

When an AC signal is applied to an antenna, it has a certain frequency. There is a corresponding wavelength for that frequency. An antenna that is half the length of this wavelength is resonant. During each phase of the applied AC, all voltage and current values experience the full range of their variability. As a result, an antenna that is half the wavelength of the corresponding AC frequency is able to allow full voltage and full current flow for the positive phase of the AC signal in one direction. The negative phase of the full AC sign wave is accommodated by the voltage and current simply changing direction in the conductor. Thus, the applied AC frequency flows through its entire wavelength, first in one direction and then in the other. This produces the strongest signal to be radiated by the transmitting antenna. It also facilitates capture of the wave and maximum induced voltage in the receiving antenna. (***Figure 4-34***)

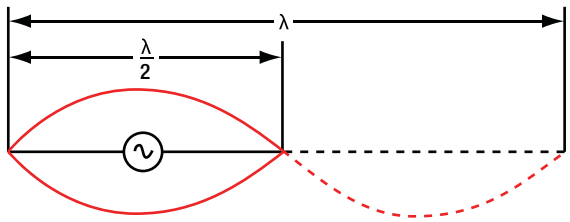


Figure 4-34. An antenna equal to the full length of the applied AC frequency wavelength would have the negative cycle current flow along the antenna as shown by the dotted line. An antenna that is 1/2 wavelength allows current to reverse its direction in the antenna during the negative cycle. This results in low current at the ends of the 1/2 wavelength antenna and high current in the center. As energy radiates into space, the field is strongest 90° to the antenna where the current flow is strongest.

Most radios, especially communication radios, use the same antenna for transmitting and receiving. Multichannel radios could use a different length antenna for each frequency, however, this is impractical. Acceptable performance can exist from a single antenna half the wavelength of a median frequency. This antenna can be made effectively shorter by placing a properly rated capacitor in series with the transmission line from the transmitter or receiver. This electrically shortens the resonant circuit of which the antenna is a part. An antenna may be electrically lengthened by adding an inductor in the circuit. Adjusting antenna length in this fashion allows the use of a single antenna for multiple frequencies in a narrow frequency range.

Many radios use a tuning circuit to adjust the effective length of the antenna to match the wavelength of the desired frequency. It contains a variable capacitor and an inductor connected in parallel in a circuit. Newer radios use a more efficient tuning circuit. It uses switches to combine frequencies from crystal controlled circuits to create a resonant frequency that matches the desired frequency. Either way, the physical antenna length is a compromise when using a multichannel communication or navigation device that must be electronically tuned for the best performance. A formula can be used to find the ideal length of a half wavelength antenna required for a particular frequency as follows:

Antenna Length (feet) = 468 / F MHz

The formula is derived from the speed of propagation of radio waves, which is approximately 300 million meters per second. It takes into account the dielectric effect of the air at the end of an antenna that effectively shortens the length of the conductor required.

VHF radio frequencies used by aircraft communication radios are 118–136.975 MHz. The corresponding half wavelengths of these frequencies are 3.96-3.44 feet (47.5–41.2 inches). Therefore, VHF antennas are relatively long. Antennas one-quarter of the wavelength of the transmitted frequency are often used. This is possible because when mounted on a metal fuselage, a ground plane is formed and the fuselage acts as the missing one-quarter length of the half wavelength antenna. This is further discussed in the following antenna types section.

Polarization, Directivity, and Field Pattern

Antennas are polarized. They radiate and receive in certain patterns and directions. The electric field cause by the voltage in the conductor is parallel to the polarization of an antenna. It is caused by the voltage difference between each end of the antenna. The electromagnetic field component of the radio wave is at 90˚ to the polarization. It is caused by changing current flow in the antenna. As radio waves radiate out from the antenna they propagate in a specific direction and in a specific pattern. This is the antenna field. The orientation of the electric and electromagnetic fields remains at 90˚ to each other, but radiate from antenna with varying strength in different directions. The strength of the radiated field varies depending on the type of antenna and the angular proximity to it. All antennas, even those that are omni-directional, radiate a stronger signal in some direction compared to other directions. This is known as the antenna field directivity.

Receiving antennas with the same polarization as the transmitting antenna generate the strongest signal. A vertically polarized antenna is mounted up and down. It radiates waves out from it in all directions. To receive the strongest signal from these waves, the receiving antenna should also be positioned vertically so the electromagnetic component of the radio wave can cross it at as close to a 90˚ angle as possible for most of the possible proximities. (***Figure 4-35***)

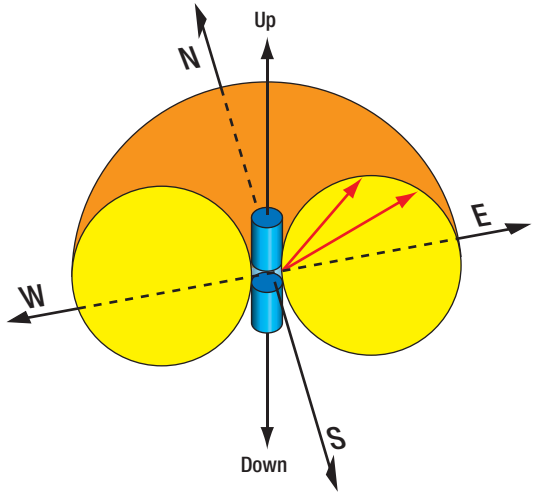


Figure 4-35. A vertically polarized antenna radiates radio waves in a donut-like pattern in all directions.

Horizontally polarized antennas are mounted side to side (horizontally). They radiate in a donut-like field. The strongest signals come from, or are received at, 90° to the length of the antenna. There is no field generated off of the end of the antenna. ***Figure 4-36*** illustrates the field produced by a horizontally polarized antenna.

Many vertical and horizontal antennas on aircraft are mounted at a slight angle off plane. This allows the antenna to receive a weak signal rather than no signal at all when the polarization of the receiving antenna is not identical to the transmitting antenna. (***Figure 4-37***)

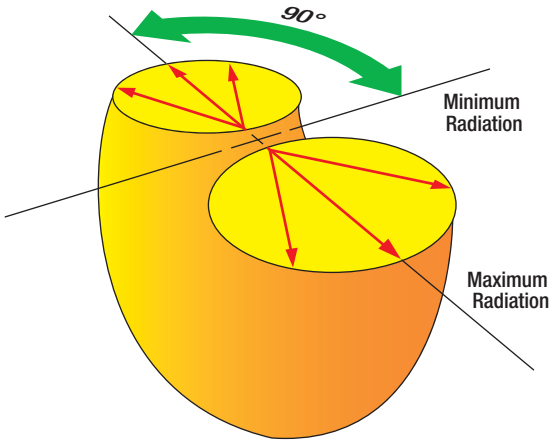


Figure 4-36. A horizontally polarized antenna radiates in a donut-like pattern. The strongest signal is at 90° to the length of the conductor.

Antenna Types

There are three basic types of antennas used in aviation:

- Dipole antenna
- Marconi antenna
- Loop antenna.



Figure 4-37. Many antenna are canted for better reception.

Dipole Antenna

The dipole antenna is the type of antenna referred to in the discussion of how a radio wave is produced. It is a conductor, the length of which is approximately equal to half the wavelength of the transmission frequency. This sometimes is referred to as a Hertz antenna. The AC transmission current is fed to a dipole antenna in the center. As the current alternates, current flow is greatest in the middle of the antenna and gradually less as it approaches the ends. Then, it changes direction and flows the other way. The result is that the largest electromagnetic field is in the middle of the antenna and the strongest radio wave field is perpendicular to the length of the antenna. Most dipole antennas in aviation are horizontally polarized.

A common dipole antenna is the V-shaped VHF navigation antenna, known as a VOR antenna, found on numerous aircraft. Each arm of the V is one-fourth wavelength creating a half wave antenna which is fed in the center. This antenna is horizontally polarized. For a dipole receiving antenna, this means it is most sensitive to signals approaching the antenna from the sides rather than head-on in the direction of flight. (***Figure 4-38***)

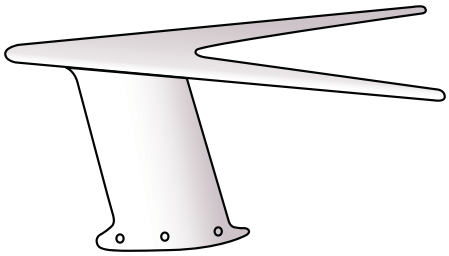


Figure 4-38. The V-shaped VOR navigation antenna is a common dipole antenna.

Marconi Antenna

A Marconi antenna is a one-fourth wave antenna. It achieves the efficiency of a half wave antenna by using the mounting surface of the conductive aircraft skin to create the second one-fourth wavelength. Most aircraft VHF communications antennas are Marconi antennas. They are vertically polarized and create a field that is omni-directional. On fabric skinned aircraft, the ground plane that makes up the second one-fourth wavelength of the antenna must be fashioned under the skin where the Marconi antenna is mounted. This can be done with thin aluminum or aluminum foil. Sometimes four or more wires are extended under the skin from the base of the vertical antenna that serve as the ground plane.

This is enough to give the antenna the proper conductive length. The same practice is also utilized on ground based antennas. (***Figure 4-39***)

Loop Antenna

The third type of antenna commonly found on aircraft is the loop antenna. When the length of an antenna conductor is fashioned into a loop, its field characteristics are altered significantly from that of a straight-half wavelength antenna. It also makes the antenna more compact and less prone to damage.

Used as a receiving antenna, the loop antenna's properties are highly direction sensitive. A radio wave intercepting the loop directly broadside causes equal current flow in both sides of the loop. However, the polarity of the current flows is opposite each other. This causes them to cancel out and produce no signal. When a radio wave strikes the loop antenna in line with the plane of the loop, current is generated first in one side, and then in the other side. This causes the current flows to have different phases and the strongest signal can be generated from this angle. The phase difference (and strength) of the generated current varies proportionally to the angle at which the radio wave strikes the antenna loop. This is useful and is discussed further in the section on automatic direction finder (ADF) navigational aids. (***Figure 4-40***)

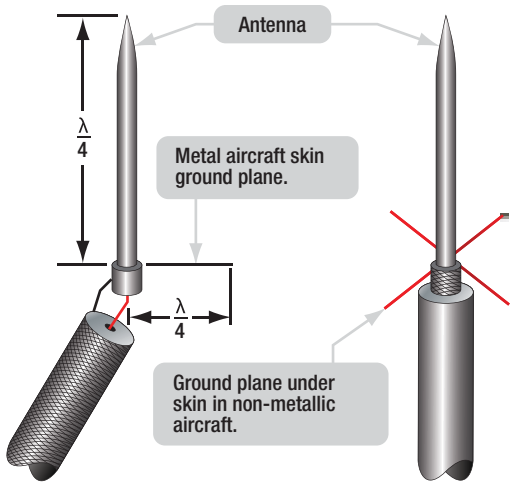


Figure 4-39. On a metal-skinned aircraft, a 1/4 wavelength Marconi antenna is used. The skin is the ground plane that creates the 2nd quarter of the antenna required for resonance (left). On a nonmetallic-skinned aircraft, wires, conductive plates or strips equal in length to the antenna must be installed under the skin to create the ground plane (right).

TRANSMISSION LINES

Transmitters and receivers must be connected to their antenna(s) via conductive wire. These transmission lines are coaxial cable, also known as coax. Coax consists of a center wire conductor surrounded by a semi-rigid insulator. Surrounding the wire and insulator material is a conductive, braided cover that runs the length of the cable. Finally, a waterproof covering is set around the braided shield to protect the entire assembly from the elements. The braided cover in the coax shields the inner conductor from any external fields. It also prevents

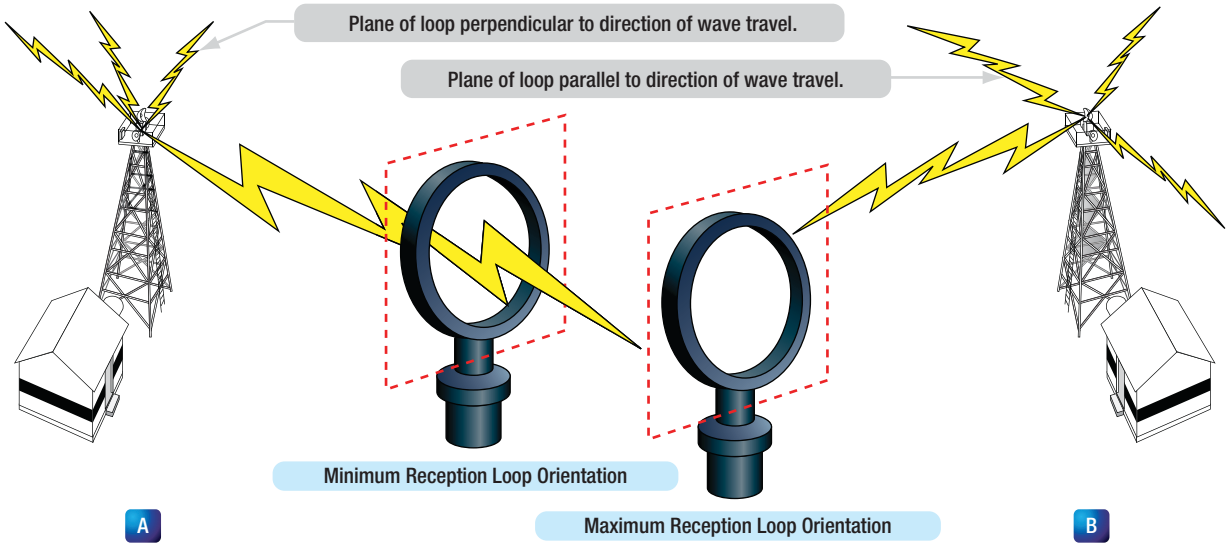


Figure 4-40. A loop antenna is highly direction-sensitive. A signal origin perpendicular or broadside to the loop creates a weak signal (A). A signal origin parallel or in the plain of the loop creates a strong signal (B).

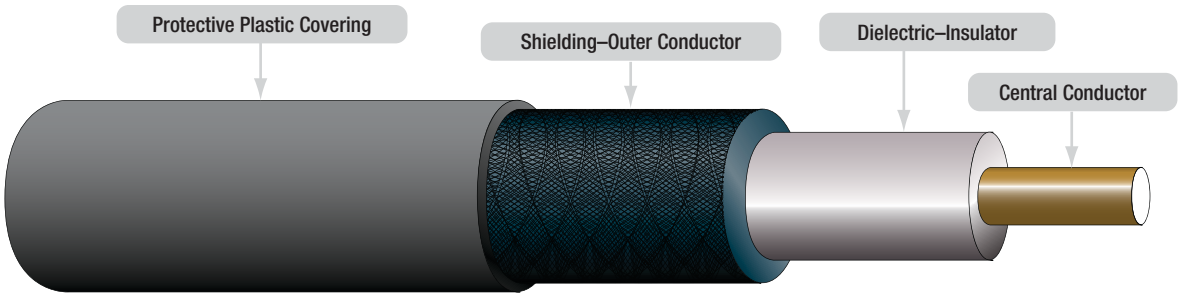


Figure 4-41. Coaxial cable is used as the transmission line between an antenna and its transmitters and/or receiver.

the fields generated by the internal conductor from radiating. For optimum performance, the impedance of the transmission line should be equal to the impedance of the antenna. In aviation antenna applications, this is often approximately 50 ohms. (***Figure 4-41***)

Special connectors are used for coaxial cable. The technician should follow all manufacturer's instructions when installing transmission lines and antenna. Correct installation is critical to radio and antenna performance.

RADIO TRANSMITTERS AND RECEIVERS

Radio transmitters and receivers are electronic devices that manipulate electricity resulting in the transmission of useful information through the atmosphere or space.

Transmitters

A transmitter consists of a precise oscillating circuit or oscillator that creates an AC carrier wave frequency. This is combined with amplification circuits or amplifiers. The distance a carrier wave travels is directly related to the amplification of the signal sent to the antenna. Other circuits are used in a transmitter to accept the input information signal and process it for loading onto the carrier wave. Modulator circuits modify the carrier wave with the processed information signal. Essentially, this is all there is to a radio transmitter. Modern transmitters are highly refined devices with extremely precise frequency oscillation and modulation. The circuitry for controlling, filtering, amplifying, modulating, and oscillating electronic signals is quite complex. A transmitter prepares and sends signals to an antenna that, in the process described above, radiates the waves out into the atmosphere. A transmitter with multiple channel (frequency) capability contains tuning circuitry that enables the user to select the frequency upon which to broadcast. This adjusts the oscillator output to the precise frequency desired. It is the oscillator frequency that is being tuned.

As shown in ***Figure 4-42***, most radio transmitters generate a stable oscillating frequency and then use a frequency multiplier to raise the AC to the transmitting frequency. This allows oscillation to occur at frequencies that are controllable and within the physical working limits of the crystal in crystal-controlled oscillators.

Receivers

Antennas are simply conductors of lengths proportional to the wavelength of the oscillated frequency put out by the transmitter. An antenna captures the desired carrier wave as well as many other radio waves that are present in the atmosphere. A receiver is needed to isolate the desired carrier wave with its information. The receiver also has circuitry to separate the information signal from the carrier wave. It prepares it for output to a device, such as speakers or a display screen. The output is the information signal originally introduced into the transmitter.

A common receiver is the super heterodyne receiver. As with any receiver, it must amplify the desired radio frequency captured by the antenna since it is weak from traveling through the atmosphere. An oscillator in the receiver is used to compare and select the desired frequency out of all of the frequencies picked up by the antenna. The undesired frequencies are sent to ground.

A local oscillator in the receiver produces a frequency that is different than the radio frequency of the carrier wave. These two frequencies are mixed in the mixer. Four

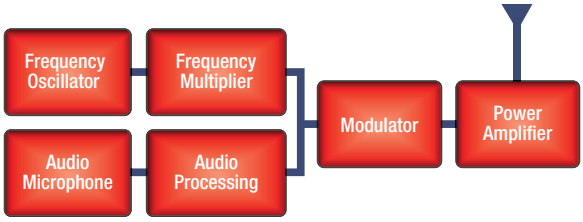


Figure 4-42. Block diagram of a basic radio transmitter.

frequencies result from this mixing. They are the radio frequency, the local oscillator frequency, and the sum and difference of these two frequencies. The sum and difference frequencies contain the information signal.

The frequency that is the difference between the local oscillator frequency and the radio frequency carrier wave frequency is used during the remaining processing. In VHF aircraft communication radios, this frequency is 10.8 MHz. Called the intermediate frequency, it is amplified before it is sent to the detector. The detector, or demodulator, is where the information signal is separated from the carrier wave portion of the signal. In AM, since both sidebands contain the useful information, the signal is rectified leaving just one sideband with a weak version of the original transmitter input signal. In FM receivers, the varying frequency is changed to a varying amplitude signal at this point. Finally, amplification occurs for the output device. (***Figure 4-43***)

Over the years, with the development of transistors, micro-transistors, and integrated circuits, radio transmitters and receivers have become smaller. Electronic bays were established on older aircraft as remote locations to mount radio devices simply because they would not fit in the flight deck. Today, many avionics devices are small enough to be mounted in the instrument panel, which is customary on most light aircraft. Because of the number of communication and navigation aids, as well as the need to present an uncluttered interface to the pilot, most complicated aircraft retain an area away from the flight deck for the mounting of avionics. The control heads of these units remain on the flight deck.

Transceiver

A transceiver is a communication radio that transmits and receives. The same frequency is used for both. When transmitting, the receiver does not function. The push to talk (PTT) switch blocks the receiving circuitry and allows the transmitter circuitry to be active. In a transceiver, some of the circuitry is shared by the transmitting and receiving functions of the device. So is the antenna. This saves space and the number of components used. Transceivers are half duplex systems where communication can occur in both directions but only one party can speak while the other must listen. VHF aircraft communication radios are usually transceivers. (***Figure 4-44***)

VERY HIGH FREQUENCY (VHF) COMMUNICATION

In commercial aircraft, there are generally two to three VHF communication systems installed. The third unit is usually optional. Each system operates in the 118 to 137 Megahertz (MHz) frequency spectrum in 8.33 Kilohertz (KHz) channel spacing. It gives short range airborne to ground (line-of-sight), ground to airborne (data), and airborne to airborne (two-way voice) communications. The VHF transceiver Amplitude Modulates (AM) the transmission.

The VHF transceiver operates in the following modes:

- Power-On Self-Test (POST)
- Operational
- Maintenance.

In a typical configuration (***Figure 4-45***), each VHF transceiver is controlled by the following equipment:

- Standby controller for VHF1 transceiver
- Audio and Radio Control Display Units (ARCDU)
- Flight Management System (FMS)
  Multi-functional Control Display Units (MCDUs).

The VHF Communication System has the following primary components:

- Transceiver
- Unit, standby control
- Antenna

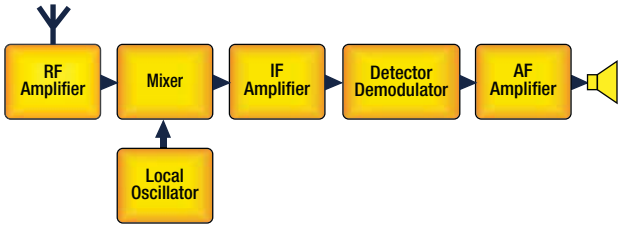


Figure 4-43. The basic stages used in a receiver to produce an output from a radio wave.



Figure 4-44. VHF aircraft communication transceivers.

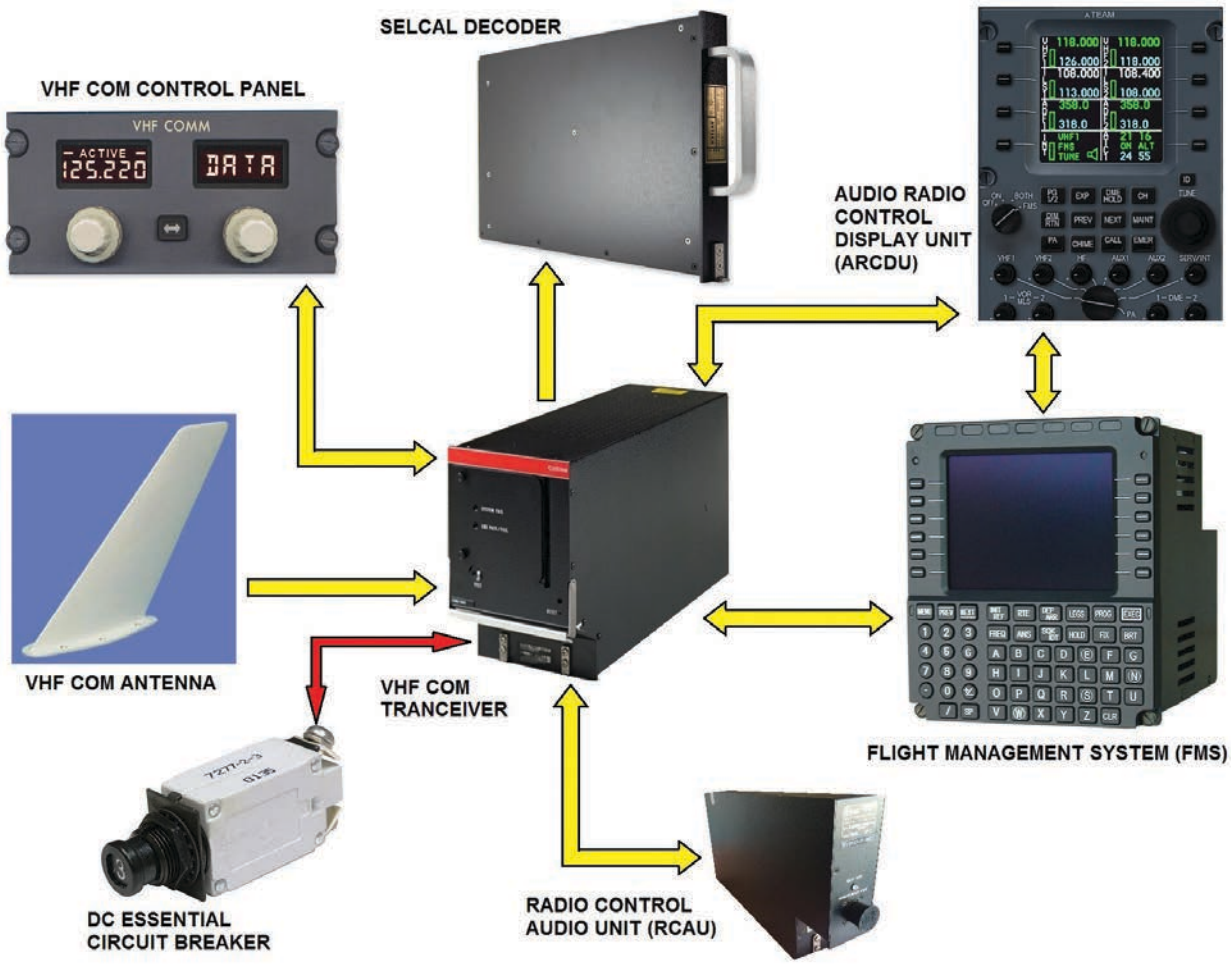


Figure 4-45. A typical VHF Communication Schematic.

The VHF COM2 system operates similar to the VHF COM1 system but cannot be tuned by the VHF COM1 standby controller. On aircraft with the VHF COM3 installed, this system is usually used with ACARS. It operates like both other VHF COM systems but cannot be tuned by the VHF COM1 standby controller or FMS either.

VHF TRANSCEIVER

A typical modern VHF transceiver unit contains sub-assemblies mechanically packaged into a lightweight casing furnished with a single recessed rack-mounted rear panel connector and an RF connector to integrate the system with aircraft wiring harness and VHF antenna. The two VHF transceivers are usually located on one of the shelves of the avionics rack compartment. The mounting tray protects and holds the wired assembly in place and helps dissipate heat.

The VHF transceiver can operate in either in operational mode or a maintenance mode.

Operational Mode

The operational mode has selections for Mode-0A, Mode-0B and Mode-1A. These modes adjust the sensitivity and selectivity of the radio.

Maintenance Mode

The Built In Test Equipment (BITE) uses the central diagnostic system to give the condition of the component. It saves faults in a non volatile memory for reporting to line and shop maintenance. The VHF transceiver has a BITE mode to monitor the condition of the system in the following two ways:

- Power-On Self Test (POST)
- Continuous Monitoring.

Power-On Self Test (POST)

The power-on self test mode automatically starts and continues for 30 seconds after power-up, or after a long power interruption, (generally more than 200 milliseconds). During POST, the VHF transceivers carry out a number of self-tests which include comprehensive

checking of the hardware, ARINC 429 bus inputs, various discrete inputs, voltage standing wave ratio, the transmission/reception feature and further displays the progress of the test itself. If the self-test senses a malfunction, it is sent to the central diagnostic system for later analysis during maintenance functions.

Continuous Monitoring

Continuous monitoring checks the status of the component in flight. It records faults in a non volatile memory for later troubleshooting using the central diagnostic system.

VHF STANDBY CONTROL PANEL

A standby VHF control panel displays and allows tuning of the VHF frequency on either VHF1 or VHF2 system. The standby control panel is generally installed in the central pedestal area, in easy reach of the pilot and the co-pilot.

VHF ANTENNA

The VHF communication system (VHF1, VHF2) has two VHF transceivers with related antennas. A single coaxial cable is routed to each VHF transceiver. The VHF antenna is a vertically-polarized antenna that operates in the VHF communications frequency spectrum of 118 to 156 MHz. The antenna blade has a preformed radiating element in a thermoplastic assembly. A Radio Frequency (RF) N-type female connector is attached to the aluminum base. An aluminum foil gasket with an elastomeric sealant makes electrical contact between the gasket and aircraft structure. The pressure applied during installation gives a contouring feature to let the gasket contact the two mating surfaces. The VHF1/3 antenna are generally attached to the upper fuselage and the VHF2 antenna is attached to the lower fuselage. (***Figure 4-46***)

AUDIO AND RADIO CONTROL DISPLAY UNIT (ARCDU)

The ARCDU front panel has the controls and indications to let the pilots control the audio and radio systems installed. The VHF transceivers are controlled by the two Audio and Radio Control Display Units (ARCDU 1, ARCDU 2). (***Figure 4-47***)

The ARCDU tunes the VHF communication system and controls various operational modes, while displaying all data that is related to the operation of the VHF communication systems. On some aircraft (especially Airbus types), the same component is referred to as Radio Tuning Unit (RTU), Radio Management Panel

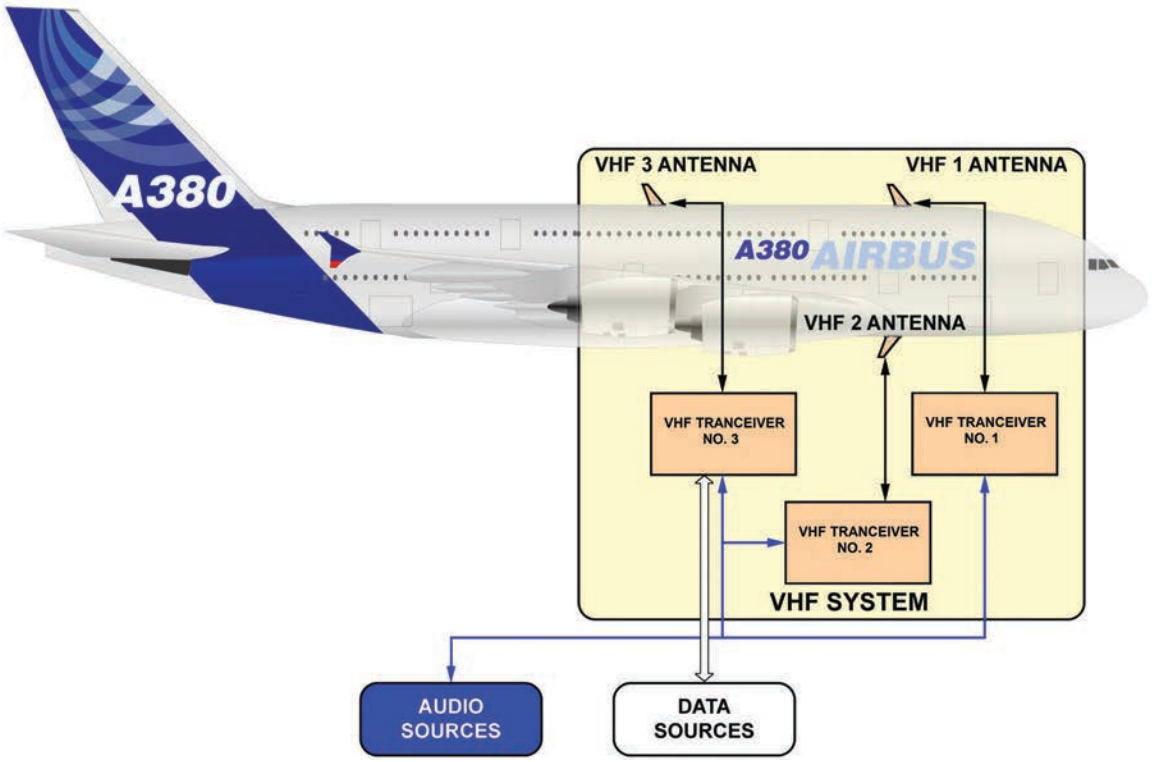


Figure 4-46. A typical VHF Communication Antenna Arrangement.



ARCDU Units on Cockpit Central Pedestal Area

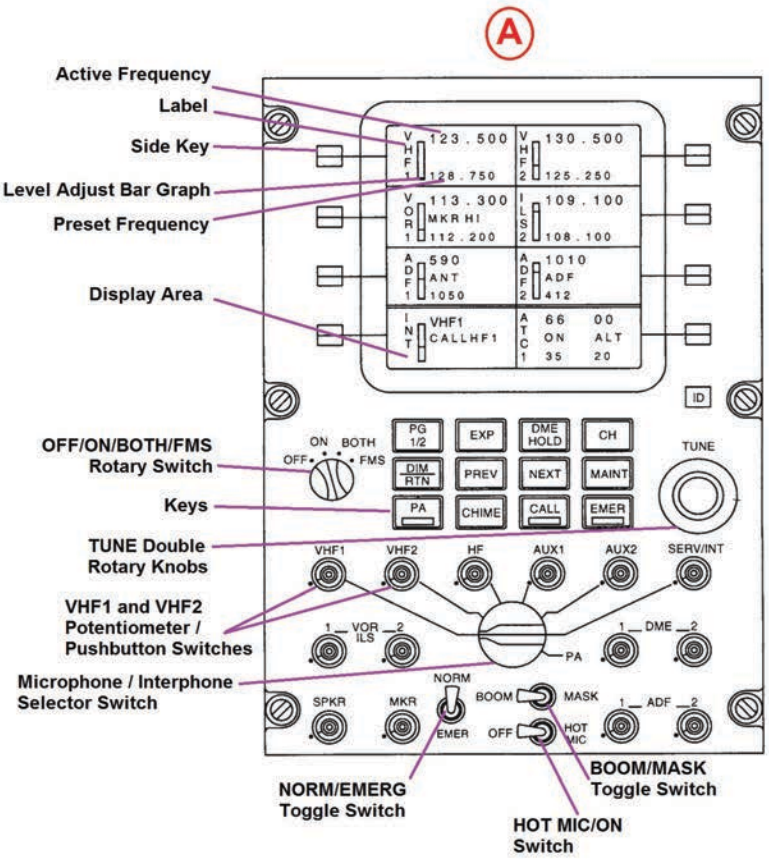


Figure 4-47. Audio & Radio Control Display Unit (ARCDU).

(RMP) or Radio & Audio Management Panel (RAMP). The ARCDU also monitors the operation of the VHF communication system to show any malfunctions, if they arise. It further controls the operation of the Audio Integration System (AIS) on some aircraft. The key or knobs provided on the panel help in tuning the desired VHF Communication frequencies and the mode to be used.

A minimum of two steps are required to change the active frequency. The side key adjacent to the VHF communication display area is pushed to highlight the preset frequency. It is pushed again to make the preset frequency active. The frequency that was active now becomes the new preset frequency. When the side key adjacent to the VHF communication display area is pushed, the preset frequency in the display area highlights. The TUNE double rotary knobs located at the lower right side of the ARCDU are turned to change the preset frequency. The side key is pushed again to set the new active frequency. The frequency that was active now becomes the new preset frequency. To meet the 8.33 KHz channel spacing requirement, an optional transceiver that is capable of tuning in 8.33 and 25 KHz increments is generally installed on modern aircraft. The ARCDU is supplied with 28 VDC through an appropriately rated circuit breaker from the aircraft ESSENTIAL DC Bus.

HIGH FREQUENCY (HF) COMMUNICATION

The HF communications system is provided for long range, air-to-ground and air-to-air communications. The HF system consists of a Receiver/Transmitter (Transceiver), Coupler, and Antenna (***Figure 4-48***). Each HF system is normally tuned and controlled by the respective radio tuning units, in case of multiple installed systems, the Radio Tuning Unit (RTU) No. 1 will control the No. 1 HF system and similarly and Radio Tuning Unit (RTU) No. 2 will control the No. 2 HF system. The HF system provides amplitude modulation (AM) and single sideband (SSB) voice communications at a frequency range of 2 to 30 MHz. The HF system, depending on configuration, may have up to 280 000 communication channels (tuned at 100 Hz steps), and to 16 programmable preset channels.

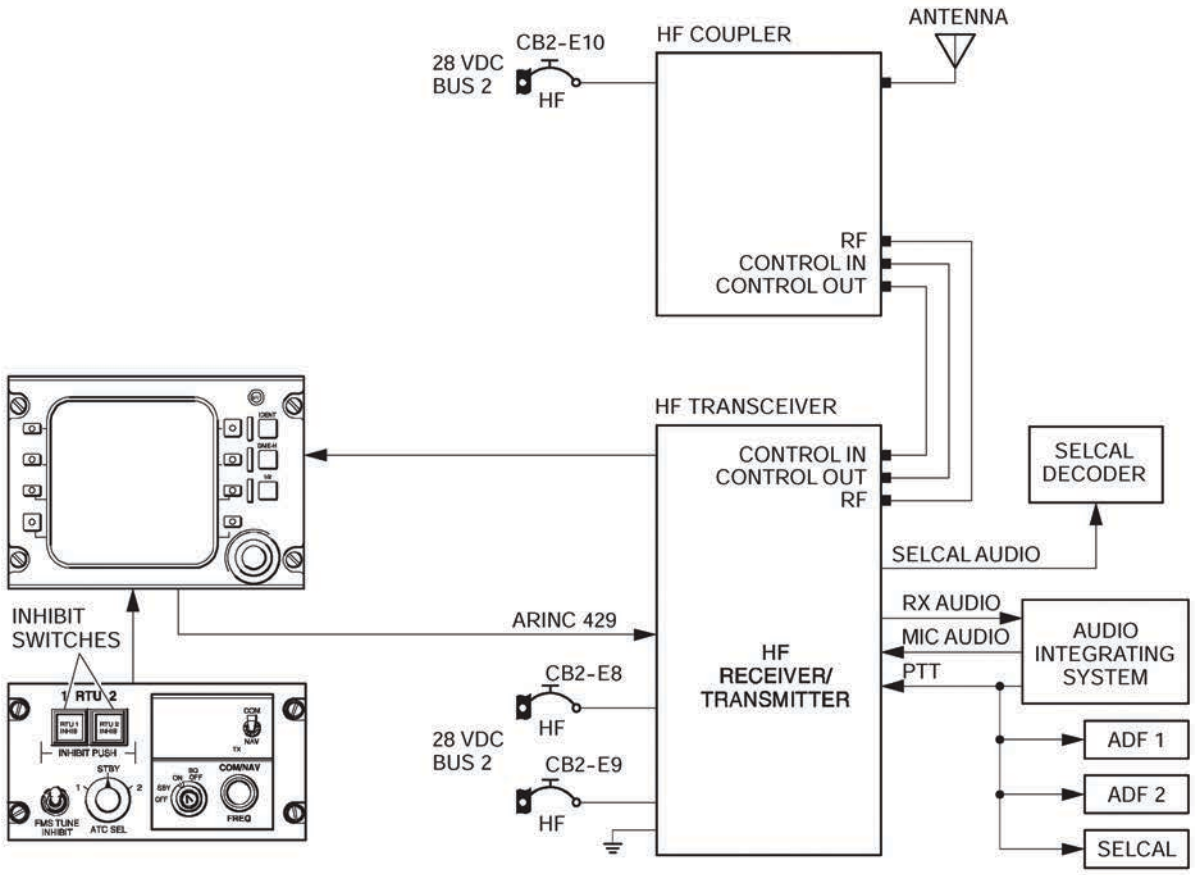


Figure 4-48. A typical HF Communication Schematic.

The HF transceiver is usually installed in the aft equipment compartment and contains the receiving, transmitting and processing circuits required for operation of the system. The unit interfaces with the antenna coupler to control the antenna tuning. It also sends signals to the audio control unit for microphone, headphone, speaker operation, ADF and SELCAL decoder.

The antenna coupler is installed in the aft equipment compartment and tunes the antenna to the selected frequencies. To reduce tuning time, the tuning data for a frequency that was tuned previously is stored in the unit memory. The antenna is an integral part of the leading edge of the vertical stabilizer on most modern aircraft (***Figure 4-49***), however, different locations and HF antenna configurations on various aircraft have been utilized for HF antenna placement by aircraft manufacturers. (***Figure 4-50***) The HF antenna is connected to the antenna coupler.

The HF system operates in simplex or half-duplex modes using the following three types of modulation:

- LV (Lower Voice Single Sideband)
- UV (Upper Voice Single Sideband)
- AM (Amplitude Modulation)

The HF frequency and modes of operation are controlled and monitored from the Top Level page of RTU No. 2, which displays the frequency, and mode of operation (***Figure 4-51***). The HF line key on the lower right side and the rotary frequency tuning knobs are used to select the frequency and control the mode of operation. Before transmitting, the antenna must be tuned to the selected frequency through the antenna coupler. The coupler tunes to the new frequency when the HF push-to-talk (PTT) switch is pushed momentarily. When the coupler is tuning, a steady 1000-Hz tone is heard in the headphone. The tone stops when the coupler is tuned. If a fault occurs, a beeping tone is heard.

There are two main HF pages HF COM and HF PRESET for system operations. These pages are accessed in sequence from the top-level page when the

HF line key is pushed. The main pages display the HF active frequencies, modes of operation, and the preset frequency channels.

The HF system is also coupled to the SELCAL decoder to monitor the selected HF frequencies for the aircraft unique SELCAL code.

AUDIO SYSTEMS

AIRCRAFT VOICE COMMUNICATIONS

Audio signals from the users are routed to the different systems capable of voice communications (***Figure 4-52***). To communicate outside the aircraft, the operators can use radio communications such as HF and VHF systems, or the satellite communication (SATCOM) system.

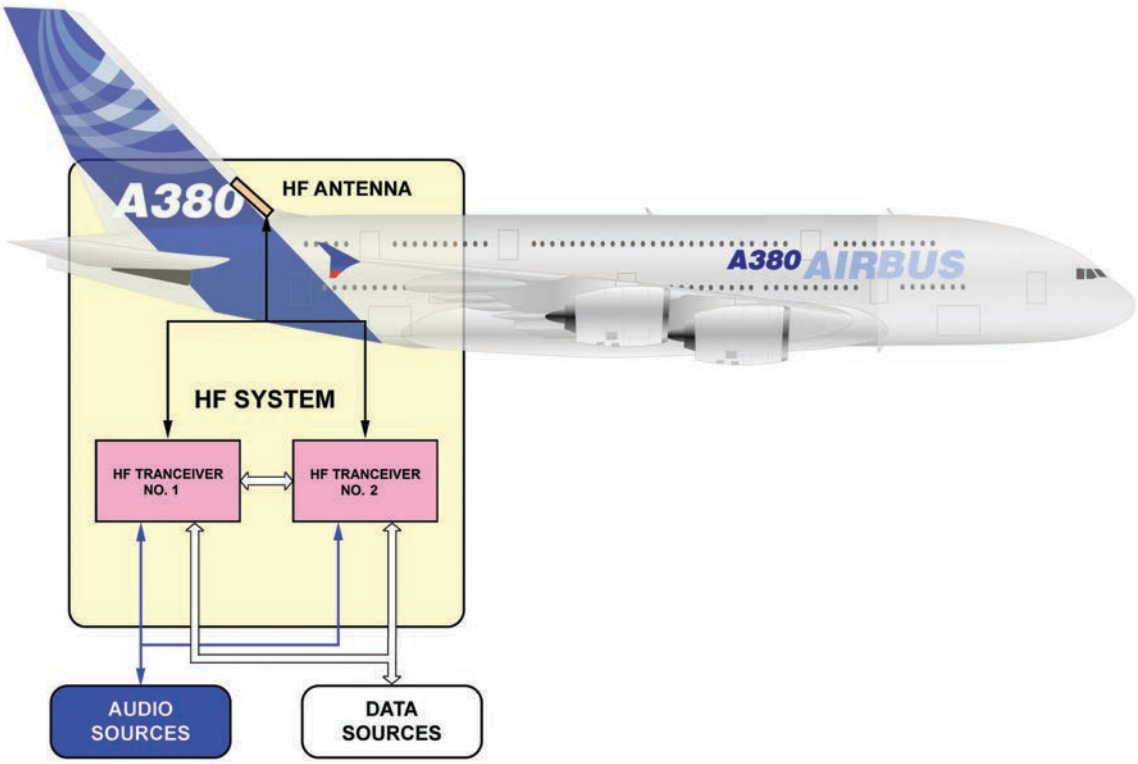


Figure 4-49. HF Communication Antenna System on Airbus A380.

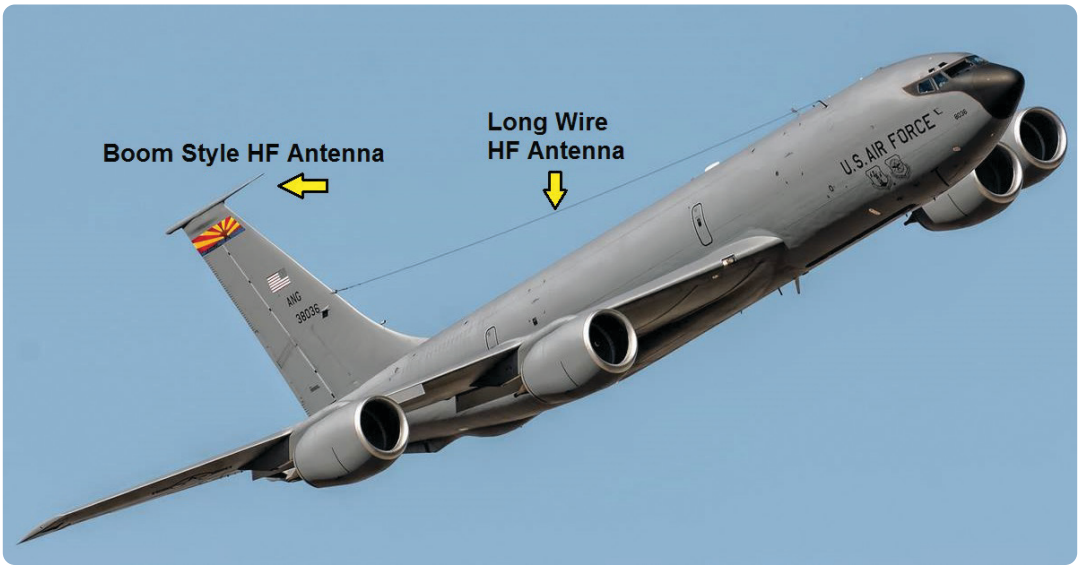


Figure 4-50. A Boom & Long Wire HF Antenna Arrangement.

The flight, service and cabin interphone systems are used for the on-board communications. The Radio and Audio Integrating Management System (RAIMS) controls voice communication in the cockpit environment. On some aircraft, the same function is performed by Audio Radio Control Display Unit (ARCDU) or a similar component which renders control for radio/audio voice communication systems.

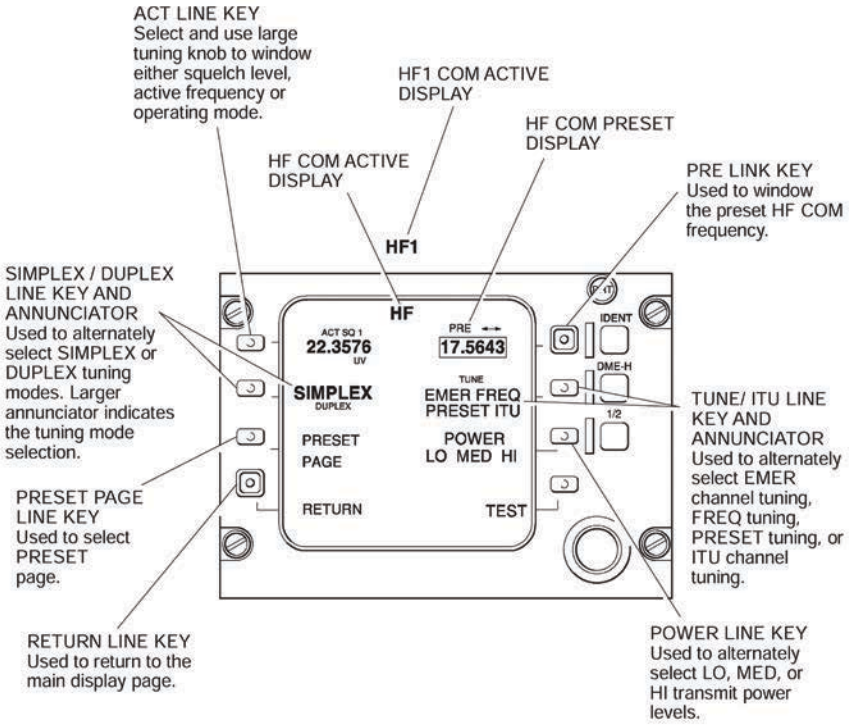


Figure 4-51. HF Communication Radio Tuning Unit.

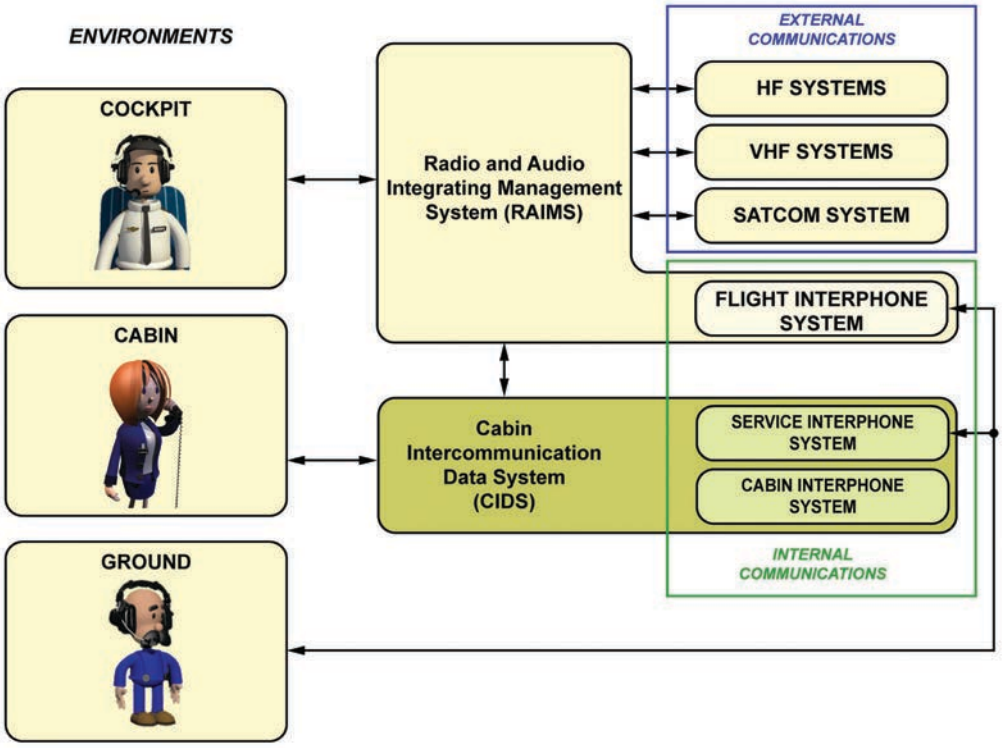


Figure 4-52. A typical aircraft voice communications layout.

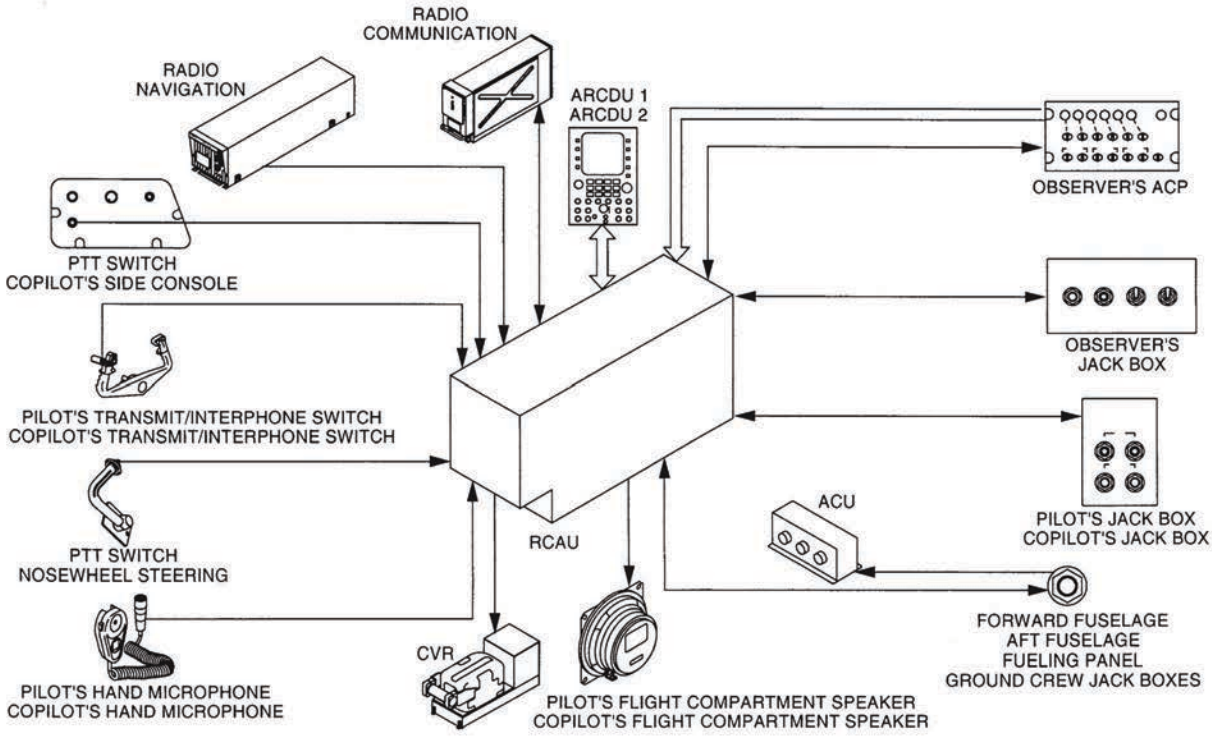


Figure 4-53. Audio Integration System Block Diagram.

The Cabin Intercommunication Data System (CIDS) is dedicated to the cabin environment and can ensure connection with ground personnel around the aircraft.

AUDIO INTEGRATION SYSTEM (AIS)

The Audio Integration System (AIS) controls all audio functions related to radio communications and navigation, inter-phone communications between pilot, copilot, observer, and ground crew members. A typical Audio Integration System (AIS) would have the following system components: (***Figure 4-53***)

- Remote Control Audio Unit (RCAU)
- Observer's Audio Control Panel (ACP)
- Pilot and copilot flight compartment speakers
- Pilot and copilot hand microphones
- Pilot and copilot jack boxes
- Observer's jack box
- Ground crew jack boxes
- Copilot's side console INPH, XMIT toggle switch
- Pilot and copilot hand-wheel PTT. INPH toggle
- Nose wheel steering PTT pushbutton switch.

The Audio Integration System (AIS) uses the audio and radio control display unit and observer's Audio Control Panel (ACP) to control the AIS. The Remote Control Audio Unit (RCAU) interfaces with communication and navigation transceivers, pilot, copilot, observer's, and ground crew jack boxes, passenger address amplifier and the flight compartment speakers. The audio from the communication transceivers and navigation receivers are supplied to the RCAU. The ARCDUs and observer's audio control panel control the Remote Control Audio Unit to select the audio and change its level to the flight compartment speakers and pilot, copilot, and observer jack boxes. To make a communication transceiver transmit, a Push To Talk (PTT) selection and microphone audio is supplied through the RCAU to the communication transceivers.

The service or communications inter-phone transmission starts with an inter-phone PTT selection. The MIC jack audio from the pilot, copilot, observer, or ground crew jack box is then supplied to the RCAU. To listen to other inter-phone transmissions, the ARCDUs and observer's audio control panel control the RCAU to select the inter-phone audio and change its level to the flight compartment speakers and the phone jacks in the jack boxes. The RCAU supplies audio data to the Cockpit Voice Recorder (CVR).

REMOTE CONTROL AUDIO UNIT (RCAU)

The Remote Control Audio Unit is used to control the Audio Integration System (AIS) from ARCDU or the observer's Audio Control Panel selections. (***Figure 4-54***)



Figure 4-54. Remote Control Audio Unit (RCAU)/ Audio Management Unit (AMU).

A typical RCAU assembly contains various sub-assemblies which include a filter, a motherboard, an input processing card, pilot/copilot audio cards and pilot/copilot speaker amplifier cards.

The installed filter provides protection against High Intensity Radio Frequency (HIRF) lightning, and other electromagnetic effects. The motherboard furnishes electrical connections between other boards and the rear panel connector. It has circuitry to protect the other cards from transient voltages as well. An input processing card amplifies the input signals and further isolates the inputs and outputs. It also supplies electrical power to the boom/mask relays and bias voltages to the attendant handset and ground crew microphones.

The Audio Cards: The pilot, copilot, and observer's audio card have input signals that pass through a low-pass filter and are digitized by a Central Processing Unit (CPU). The CPU receives selection and volume level inputs from the ARCDU and the ACP. It uses digital processing to give applicable analog outputs to headsets and flight compartment speakers. The ARCDU and ACP potentiometers have a voltage at their wipers that changes with the potentiometer position. The analogue voltages are changed into digital words by the multiplexer in the analogue to digital converter. The pilot and copilot speaker amplifier cards are used to amplify low level inter-phone, radio and aural alerts signals for the flight compartment speakers.

AUDIO CONTROL PANEL (ACP)

There may be multiple ACP's in various locations inside the aircraft to provide ease of communication to crew members. Each panel controls an independent audio system and allows the flight crew to select the desired radios, navigation aids, interphones and PA systems for monitoring and transmission. The audio panel receives inputs from all audio communication channels and aural warnings. Audio warning for altitude alert, ground proximity warning system, traffic collision avoidance system, and wind shear among others, are also heard through the speakers and headsets. These warnings cannot be controlled or turned off by the flight crew. Audio from each ACP is monitored using a headset, headphone or the related speaker, except for the observer speaker.

From a functionality stand point, the observer's ACP changes the front panel audio and PTT selections into ARINC 429 data for transmission to the RCAU. The ACP normally has two circuit card assemblies attached together and to the chassis. The top card assembly has the mechanical section of the transmitter selector keyboard. This circuit card assembly and the potentiometers are electrically connected to the lower circuit card. The lower card assembly holds the rear connector (***Figure 4-55***).

INT/RAD SWITCH

This switch is located on the ACP panel and operates as a press-to-talk switch for boom or oxygen microphone. The switch when selected to INT position, the boom and mask microphones, transmit on inter-phone, regardless of which transmission key is selected. However, for reception on inter-phone, the crew member must have the INT reception knob out, on his/her ACP panel. The other position is the RAD position (press and hold), which allows voice transmission on boom and mask microphones on the radio selected on the audio control panel.

FLIGHT COMPARTMENT SPEAKER

Generally, individual speakers installed above the pilot and copilot, are used to monitor audio selected at the audio control panels (***Figure 4-56***). The speaker converts electrical signal to audio. The audio sources selected on the audio control panel (ACP) can be routed to the flight compartment speakers by pressing in the speaker switch. Speaker volume is controlled by rotating the speaker control knob.

HAND MICROPHONE

The hand microphone is placed in such a manner that it is accessible to the flight crew members for making desired announcements. In some aircraft, it is connected to the base of the control column with a cable and in others it

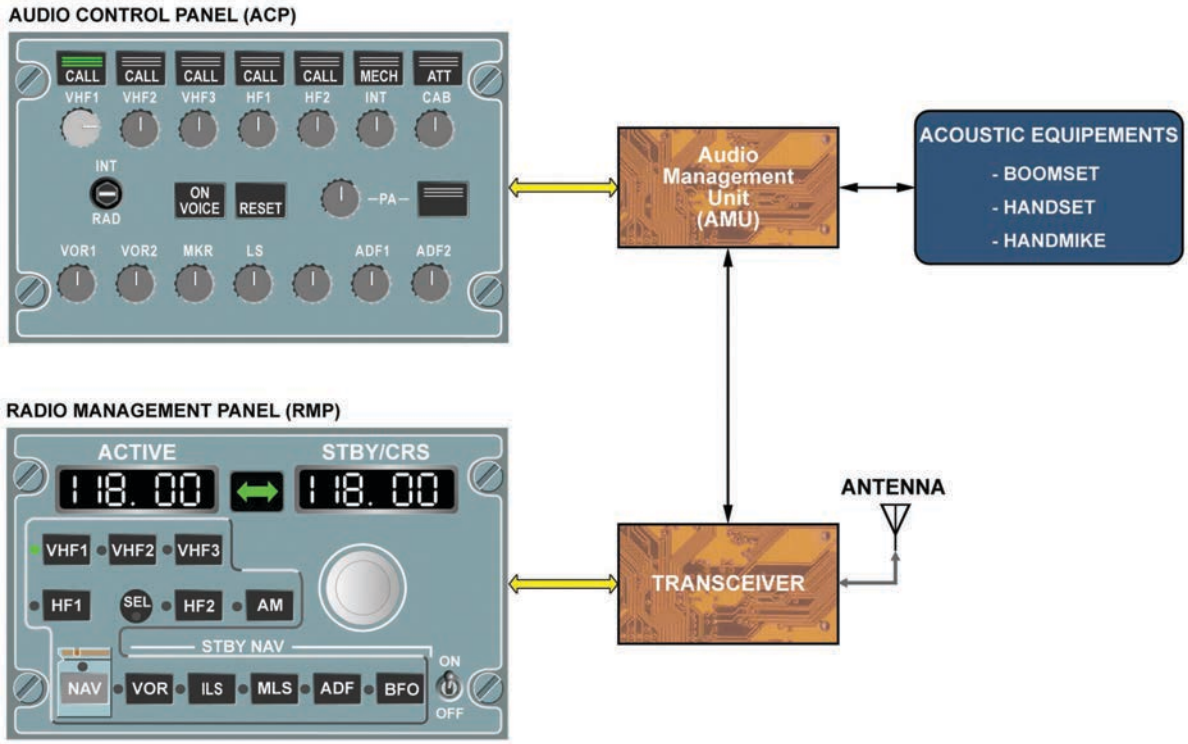


Figure 4-55. The Audio Control Panel (ACP) and its interface with aircraft communication systems.



Figure 4-56. The Flight Compartment Speaker.



Figure 4-57. The Flight Compartment Hand Microphone.

is appropriately placed on the central or side pedestal panels for easy access of the aircrew. (***Figure 4-57***) It has a mounting bracket and PTT switch.

The hand microphone is a device that changes sound waves into variations of electric currents for transmission. It has a condenser to remove much of the electromagnetic and radio frequency interference and magnetic signals.

JACK BOX CONSOLE (FLIGHT COMPARTMENT, OBSERVER, GROUND)

The jack box consoles are used to connect the aircrew headsets and oxygen mask microphone to the RCAU or AMU, to enable voice communication with other aircrew members or the ground crew. The jacks are connected to headset and microphone plugs. A notch in the plugs aligns with a "V" bend in the jack tips to hold them firmly together. The jacks have solder lug terminals for wiring connections and are attached to the front surface of the jack panel with mounting nuts and lock washers. Each jack box contains jacks for Boom Set or Oxygen Mask (BOOM/MASK) microphones and NORMAL/AUX phones. (***Figure 4-58***)

PUSH-TO-TALK (PTT) SWITCH

The Push-to-talk (PTT), also known as Press-to-Transmit, is a method of having conversations or talking

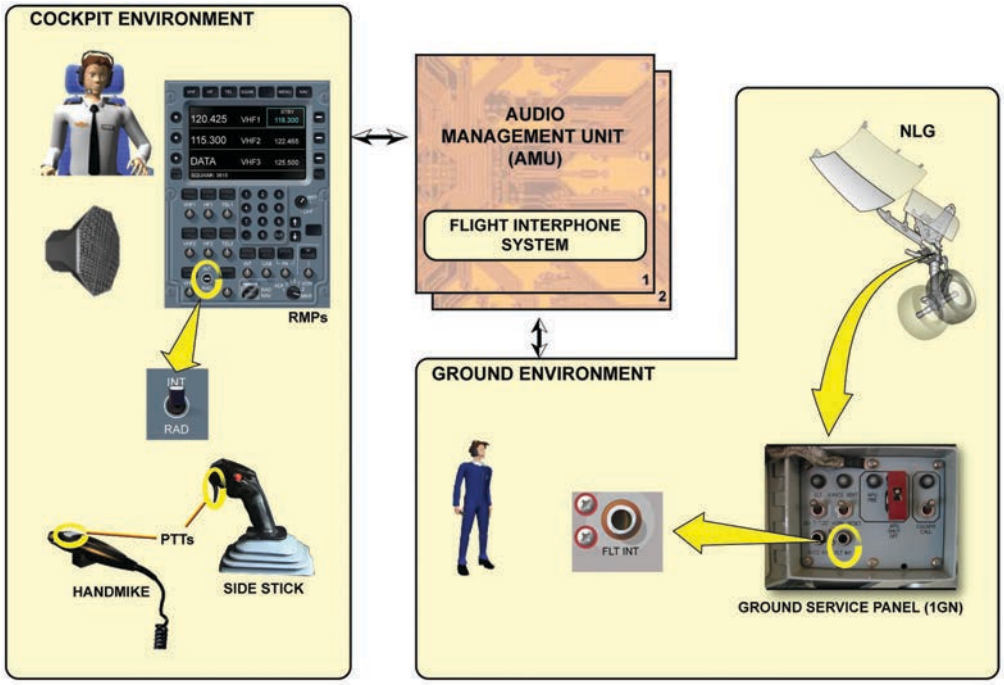


Figure 4-58. Jack box consoles for communication between aircrew and ground crew.

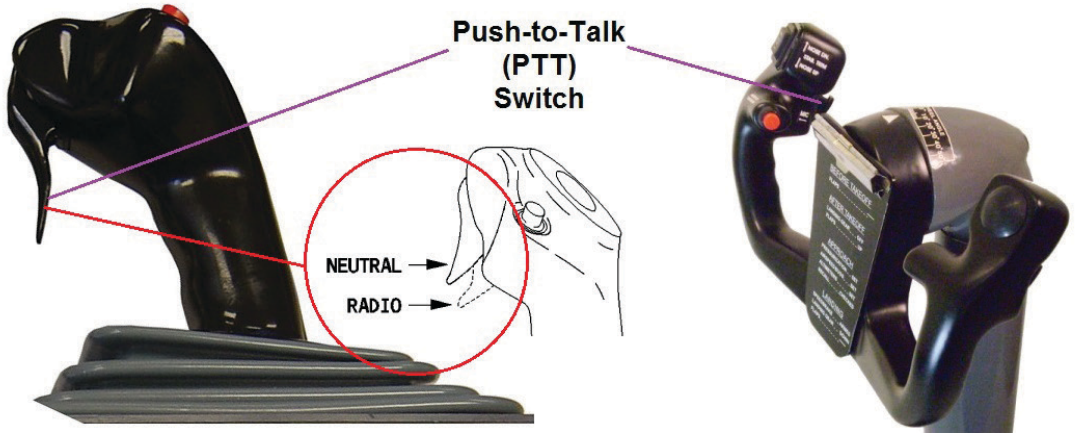


Figure 4-59. Push-to-Talk (PTT) switch on control column.

on half-duplex communication lines, including two-way radio, using a momentary button to switch from voice reception mode to transmit mode (***Figure 4-59***). The PTT switch on the yoke column or the side-stick control column operates in a similar manner as the INT/RAD switch on the ACP. In NEUTRAL (spring-loaded) condition, the boom and mask mikes do not transmit.

However, the PTT switch is pressed, the boom and mask microphones transmit on the desired radio channel as selected on the ACP. It must be noted, even if the INT/RAD switch is on INT position on the ACP panel, pressing the control column PTT would cause radio transmission to supersede intercom transmission, as radio function has priority over the inter-phone function.

CABIN ATTENDANT HANDSET

Each attendant station has a handset for public announcement, inter-phone dialing and communication. The handset rests in a cradle. The handset and cradle fulfill the following functions:

- The hook-off sensing.
- The Push-To-Talk (PTT) switching.
- The Passenger Address (PA) announcement.
- The inter-phone.

After removing the keyboard cover, two potentiometers under the PTT button can be adjusted for setting phone and microphone sensitivity (***Figure 4-60***).

CABIN INTERCOMMUNICATION DATA SYSTEM (CIDS)

The CIDS has several communication functions, which are: (***Figure 4-61***)

- Passenger Address
- Cabin inter-phone
- Service inter-phone
- Crew signaling and alerting

Passenger Address (PA)

The Passenger Address (PA) is one of the main functions of the CIDS. It supplies one-way voice communication to do announcements from the cockpit or from a cabin crew station to the passengers. These announcements are initiated from the cockpit using either a handset or acoustic devices and from the cabin using cabin crew stations handsets. They are then broadcasted to the passengers either through passenger's headsets via In-Flight entertainment (IFE) and through all cabin loudspeakers. A PA announcement can be direct or prerecorded voice information and it can also be used to supply prerecorded music.

Cabin Inter-phone

The cabin inter-phone system is used for communication between all cabin crew stations or between the cockpit and the cabin crew stations. From the cockpit the communication is established via the cockpit handset or via acoustic device. From the cabin the communication is established via any cabin crew station handset.

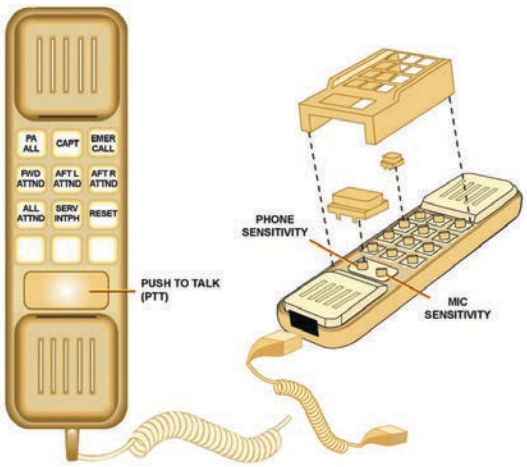


Figure 4-60. Cabin attendant handset.

Service Inter-Phone

The service inter-phone system is used, on ground only, for communications between the service inter-phone jacks or between the service inter-phone jacks, the cockpit and the cabin crew stations. The service inter-phone jacks are located within the major service areas.

Crew Signaling And Alerting

There are different kinds of cockpit and cabin crew signaling and alerting functions depending on the situation. The "area ready" function lets the cabin crew inform the purser that a dedicated cabin area is ready for takeoff/landing. This function is activated through the Flight Attendant Panels (FAPs) and the optional MINI-FAPs. The "cabin ready" signaling function lets the purser inform the cockpit crew that all the cabin areas are ready for takeoff/landing. The optional "sterile cockpit" lets the cockpit crew inform the purser that they do not want to be disturbed. This function is activated by the cockpit crew through a sterile cockpit P/B. The optional Emergency Crew Alerting System (ECAS) is used to indicate an unusual behavior in cabin or cockpit. This function is activated from the cabin thanks to a cabin alert P/B and from the cockpit via a cockpit alert P/B.

EMERGENCY LOCATER TRANSMITTER (ELT)

An Emergency Locater Transmitter is an independent battery powered transmitter activated by the excessive G-forces experienced during an aircraft crash. It transmits a digital signal every 50 seconds on a frequency of 406.025 MHz at 5 watts for at least 24 hours. The signal is received anywhere in the world by satellites in the COSPAS-SARSAT (a Russian acronym translated to Space System for the Search of Vessels in Distress - Search and Rescue Satellite-Aided Tracking) satellite system.

Two types of satellites, Low-Earth Orbiting (LEOSATs) and Geostationary Satellites (GEOSATs) are used with different, complimentary capability. The signal is partially processed and stored in the satellites and then relayed to ground stations known as Local User Terminals (LUTs). Further deciphering of a signal takes place at the LUTs, and appropriate search and rescue operations are notified through Mission Control Centers (MCCs) set up for this purpose.

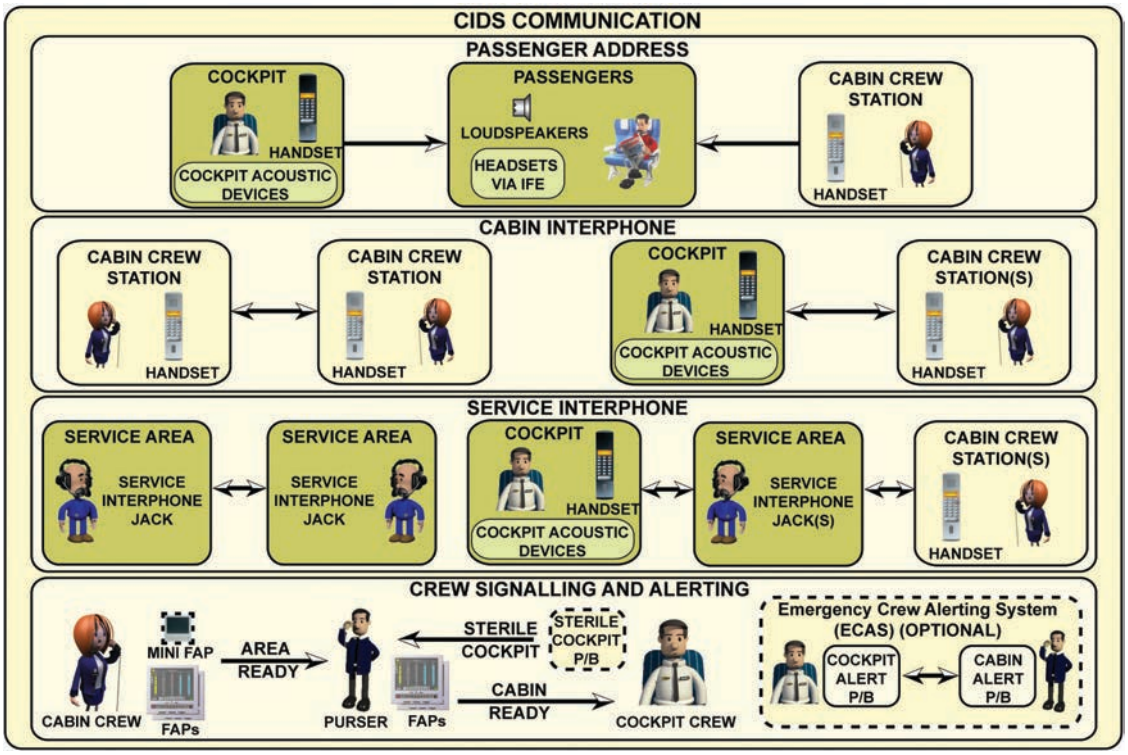


Figure 4-61. CIDS Communication functions.

Maritime vessel Emergency Position-Indicating Radio Beacons (EPIRBs) and Personal Locator Beacons (PLBs) use the exact same system. EPIRB beacons with built-in GPS are called GPIRBs, for GPS Position-Indicating Radio Beacon. The U.S. portion of the COSPAS-SARSAT system is maintained and operated by the National Oceanic and Atmospheric Administration (NOAA). ***Figure 4-62*** illustrates the basic components in the COSPAS-SARSAT system.

ELTs must be inspected within 12 months of previous inspection for proper installation, battery corrosion, operation of the controls and crash sensor, and the presence of a sufficient signal at the antenna. Bui lt-in test equipment facilitates testing without transmission of an emergency signal. The remainder of the inspection is visual. Technicians are cautioned not to activate the ELT and transmit an emergency distress signal. Inspection must be recorded in maintenance records including the new expiration date of the battery. This must also be recorded on the outside of the ELT.

ELTs are typically installed as far aft in the fuselage of an aircraft as is practicable just for ward of the empennage. The built-in G-force sensor is aligned with the longitudinal axis of the aircraft. Helicopter ELTs may be located elsewhere on the airframe as they are equipped with multidirectional activation devices. Follow ELT and airframe manufacturer's instructions for proper installation, inspection, and maintenance of all ELTs. ***Figure 4-63*** illustrates ELT mounted locations.

Use of Doppler technology enables the origin of the 406 MHz ELT signal to be calculated within 2 to 5 kilometers. Second generation 406 MHz ELT digital signals are loaded with GPS location coordinates from a receiver inside the ELT unit or integrated from an outside unit. This reduces the location accuracy of the crash site to within 100 meters. The digital signal is also loaded with unique registration information. It identifies the aircraft, the owner, and contact information, etc. When a signal is received, this is used to immediately research the validity of the alert to ensure it is a true emergency transmission so that rescue resources are not deployed needlessly.

ELTs with automatic G-force activation mounted in aircraft are easily removable. They often contain a portable antenna so that crash victims may leave the site and carry the operating ELT with them. A light deck mounted panel is required to alert the pilot if the ELT is activated. It also allows the ELT to be armed, tested, and manually activated if needed. (***Figure 4-64***)

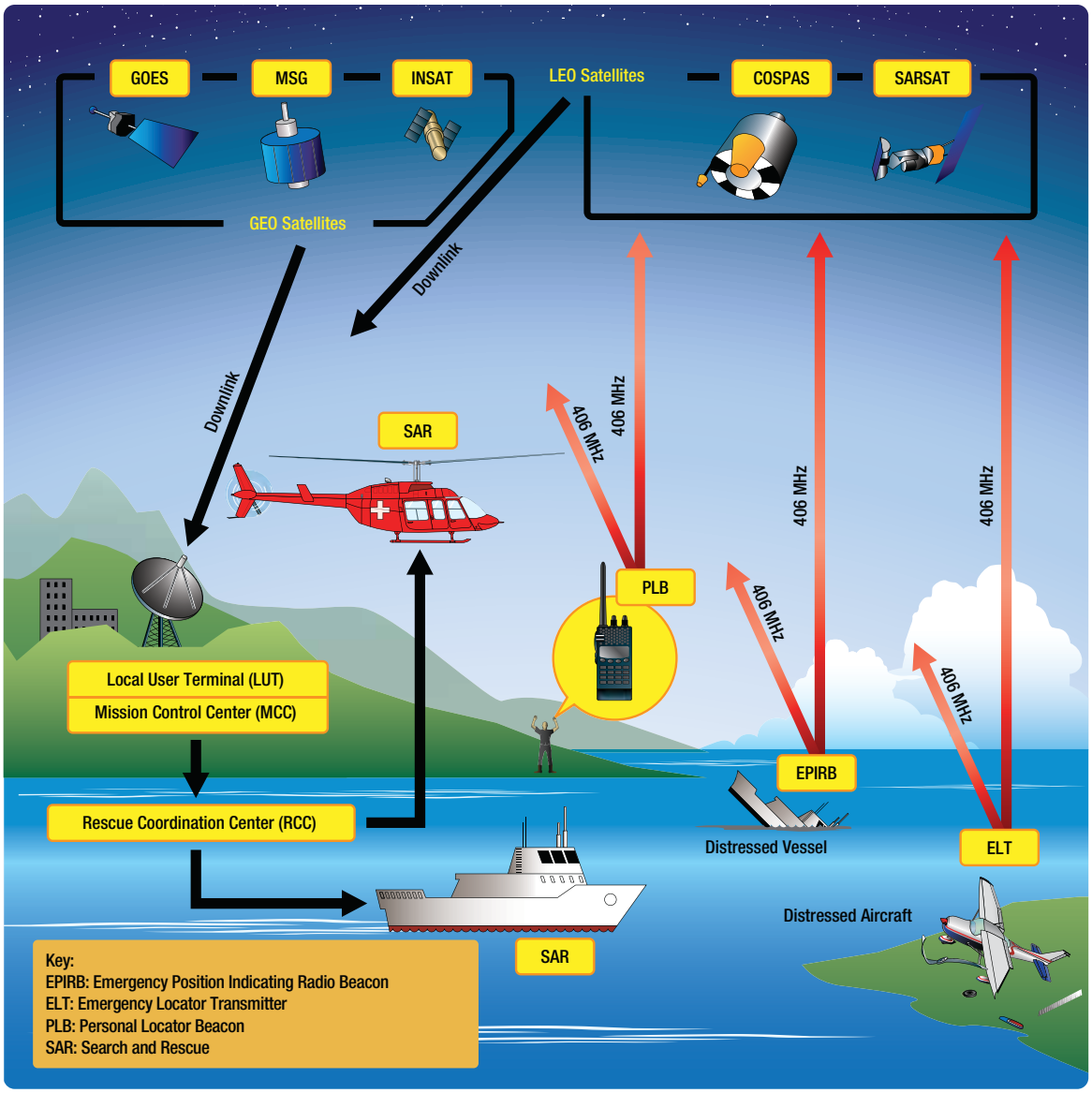


Figure 4-62. The basic operating components of the satellite-based COSPAS-SARSAT rescue system of which aircraft ELTs are a part.



Figure 4-63. An ELT mounting location is generally far aft in a fixed-wing aircraft fuselage in line with the longitudinal axis.



Figure 4-64. An ELT and its components including a cockpit mounted panel, the ELT, a permanent mount antenna, and a portable antenna.

Modern ELTs may also transmit a signal on 121.5 MHz. This is an analog signal transmission that can be used for homing. Prior to 2009, 121.5 MHz was a worldwide emergency frequency monitored by the CORPASSARSAT satellites. However, it has been replaced by the 406 MHz standard. Transmission on 121.5 MHz are no longer received and relayed via satellite. The use of a 406 MHz ELT has not been mandated by the FAA. An older 121.5 MHz ELT satisfies the requirements in all except new aircraft. Thousands of aircraft remain equipped with ELTs that transmit a .75 watt analog 121.5 MHz emergency signal when activated. The 121.5 MHz frequency is still an active emergency frequency and is monitored by over-lying aircraft and control towers.

Technicians are required to perform an inspection and test of 121.5 MHz ELTs within 12 months of the previous one and inspect for the same integrity as required for the 406MHz ELTs mentioned above. However, older ELTs of ten lack the built-in-test circuitry of modern ELTs certified to TSO C-126. Therefore, a true operational test may include activating the signal. This can be done by removing the antenna and installing a dummy load. Any activation of an ELT signal is required to only be done between the top of each hour and 5 minutes after the hour. The duration of activation must be no longer than three audible sweeps. Contact of the local control tower or flight service station before testing is recommended. It must be noted that older 121.5 MHz analog signal ELTs often also transmit an emergency signal on a frequency of 243.0 MHz. This has long been the military emergency frequency. Its use is being phased out in favor of digital ELT signal s and satellite monitoring. Improvements in coverage, location accuracy, identification of false alerts, and shortened response times are so significant with 406 MHz ELTs, they are currently the industry standard worldwide.

COCKPIT VOICE RECORDERS

The cockpit voice recorder (CVR) and Flight Data Recorder (FDR), commonly referred to as the "Black Box", is a device that records conversation of pilots and logs flight data. A black box is in fact two pieces of separate equipment: the Cockpit Voice Recorder (CVR) and the Flight Data Recorder (FDR). Both pieces are mandatory for any commercial aircraft or corporate jet.

The cockpit contains several microphones to track the conversations of the flight crew as well as the ambient noises, like possible malfunctions in the aircraft. All the audio is sent to the CVR which has a maximum storage for two hours of data. After reaching that limit it starts to record again over the previous material.

The Flight Data Recording system collects mandatory parameters required by the airworthiness authorities and records this data on the Solid State Flight Data Recorder (SSFDR). A time signal is transmitted to the CVR for synchronization purposes. The sources of these parameters come from several aircraft systems which are connected to the Flight Data Recording system. Recorded data are for example engine data (e.g. EGT, EPR, N1, N2 and fuel), air data (e.g. temperature, altitude and speed), flight control data, navigation data, hydraulic data, etc. Around 1 000 different parameters like airspeed, altitude, engine power, fuel flow, time and direction of the plane are constantly recorded in the FDR, which can store 25 hours of data. To record the data during the whole flight profile, the recorder is switched on automatically as soon as one engine is running and it stops five minutes after the last engine is shut down.

The CVR/FDR are always located in the tail of the plane to improve the chances of surviving the crash, mainly because it is the last part of the aircraft to impact. ***Figure 4-65*** shows the typical placement of a CVR in a plane and general communication/data flow to the device.

The modern CVR is a solid state assembly housed in an ARINC 404A, 1/2-ATR short case. The CVR, the chassis, and Crash Survivable Memory Unit (CSMU) are painted international ORANGE (***Figure 4-66***) though still referred to as the "Black" box in public. The prime reason for painting the device in "orange" is that the device must be easily traceable in the aftermath of a plane crash. Aside the chassis color, these devices are also pasted with strips of reflective tape which are very helpful in locating them when a plane lands over water. There are several possible origins for the term "black box". Some people believe it is simply because the early devices were painted black, while others postulate that it refers to the charring on the device that occurs in post-accident fires.

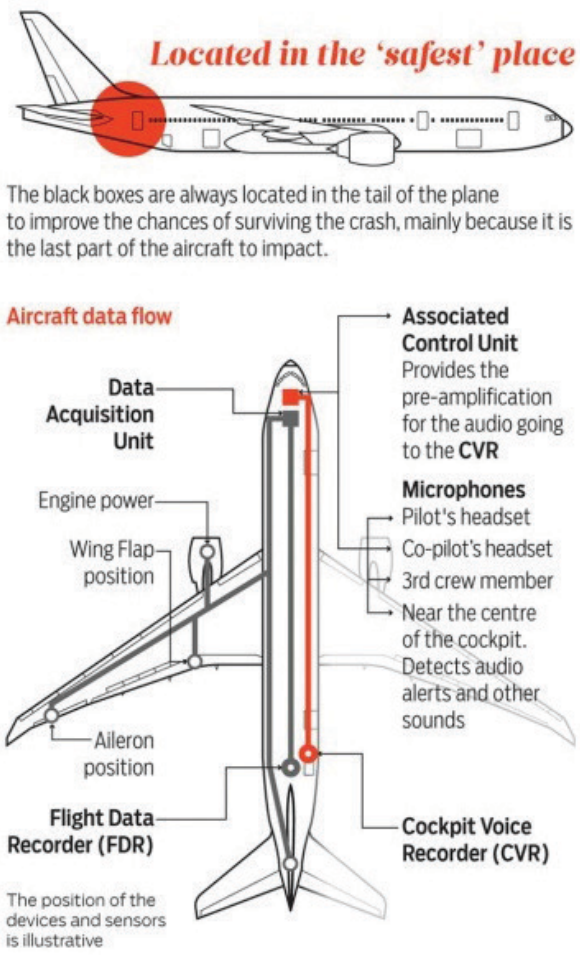


Figure 4-65. A typical placement of CVR/FDR in an aircraft.

The CVR consists of a chassis and front panel, three Printed Wiring Assemblies (Aircraft Interface PWA, Audio Compressor PWA, and Acquisition Processor PWA), and the Crash Survivable Memory Unit (CSMU). The CSMU contains the solid state flash memory used as the recording medium.

An Underwater Locator Device (ULD), also referred to as an underwater acoustic beacon, is mounted horizontally on the front of the CSMU and is also used as the recorder's carrying handle. The ULD is equipped with a battery that has an expected life of six years.

The Ground Support Equipment (GSE) connector is located on the front of the CVR. This connector provides the interface from the recorder to the GSE for checkout of the recorder, or to transfer data to a readout device. The CVR is connected to aircraft wiring via a 57 pin, DPXB-style connector. Both are tray mountable.

Following are salient functionality features of a modern CVR:

- 2 hours, 4 channel, high quality audio recording.
- Three voice channel inputs (150-3500 Hz).
- One cockpit area microphone input (150-6000 Hz).
- 32 Mbytes of dedicated data-link recording memory.
- Dedicated ARINC 429, high speed, data-link input.
- 28 VDC (10 W max) power input.

POST-CRASH CVR/FDR UNIT RECOVERY

All modern aircraft are equipped with an Emergency Locator Transmitter (ELT). This radio beacon with an external fixed antenna interfaces with a dedicated Search and Rescue non-geostationary satellite system (COSPAS-SARSAT). When activated, such beacons send out (50 seconds after activation) a worldwide monitored distress signal on 406 MHz (formerly 121.5 MHz), that either can be located by triangulation or by a GPS (Global Positioning System) signal, whenever equipped. The ELT, can be:

- Automatic Fixed: Rigidly mounted inside the fuselage of the aircraft and automatically triggered by the G-force sensing switch, or manually from the cockpit (it cannot send a signal to the satellite system when underwater).
- Portable Survival (as a complement of the automatic fixed): Generally carried in the cabin or packed into the escape slides/rafts of the aircraft. It floats and can be manually activated, water activated or both, depending on the model. The cabin crew would normally activate these beacons on evacuating the aircraft or requires a survivor to manually do so when the model is not water activated.

When the ELT signal is properly received, it indicates the accident occurrence and the on-ground localization. As it doesn't work under water, the sunken wreckage can be located thanks to the Underwater Locator Beacon (ULB). The ULB is attached to the memory unit of each recorder and emits, upon immersion, an acoustic signal of 37.5 kHz at an interval of one per second.

POST-CRASH DATA RECOVERY

After the recorder recovery, the Crash Survivable Memory Unit (CSMU) is extracted from its crash protection housing for data recovery for further investigation. The FDR and CVR have brought a great

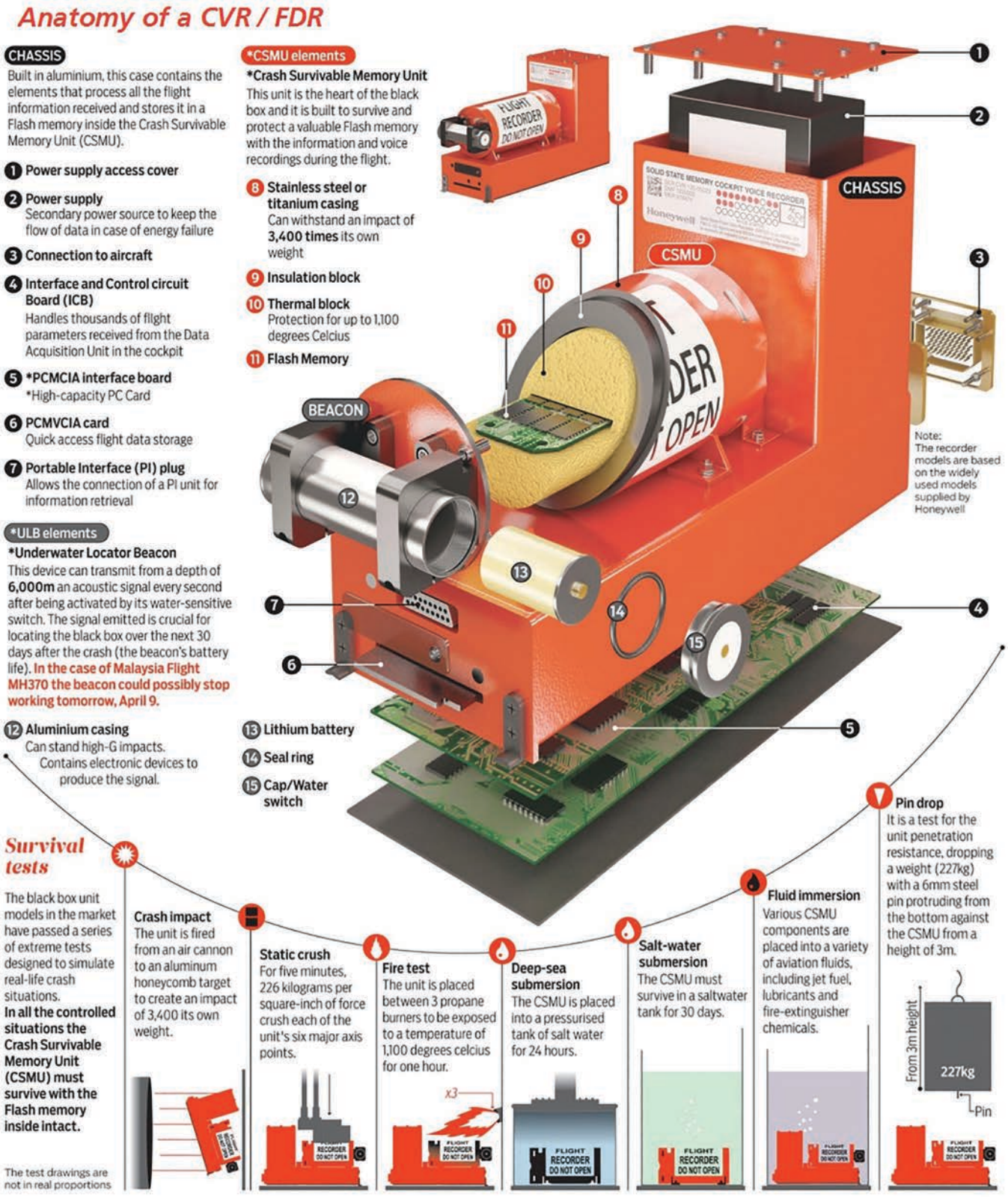


Figure 4-66. The anatomy of a Cockpit Voice Recorder (CVR)/Flight Data recorder (FDR).

contribution to safety by allowing experience feedback from accidents for improving designs of aircraft, air traffic management, improved operational procedures and trainings. However, there are still possibilities for improvement. Data and voice recordings sometimes do not completely allow the root cause analysis of an accident. Additional recordings are under study to enhance the investigation process.



Figure 4-67. A VOR ground station.

VERY HIGH FREQUENCY OMNI-DIRECTIONAL RANGE (VOR)

One of the oldest and most useful navigational aids is the VOR system. The system was constructed after WWII and is still in use today. It consists of thousands of land-based transmitter stations, or VORs, that communicate with radio receiving equipment on board aircraft. Many of the VORs are located along airways. The Victor airway system is built around the VOR navigation system. Ground VOR transmitter units are also located at airports where they are known as TVOR (terminal VOR). The U.S. Military has a navigational system known as TACAN that operates similarly to the VOR system. Sometimes VOR and TACAN transmitters share a location. These sites are known as VORTACs.

The position of all VORs, TVORs, and VORTACs are marked on aeronautical charts along with the name of the station, the frequency to which an airborne receiver must be tuned to use the station, and a Morse code designation for the station. Some VORs also broadcast a voice identifier on a separate frequency that is included on the chart. (***Figure 4-67***)

VOR uses VHF radio waves (108–117.95 MHz) with 50 kHz separation between each channel. This keeps atmospheric interference to a minimum but limits the VOR to line-of-sight usage. To receive VOR VHF radio waves, generally a V-shaped, horizontally polarized, bi-pole antenna is used. A typical location for the V dipole is in the vertical fin. Other type antennas are also certified. Follow the manufacturer's instructions for installation location. (***Figure 4-68***)

The signals produced by a VOR transmitter propagate 360° from the unit and are used by aircraft to navigate to and from the station with the help of an onboard VOR receiver and display instruments. A pilot is not required to fly a pattern to intersect the signal from a VOR station since it propagates out in every direction. The radio waves are received as long as the aircraft is in range of the ground unit and regardless of the aircraft's direction of travel. (***Figure 4-69***)

A VOR transmitter produces two signals that a receiver on board an aircraft uses to locate itself in relation to the ground station. One signal is a reference signal. The second is produced by electronically rotating a variable signal. The variable signal is in phase with the reference signal when at magnetic north, but becomes increasingly



Figure 4-68. V-shaped, horizontally polarized, bi-pole antennas are commonly used for VOR and VOR/glideslope reception. All antenna shown are VOR/glideslope antenna.

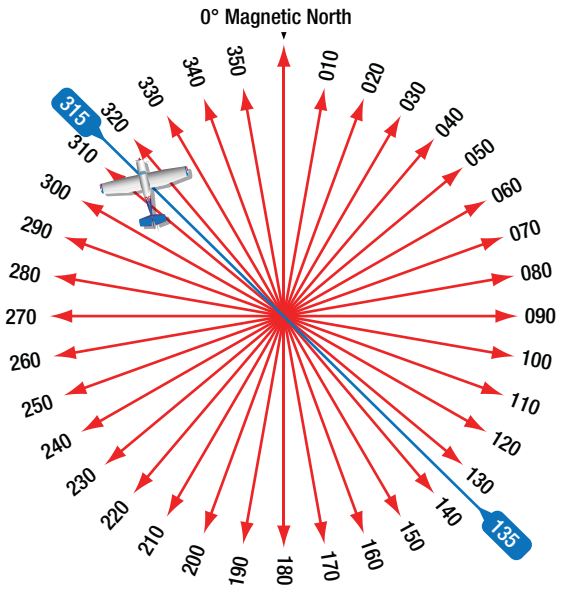


Figure 4-69. A VOR transmitter produces signals for 360° radials that an airborne receiver uses to indicate the aircraft's location in relation to the VOR station regardless of the aircraft's direction of flight. The aircraft shown is on the 315° radial even though it does not have a heading of 315°.

out of phase as it is rotated to 180°. As it continues to rotate to 360° (0°), the signals become increasingly in phase until they are in phase again at magnetic north. The receiver in the aircraft deciphers the phase difference and determines the aircraft's position in degrees from the VOR ground based unit. (***Figure 4-70***)

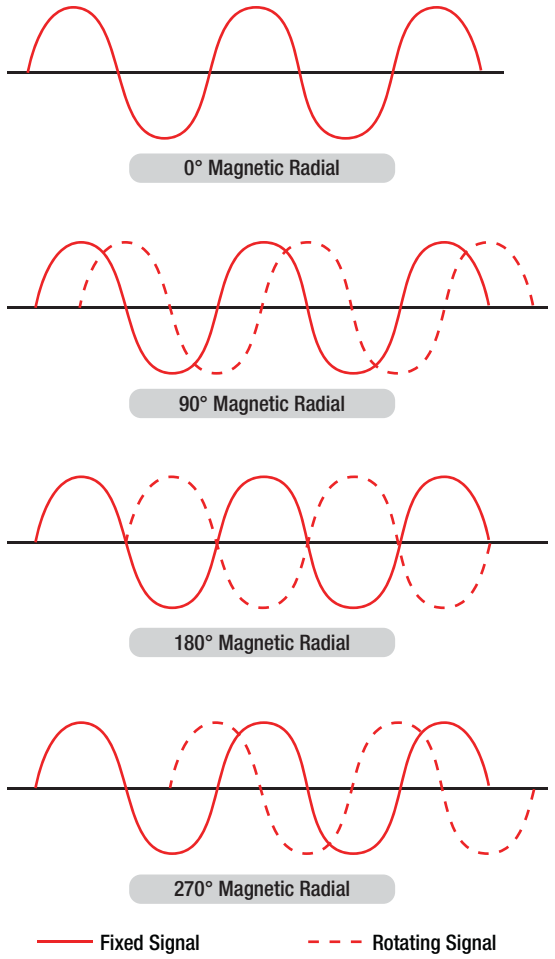


Figure 4-70. The phase relationship of the two broadcast VOR signals.



Figure 4-71. A NAV/COM receiver typically found in light aircraft.

Most aircraft carry a dual VOR receiver. Sometimes, the VOR receivers are part of the same avionics unit as the VHF communication transceiver(s). These are known as NAV/COM radios. Internal components are shared since frequency bands for each are adjacent. (***Figure 4-71***)

Large aircraft may have two dual receivers and even dual antennas. Normally, one receiver is selected for use and the second is tuned to the frequency of the next VOR station to be encountered en route. A means for switching between NAV1 and NAV2 is provided as is a switch for selecting the active or standby frequency. (***Figure 4-73***) VOR receivers are also found coupled with instrument landing system (ILS) receivers and glideslope receivers.

A VOR receiver interprets the bearing in degrees to (or from) the VOR station where the signals are generated. It also produces DC voltage to drive the display of the deviation from the desired course centerline to (or from) the selected station. Additionally, the receiver decides whether or not the aircraft is flying toward the VOR or away from it. These items can be displayed a number of different ways on various instruments. Older aircraft are often equipped with a VOR gauge dedicated to display only VOR information. This is also called an omni-bearing selector (OBS) or a course deviation indicator (CDI). (***Figure 4-72***)

The CDI linear indicator remains essentially vertical but moves left and right across the graduations on the instrument face to show deviation from being on course. Each graduation represents 2°. The OBS knob rotates the azimuth ring. When in range of a VOR, the pilot rotates



Figure 4-72. An airliner VOR control head with two independent NAV receivers each with an active and standby tuning circuit controlled by a toggle switch.

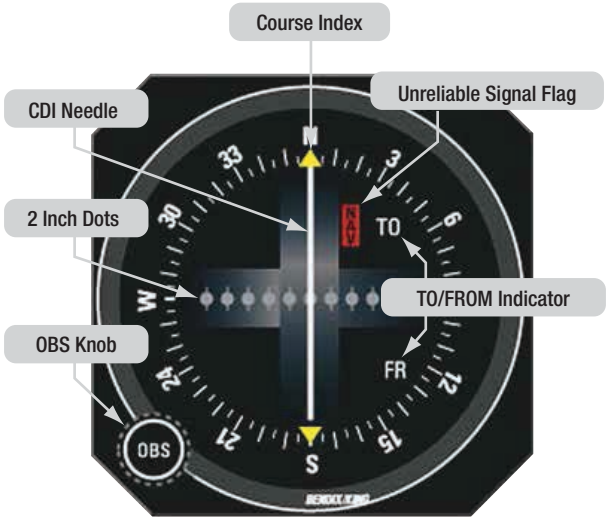


Figure 4-73. A traditional VOR gauge, also known as a course deviation indicator (CDI) or an omni-bearing selector (OBS).

the OBS until the course deviation indicator centers. For each location of an aircraft, the OBS can be rotated to two positions where the CDI will center. One produces an arrow in the TO window of the gauge indicating that the aircraft is traveling toward the VOR station. The other selectable bearing is 180° from this. When chosen, the arrow is displayed in the FROM window indicating the aircraft is moving away from the VOR on the course selected. The pilot must steer the aircraft to the heading with the CDI centered to fly directly to or from the VOR. The displayed VOR information is derived from deciphering the phase relationship between the two simultaneously transmitted signals from the VOR ground station. When power is lost or the VOR signal is weak or interrupted, a NAV warning flag comes into view. (***Figure 4-73***)

A separate gauge for the VOR information is not always used. As flight instruments and displays have evolved, VOR navigation information has been integrated into other instruments displays, such as the radio magnetic indicator (RMI), the horizontal situation indicator (HSI), an EFIS display or an electronic attitude director indicator (EADI). Flight management systems and automatic flight control systems are also made to integrate VOR information to automatically control the aircraft on its planned flight segments. Flat panel MFDs integrate VOR information into moving map presentations and other selected displays. The basic information of the radial bearing in degrees, course deviation indication, and to/from information remains unchanged however. (***Figure 4-74***)

At large airports, an instrument landing system (ILS) guides the aircraft to the runway while on an instrument landing approach. The aircraft's VOR receiver is used to interpret the radio signals. It produces a more sensitive course deviation indication on the same instrument display as the VOR CDI display. This part of the ILS is known as the localizer and is discussed below. While tuned to the ILS localizer frequency, the VOR circuitry of the VOR/ILS receiver is inactive.

It is common at VOR stations to combine the VOR transmitter with distance measuring equipment (DME) or a non-directional beacon (NDB) such as an ADF transmitter and antenna. When used with a DME, pilots can gain an exact fix on their location using the VOR and DME together. Since the VOR indicates the aircraft's bearing to the VOR transmitter and a co-located DME indicates how far away the station is, this relieves the pilot from having to fly over the station to know with certainty his or her location. These navigational aids are discussed separately in the following sections.

Functional accuracy of VOR equipment is critical to the safety of flight. VOR receivers are operationally tested using VOR test facilities (VOT). These are located at numerous airports that can be identified in the Airport Facilities Directory for the area concerned. Specific points on the airport surface are given to perform the test. Most VOTs require tuning 108.0 MHz on the VOR receiver and centering the CDI. The OBS should indicate 0° showing FROM on the indicator or 180°

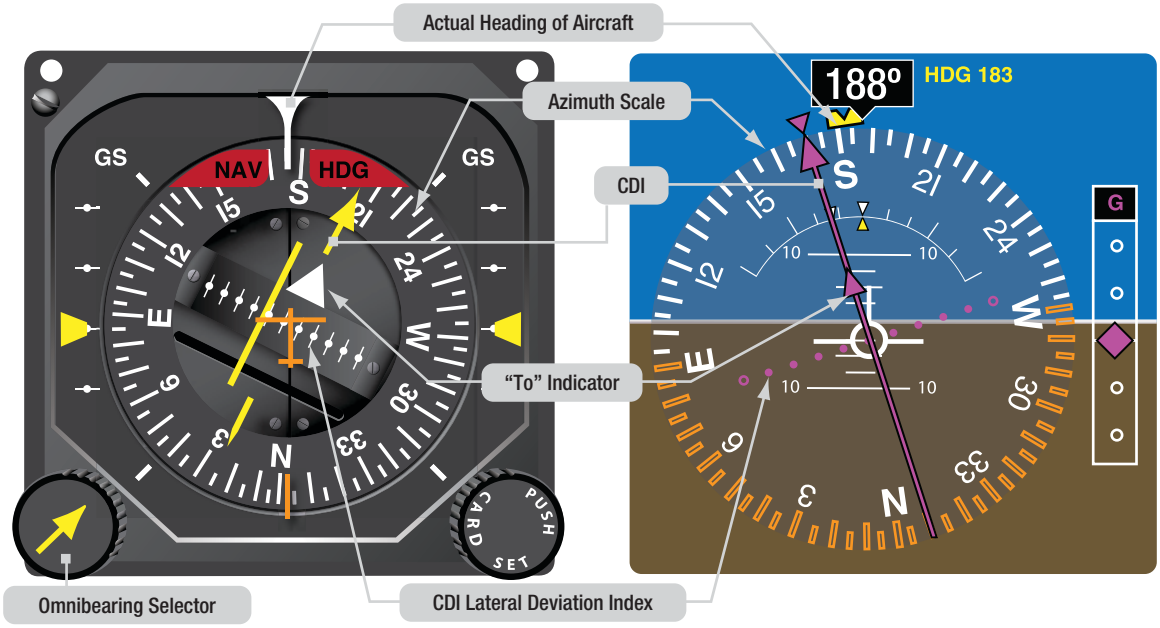


Figure 4-74. A mechanical HSI (left) and an electronic HSI (right) both display VOR information.

when showing TO. If an RMI is used as the indicator, the test heading should always indicate 180˚. Some repair stations can also generate signals to test VOR receivers although not on 108.0 MHz. Contact the repair station for the transmission frequency and for their assistance in checking the VOR system. A logbook entry is required. It may be noted that some airborne testing using VOTs is possible by the pilot.

An error of ±4˚ should not be exceeded when testing a VOR system with a VOT. An error in excess of this prevents the use of the aircraft for IFR fight until repairs are made. Aircraft having dual VOR systems where only the antenna is shared may be tested by comparing the output of each system to the other. Tune the VOR receivers to the local ground VOR station. A bearing indication difference of no more than ±4˚ is permissible.

AUTOMATIC DIRECTION FINDING (ADF)

An automatic direction finder (ADF) operates off of a ground signal transmitted from a NDB. Early radio direction finders (RDF) used the same principle. A vertically polarized antenna was used to transmit LF frequency radio waves in the 190 kHz to 535 kHz range. A receiver on the aircraft was tuned to the transmission frequency of the NDB. Using a loop antenna, the direction to (or from) the antenna could be determined by monitoring the strength of the signal received. This was possible because a radio wave striking a loop antenna broadside induces a null signal. When striking it in the plane of the loop, a much stronger signal is induced. The NDB signals were modulated with unique Morse code pulses that enabled the pilot to identify the beacon to which he or she was navigating.

With RDF systems, a large rigid loop antenna was installed inside the fuselage of the aircraft. The broadside of the antenna was perpendicular to the aircraft's longitudinal axis. The pilot listened for variations in signal strength of the LF broadcast and maneuvered the aircraft so a gradually increasing null signal was maintained. This took them to the transmitting antenna. When over flown, the null signal gradually faded as the aircraft became farther from the station. The increasing or decreasing strength of the null signal was the only way to determine if the aircraft was flying to or from the NDB. A deviation left or right from the course caused the signal strength to sharply increase due to the loop antenna's receiving properties.

The ADF improved on this concept. The broadcast frequency range was expanded to include MF up to about 1 800 kHz. The heading of the aircraft no longer needed to be changed to locate the broadcast transmission antenna. In early model ADFs, a rotatable antenna was used instead. The antenna rotated to seek the position in which the signal was null. The direction to the broadcast antenna was shown on an azimuth scale of an ADF indicator in the flight deck. This type

of instrument is still found in use today. It has a fixed card with 0˚ always at the top of a non-rotating dial. A pointer indicates the relative bearing to the station. When the indication is 0˚, the aircraft is on course to (or from) the station. (***Figure 4-75***)

As ADF technology progressed, indicators with rotatable azimuth cards became the norm. When an ADF signal is received, the pilot rotates the card so that the present heading is at the top of the scale. This results in the pointer indicating the magnetic bearing to the ADF transmitter. This is more intuitive and consistent with other navigational practices. (***Figure 4-76***)

In modern ADF systems, an additional antenna is used to remove the ambiguity concerning whether the aircraft is heading to or from the transmitter. It is called a sense antenna. The reception field of the sense antenna is omni-directional. When combined with the fields of the loop antenna, it forms a field with a single significant null reception area on one side. This is used for tuning and produces an indication in the direction toward the ADF station at all times. The onboard ADF receiver needs only to be tuned to the correct frequency of the broadcast transmitter for the system to work. The loop and sense antenna are normally housed in a single, low profile antenna housing. (***Figure 4-77***)

Any ground antenna transmitting LF or MF radio waves in range of the aircraft receiver's tuning capabilities can be used for ADF. This includes those from AM radio stations. Audible identifier tones are loaded on the NDB carrier waves. Typically, a two-character Morse code designator is used.

With an AM radio station transmission, the AM broadcast is heard instead of a station identifier code. The frequency for an NDB transmitter is given on an aeronautical chart next to a symbol for the transmitter. The identifying designator is also given. (***Figure 4-78***)

ADF receivers can be mounted in the flight deck with the controls accessible to the user. This is found on many general aviation aircraft. Alternately, the ADF receiver is mounted in a remote avionics bay with only the control head in the flight deck. Dual ADF receivers are common. ADF information can be displayed on the ADF indicators mentioned or it can be digital. Modern, flat, multipurpose electronic displays usually display the

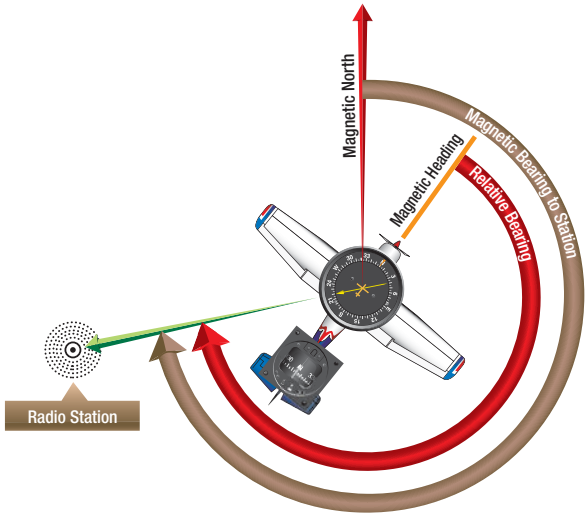


Figure 4-75. Older ADF indicators have nonrotating azimuth cards. 0˚ is fixed at the top of the instrument and the pointer always indicates the relative bearing to the ADF transmission antenna. To fly to the station, the pilot turns the aircraft until the ADF pointer indicates 0˚.



Figure 4-76. A movable card ADF indicator can be rotated to put the aircraft's heading at the top of the scale. The pointer then points to the magnetic bearing the ADF broadcast antenna.

ADF digitally. (***Figure 4-79***) When ANT is selected on an ADF receiver, the loop antenna is cut out and only the sense antenna is active. This provides better multi-directional reception of broadcasts in the ADF frequency range, such as weather or AWAS broadcasts.

When the best frequency oscillator (BFO) is selected on an ADF receiver/controller, an internal beat frequency oscillator is connected to the IF amplifier inside the ADF receiver. This is used when an NDB does not transmit a modulated signal. Continued refinements to ADF technology has brought it to its current state. The

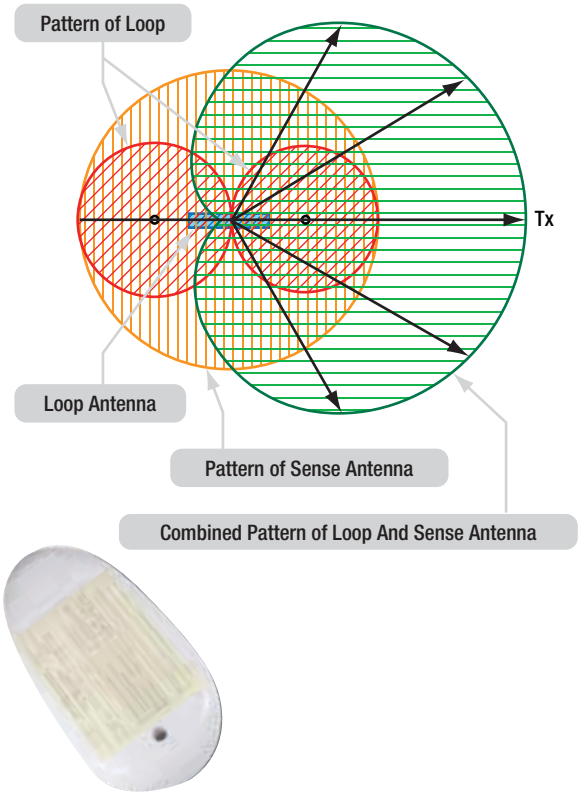


Figure 4-77. The reception fields of a loop and sense antenna combine to create a field with a sharp null on just one side. This removes directional ambiguity when navigating to an ADF station.



Figure 4-78. Non-directional broadcast antenna in the LF and medium frequency range are used for ADF navigation.

rotating receiving antenna is replaced by a fixed loop with a ferrite core. This increases sensitivity and allows a smaller antenna to be used. The most modern ADF systems have two loop antennas mounted at 90° to each other. The received signal induces voltage that is sent to two stators in a resolver or goniometer. The goniometer stators induce voltage in a rotor that correlates to the signal of the fixed loops. The rotor is driven by a motor to seek the null. The same motor rotates the pointer in the flight deck indicator to show the relative or magnetic bearing to the station. (***Figure 4-80***)



Figure 4-79. A cockpit mountable ADF receiver used on general aviation aircraft.

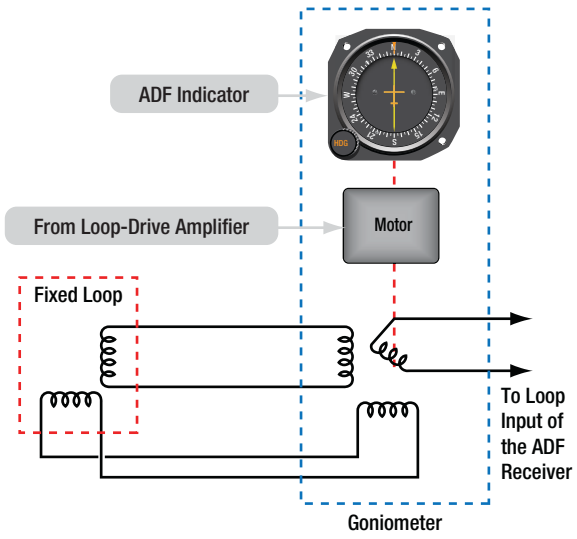


Figure 4-80. In modern ADF, a rotor in a goniometer replaces the rotating loop antenna used in earlier models.

ADF antenna is critical to a correct indication since it is a directional device. Calibration with the longitudinal axis of the fuselage or nose of the aircraft is important. A single null reception area must exist in the correct direction. The antenna must be oriented so the ADF indicates station location when the aircraft is flying toward it rather than away. Follow all manufacturer's instructions.

INSTRUMENT LANDING SYSTEMS (ILS)

An ILS is used to land an aircraft when visibility is poor. This radio navigation system guides the aircraft down a slope to the touch down area on the runway. Multiple radio transmissions are used that enable an exact approach to landing with an ILS. A localizer is one of the radio transmissions. It is used to provide horizontal guidance to the center line of the runway. A separate glideslope broadcast provides vertical guidance of the aircraft down the proper slope to the touch down point. Compass locator transmissions for outer and middle approach marker beacons aid the pilot in intercepting the approach navigational aid system. Marker beacons provide distance from the runway information. Together, all of these radio signals make an ILS a very accurate and reliable means for landing aircraft. (***Figure 4-81***)

LOCALIZER

The localizer broadcast is a VHF broadcast in the lower range of the VOR frequencies (108 MHz to 111.95 MHz) on odd frequencies only. Two modulated signals are produced from a horizontally polarized antenna complex beyond the far end of the approach runway. They create an expanding field that is 2-1/2° wide (about 450 meters) 8 km from the runway. The field tapers to runway width near the landing threshold. The left side of the approach area is filled with a VHF carrier wave modulated with a 90 Hz signal. The right side of the approach contains a 150 MHz modulated signal. The aircraft's VOR receiver is tuned to the localizer VHF frequency that can be found on published approach plates and aeronautical charts.

The circuitry specific to standard VOR reception is inactive while the receiver uses localizer circuitry and components common to both. The signals received are passed through filters and rectified into DC to drive the course deviation indicator. If the aircraft receives a 150 Hz signal, the CDI of the VOR/ILS display deflects to the left. This indicates that the runway is to the left. The pilot must correct course with a turn to the left. This centers course deviation indicator on the display and centers the aircraft with the centerline of the runway. If the 90 Hz signal is received by the VOR receiver, the CDI deflects to the right. The pilot must turn toward the right to center the CDI and the aircraft with the runway center line. (***Figure 4-82***)

GLIDESLOPE

The vertical guidance required for an aircraft to descend for a landing is provided by the glideslope of the ILS. Radio signals funnel the aircraft down to the touchdown point on the runway at an angle of approximately 3°. The transmitting glideslope antenna is located off to the side of the approach runway approximately 1 000 feet from the threshold. It transmits in a wedge-like pattern with the field narrowing as it approaches the runway. (***Figure 4-83***)

The glideslope transmitter antenna is horizontally polarized. The transmitting frequency range is UHF between 329.3 MHz and 335.0 MHz. The frequency is paired to the localizer frequency of the ILS. When the VOR/ILS receiver is tuned for the approach, the glideslope receiver is automatically tuned. Like the localizer, the glideslope transmits two signals, one modulated at 90 Hz and the other modulated at 150 Hz. The aircraft's glideslope receiver deciphers the signals similar to the method of the localizer receiver. It drives a vertical course deviation indicator known as the glideslope indicator. The glideslope indicator operates identically to the localizer CDI only 90° to it. The VOR/ILS localizer CDI and the glideslope are displayed together on whichever kind of instrumentation is in the aircraft. (***Figure 4-84***)

The UHF antenna for aircraft reception of the glideslope signals comes in many forms. A single dipole antenna mounted inside the nose of the aircraft is a common option. Antenna manufacturers have also incorporated glideslope reception into the same dipole antenna used for the VHS VOR/ILS localizer reception. Blade type antennas are also used. (***Figure 4-85***) ***Figure 4-86*** shows a VOR and a glideslope receiver for a GA aircraft ILS.

COMPASS LOCATORS

It is imperative that a pilot be able to intercept the ILS to enable its use. A compass locator is a transmitter designed for this purpose. There is typically one located at the outer marker beacon 6-11 km from the runway threshold. Another may be located at the middle marker beacon about 1 km from the threshold. The outer marker compass locator is a 25 watt NDB with a range of about 24 km. It transmits omnidirectional LF radio waves (190 Hz to 535 Hz) keyed with the first two letters of the ILS identifier. The ADF receiver is used to intercept the locator so no additional equipment is required. If a middle marker compass locator is in place, it is similar but is identified with the last two letters of the ILS identifier. Once located, the pilot maneuvers the aircraft to fly down the glide path to the runway.

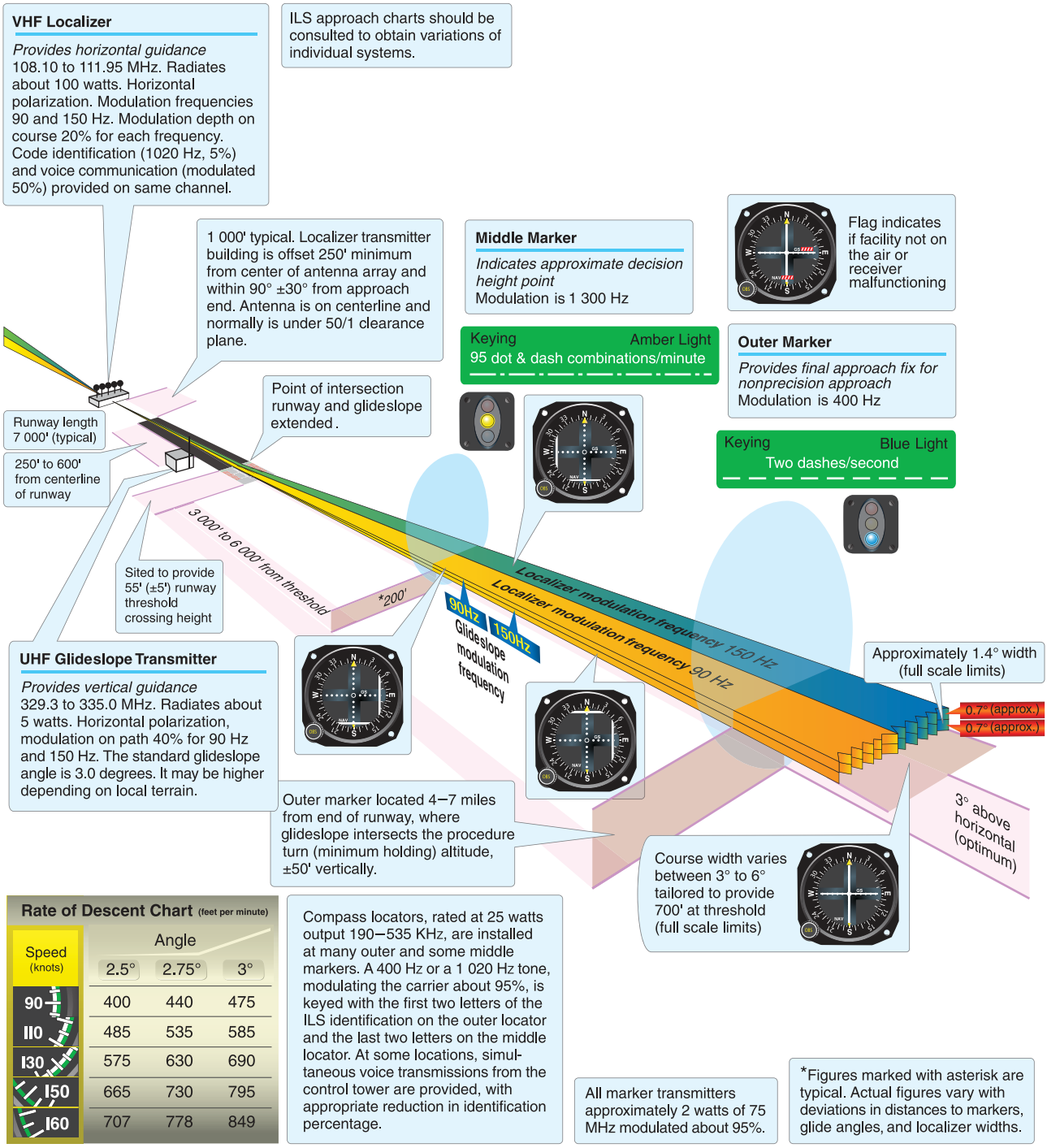


Figure 4-81. Components of an instrument landing system (ILS).



Figure 4-82. An ILS localizer antenna.

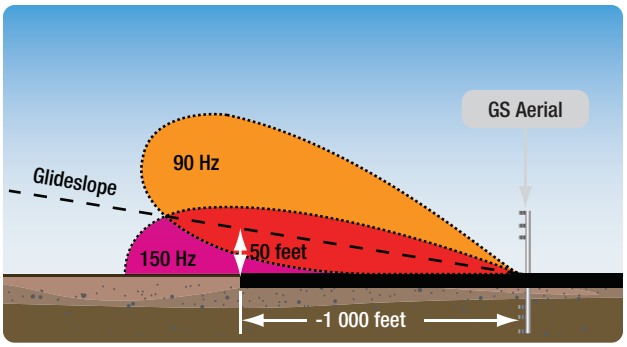


Figure 4-83. A glideslope antenna broadcasts radio signals to guide an aircraft vertically to the runway.



Figure 4-84. A traditional course deviation indicator is shown on the left. The horizontal white line is the deviation indicator for the glideslope. The vertical line is for the localizer. On the right, a Garmin G-1000 PFD illustrates an aircraft during an ILS approach. The narrow vertical scale on the right of the attitude indicator with the "G" at the top is the deviation scale for the glideslope. The green diamond moves up and down to reflect the aircraft being above or below the glidepath. The diamond is shown centered indicating the aircraft is on course vertically. The localizer CDI can be seen at the bottom center of the display. It is the center section of the vertical green course indicator. LOC1 is displayed to the left of it.

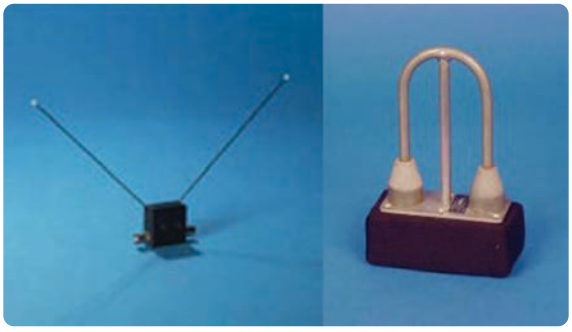


Figure 4-85. Glideslope antennas—designed to be mounted inside a non-metallic aircraft nose (left), and mounted inside or outside the aircraft (right).

MARKER BEACONS

Marker beacons are the final radio transmitters used in the ILS. They transmit signals that indicate the position of the aircraft along the glide-path to the runway. As mentioned, an outer marker beacon transmitter is located 6-11 km from the threshold. It transmits a 75 MHz carrier wave modulated with a 400 Hz audio tone in a series of dashes. The transmission is very narrow and directed straight up. A marker beacon receiver receives the signal and uses it to light a blue light on the instrument panel. This, plus the oral tone in combination with the localizer and the glideslope indicator, positively locates the aircraft on an approach. (***Figure 4-86***)

A middle marker beacon is also used. It is located on approach approximately 1 km from the runway. It also transmits at 75 MHz. The middle marker transmission is modulated with a 1 300 Hz tone that is a series of dots and dashes so as to not be confused with the all dash tone of the outer marker. When the signal is received, it is used in the receiver to illuminate an amber-colored light on the instrument panel. (***Figure 4-87***)

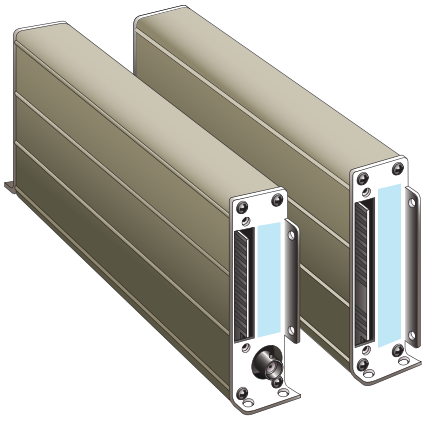


Figure 4-86. A localizer and glideslope receiver for a general aviation aircraft ILS.



Figure 4-87. Various marker beacon instrument panel display lights.

Some ILS approaches have an inner marker beacon that transmits a signal modulated with 3 000 Hz in a series of dots only. It is placed at the land-or-go-around decision point of the approach close to the runway threshold. If present, the signal when received is used to illuminate a white light on the instrument panel. The three marker beacon lights are usually incorporated into the audio panel of a general aviation aircraft or may exist independently on a larger aircraft. Electronic display aircraft usually incorporate marker lights or indicators close to the glideslope display near attitude director indicator. (***Figure 4-88***)

ILS radio components can be tested with an ILS test unit. Localizer, glideslope, and marker beacon signals are generated to ensure proper operation of receivers and correct display on flight deck instruments. (***Figure 4-89***)

FLIGHT DIRECTOR SYSTEMS (FDS)

A flight director system is an instrument system consisting of electronic components that compute and indicate the aircraft attitude required to attain and maintain a preselected flight condition. A command bar on the aircraft's attitude indicator shows the pilot how much and in what direction the attitude of the aircraft must be changed to achieve the desired result. The computed command indications relieve the pilot of many of the mental calculations required for instrument flights, such as interception angles, wind drift correction, and rates of climb and descent. Essentially, a flight director system is an autopilot system without the servos.

All of the same sensing and computations are made, but the pilot controls the aeroplane and makes maneuvers by following the commands displayed on the instrument panel. Flight director systems can be part of an autopilot system or exist on aircraft that do not possess full autopilot systems. Many autopilot systems allow for the option of engaging or disengaging a flight director display. Flight director information is displayed on the instrument that displays the aircraft's attitude. The process is accomplished with a visual reference



Figure 4-88. An outer marker transmitter antenna 4-7 miles from the approach runway transmits a 75 MHz signal straight up (left). Aircraft mounted marker beacon receiver antennas are shown (center and right).

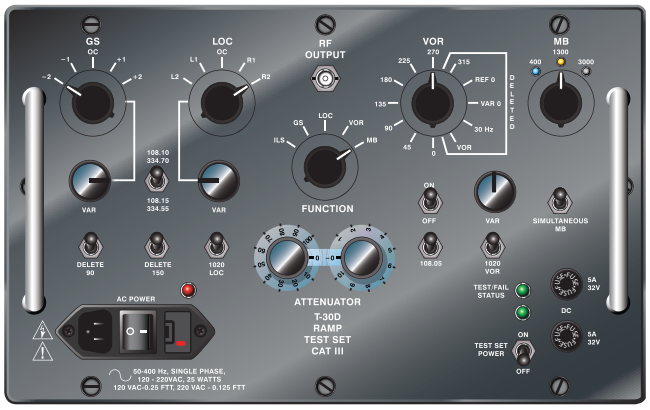


Figure 4-89. An ILS Test Unit.

technique. A symbol representing the aircraft is fit into a command bar positioned by the flight director in the proper location for a maneuver to be accomplished. The symbols used to represent the aircraft and the command bar vary by manufacturer. Regardless, the object is always to fly the aircraft symbol into the command bar symbol. (***Figure 4-90***).

The instrument that displays the flight director commands is known as a flight director indicator (FDI), attitude director indicator (ADI), or electronic attitude director indicator (EADI). It may even be referred to as an artificial horizon with flight director. This display element combines with the other primary components of the flight director system. Like an autopilot, these consist of the sensing elements, a computer, and an interface panel. Integration of navigation features into the attitude indicator is highly useful. The flight director contributes to this usefulness by indicating to the pilot how to maneuver the aeroplane to navigate a desired course. Selection of the VOR function on the flight director control panel links the computer to the Omni range receiver. The pilot selects a desired course and the flight director displays the bank attitude necessary to intercept and maintain this course. Allocations for wind drift and calculation of the intercept angle is performed automatically.

Flight director systems vary in complexity and features. Many have altitude hold, altitude select, pitch hold, and other features. But flight director systems are designed to offer the greatest assistance during the instrument approach phase of flight. ILS localizer and glideslope signals are transmitted through the receivers to the computer and are presented as command indications. This allows the pilot to fly the aeroplane down the optimum approach path to the runway using the flight director system. With the altitude hold function engaged, level flight can be maintained during the maneuvering and procedure turn phase of an approach. Altitude hold automatically disengages when the glideslope is intercepted.

Once inbound on the localizer, the command signals of the flight director are maintained in a centered or zero condition. Interception of the glideslope causes a downward indication of the command pitch indicator. Any deviation from the proper glideslope path causes a fly-up or fly-down command indication. The pilot needs only to keep the aeroplane symbol fit into the command bar.

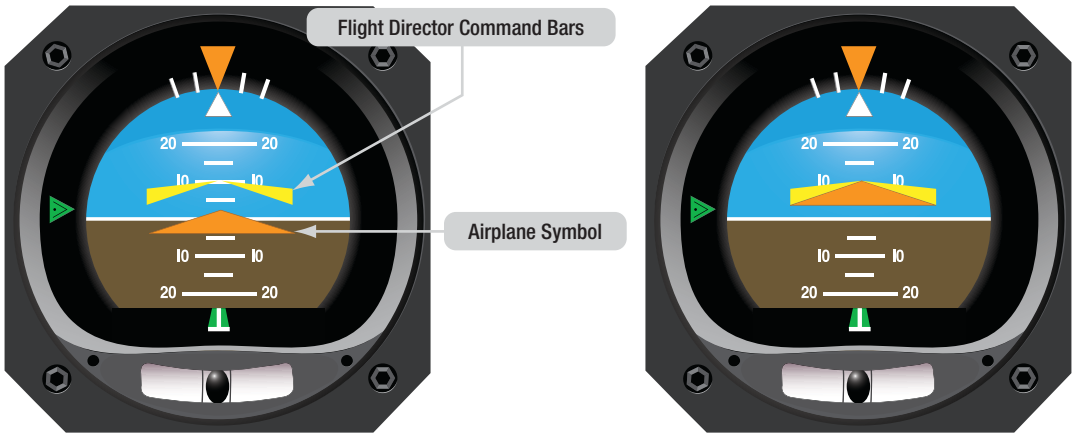


Figure 4-90. The fight director command bar signals the pilot how to steer the aircraft for a maneuver. By flying the aircraft so the triangular aeroplane symbol fits into the command bar, the pilot performs the maneuver calculated by the flight director. The instrument shown on the left is commanding a climb while the aeroplane is flying straight and level. The instrument on the right shows that the pilot has accomplished the maneuver.

DISTANCE MEASURING EQUIPMENT (DME)

Many VOR stations are co-located with the military version of the VOR station, which is known as TACAN. When this occurs, the navigation station is known as a VORTAC station. Civilian aircraft make use of one of the TACAN features not originally installed at civilian VOR stations - distance measuring equipment (DME). A DME system calculates the distance from the aircraft to the DME unit at the VORTAC ground station and displays it on the flight deck. It can also display calculated aircraft speed and elapsed time for arrival when the aircraft is traveling to the station.

DME ground stations have subsequently been installed at civilian VORs, as well as in conjunction with ILS localizers. These are known as VOR/DME and ILS/DME or LOC/DME. The latter aid in approach to the runway during landings. The DME system consists of an airborne DME transceiver, display, and antenna, as well as the ground based DME unit and its antenna. (***Figure 4-91***)

The DME is useful because with the bearing (from the VOR) and the distance to a known point (the DME antenna at the VOR), a pilot can positively identify the location of the aircraft. DME operates in the UHF frequency range from 962 MHz to 1 213 MHz. A carrier signal transmitted from the aircraft is modulated with a string of integration pulses. The ground unit receives the pulses and returns a signal to the aircraft. The time that transpires for the signal to be sent and returned is calculated and converted into nautical miles for display. Time to station and speed are also calculated and displayed. DME readout can be on a dedicated DME display or it can be part of an EHSI, EADI, EFIS, or on the primary flight display in a glass cockpit. (***Figure 4-92***)

The DME frequency is paired to the co-located VOR or VORTAC frequency. When the correct frequency is tuned for the VOR signal, the DME is tuned automatically. Tones are broadcast for the VOR station identification and then for the DME. The hold selector on a DME panel keeps the DME tuned in while the VOR selector is tuned to a different VOR. In most cases, the UHF of the DME is transmitted and received via a small blade-type antenna mounted to the underside of the fuselage centerline. (***Figure 4-93***)



Figure 4-92. Distance information from the DME can be displayed on a dedicated DME instrument or integrated into any of the electronic navigational displays found on modern aircraft. A dual display DME is shown with its remote mounted receiver.



Figure 4-91. A VOR with DME ground station.



Figure 4-93. A typical aircraft mounted DME antenna.

A traditional DME displays the distance from the DME transmitter antenna to the aircraft. This is called the slant distance. It is very accurate. However, since the aircraft is at altitude, the distance to the DME ground antenna from a point directly beneath the aircraft is shorter. Some modern DMEs are equipped to calculate this ground distance and display it. (***Figure 4-94***)

AREA NAVIGATION (RNAV)

Area navigation (RNAV) is a general term used to describe the navigation from point A to point B without direct over flight of navigational aids, such as VOR stations or ADF non-directional beacons. It includes VORTAC and VOR/DME based systems, as well as systems of RNAV based around LORAN, GPS, INS, and the FMS of transport category aircraft. However, until recently, the term RNAV was most commonly used to describe the area navigation or the process of direct flight from point A to point B using VORTAC and VOR/DME based references which are discussed in this section.

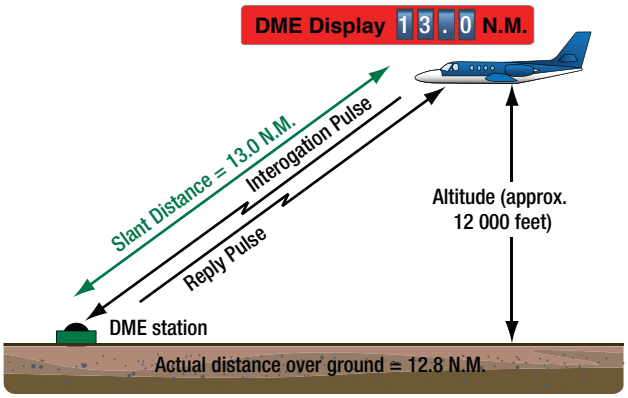


Figure 4-94. Many DME's only display the slant distance, which is the actual distance from the aircraft to the DME station. This is different than the ground distance due to the aircraft being at altitude. Some DMEs compute the ground distance for display.

All RNAV systems make use of waypoints. A waypoint is a designated geographical location or point used for route definition or progress-reporting purposes. It can be defined or described by using latitude/longitude grid coordinates or, in the case of VOR based RNAV, described as a point on a VOR radial followed by that point is distance from the VOR station (i.e., 200/25 means a point 25 nautical miles from the VOR station on the 200° radial). ***Figure 4-95*** illustrates an RNAV route of flight from airport A to airport B. The VOR/DME and VORTAC stations shown are used to create phantom waypoints that are overflown rather than the actual stations. This allows a more direct route to be taken. The phantom waypoints are entered into the RNAV course-line computer (CLC) as a radial and distance number pair. The computer

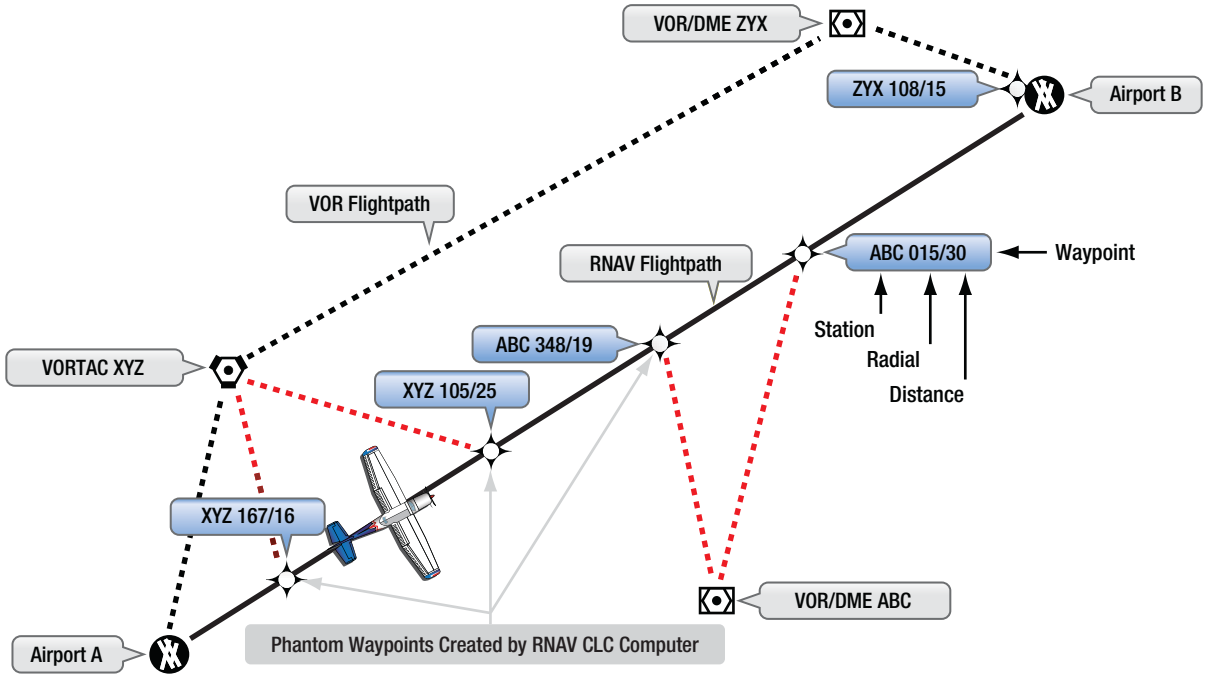


Figure 4-95. The pilot uses the aircraft's course deviation indicator to fly to and from RNAV phantom waypoints created by computer. This allows direct routes to be created and flown rather than flying from VOR to VOR.

creates the waypoints and causes the aircraft's CDI to operate as though they are actual VOR stations. A mode switch allows the choice between standard VOR navigation and RNAV.

VOR based RNAV uses the VOR receiver, antenna, and VOR display equipment, such as the CDI. The computer in the RNAV unit uses basic geometry and trigonometry calculations to produce heading, speed, and time readouts for each waypoint. VOR stations need to be within line-of sight and operational range from the aircraft for RNAV use. (***Figure 4-96***)

RNAV has increased in flexibility with the development of GPS. Integration of GPS data into a planned VOR RNAV flight plan is possible as is GPS route planning without the use of any VOR stations.

FLIGHT MANAGEMENT SYSTEM (FMS)

Flight Management Systems, which became popular in the mid-1980s, automates a wide variety of in-flight tasks previously performed by a human navigator; thereby, reducing the workload on the flight crew. Not only can it define a desired route from the aircraft's current position to any point in the world, but the route will be based on the aircraft's operating characteristics to manage overall flight performance.

On-time take-offs and landings, fuel conservation, and long engine and component life all contribute to airline profitability. The FMS helps achieve these results by operating the aircraft with much greater precision than humanly possible.

As shown in ***Figure 4-97***, the FMS consists of dual-redundant Flight Management Computers (FMC) and Control Display Units (CDU). During normal operation, the computers crosstalk with each other to share and compare information. Each FMC is capable of operating independently in the event the other FMC fails. However, only one FMC can provides commands to the other aircraft systems at any one time.

Prior to take-off, the flight crew inputs their initial geographic location (latitude and longitude) and flight plans on one of the CDUs located in the center console. The FMC receives navigation and guidance information from the air data, inertial navigation, and global positioning system, in addition to ground-based radio navigational aids, and uses that information to compute ground speed, track, wind direction and velocity. The FMC then computes the optimum flight path, using a stored digital map terrain data base, to guide the aircraft to its final destination.



Figure 4-96. RNAV unit from a general aviation aircraft.

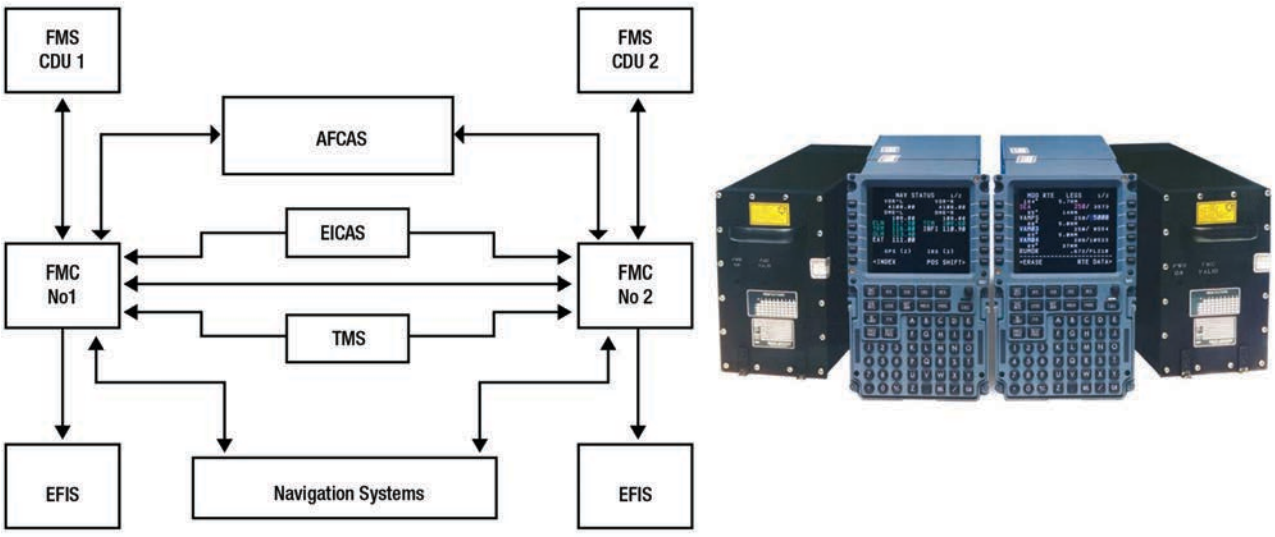


Figure 4-97. FMS Block Diagram (left) and FMS LRUs (right).

The FMC provides waypoints along the intended route on the EFIS Navigation Display and drives the flight director steering command bars on the EFIS Primary Flight Display. The FMC communicates with the Thrust Management System (TMS) that drives the auto-throttle to control aircraft speed, and the automatic

Flight Control System (FCS or AFCAS), that drives the actuators that move the control surfaces for lateral and vertical control along the computed flight path. The FMC continually optimizes the flight path as variables, such as wind speed and direction change during flight.

Each flight path has vertical, horizontal, and navigational components, which are maintained by manipulating the engine and airframe controls. While doing so, numerous options are available to the pilot. Rate of climb, thrust settings, EPR (engine pressure ratio) levels, airspeed, descent rates, etc., can be varied. Commercial air carriers use the FMS to establish guidelines by which flights can be flown to promote the company's goals for fuel and equipment conservation. The flight crew need only enter variables as requested and respond to suggested alternatives as the FMS presents them. The FMS has stored in its database literally hundreds of flight plans with predetermined operational parameters that can be selected and implemented.

Integration with onboard radio navigational aids allows the FMC to change VOR, DME and ILS frequencies as the flight plan is enacted. Internal computations, using direct input from fuel flow and fuel quantity systems, allow the FMC to carry out lean operations or pursue other objectives, such as high performance operations if making up time is paramount on a particular flight. Weather and traffic considerations are also integrated. The FMC communicates to the crew via the CDU to present its planned action, gain consensus, or ask for an input or decision. The FMC continuously monitors its inputs for faults during operation. Maintenance personal can retrieve BITE generated information and pilot recorded fault messages. They may also access maintenance pages that identify faulty LRUs that have been identified by BITE.

GLOBAL POSITIONING SYSTEM (GPS)

The global positioning system is a space-based navigation system that was launched by the U.S. Department of Defense in the early 1990's, and has since become very popular commercially, not only for aircraft navigation, but for maritime, automotive and personal applications as well. It provides position data, velocity rate, and system time to GPS receivers all around the world.

GPS consists of three segments: a space system segment, a control system segment, and a user system segment. The space system segment consists of 6 planes of satellites, each plane containing 4 satellites, in precise 10 898 mile geostationary orbits. The control system segment has several monitoring stations located in U.S. controlled territory that track all satellites in view and accumulates ranging data to provide orbit determination and eliminate any errors. The user's GPS receiver computes its distance from the satellites by measuring the travel time of the satellite's signal. By using the distances from at least three satellites, the GPS receiver can triangulate the user's current position. With measurements from four satellites, the GPS receiver can determine elevation as well. (***Figure 4-98***)

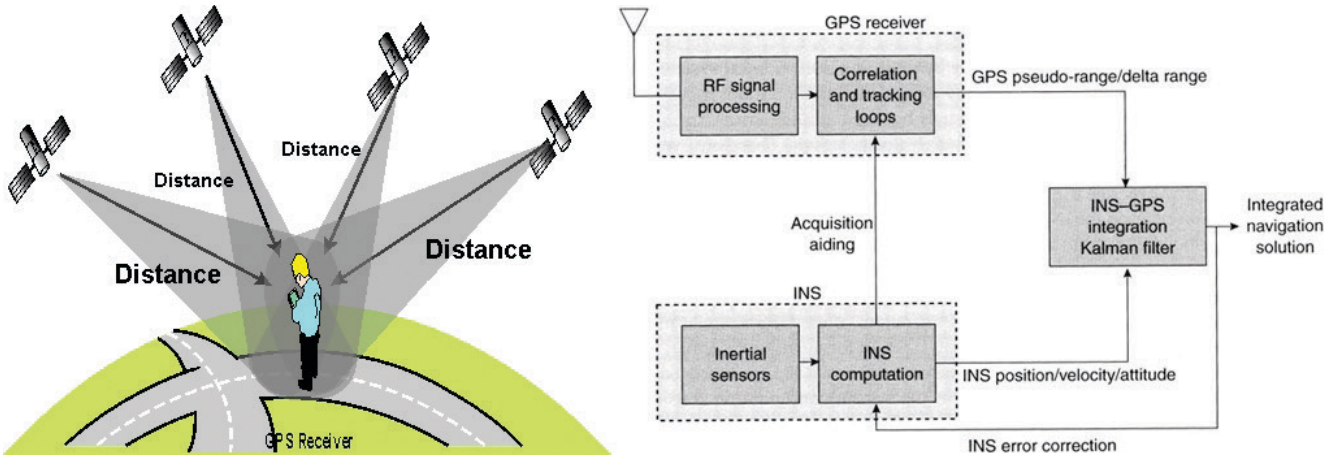


Figure 4-98. GPS Triangulation (left) and Integrated Navigation Solution (right).

The INS system only measures acceleration and angular rates and mathematically integrates these into velocity and position. However, since the inertial system outputs are obtained through integration, they drift at a very low frequency. The GPS is an ideal reference for realigning and recalibrating the INS during flight.

As shown in ***Figure 4-98***, an algorithm, known as a Kalman Filter, is used to provide error correction to the INS. In addition, the INS provides prepositioning data for reacquiring the GPS signal should the signal be lost.

WIDE AREA AUGMENTATION SYSTEM (WAAS)

Unlike traditional ground-based navigation aids, the WAAS covers nearly all of the airspace systems. The WAAS provides augmentation information to GPS receivers to enhance the accuracy and reliability of position estimates. The signals from GPS satellites are received at many widely spaced Wide Area Reference Stations (WRS) sites. The WRS locations are precisely surveyed so that any errors in the received GPS signals can be detected. The GPS information collected by the WRS sites is forwarded to the WAAS Master Station (WMS) via a terrestrial communications network. At the WMS, the WAAS augmentation messages are generated. These messages contain information that allows GPS receivers to remove errors in the GPS signal, allowing for a significant increase in location accuracy and reliability. The augmentation messages are sent from the WMS to uplink stations to be transmitted to navigation payloads on geo-stationary communications satellites. (***Figure 4-99***)

The geo-satellite navigation transponders broadcast the augmentation messages on a GPS-like signal. The aircraft's GPS/WAAS receiver processes the WAAS augmentation message as part of estimating position. The GPS-like signal from the satellite navigation transponder can also be used by the receiver as an additional source for calculation of the user's position. WAAS also provides indications to GPS/ WAAS receivers of where the GPS system is unusable due to system errors or other effects. Further, the WAAS system was designed to the strictest of safety standards hazardously misleading information that would cause an error in the GPS position estimate.

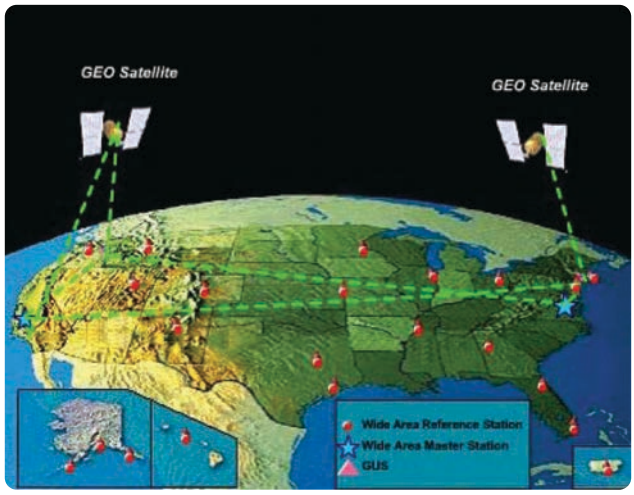


Figure 4-99. The WMS generates augmentation messages for GPS error correction.

GLOBAL NAVIGATION SATELLITE SYSTEMS (GNSS)

In 2001, the International Civil Aviation Organization (ICAO) approved an international standard for a landing system based on local correction of GNSS data to a level that would support instrument approaches. The ICAO Standards and Recommended Practices (SARPS) define the characteristics of a Ground-Based Augmentation System (GBAS) service that can be provided by an airport authority or an Air Traffic Service provider. The GBAS service provides the radiated signal in space that can be used by suitably equipped airplanes as the basis of a GNSS Landing System (GLS). The initial SARPS support an approach service. Future refinements should lead to full low-visibility service (i.e., takeoff, approach, and landing) and low-visibility taxi operations.

The GLS consists of three major elements—a global satellite constellation that supports worldwide navigation position fixing, a GBAS facility at each equipped airport that provides local navigation satellite correction signals, and avionics in each airplane that process and provide guidance and control based on the satellite and GBAS signals (***Figure 4-100***). The GLS uses a navigation satellite constellation which include the U.S. Global Positioning System (GPS), and the European Galileo System for the basic positioning service.

The GPS constellation already is in place and improvements are planned over the coming decades. Galileo is the global navigation satellite system (GNSS) that is currently being created by the European Union (EU) through the European Space Agency (ESA) and

the European GNSS Agency (GSA) headquartered in Prague in the Czech Republic, with two ground operations centers. One of the aims of Galileo is to provide an indigenous alternative high-precision positioning system upon which European nations can rely, independently from the Russian GLONASS and US GPS systems. Galileo is intended to provide horizontal and vertical position measurements within 1-metre precision, and better positioning services at high latitudes than other positioning systems. The complete 30-satellite Galileo system (24 operational and 6 active spares) is expected by 2020.

GNSS Operational Elements

The basic positioning service is augmented locally—at or near the airport—through a GBAS radio transmitter facility. Because the ground facility is located at a known surveyed point, the GBAS can estimate the errors contained in the basic positioning data. Reference receivers in the GBAS compare the basic positioning data with the known position of the facility and compute corrections on a satellite-by-satellite basis. The corrections are called pseudo-range corrections because the primary parameter of interest is the distance between the GBAS facility and individual satellites.

The satellite constellation is continuously in motion, and satellites ascend and descend over the horizon when observed from any point on Earth. The GBAS calculates corrections for all the satellites that meet the specified in-view criteria and transmits that information to the nearby airplanes over a VHF Data Broadcast (VDB) data link. Boeing airplanes that are currently being produced contain Multi-Mode Receivers (MMR) that support Instrument Landing System (ILS) and basic GPS operations. These MMRs can be modified to support GLS and potentially Microwave Landing System operations. The GLS capability is supported through the addition of a receiver and processing in the MMRs of the GBAS data provided through the VDB data link. The MMRs apply the local correction data received from the GBAS to each satellite that

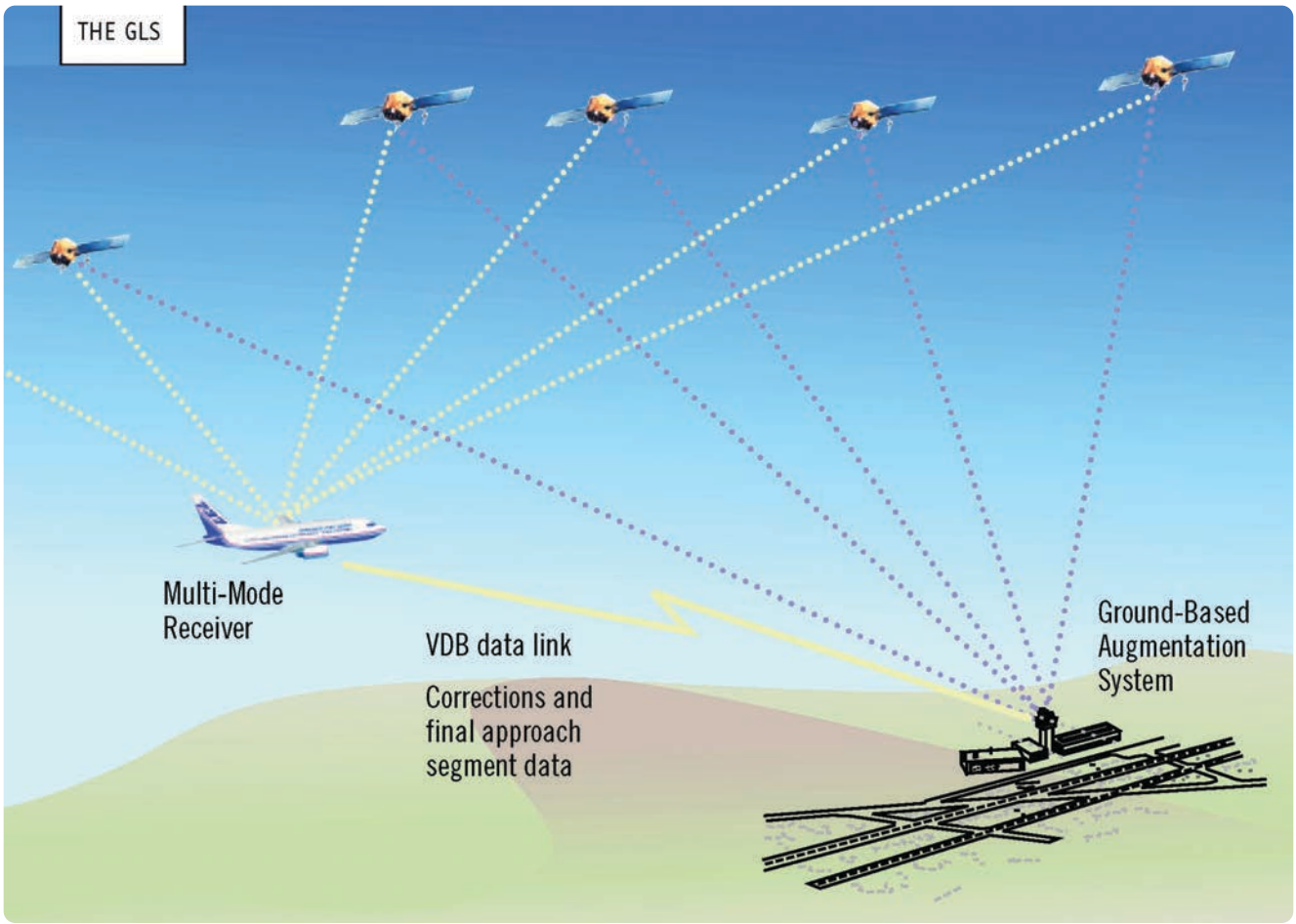


Figure 4-100. The GNSS Landing System (GLS).

the airplane and GBAS share in common. Because of position and altitude differences and local terrain effects, the GBAS and the airplane may not necessarily be observing the same combination of satellites.

The airplane systems only use satellite information that is supported by correction data received from the GBAS. When the airplane is relatively close to the GBAS station, the corrections are most effective, and the MMRs can compute a very accurate position. Typical lateral accuracy should be ≤1 m.

A single GBAS ground station typically provides approach and landing service to all runways at the airport where it is installed. The GBAS may even provide limited approach service to nearby airports. Each runway approach direction requires the definition of a final approach segment (FAS) to establish the desired reference path for an approach, landing, and rollout. The FAS data for each approach are determined by the GBAS service provider and typically are verified after installation of the GBAS ground station.

One feature that differentiates the GLS from a traditional landing system such as the ILS is the potential for multiple final approach paths, glideslope angles, and missed approach paths for a given runway. Each approach is given a unique identifier for a particular FAS, glideslope, and missed approach combination. FAS data for all approaches supported by the particular GBAS facility are transmitted to the airplane through the same high-integrity data link as the satellite range correction data (i.e., through the VDB data link).

The MMRs process the pseudo-range correction and FAS data to produce an ILS-like deviation indication from the final approach path. These deviations are then displayed on the pilot's flight instruments (e.g., Primary Flight Display (PFD)) and are used by airplane systems such as the flight guidance system (e.g., autopilot and flight director) for landing guidance. The ILS-like implementation of the GLS was selected to support common flight deck and airplane systems integration for both safety and economic reasons. This implementation helps provide an optimal pilot and system interface while introducing the GLS at a reasonable cost.

The use of operational procedures similar to those established for ILS approach and landing systems minimizes crew training, facilitates the use of familiar instrument and flight deck procedures, simplifies flight crew operations planning, and ensures consistent use of flight deck displays and annunciations. For example, the source of guidance information (shown on the PFD in ***Figure 4-101***) is the GLS rather than the ILS. The scaling of the path deviation information on the pilot's displays for a GLS approach can be equivalent to that currently provided for an ILS approach. Hence, the pilot can monitor a GLS approach by using a display that is equivalent to that used during an ILS approach.

DATA LINK

A data link is the means of connecting one location to another for the purpose of transmitting and receiving digital information. It can also refer to a set of electronic devices consisting of a transmitter and a receiver and an interconnecting circuit.

There are three types of data-link configurations:

- Simplex communications means all communications are in one direction only.
- Half-duplex communications means that communications may occur in both directions, but only one direction at a time.
- Duplex communication means that communications may occur in both directions at the same time.



Figure 4-101. GLS approach indications on the Primary Flight Display (PFD).

The International Telecommunication Union defines simplex as a communication channel that operates in one direction at a time. If that direction is reversible, the system becomes known as half duplex. A full duplex channel requires two simplex channels operating simultaneously in opposite directions.

CONTROLLER PILOT DATA LINK COMMUNICATIONS (CPDLC)

In civil aviation a data link system known as Controller Pilot Data Link Communications (CPDLC) is a two-way data-link system by which controllers can transmit messages to an aircraft as an alternative to voice communications. The short text message is displayed on a flight deck visual display.

Air traffic controllers are provided with the capability to issue ATC instructions such as altitude assignments, course changes, speed assignments, radio frequency assignments, and various other requests for information. ***Figure 4-102*** shows a typical CPDLC message display advising the pilot that a revised route change has been approved and to maintain an altitude of 3 000 feet. The pilots are provided with the capability to respond to messages, to request or receive clearances and information, and to report information. A "free text" capability is also provided to exchange information not conforming to standard defined formats.

CPDLC is most commonly used when the aircraft is too far from the traffic control to make voice communication possible, such as during crossing oceans. One such Canadian system, uses a five-digit sequence number confirmed between air traffic control and the flight crew before the aircraft proceeds to cross the ocean. This system uses the aircraft's flight management computer (FMC) to send location, speed and altitude information about the aircraft to the ATC. ATC can then send messages to the aircraft regarding any necessary change of course.

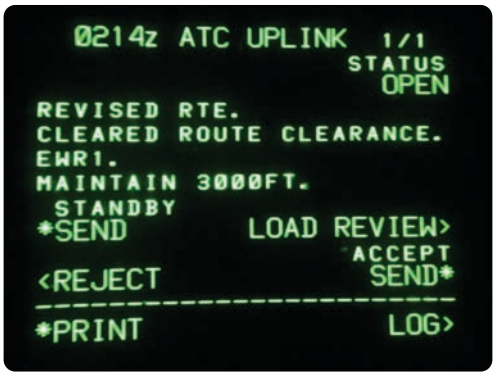


Figure 4-102. What a pilot sees on a cockpit display when using CPDLC datalink.

SECTION B

AIR TRAFFIC CONTROL TRANSPONDER

A radar beacon transponder, or just 'transponder', provides Air Traffic Control (ATC) with identification and location of an aircraft on the radar screens. For each aircraft equipped with an altitude encoder, the transponder also provides the pressure altitude of the aircraft to be displayed adjacent to the on-screen blip that represents the aircraft. (***Figure 4-103***)

A pilot verbally is instructed by ATC to select one of 4 096 discrete octal codes on the transponder. The ground station then transmits a pulse of energy at 1 030 MHz and the transponder transmits a reply with the assigned code at 1 090 MHz. This confirms the aircraft's location by altering its target symbol on the radar screen. As the screen may be filled with many aircraft, ATC can also ask the pilot to identify. By pressing the IDENT button on the transponder, it transmits in such a way that the aircraft's target symbol is highlighted on the controllers screen to be distinguishable.

To gain altitude clarification, the transponder control must be placed in the ALT or Mode C position. The signal transmitted back to ATC is then modified with a code that places the pressure altitude of the aircraft next to its target symbol on the radar screen. The transponder gets the pressure altitude off the aircraft from an altitude encoder that is electrically connected to the transponder.

PRIMARY SURVEILLANCE RADAR (PSR)

Two types of radar are used by ATC. The primary radar (PSR) transmits directional UHF or SHF radio waves sequentially in all directions. When the radio waves encounter an aircraft, part of those waves reflect back to a ground antenna. Calculations are made in a receiver to determine the direction and distance of the aircraft from the transmitter. A blip representing the aircraft direction and distance from the tower is displayed on a radar screen which is known as a plan position indicator. (***Figure 4-104***)



Figure 4-103. A traditional transponder control head (A), a lightweight digital transponder (B), and a remote altitude encoder (C) that connects to a transponder to provide ATC with an aircraft's altitude displayed on a PPI radar screen next to the target that represents the aircraft.

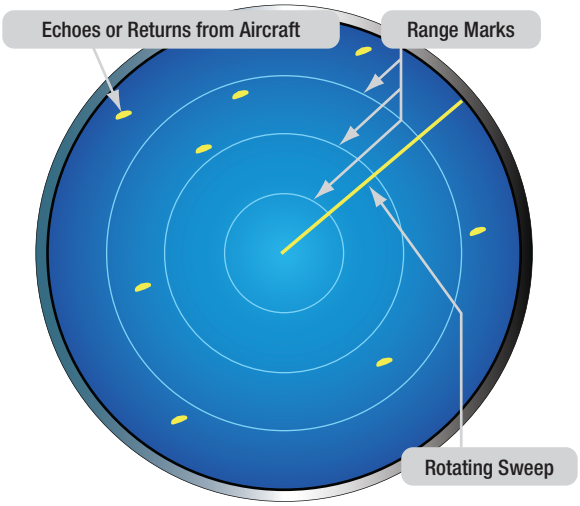


Figure 4-104. A plan position indicator (PPI) for ATC primary radar locates target aircraft on a scaled field.

SECONDARY SURVEILLANCE RADAR (SSR)

A secondary surveillance radar (SSR) adds the third dimension of altitude. SSR radar transmits two sets of pulses. The interrogation modes transmitted are determined by the pulse spacing. The two modes are Mode A, spaced 8 microseconds apart for basic identity and Mode C, spaced 21 microseconds apart for requesting digitally encoded altitude information. The pulses in both modes are identical except for their spacing.

Mode A and C Interrogation and Reply

A ground station verifies that the reply signal it receives is in response to that interrogation by a third pulse which prevents the transponder from replying to a side lobe transmission of the SSR. The main two pulses (P1 and P3) are transmitted by a rotating directional antenna and the third pulse (P2) by an omnidirectional antenna sent two microseconds after the P1 pulse. The transponder detects the amplitude of the P2 relative to the P1.

An Identification Pulse (IP) is transmitted upon the interrogator's request. The IP pulse is activated by the pilot pushing the IDENT switch and is then transmitted for a period of 18 seconds. If the interrogation of the transponder occurred in Mode C, the encoded altitude is also replied to ATC.

MODE S

The Mode S transponder adds a data link feature to improve the flow of data between aircraft and ATC. Mode S capabilities include bi-directional air-air information exchange, as well as a ground-to-air bi-directional uplink. Mode S may also function as an airborne separation system when interfaced with a TCAS collision avoidance system.

With Mode S, each aircraft is pre-assigned a unique identity code that displays along with its pressure altitude on ATC radar when the transponder responds to an SSR interrogation. Since no other aircraft respond with this code, the chance of two pilots selecting the same response code on the transponder is eliminated. (***Figure 4-105***)

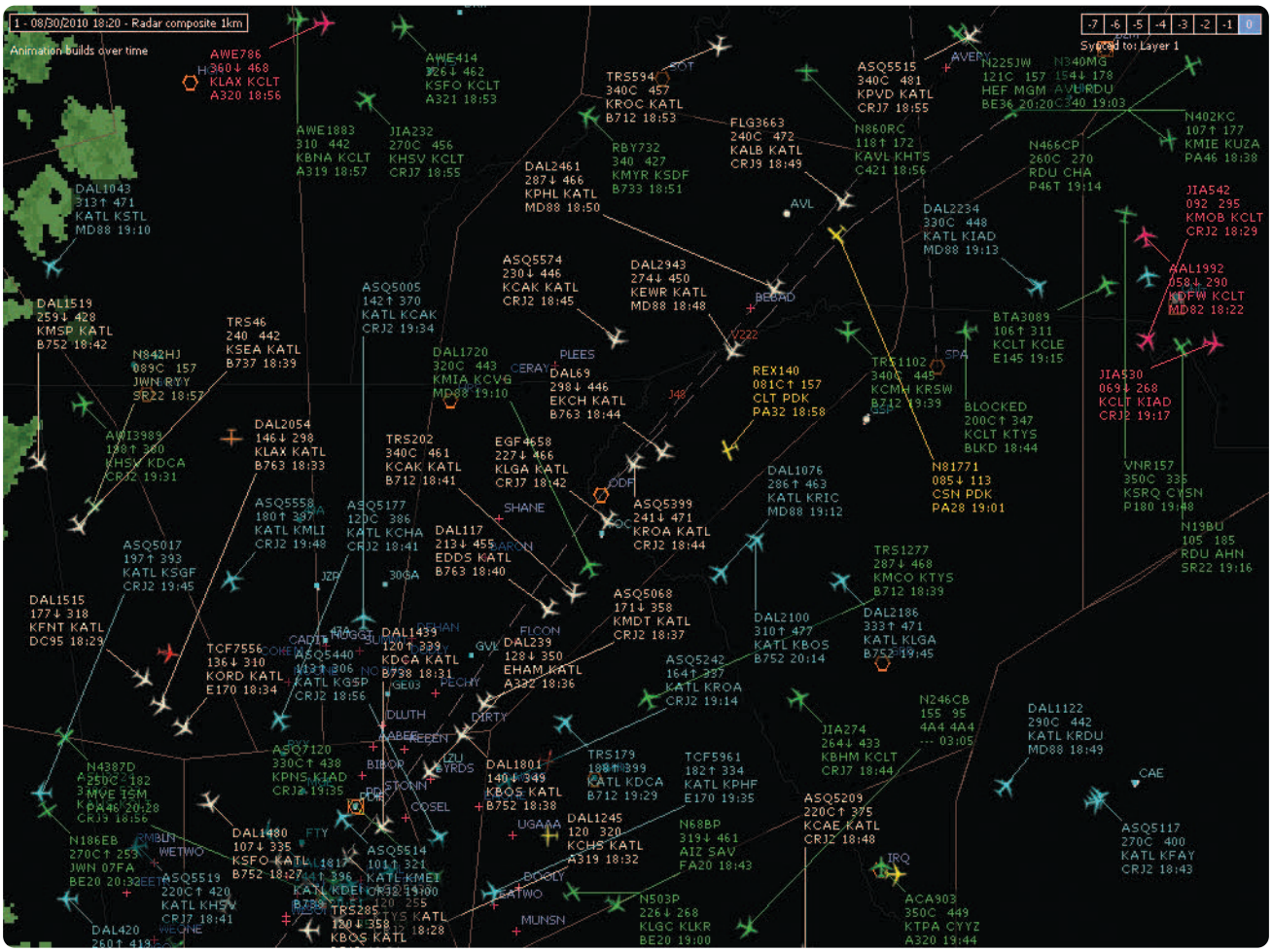


Figure 4-105. Air traffic control radar technology and an onboard radar beacon transponder work together to convey and display air traffic information on a PPI radar screen. A modern approach ATC PPI is shown. Targets representing aircraft are shown as little aircraft on the screen. The nose of the aircraft indicates the direction of travel. Most targets shown above are airliners. The data block for each target includes the following information either transmitted by the transponder or matched and loaded from flight plans by a flight data processor computer: call sign, altitude/speed, origination/destination, and aircraft type/ETA (ZULU time). A "C" after the altitude indicates the information came from a Mode C equipped transponder. The absence of a C indicates Mode S is in use. An arrow up indicates the aircraft is climbing. An arrow down indicates a descent. White targets are arrivals, light blue targets are departures, all other colors are for arrivals and departures to different airports in the area.

Mode S is sometimes referred to as 'mode select'. It is a data packet protocol that is also used in onboard collision avoidance systems. When used by ATC, Mode S interrogates one aircraft at a time. Controller workload is reduced by not having to respond to all interrogations in an airspace. Additionally, location information is more accurate with Mode S. A single reply in which the phase of the transponder reply is used to calculate position, called monopulse, and is sufficient to locate the aircraft.

TRANSPONDER TESTS AND INSPECTIONS

Because of the danger involved should a transponder malfunction, for example report the wrong altitude information, the functional condition of all transponders is of great concern to aviators. Transponder testing is typically required every 24 months by an approved facility using appropriate (and calibrated) ramp testing equipment in accordance with applicable maintenance manual procedures.

For Mode C equipped aircraft, testing for correct functionality includes:

1. Correct operation of the Mode A code
2. Altitude reporting including the check of the altitude sensor from ground to the ceiling of the aircraft.

For Mode S equipped aircraft, testing for correct functionality includes:

1. The Mode S 24-Bit aircraft address.
2. Altitude reporting including a check of the altitude sensor from ground to the ceiling of the aircraft.
3. Mode S Elementary Surveillance & Downlink Parameters including:
   - Aircraft Identification
   - Capability Report
   - Pressure Altitude
   - Flight Status
4. Mode S Enhanced Surveillance & Downlink Aircraft Parameters including:
   - Magnetic Heading
   - Indicated Airspeed
   - Mach Number
   - Vertical Rate
   - Roll Angle
   - Track Angle Rate Or True Airspeed
   - True Track Angle
   - Ground Speed
   - Selected Altitude (and barometric pressure setting)

Guidelines For All Transponder Testing Includes:

- Ensure all transponders are selected to 'OFF' or 'Standby'.
- Before starting any test, contact the local Air Traffic Control Unit and advise them of your intention to conduct transponder testing, your start time and test duration, the altitude(s) at which you will be testing, your intended Aircraft Identification, and your intended Mode A code.
- Set the Mode A code to 7776 (or other Mode A code agreed with ATC).
- For Mode S aircraft, set the Aircraft Identification with the first 8 characters of the name of the company conducting the tests.

For Mode S aircraft, set on-the-ground status for all Mode S replies, except when an airborne reply is required for altitude testing).

- Where possible, perform the testing inside a hanger to take advantage of any shielding properties it may provide.
- As a precaution, use antenna transmission covers.
- When testing the altitude (Mode C or S) parameter, radiate directly into the ramp test set via the prescribed attenuator.
- In between testing from one altitude to another, select the transponder to 'standby' mode.
- If testing transponder parameters other than 'altitude', set altitude to 1 000 feet or over 60 000 feet. This will minimize the possibility of warnings to overflying aircraft.
- When testing is complete select the transponder(s) to 'OFF' or 'Standby'.

When selecting a testing code during a transponder test, avoid setting or passing through the following special purpose codes:

0000 – For Military Use Only
1200 – For VFR Traffic Use Only
7700 – In-Flight Emergency
7600 – Voice Communication Failure
7500 – Hijacking Emergency

ALTITUDE ENCODERS

Altitude encoders convert the aircraft's pressure altitude into a code sent by the transponder to ATC. Increments of 100 feet are usually reported. Encoders have varied over the years.

Some are built into the altimeter used in the panel and connected by wires to the transponder. Others are mounted out of sight on an avionics rack or other out of the way place. These are known as blind encoders. On transport aircraft, the altitude encoder may be a large black box with a static line connection to an internal aneroid. General aviation encoders are smaller and more lightweight, but still often feature an internal aneroid and static line connection. Some encoders use microtransistors and are completely solid state including the pressure sensing device from which the altitude is derived. No static port connection is required. Data exchanges with GPS and other systems is becoming common.

When a transponder selector is set on ALT, the digital pulse message sent in response to the secondary surveillance radar interrogation becomes the digital representation of the pressure altitude of the aircraft. There are 1 280 altitude codes, one for each 100 feet of altitude between 1 200 feet mean sea level (MSL) and 126 700 feet MSL. Each altitude increment is assigned a code. While these would be 1 280 of the same codes used for location and IDENT, the Mode C

(or S) interrogation deactivates the 4 096 location codes and causes the encoder to become active. The correct altitude code is sent to the transponder that replies to the interrogation. The SSR receiver recognized this as a response to a Mode C (or S) interrogation and interprets the code as altitude code.

ADS-B

Collision avoidance is a significant part of the NextGen plan for transforming airspace systems. Increasing the number of aircraft using the same quantity of airspace and ground facilities requires the implementation of new technologies to maintain a high level of performance and safety. The successful proliferation of global navigation satellite systems (GNSS), such as GPS, has led to the development of a collision avoidance system known as automatic dependent surveillance broadcast (ADS-B). ADS-B is an integral part of NextGen program. The implementation of its ground and airborne infrastructure is currently underway. ADS-B is active in parts of the United States and around the world. (***Figure 4-106***)



Figure 4-106. Low power requirements allow remote ADS-B stations with only solar or propane support. This is not possible with ground radar due to high power demands which inhibit remote area radar coverage for air traffic purposes.

ADS-B is considered in two segments: ADS-B OUT and ADS-B IN. ADS-B OUT combines the positioning information available from a GPS receiver with on-board flight status information, i.e. location including altitude, velocity, and time. It then broadcasts this information to other ADS-B equipped aircraft and ground stations. (***Figure 4-107***)

Two different frequencies are used to carry these broadcasts with data link capability. The first is an expanded use of the 1 090 MHz Mode-S transponder protocol known as 1 090 ES. The second, largely being introduced as a new broadband solution for general aviation implementation of ADS-B, is at 978 MHz. A 978 universal access transceiver (UAT) is used

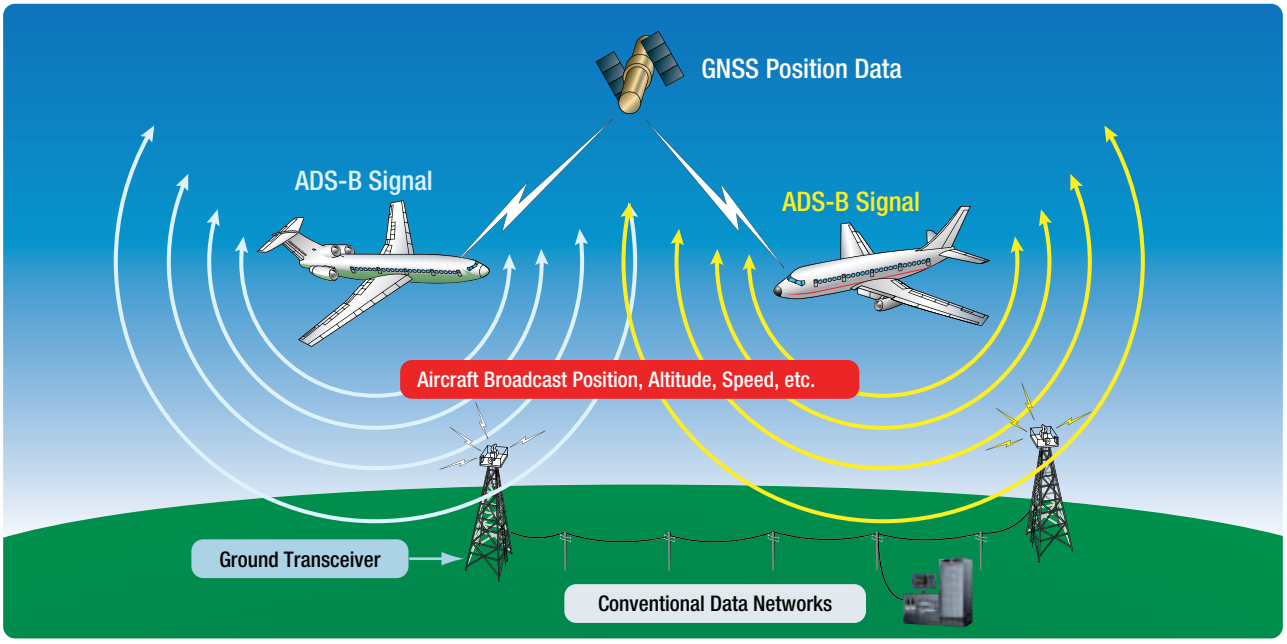


Figure 4-107. ADS-B OUT uses satellites to identify the position aircraft. This position is then broadcast to other aircraft and to ground stations along with other flight status information.



Figure 4-108. A cockpit display of ADS-B generated targets (left) and an ADS-B airborne receiver with antenna (right).

to accomplish this. An omni-directional antenna is required in addition to the GPS antenna and receiver. Airborne receivers of an ADS-B broadcast use the information to plot the location and movement of the transmitting aircraft on a flight deck display similar to TCAS. (***Figure 4-108***)

Inexpensive ground stations (compared to radar) are constructed in remote and obstructed areas to proliferate ADS-B. Ground stations share information from airborne ADS-B broadcasts with other ground stations that are part of the air traffic management system (ATMS). Data is transferred with no need for human acknowledgement. Microwave and satellite transmissions are used to link the network.

For traffic separation and control, ADS-B has several advantages over conventional ground-based radar. The first is the entire airspace can be covered with a much lower expense. The aging ATC radar system that is in place is expensive to maintain and replace. Additionally, ADS-B provides more accurate information since the vector state is generated from the aircraft with the help of GPS satellites. Weather is a greatly reduced factor with ADS-B. Ultra high frequency GPS transmissions are not affected. Increased positioning accuracy allows for higher density traffic flow and landing approaches, an obvious requirement to operate more aircraft in and out of the same number of facilities.

The higher degree of control available also enables routing for fewer weather delays and optimal fuel burn rates. Collision avoidance is expanded to include runway incursion from other aircraft and support vehicles on the surface of an airport. ADS-B IN offers features not available in TCAS. Equipped aircraft are able to receive abundant data to enhance situational awareness. Traffic information services-broadcast (TIS-B) supply traffic information from non-ADS-B aircraft and ADS-B aircraft on a different frequency. Ground radar monitoring of surface targets, and any traffic data in the linked network of ground stations is sent via ADS-B IN to the flight deck. This provides a more complete picture than air-to-air only collision avoidance. Flight information services-broadcast (FIS-B) are also received by ADS-B IN. Weather text and graphics, ATIS information, and NOTAMS (notices to airmen) are able to be received in aircraft that have 987 UAT capability. (***Figure 4-109***)

ADS-B test units are available for trained maintenance personnel to verify proper operation of ADS-B equipment. This is critical since close tolerance of air traffic separation depends on accurate data from each aircraft and throughout all components of the ADS-B system. (***Figure 4-110***)

SECTION C

MICROWAVE LANDING SYSTEM (MLS)

The Microwave Landing System (MLS) was designed to replace ILS with an advanced precision approach system that would overcome the disadvantages of ILS and also provide greater flexibility to its users. However, there are few MLS installations in use at present and they are likely to co-exist with ILS for a long time. MLS is a precision approach and landing system that

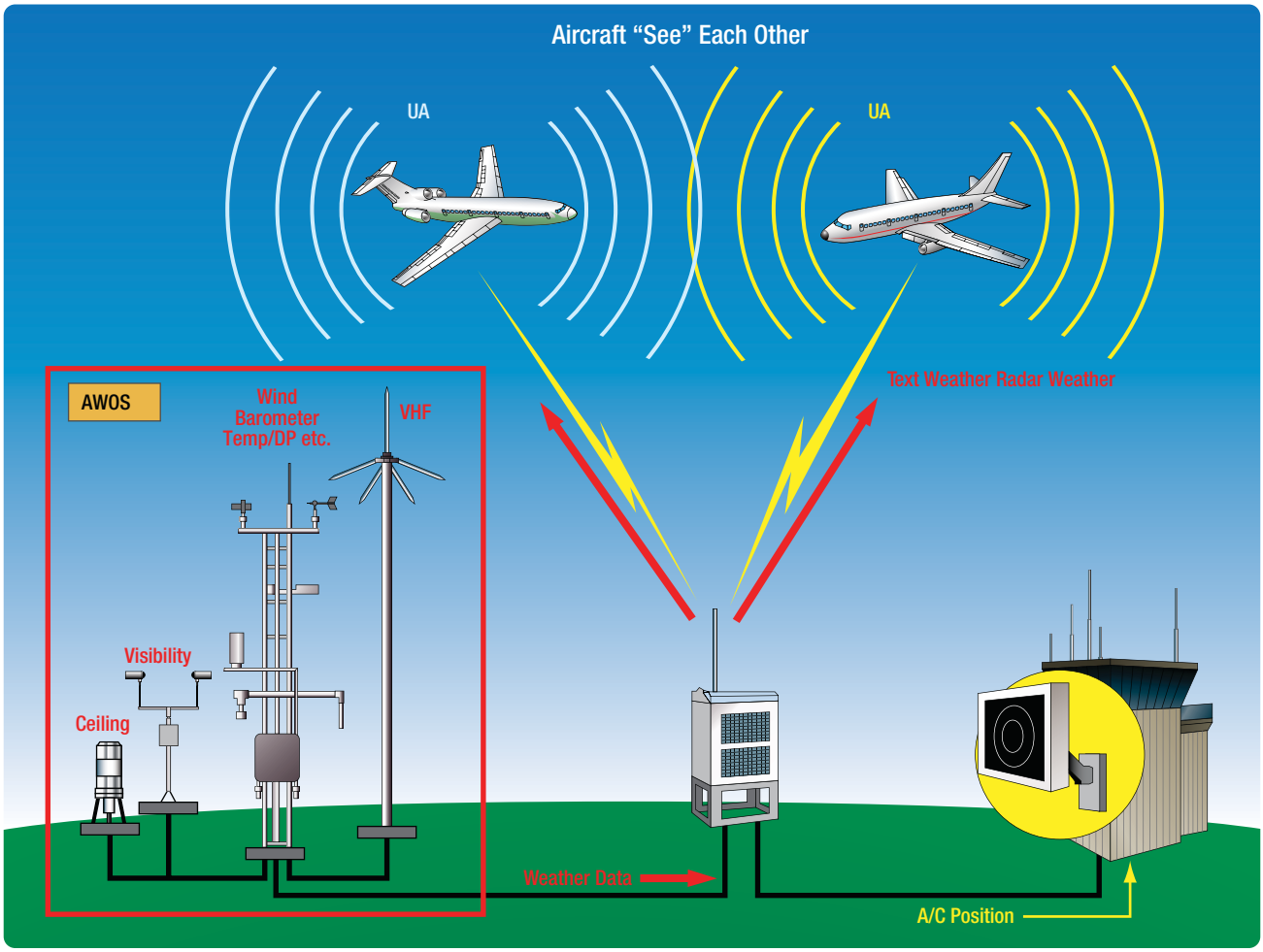


Figure 4-109. ADS-B IN enables weather and traffic information to be sent into the flight deck. In addition to AWOS weather, NWS can also be transmitted.



Figure 4-110. An ADS-B test unit.

provides position information and various ground to air data. The position information is provided in a wide coverage sector and is determined by an azimuth angle measurement, an elevation measurement and a range measurement.

The ILS system inherently has certain disadvantages, which include the limitation of availability of 40 channels worldwide, the azimuth and glideslope beams are fixed and narrow. As a result, aircraft have to be sequenced and adequately separated which causes landing delays. There are no special procedures available for slower aircraft, helicopters, and Short Take Off and Landing (STOL) aircraft. The ILS system cannot be installed in hilly areas and it requires large expanses of flat, cleared land to minimize interference with the localizer and glideslope beams. Vehicles, taxiing aircraft, low-flying aircraft and buildings have to be kept well away from the transmission sites to minimize localizer and glideslope course deviations (bending of the beams).

The Microwave Landing System (MLS) on the other hand offers certain significant advantages such as 200 channels availability worldwide. The azimuth coverage is at least ± 40° off the runway on-course line (QDM) and glideslopes from .9° to 20° can be selected. The MLS Azimuth and Elevation systems are shown in ***Figure 4-111 and Figure 4-112*** respectively. The usable range is 20-30 nm from the MLS site. There is no problem with back-course transmissions; a secondary



Figure 4-111. An MLS azimuth guidance station with rectangular azimuth scanning antenna with DME antenna at left.

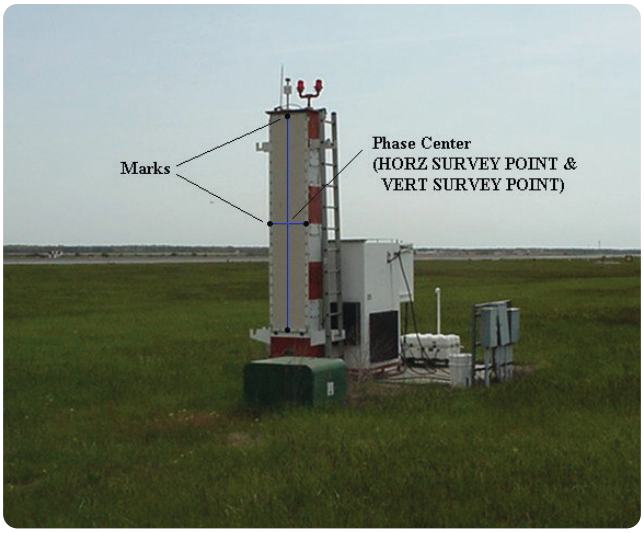


Figure 4-112. An MLS elevation guidance station.

system is provided to give overshoot and departure guidance ± 20° of runway direction up to 15° in elevation to a range of 10 nm and a height of 10 000 ft. It operates in the SHF band, 5 031 - 5 090 MHZ. This enables it to be sited in hilly areas without having to level the site. Course deviation errors (bending) of the localizer and glide path caused by aircraft, vehicles and buildings are no longer a problem because the MLS scanning beam can be interrupted and therefore avoids the reflections.

Because of its increased azimuth and elevation coverage aircraft can choose their own approaches. This will increase runway utilization and be beneficial to helicopters and STOL aircraft. The MLS has a built-in DME (***Figure 4-111***). The MLS is also compatible with conventional localizer and glidepath instruments, EFIS, auto-pilot systems and area navigation equipment. The MLS is capable of giving positive automatic landing indications plus definite and continuous ON/OFF flag indications for the localizer and glideslope needles. The identification prefix for the MLS is an 'M' followed by two letters. The aim is for all MLS equipped aircraft to operate to CAT III criteria. ***Figure 4-113*** and ***Figure 4-114*** and ***Figure 4-115*** below show some of these features.

OPERATIONAL FUNCTIONS

The Microwave Landing System (MLS) employs the principle of Time Division Multiplexing (TDM) whereby only one frequency is used on a channel but the transmissions from the various angle and data ground equipment are synchronized to assure interference free operations on the common radio frequency.

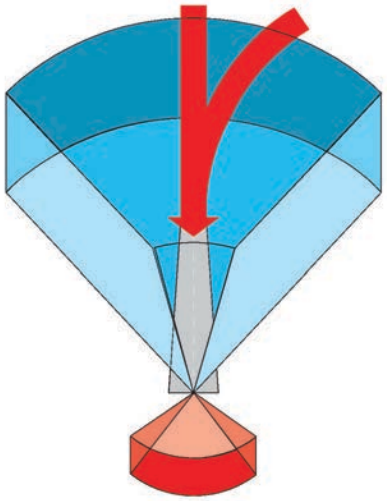


Figure 4-113. MLS Coverage Volumes 3-D Representation.

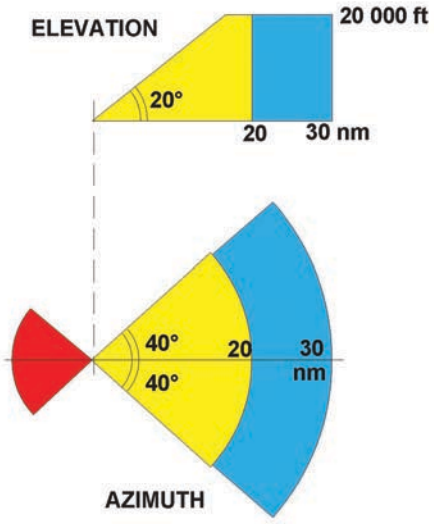


Figure 4-114. MLS Coverage Volumes of the Elevation station.

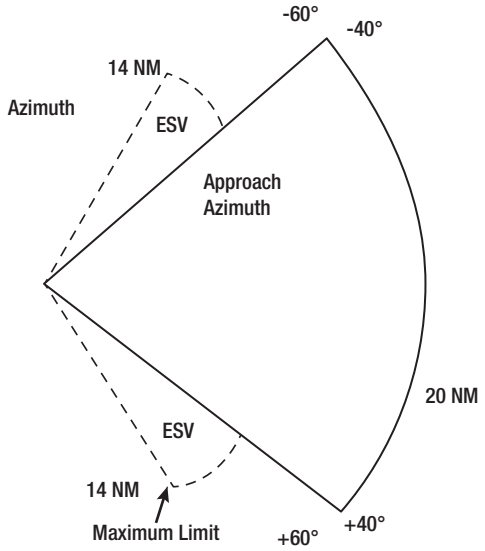


Figure 4-115. MLS Coverage Volumes of the Azimuth station.

AZIMUTH GUIDANCE

A technique called time referenced scanning beam (TRSB) is effectively utilized in azimuth and elevation guidance. The aircraft computes its azimuth position in relation to the runway center-line by measuring the time interval in microseconds between the reception of the 'TO' and 'FRO' scanning beams. The beam starts the 'TO' sweep at one extremity of its total scan and travels at a uniform speed to the other extremity. It then starts its 'FRO' scan back to its start position. The time interval between the reception of the 'to' and 'fro' pulses is proportional to the angular position of the aircraft in relation to the runway on-course line. The pilot can choose to fly the runway on-course line (QDM) or an approach path which he selects as a pre-determined number of degrees ± the runway direction. (***Figure 4-116***). The MLS also incorporates a back azimuth feature, which provides overshoot and departure guidance ± 20° of runway direction up to 15° in elevation. A DME system is mostly coupled with the MLS. The DME range along the MLS course is provided not by markers but by the associated DME system. For CAT II and III approaches, a precision DME (DME/P) that is accurate to within 100 feet must be available.

GLIDESLOPE GUIDANCE

For glideslope guidance, another beam scans up and down at a uniform speed within its elevation limits. The aircraft's position in relation to its selected glideslope angle is thus calculated in the same manner by measuring the time difference between the reception of the pulses from the up and down sweep. The transmissions from the two beams and the transmissions from the other components of the MLS system are transmitted at different intervals i.e. it uses "time multiplexing".

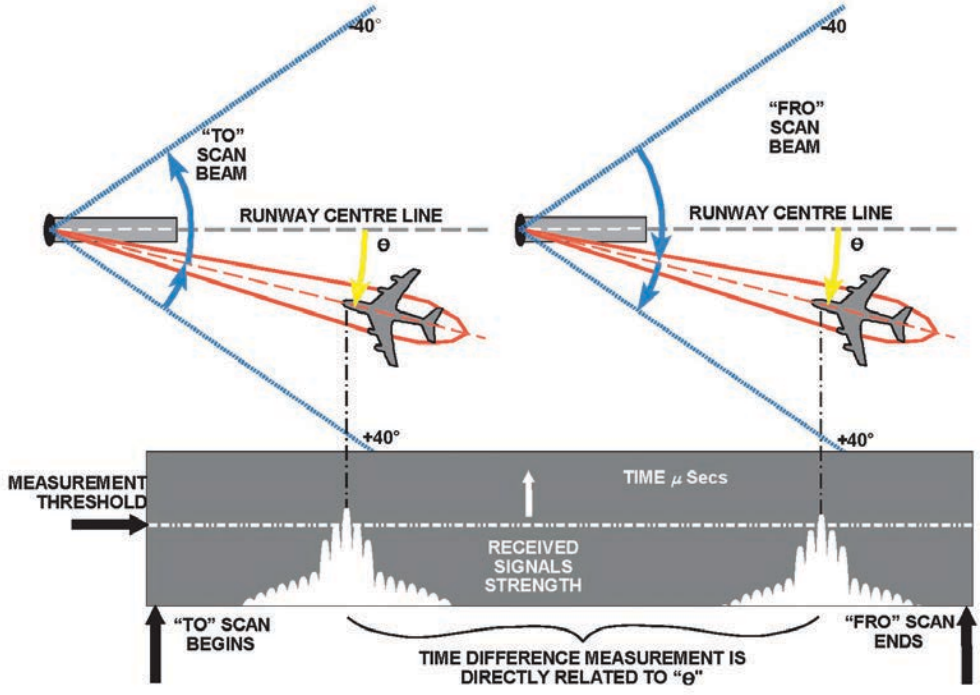


Figure 4-116. MLS approach path configured by the pilot on a pre-determined number of degrees ± of the runway direction.

AIRBORNE EQUIPMENT

The MLS airborne equipment is designed to continuously display the position of the aircraft in relation to the preselected course and glide path along with distance information during approach as well as during departure. The display consists of two cross bars similar to an ILS display except that the indications are given relative to the selective course. It is possible to program the computer to give segmented approaches and curved approaches for which a DME-P must be installed on the ground. (***Figure 4-117***)

In order to receive ILS, MLS and GPS transmissions, aircraft are equipped with multi-mode receivers and a combined control unit for ease of use by the flight crew. An example of a control unit is shown at ***Figure 4-118***.

VERY LOW FREQUENCY AND HYPERBOLIC NAVIGATION (VLF/OMEGA)

Omega is a worldwide, internationally operated radio navigation system. It operates in the Very Low Frequency (VLF) band between 10 and 14 kHz. It provides an all-weather, medium-accuracy navigation service. OMEGA was originally developed by the United States Navy for military aviation users. It was approved for development in 1968 and promised a true worldwide oceanic coverage capability with only eight transmitters and the ability to achieve a four-mile accuracy when fixing a position. Each Omega station transmitted a very low frequency signal which consisted of a pattern of four tones unique to the station that was repeated every ten seconds. Because of this and radio navigation principles, an accurate fix of the receiver's position could be calculated. OMEGA employed hyperbolic radio navigation techniques and the chain operated in the VLF portion of the spectrum between 10 to 14 kHz. Over time, it evolved into a system used primarily by the civil community. By receiving signals from three stations, an Omega receiver could locate a position to within 4 nautical miles using the principle of phase comparison of signals. Omega stations used very extensive antennas in order to transmit their extremely low frequencies. Specifically, they used grounded or insulated guyed masts with umbrella antennas. Some Omega antennas were the tallest constructions on the continent where they stood or still stand. (***Figure 4-119***)

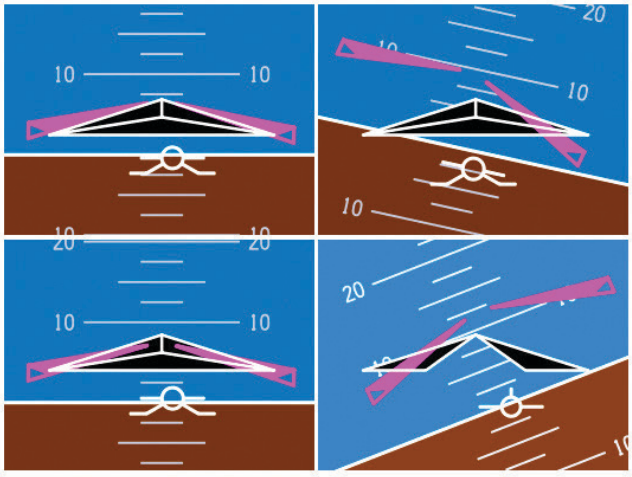


Figure 4-117. MLS crossbars for azimuth and elevation guidance.

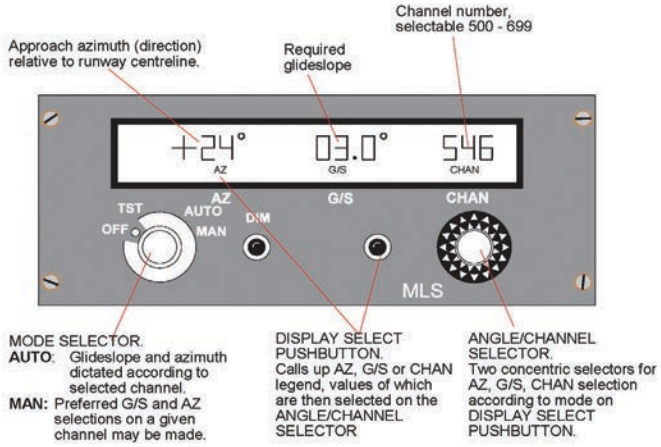


Figure 4-118. The MLS Control Unit.



Figure 4-119. A VLF/OMEGA Transmitter (Woodside Station G) in Woodside, Victoria (Australia).

When six of the eight station chain became operational in 1971, day to day operations were managed by the United States Coast Guard in partnership with Argentina, Norway, Liberia, and France. The Japanese and Australian stations became operational several years later. Coast Guard personnel operated two US stations: one in LaMoure, North Dakota and the other in Kaneohe, Hawaii on the island of Oahu. The station list is depicted in ***Figure 4-120***.

Due to the success of the Global Positioning System, the use of Omega declined during the 1990s, to a point where the cost of operating Omega could no longer be justified. Omega was permanently terminated on September 30, 1997 and all stations ceased operation. Some of the stations, such as the La Moure station, are now used for submarine communications.

Common Frequencies

10.2 kHz	11.05 kHz	11-1/3 kHz	13.6 kHz

Unique Frequencies

Station Location	Frequency (kHz)
A: Norway	12.1 kHz
B: Liberia	12.0 kHz
C: Hawaii	11.8 kHz
D: North Dakota	13.1 kHz
E: La Reunion	12.3 kHz
F: Argentina	12.9 kHz
G: Australia	13.0 kHz
H: Japan	12.8 kHz

Figure 4-120. OMEGA Stations Worldwide and Frequencies.

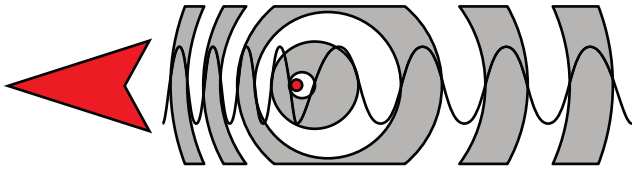


Figure 4-121. The Doppler Effect.

DOPPLER NAVIGATION

A Doppler Navigation System uses the Doppler principle (***Figure 4-121***) to measure an aircraft's ground speed and drift. The Doppler radar functions by continuous measurement of Doppler shift and converting the measured values to groundspeed and drift angle. In early systems the aircraft's departure point was loaded into a navigation computer, which then converted the aircraft's heading and Doppler ground speed/drift inputs into a continuous display of aircraft position; this was then displayed as latitude and longitude, and/or as distance to go along track and position left or right of track, in nautical miles.

A Doppler navigation system is a completely self contained and requires no ground based navigation aids. It is usable worldwide and most accurate overland. The system accuracy drops during flight over the sea because the surface winds, tides and currents move the surface in random directions. At times, the Doppler system also fails to measure a ground speed and drift during flight over a smooth, glassy sea. The latest improved Doppler Navigation Systems combine the inherent accuracy of Doppler ground speed and drift measurement with information from Decca, Inertial Reference Units, Loran C, Global Positioning Systems and VOR/DME, in various combinations to suit customer requirements. These navigational inputs also help to eradicate the errors of the original Doppler Navigation Systems, caused by inaccurate heading reference and degradation, or loss, of Doppler ground speed and drift when flying over large expanses of water. The Doppler principle is also utilized in other navigation systems, such as VOR and VDF, and some radar equipment.

DOPPLER RADAR

Doppler radar measures groundspeed and drift angle using the Doppler Effect. The Doppler Effect is the frequency shift induced into a signal when the transmitter or receiver, or both, are in motion in relation to each other. The amount of Doppler shift is directly proportional to the aircraft velocity (groundspeed) and can be expressed by the mathematical formula:

$$\text{Doppler Shift} = FV/C$$

Where:

F = frequency transmitted
V = velocity of aircraft (ground speed)
C = Speed of Light (Propagation Velocity of Radar Wave)

The Doppler shift is upward (Up-Doppler) on a beam transmitted forward of a moving aircraft and downward (Down-Doppler) on a beam transmitted aft. Two other factors must be considered when the Doppler shift is measured by a radar set in an aircraft. First, since both the transmitter and receiver are located in the aircraft,

two Doppler shifts are created. One when the wave is transmitted and the second as it is reflected back to the aircraft. A factor of two must now be added to the Doppler formula.

$$\text{Doppler Shift} = 2\,FV/C$$

The second factor that must be considered is the angle of antenna radiation. Maximum Doppler shift would be received if the antenna beam was transmitted dead ahead. If the beam was directed straight down at the earth, the Doppler shift would be minimum. Since in the aircraft the beam is directed to the ground at an angle between dead ahead (0 degrees) and straight down (90 degrees), we introduce a factor to compensate for this. The factor used is the cosine of the radiating angle (θ). The complete Doppler formula then becomes:

$$\text{Doppler Shift} = 2\,FV\cos\theta/C$$

The Doppler shifts incurred, above and below the transmitted frequency, are detected and converted into an audio signal within the Doppler system at the nominal frequency of 10.33 Hz per knot of groundspeed. The audio signal is used for groundspeed and drift angle computations. The Doppler system transmits one fan-shaped beam forward and one aft. Generally, these beams are angled below the aircraft, fore and aft at a 70-degree depression angle (***Figure 4-122***). This creates an RF energy pattern on the ground much like the illumination pattern of a flashlight beamed at· an angle onto a tabletop.

Two receiver beams are utilized simultaneously, one looking forward, the other looking aft. This ability of a radar set to look in two directions at once is called "Janus" operation. The fore and aft beams are offset from the centerline of the aircraft and are switched from side to side at a specified rate, for example at 3/4 second rate. During the first 3/4 second period, the beams cover the right forward and left aft segments of the transmitted patterns. In the following 3/4 second period, the beams cover the left forward and right aft segments. (***Figure 4-123***)

The tracker circuits measure ground speed by determining (frequency tracking) the Doppler shift. The difference between the Up-Doppler and Down-Doppler frequencies is detected to produce a Janus Doppler audio signal which is the sum of the two Doppler shifts and is representative of ground speed. In older analogue instrument systems, the Janus Doppler signal was used to drive a servo. The rotation of the servo shaft, was analogous to the ground speed, which in turn controlled a servo loop responsible for positioning the digital ground speed readout on the indicator.

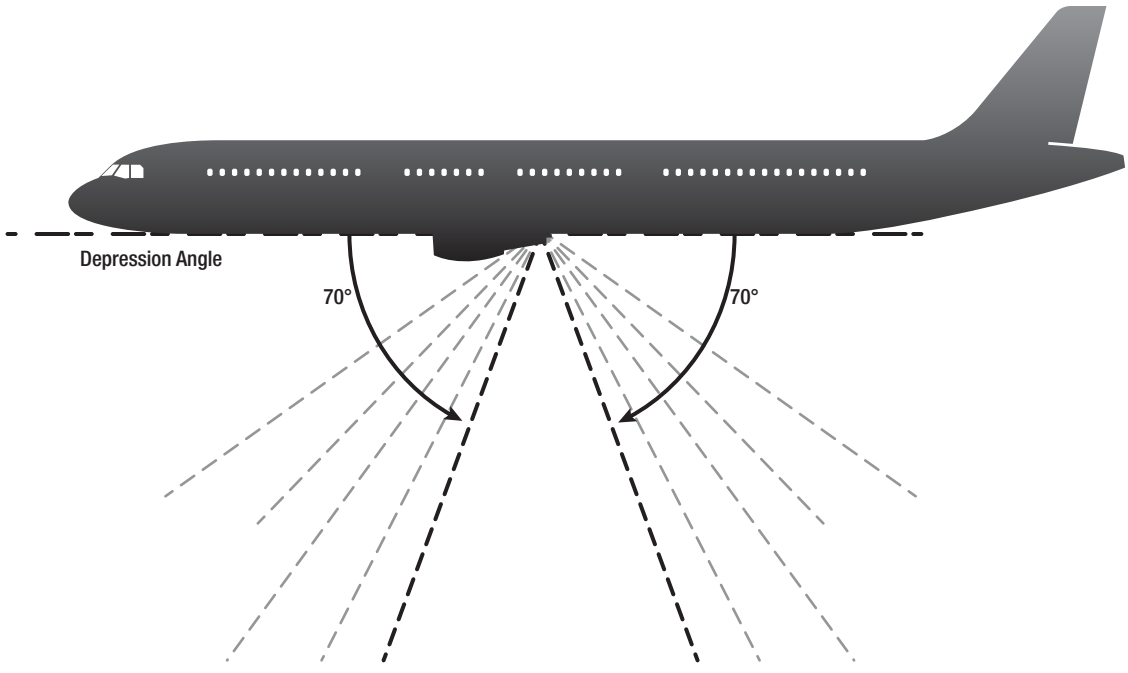


Figure 4-122. The Doppler Radar Transmitter Beam Pattern.

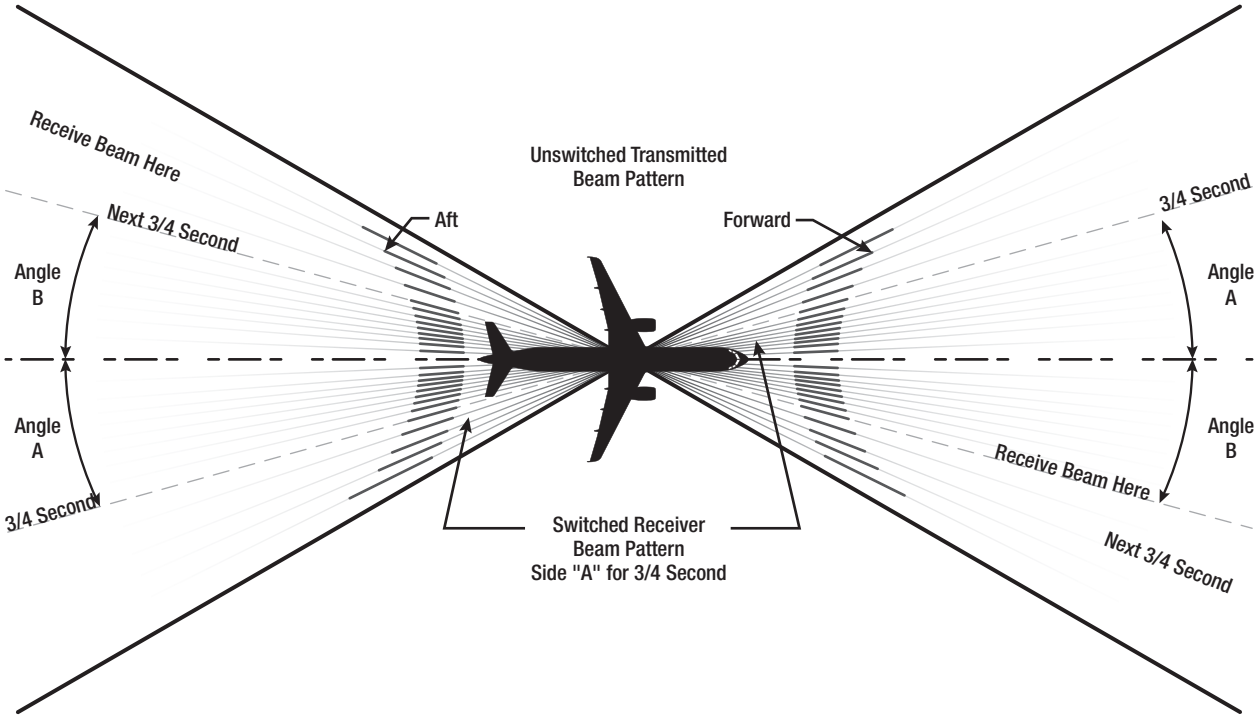


Figure 4-123. The Doppler Radar Transmitter Beam Pattern.

INERTIAL NAVIGATION SYSTEM (INS)

An inertial navigation system (INS) is used on some large aircraft for long range navigation. This may also be identified as an inertial reference system (IRS), although the IRS designation is generally reserved for more modern systems.

An INS/IRS is a self-contained system that does not require input radio signals from a ground navigation facility or transmitter. The system derives attitude, velocity, and direction information from measurement of the aircraft's accelerations given a known starting point. The location of the aircraft is continuously updated through calculations based on the forces experienced by INS accelerometers. A minimum of two accelerometers is used, one referenced to north, and the other referenced to east. In older units, they are mounted on a gyro-stabilized platform. This averts the introduction of errors that may result from acceleration due to gravity. An INS uses complex calculation made by an INS computer to convert applied forces into location information.

An interface control head is used to enter starting location position data while the aircraft is stationary on the ground. This is called initializing. (***Figure 4-124***)



Figure 4-124. An interface panel for three air data and inertial reference systems on an Airbus. The keyboard is used to initialize the system. Latitude and longitude position is displayed at the top.

From then on, all motion of the aircraft is sensed by the built-in accelerometers and run through the computer. Feedback and correction loops are used to correct for accumulated error as flight time progresses.

The amount an INS is off in one hour of flight time is a reference point for determining performance. Accumulated error of less than one mile after one hour of

flight is possible. Continuous accurate adjustment to the gyro-stabilized platform to keep it parallel to the Earth's surface is a key requirement to reduce accumulated error. A latitude/longitude coordinate system is used when giving the location output.

INS is integrated into an airliner's flight management system and automatic flight control system. Waypoints can be entered for a predetermined flight path and the INS will guide the aircraft to each waypoint in succession. Integration with other NAV aids is also possible to ensure continuous correction and improved accuracy but is not required. Modern INS systems are known as IRS. They are completely solid-state units with no moving parts.

Three-ring, laser gyros replace the mechanical gyros in the older INS platform systems. This eliminates precession and other mechanical gyro shortcomings. The use of three solid-state accelerometers, one for each plane of movement, also increases accuracy. The accelerometer and gyro output are input to the computer for continuous calculation of the aircraft's position.

The most modern IRS integrate is the satellite GPS. The GPS is extremely accurate in itself. When combined with IRS, it creates one of the most accurate navigation systems available. The GPS is used to initialize the IRS so the pilot no longer needs to do so. GPS also feeds data into the IRS computer to be used for error correction. Occasional service interruptions and altitude inaccuracies of the GPS system pose no problem for IRS/GPS. The IRS functions continuously and is completely self-contained within the IRS unit. Should the GPS falter, the IRS portion of the system continues without it. The latest electronic technology has reduced the size and weight of INS/IRS avionics units significantly

AIRCRAFT COMMUNICATION ADDRESSING AND REPORTING SYSTEM (ACARS)

Airlines pay their flight and cabin crews based on the times reported for out of the gate, off the ground, on the ground, and in the gate (OOOI). Prior to the introduction of data links to aircraft for sending text based messages, the flight crew would verbally report their OOOI times to the Dispatch radio operators using their VHF or HF communication transceivers.

ACARS was originally called ARINC Communication Addressing and Reporting System because it was developed by ARINC in 1978 as a digital data link for transmission of short text messages between aircraft and ground stations to relieve the crew from having to call in their times and to provide more accurate reporting. The ARINC 597 standard defines the operation of the ACARS management unit, which uses discrete (on/off) inputs from the landing gear doors, parking brake and weight-on-wheels sensors to automatically determine the OOOI times and send these as text messages. ACARS also has a medium shift keying modem that is used to transmit the OOOI reports over existing VHF or HF radios.

Figure 4-125 shows a typical ACARS display format prior to landing. As the plane comes out of cruise altitude, ACARS begins updating the dispatch office with a new estimated time of arrival and other changing details so that appropriate preparations, such as gate assignment and baggage handling, can be made. On touchdown, the ACARS report is automatically sent to dispatch and the next crew heads for the gate. When the plane comes to a final stop, the entire report is sent to a printer. The Captain signs the report securing the crew's pay for the day.

COMMUNICATION AND NAVIGATION AVIONICS INSTALLATIONS

The aircraft maintenance technician may remove, install, inspect, maintain, and troubleshoot avionics equipment. It is imperative to follow all equipment and airframe manufacturers' instruction when dealing with an aircraft's avionics.



Figure 4-125. Typical ACARS display format.

The installation of avionics equipment is partially mechanical, involving sheet metal work to mount units, racks, antennas, and controls. Routing of the interconnecting wires, cables, antenna leads, etc. is also an important part of the installation process. When a location for the equipment is selected by the manufacturer avionics radio equipment is securely mounted to the aircraft. All mounting bolts must be secured by locking devices to prevent loosening from vibration. Adequate clearance between all units and adjacent structure is provided to prevent mechanical damage to electric wiring or to the avionic equipment from vibration, chafing, or landing shock. Combustible materials are kept away from avionics.

The performance and service life of most avionics equipment is seriously limited by excessive ambient temperatures. High performance aircraft with avionics equipment racks typically route air-conditioned air over the avionics to keep them cool. It is also common for non-air conditioned aircraft to use a blower or scooped ram air to cool avionics installations. Measures are also taken to prevent moisture from reaching the avionics equipment. The presence of water in avionics equipment areas promotes deterioration of the exposed components and could lead to failure.

Avionics equipment is sensitive to mechanical shock and vibration and is normally shock mounted to provide some protection against in-flight vibration and landing shock. Vibration is a continued motion by an oscillating force. The amplitude and frequency of vibration of the aircraft structure will vary considerably with the type of aircraft. Special shock mounted racks are often used to isolate avionics equipment from vibrating structure. (***Figure 4-126***)

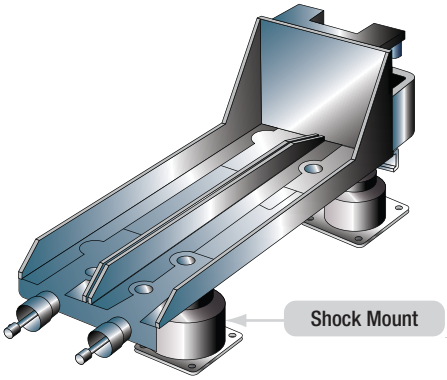


Figure 4-126. A shock mounted equipment rack is often used to install avionics.

Such mounts provide adequate isolation over the entire range of expected vibration frequencies. Periodic inspection of the shock mounts is required and defective mounts should be replaced with the proper type. The following factors to observe during the inspection are:

- Deterioration of the shock-absorbing material;
- Stiffness and resiliency of the material;
- Overall rigidity of the mount.

If the mount is too stiff, it may not provide adequate protection against the shock of landing. If the shock mount is not stiff enough, it may allow prolonged vibration following an initial shock.

Shock-absorbing materials commonly used in shock mounts are usually electrical insulators. For this reason, each electronic unit mounted with shock mounts must be electrically bonded to a structural member of the aircraft to provide a current path to ground. This is accomplished by secure attachment of a tinned copper wire braid from the component, across the mount, to the aircraft structure as shown in ***Figure 4-127***. Occasional bonding is accomplished with solid aluminum or copper material where a short flexible strap is not possible.

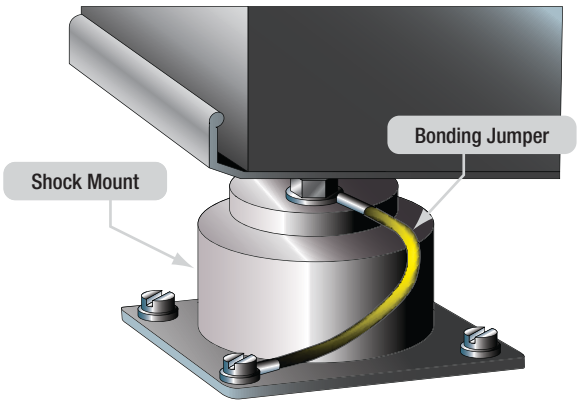


Figure 4-127. A bonding jumper is used to ground an equipment rack and avionics chassis around the non-conductive shock mount material.

Question: 4-1
The ability of an aircraft doppler radar to look in two directions at once is called _____ operation.

Question: 4-2
In ILS system, the glideslope transmits two signals, the upper lobe modulated at _____ Hz and the lower lobe modulated at _____ Hz.

Question: 4-3
An aircraft High Frequency system can operate in simplex or half-duplex modes. List the three types of modulation an HF system may deploy:

Question: 4-4
List the three basic types of antennas used on aircraft.

Question: 4-5
What is meant by the terms "Compression" and "Rarefaction" in the context of Longitudinal Waves?

Question: 4-6
In an electromagnetic wave, the electrical function is a measure of _____ while the magnetic function is a measure of _____.

Question: 4-7
The three basic ways of modulating radio waves are?

Question: 4-8
The radio frequencies useful for satellite based communications are _____.

Question: 4-9
What determines the size or shape of an antenna needed for a particular application?

Question: 4-10
The battery life expiration date of an Emergency Locator Transmitter is recorded in two places. They are?

ANSWERS

Answer: 4-1
Janus.

Answer: 4-2
90Hz; 150Hz.

Answer: 4-3
LV (Lower Voice Single Sideband);
UV (Upper Voice Single Sideband);
AM (Amplitude Modulation)

Answer: 4-4
Dipole antenna;
Marconi antenna;
Loop antenna.

Answer: 4-5
Longitudinal waves are characterized by regular regions of high and low pressure along the wave. High pressure regions are called compressions and low pressure regions are called rarefactions.

Answer: 4-6
voltage; current

Answer: 4-7
amplitude modulation; frequency modulation; phase modulation.

Answer: 4-8
VHF, UHF, and SHF; frequencies above 30MHz

Answer: 4-9
the wavelength of the frequency being transmitted

Answer: 4-10
aircraft maintenance records; written on the outside of the ELT



PART-66 SYLLABUS **LEVELS**

CERTIFICATION CATEGORY → **B2**

Sub-Module 05

ELECTRICAL POWER (ATA 24)

Knowledge Requirements

13.5 - Electrical Power (ATA 24)

Batteries Installation and Operation; 3
DC power generation;
AC power generation;
Emergency power generation;
Voltage regulation;
Power distribution;
Inverters, transformers, rectifiers, circuit protection;
External/Ground power.

BATTERIES

A battery is an energy storage device that relies on electrochemical reactions to deliver energy.

PRIMARY CELL BATTERIES

The dry cell is the most common type of primary-cell battery and is similar in its characteristics to that of an electrolytic cell. This type of a battery is basically designed with a metal electrode or graphite rod acting as the cathode (+) terminal, immersed in an electrolytic paste. This electrode/electrolytic build-up is then encased in a metal container, usually made of zinc, which itself acts as the anode (–) terminal. When the battery is in a discharge condition an electrochemical reaction takes place resulting in one of the metals being consumed. Because of this consumption, the charging process is not reversible. Attempting to reverse the chemical reaction in a primary cell by way of recharging is usually dangerous and can lead to a battery explosion.

These batteries are commonly used to power items such as flashlights. The most common primary cells today are found in alkaline batteries, silver-oxide and lithium batteries. The earlier carbon-zinc cells, with a carbon post as cathode and a zinc shell as anode were once prevalent but are not as common.

SECONDARY CELL BATTERIES

A secondary cell is any kind of electrolytic cell in which the electrochemical reaction that releases energy is reversible. The lead-acid car battery is a secondary-cell battery as are some aircraft batteries. The electrolyte is sulphuric acid (battery acid), the positive electrode is lead peroxide, and the negative electrode is lead. A typical lead-acid battery consists of six lead-acid cells in a case. Each cell produces 2 volts, so the whole battery produces a total of 12 volts. Other commonly used secondary cell chemistry types are nickel cadmium (NiCd), nickel metal hydride (NiMH), lithium ion (Li-ion), and Lithium ion polymer (Li-ion polymer).

Lead-acid batteries used in aircraft are similar to automobile batteries. The lead acid battery is made up of a series of identical cells each containing sets of positive and negative plates. ***Figure 5-1*** illustrates each cell contains positive plates of lead dioxide (PbO_2), negative plates of spongy lead, and electrolyte (sulfuric acid and water).

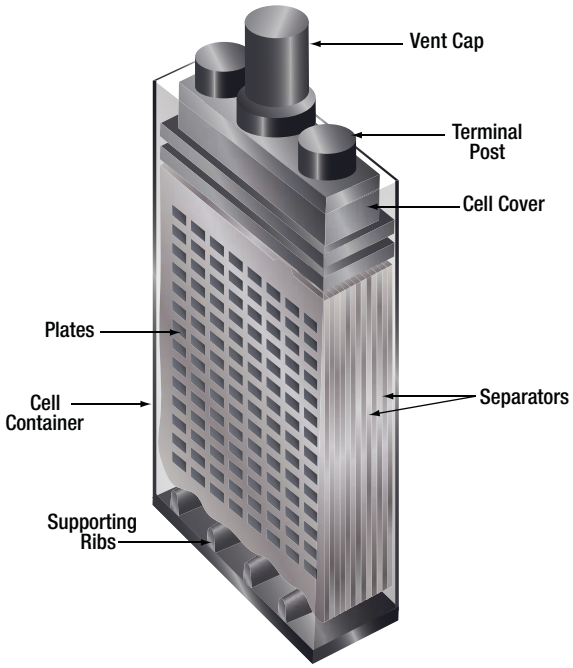


Figure 5-1. Lead-acid cell construction.

A practical cell is constructed with many more plates than just two in order to get the required current output. All positive plates are connected together as well as all the negatives. Because each positive plate is always positioned between two negative plates, there are always one or more negative plates than positive plates.

Between the plates are porous separators that keep the positive and negative plates from touching each other and shorting out the cell. The separators have vertical ribs on the side facing the positive plate. This construction permits the electrolyte to circulate freely around the plates. In addition, it provides a path for sediment to settle to the bottom of the cell.

Each cell is seated in a hard rubber casing through the top of which are terminal posts and a hole into which is screwed a non-spill vent cap. The hole provides access for testing the strength of the electrolyte and adding water. The vent plug permits gases to escape from the cell with a minimum of leakage of electrolyte, regardless of the position the airplane might assume.

Figure 5-2 shows the construction of the vent plug. In level flight, the lead weight permits venting of gases through a small hole. In inverted flight, this hole is covered by the lead weight.

DRY CHARGED (FLOODED) LEAD ACID BATTERIES

The electrolyte is added to the battery when it is placed in service, and battery life begins when the electrolyte is added. An aircraft storage battery consists of 6 or 12 lead-acid cells connected in series. The open circuit voltage of the 6 cell battery is approximately 12 volts, and the open circuit voltage of the 12-cell battery is approximately 24 volts. Open circuit voltage is the voltage of the battery when it is not connected to a load. When flooded (vented) batteries are on charge, the oxygen generated at the positive plates escapes from the cell. Concurrently, at the negative plates, hydrogen is generated from water and escapes from the cell. The overall result is the gassing of the cells and water loss. Therefore, flooded cells require periodic water replenishment. (***Figure 5-3***)

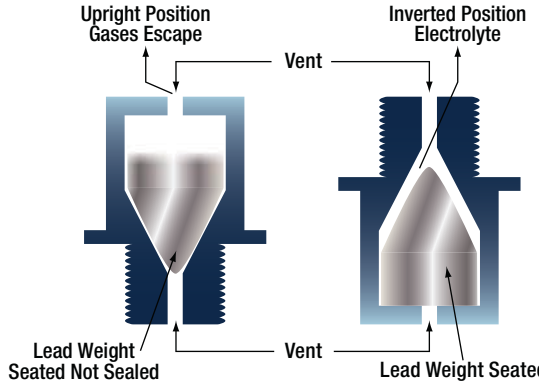


Figure 5-2. Nonspill battery vent plug.



Figure 5-3. Lead acid battery installation.

VALVE-REGULATED (SEALED) LEAD-ACID BATTERIES (VRLA)

VRLA batteries contain all electrolyte absorbed in glass-mat separators with no free electrolyte and are sometimes referred to as sealed batteries. (***Figure 5-4***) The electrochemical reactions for VRLA batteries are the same as flooded batteries, except for the gas recombination mechanism that is predominant in VRLA batteries. These types of battery are used in general aviation and turbine powered aircraft and are sometimes authorized replacements for NiCd batteries.

When VRLA batteries are on charge, oxygen combines chemically with the lead at the negative plates in the presence of H_2SO_4 to form lead sulfate and water. This oxygen recombination suppresses the generation of hydrogen at the negative plates. Overall, there is no water loss during charging.

A very small quantity of water may be lost as a result of self-discharge reactions; however, such loss is so small that no provisions are made for water replenishment. The battery cells have a pressure relief safety valve that may vent if the battery is overcharged.



Figure 5-4. Valve-regulated lead-acid battery (sealed battery).

NICKEL CADMIUM (NICD) BATTERIES

A NiCd battery consists of a metallic box, usually stainless steel, plastic-coated steel, painted steel, or titanium containing a number of individual cells. (***Figure 5-5***) These cells are connected in series to obtain 12 volts or 24 volts. The cells are connected by highly conductive nickel copper links. Inside the battery box, the cells are held in place by partitions, liners, spacers, and a cover assembly. The battery has a ventilation system to allow the escape of the gases produced during an overcharge condition and provide cooling during normal operation. NiCd cells installed in an aircraft battery are typical of the vented cell type.

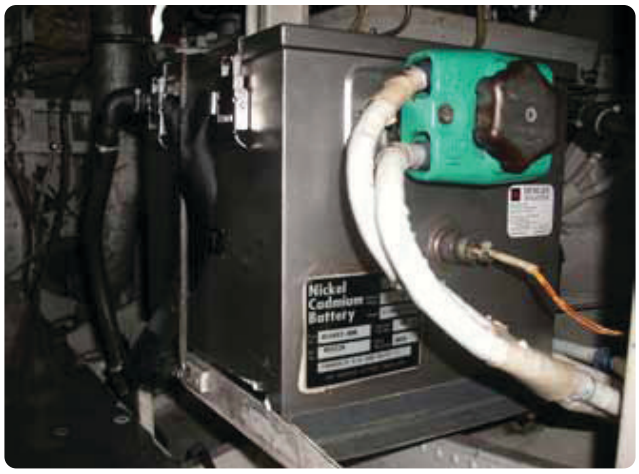


Figure 5-5. NiCd battery installation.



Figure 5-6. Thermal runaway damage.

The vented cells have a vent or low pressure release valve that releases any generated oxygen and hydrogen gases when overcharged or discharged rapidly. This also means the battery is not normally damaged by excessive rates of overcharge, discharge, or even negative charge. The cells are rechargeable and deliver a voltage of 1.2 volts during discharge.

Aircraft that are outfitted with NiCd batteries typically have a fault protection system that monitors the condition of the battery. The battery charger is the unit that monitors the condition of the battery and the following conditions are monitored.

1. Overheat condition
2. Low temperature condition (below –40 °F)
3. Cell imbalance
4. Open circuit
5. Shorted circuit

If the battery charger finds a fault, it turns off and sends a fault signal to the Electrical Load Management System (ELMS).

NiCd batteries are capable of performing to its rated capacity when the ambient temperature of the battery is in the range of approximately 60–90 °F. An increase or decrease in temperature from this range results in reduced capacity. NiCd batteries have a ventilation system to control the temperature of the battery. A combination of high battery temperature (in excess of 160 °F) and overcharging can lead to a condition called thermal runaway. (***Figure 5-6***) The temperature of the battery has to be constantly monitored to ensure safe operation. Thermal runaway can result in a NiCd chemical fire and/or explosion of the NiCd battery under recharge by a constant-voltage source and is due to cyclical, ever-increasing temperature and charging current. One or more shorted cells or an existing high temperature and low charge can produce the following cyclical sequence of events:

1. Excessive current,
2. Increased temperature,
3. Decreased cell(s) resistance,
4. Further increased current, and
5. Further increased temperature.

This does not become a self-sustaining thermal-chemical action if the constant-voltage charging source is removed before the battery temperature is in excess of 160 °F.

LITHIUM ION BATTERIES

The most recent type of battery to be certified in aircraft is the lithium ion battery. These batteries have greater capacity and weigh less than NiCd or lead acid types. They have no memory as NiCd batteries have and discharge less than half as slowly when not being used. The anode is a graphite layered structure capable of storing and releasing lithium ions. Cathode materials vary. The certified battery used on the Boeing 787 is made of lithium cobalt oxide (LiCoO2) with an aluminum core. A water-free electrolyte composed of organic carbonates resides between the anode and cathode. It functions as a transport medium for the lithium ions moving from the anode to the cathode during discharge and from the cathode to the anode during charging. A separator porous to the Li+ ions is between the anode and cathode in each cell. The electrons that cause the lithium to be ions are the current used in the external circuit to power aircraft electrical buses and components.

Typical cell output voltage is between 3 and 4.2 volts depending primarily on the materials used to construct the cathode. Eight cells connected in series are typical as shown in ***Figure 5-7***.

Lithium ion aircraft batteries require built-in safety devices to prevent overheating and thermal runaway. They are constructed with a wide variety of material choices that result in compromise between capacity, longevity, environmental endurance and operating range, current loading, specific energy, size and weight, etc. Additional current monitoring and other safety and alerting devices are included to warn flight crew of battery status and malfunction. Technicians must follow all manufacturer instructions when maintaining lithium ion batteries.

BATTERY INSTALLATIONS AND OPERATION

There are numerous ways to install and integrate batteries into an aircraft electrical system. DC battery power is used to supply power when other power sources are not available (i.e. ground power or engine-driven generator power). The Boeing 777 airliner battery installation is described here. Airbus and other modern aircraft have slightly different installations.

Applicable to all installations are redundancy and the ability to maintain electric power to essential systems in case of electrical failure. Most modern airliners have at least two batteries - one for the aircraft and a separate, dedicated battery for APU starting. The main aircraft battery supplies the power for DC loads while the aircraft is on the ground. In the air, it supplies standby system power. This includes power to all components required to deployed the ram air turbine (RAT). Because of it's independent nature, powered by ram air and not an engine, the RAT is an important component of emergency power production for the standby system. When the RAT supplies electrical power from it's generator, the aircraft battery no long supplies the standby bus.

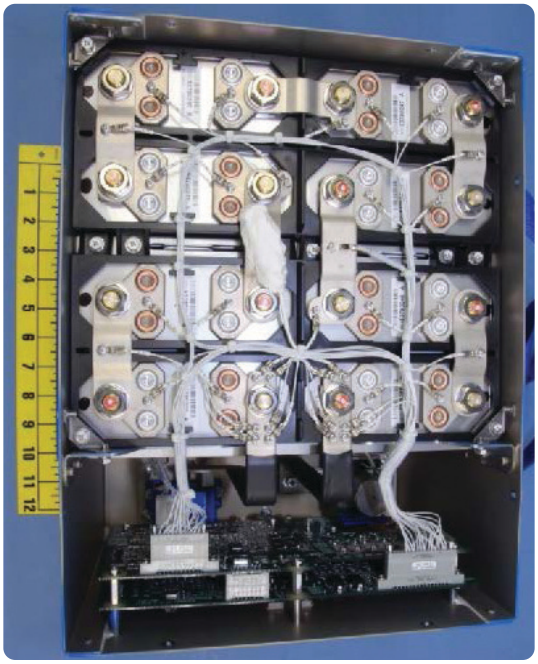


Figure 5-7. Cells and wring in a lithium ion aircraft battery.

Usually, the same nickel-cadmium battery is used for both the main aircraft battery and the APU battery. They are interchangeable. These batteries weigh near 100 pounds (45 kg). Attach fittings on a battery facilitate the use of lifting equipment during removal and installation. Most aircraft use 28V DC batteries but configurations exist where two 14V batteries are connected in series to arrive at 28V for bus use.

Typical transport aircraft batteries have two connectors. The large connector is a terminal block which connects the high power output of the battery into the bus system. The smaller electrical connector is for battery control and status signals. Temperature sensors and overheat sensing are common. A cooling fan may be included in the installation. (***Figure 5-8***)

On the 777, the main aircraft battery is wired so that it supplies the hot battery bus. This bus supplies the aircraft information management system (AIMS) that needs power to maintain all data stored in its static random access memory (SRAM).

The battery system installation includes a battery charger for each battery. The chargers change AC power into DC power for DC buses in addition to keeping the batteries charged. They are LRU's that have the same connectors as the batteries. Status monitoring and control signals are AC powered throughout the small electrical connector. The battery chargers have multiple charging modes. Boeing chargers in the 777 also have a transformer rectifier (TR) mode which converts the 115V AC input to DC power.

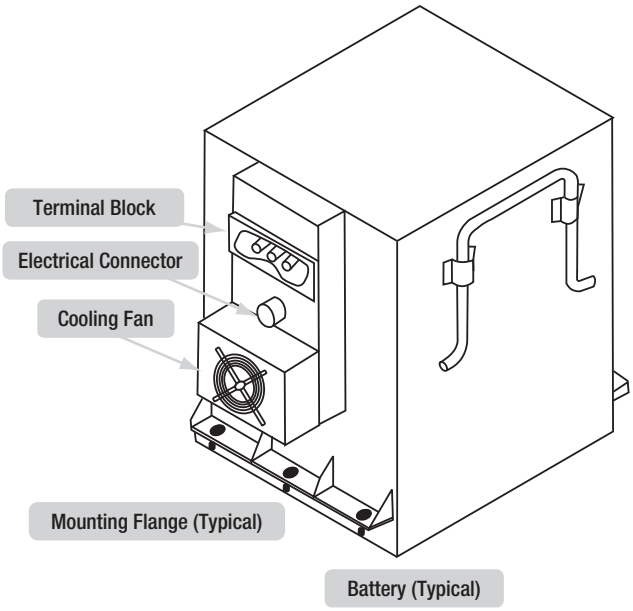


Figure 5-8. A typical aircraft battery and connectors.

Up to 65 amps may be drawn directly from the tightly controlled voltage output of this TR. Temperature sensing is important during battery charging. Built in sensors are used to isolate the battery charger should an over temp occur. Most airliner battery chargers are controlled by a series of relays so that the chargers only charge when power production is normal. In addition to a control relay in the charger, a battery thermal switch controlled by battery temperature interrupt charging when the battery or the charger are hot. Charging may also be interrupted by relays controlling connectivity to the APU starter or the fueling panel. The manufacturer's wiring diagrams are the source for learning exactly how the batteries and charger are configured in relationship to other electrical components.

On a Boeing 777, an electrical load management system (ELMS) is used to control many aircraft electrical components. It is comprised of power distribution panels in a centralized location, each designed with a specific function within the entire power system of the aircraft. The panel are the left and right power panels, the left and right power management panels, the auxiliary power panel, the ground service/ handling power panel and the standby power management panel. (***Figure 5-9***) The main aircraft battery is controlled through the standby power management panel. It is also directly or indirectly connected to all panels in the ELMS.

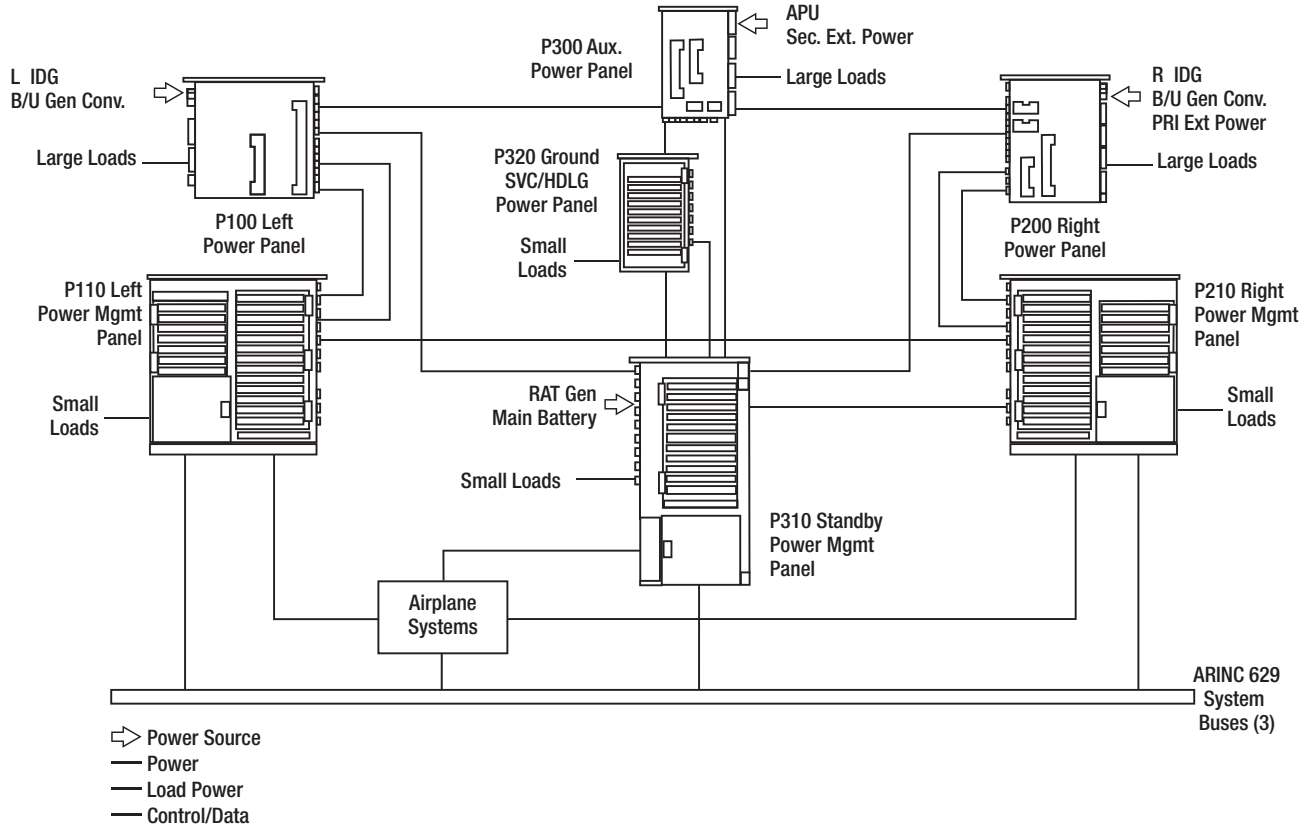


Figure 5-9. The main battery position in the aircraft electrical load management system is through the standby power management panel.

AC POWER GENERATION

An AC power system is the primary source of power on most transport aircraft.

These extremely reliable power distribution systems are computer controlled. Multiple power sources (AC alternators also known as AC generators) and a variety of distribution busses are used on a single aircraft for redundancy. A typical airliner contains two or more main AC generators driven by the aircraft turbine engines, as well as more than one backup AC generator. DC systems are also employed on large aircraft and the aircraft battery may be used to supply emergency power in case of a multiple failures.

The typical AC generator produces three-phase 115-volt AC at 400 Hz. AC generators were discussed previously in this module series. Since most modern transport category aircraft are designed with two engines, there are two main AC generators. The APU also drives an AC generator. This unit is available during flight if one of the main generators fails. The main and auxiliary generators may be similar in output capacity or back-up generators may have a low output. The main generators on modern transport aircraft typically supply a maximum of 120 kilovolt amps (KVA). A fourth generator, driven by an emergency ram air turbine, is also available in the event the two main generators and one auxiliary generator fail. This emergency generator is typically smaller and produces less power. With four AC generators available, it is highly unlikely that a complete power failure occurs. However, if all AC generators are lost, the aircraft battery continues to supply DC electrical power to operate vital systems.

Note that on the latest generation of aircraft, there is even more redundancy for AC power production. The Boeing 777 has 2 integrated drive generators that are the primary AC generators but the aircraft also has 2 back-up engine-driven AC generators in case of failure of the primary generators. Each back-up generator also contains two permanent magnet DC generators for powering the flight controls power supply assemblies. Airbus aircraft have at least one back-up AC generator that is powered by a hydraulic motor.

Transport category aircraft use large amounts of electrical power for a variety of systems. Passenger comfort requires power for lighting, audio visual systems, and galley power for food warmers and beverage coolers. A variety of electrical systems are required to fly the aircraft, such as flight control systems, electronic engine controls, communication, and navigation systems. The output capacity of one engine-driven AC generator can typically power all necessary electrical systems. A second engine-driven generator is operated during flight to share the electrical loads and provide redundancy. ***Figure 5-10*** illustrates the electrical power system of a Boeing 777.

The complexity of multiple generators and a variety of distribution busses requires several control units to maintain a constant supply of safe electrical power. The AC electrical system must maintain a constant output of 115 to 120 volts at a frequency of 400 Hz (±10 percent). The system must ensure power limits are not exceeded. AC generators are connected to the appropriate distribution busses at the appropriate time, and generators are in phase when needed. There is also the need to monitor and control any external power supplied to the aircraft, as well as control of all DC electrical power.

Two electronic line replaceable units are used to control the electrical power on a typical large aircraft. The generator control unit (GCU) is used for control of AC generator functions, such as voltage regulation and frequency control. The bus power control unit (BPCU) is used to control the distribution of electrical power between the various distribution busses on the aircraft. The GCU and BPCU work together to control electrical power, detect faults, take corrective actions when needed, and report any defect to the pilots and the aircraft's central maintenance system. There is typically one GCU for each AC generator and at least one BPCU to control bus connections. These LRUs are located in the aircraft's electronics equipment bay and are designed for easy replacement.

When the pilot calls for generator power by activating the generator control switch on the flight deck, the GCU monitors the system to ensure correct operation. If all systems are operating within limits, the GCU energizes the appropriate generator circuits and provides voltage regulation for the system. The GCU also monitors AC output to ensure a constant 400-Hz

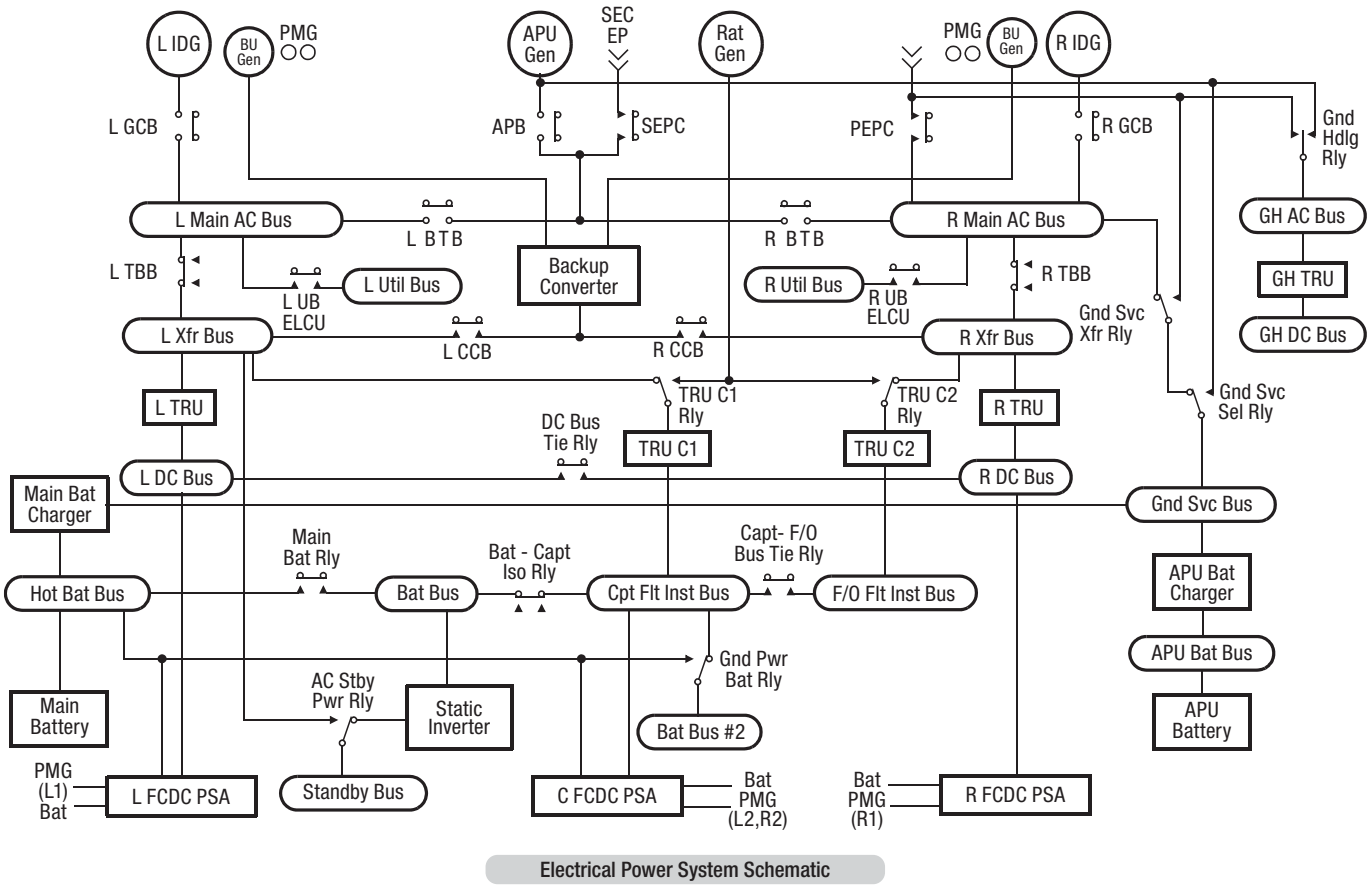


Figure 5-10. The main battery position in the aircraft electrical load management system is through the standby power management panel.

frequency. If the generator output is within limits, the GCU then connects the electrical power to the main generator bus through an electrical contactor (solenoid). These contactors are often called generator breakers (GB) since they break (open) or make (close) the main generator circuit.

After generator power is available, the BPCU activates various contactors to distribute the electrical power. The BPCU monitors the complete electrical system and communicates with the GCU to ensure proper operation. The BPCU employs remote current sensors known as a current transformers (CT) to monitor the system. (***Figure 5-11***)

A CT is an inductive unit that surrounds the main power cables of the electrical distribution system. As AC power flows through the main cables, the CT receives an induced voltage. The amount of CT voltage is directly related to the current flowing through the cable. The CT connects to the BPCU, which allows accurate current monitoring of the system. A typical aircraft employs several CTs throughout the electrical system.

The BPCU is a dedicated computer that controls the electrical connections between the various distribution busses found on the aircraft.

The BPCU uses contactors (solenoids) called bus tie breakers (BTB) for connection of various circuits. These BTBs open/close the connections between the busses as needed for system operation as called for by the flight crew and the BPCU. This sounds like a simple task, yet to ensure proper operation under a variety of conditions, the bus system becomes very

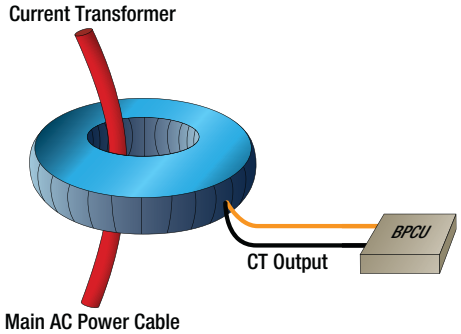


Figure 5-11. Current transformer.

complex. There are three common types of distribution bus systems found on transport category aircraft: split bus, parallel bus, and split parallel. These are examined in the section on power distribution below.

AC ALTERNATORS

AC alternators produce a three-phase AC output. For each revolution of the alternator, the unit produces three separate voltages. The sine waves for these voltages are separated by 120˚. (***Figure 5-12***) This wave pattern is similar to those produced internally by a DC alternator; however, in this case, the AC alternator does not rectify the voltage and the output of the unit is AC.

The modern AC alternator does not utilize brushes or slip rings and is often referred to as a brushless AC alternator. This brushless design is extremely reliable and requires very little maintenance. In a brushless alternator, energy to or from the alternator's rotor is transferred using magnetic energy. In other words, energy from the stator to the rotor is transferred using magnetic flux energy and the process of electromagnetic induction. A typical large aircraft AC alternator is shown in ***Figure 5-13***.

As seen in ***Figure 5-14***, the brushless alternator actually contains three generators: the Exciter generator (armature and permanent magnet field), the Pilot exciter generator (armature and fields windings), and the main AC alternator (armature winding and field windings.

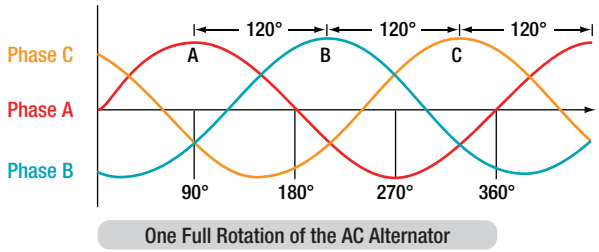


Figure 5-12. AC alternator sine waves.



Figure 5-13. Large aircraft AC alternator.

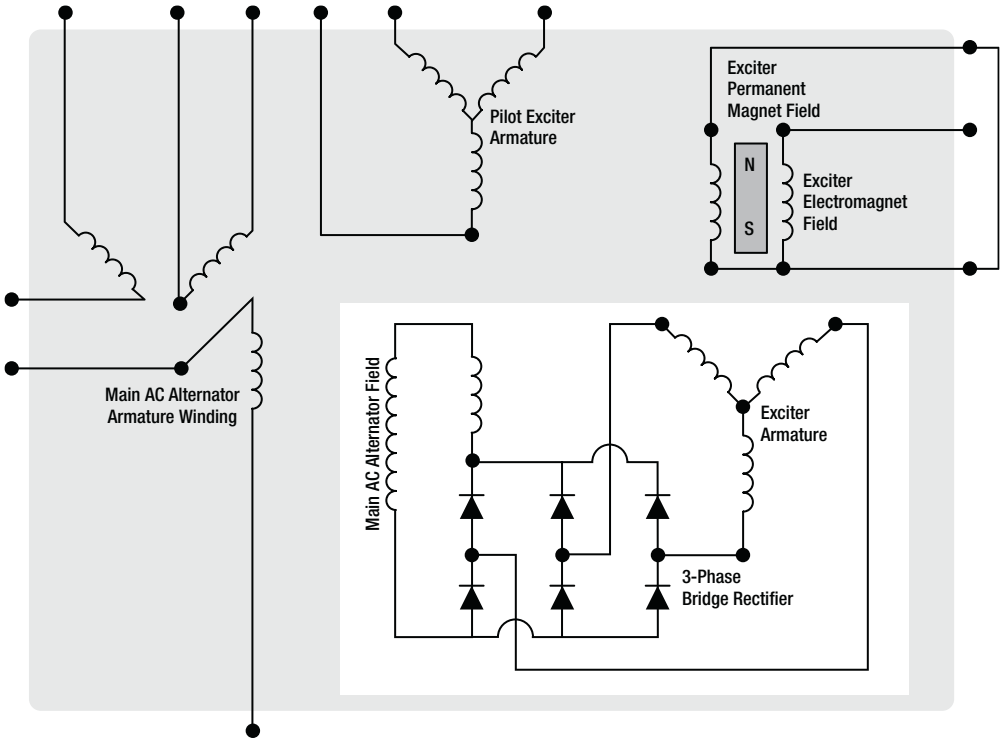


Figure 5-14. Schematic of an AC alternator.

The need for brushes is eliminated by using a combination of these three distinct generators. The exciter is a small AC generator with a stationary field made of a permanent magnet and two electromagnets. The exciter armature is three phase and mounted on the rotor shaft. The exciter armature output is rectified and sent to the pilot exciter field and the main generator field.

The pilot exciter field is mounted on the rotor shaft and is connected in series with the main generator field. The pilot exciter armature is mounted on the stationary part of the assembly. The AC output of the pilot exciter armature is supplied to the generator control circuitry where it is rectified, regulated, and then sent to the exciter field windings. The current sent to the exciter field provides the voltage regulation for the main AC alternator. If greater AC alternator output is needed, there is more current sent to the exciter field and vice versa.

In short, the exciter permanent magnet and armature starts the generation process, and the output of the exciter armature is rectified and sent to the pilot exciter field. The pilot exciter field creates a magnetic field and induces power in the pilot exciter armature through electromagnetic induction. The output of the pilot exciter armature is sent to the main alternator control unit and then sent back to the exciter field. As the rotor continues to turn, the main AC alternator field generates power into the main AC alternator armature, also using electromagnetic induction. The output of the main AC armature is three-phase AC and used to power the various electrical loads.

Some alternators are cooled by circulating oil through the internal components of the alternator. The oil used for cooling is supplied from the constant speed drive assembly and often cooled by an external oil cooler assembly. Located in the flange connecting the generator and drive assemblies, ports make oil flow between the constant speed drive and the generator possible. This oil level is critical and typically checked on a routine basis.

ALTERNATOR DRIVE

The unit shown in ***Figure 5-15*** contains an alternator assembly combined with an automatic drive mechanism. The automatic drive controls the alternator's rotational speed which allows the alternator to maintain a constant 400-Hz AC output.

All AC alternators must rotate at a specific rpm to keep the frequency of the AC voltage within limits. Aircraft AC alternators should produce a frequency of approximately 400 Hz. If the frequency strays more than 10 percent from this value, the electrical systems do not operate correctly. A unit called a constant-speed drive (CSD) is used to ensure the alternator rotates at the correct speed to ensure a 400-Hz frequency. The CSD can be an independent unit or mounted within the alternator housing. When the CSD and the alternator are contained within one unit, the assembly is known as an integrated drive generator (IDG).

The CSD is a hydraulic unit similar to an automatic transmission found in a modern automobile. The engine of the automobile can change rpm while the speed of the car remains constant. This is the same process that occurs for an aircraft AC alternator. If the aircraft engine changes speed, the alternator speed remains constant.

A typical hydraulic-type drive is shown in ***Figure 5-16***. This unit can be controlled either electrically or mechanically. Modern aircraft employ an electronic system. The constant-speed drive enables the alternator to produce the same frequency at slightly above engine idle rpm as it does at maximum engine rpm.

The hydraulic transmission is mounted between the AC alternator and the aircraft engine. Hydraulic oil or engine oil is used to operate the hydraulic transmission, which creates a constant output speed to drive the alternator. In some cases, this same oil is used to cool the alternator as shown in the CSD cutaway view of ***Figure 5-16***.

The input drive shaft is powered by the aircraft engine gear case. The output drive shaft, on the opposite end of the transmission, engages the drive shaft of the alternator. The CSD employs a hydraulic pump assembly, a mechanical speed control, and a hydraulic drive. Engine rpm drives the hydraulic pump, the hydraulic drive turns the alternator. The speed control unit is made up of a wobble plate that adjusts hydraulic pressure to control output speed.

Figure 5-17 shows a typical electrical circuit used to control alternator speed. The circuit controls the hydraulic assembly found in a typical CSD. As shown, the alternator input speed is monitored by a tachometer (tach) generator. The tach generator signal is rectified

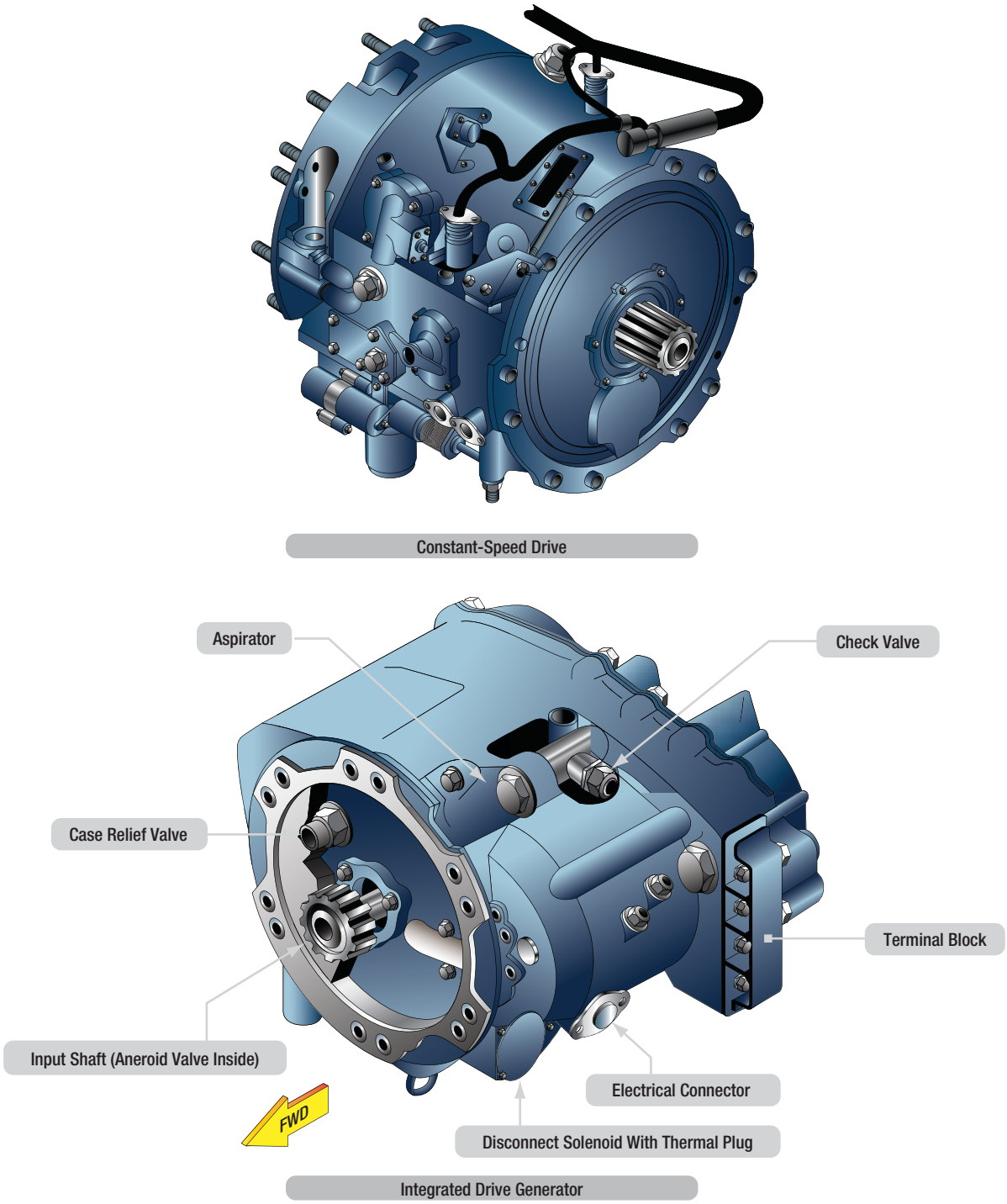


Figure 5-15. Constant-speed drive (top) and integrated drive generator (bottom).

and sent to the valve assembly. The valve assembly contains three electromagnetic coils that operate the valve. The AC alternator output is sent through a control circuit that also feeds the hydraulic valve assembly. By balancing the force created by the three electromagnets, the valve assembly controls the flow of fluid through the automatic transmission and controls the speed of the AC alternator.

It should be noted that an AC alternator also produces a constant 400 Hz if that alternator is driven directly by an engine that rotates at a constant speed.

On many aircraft, the auxiliary power unit operates at a constant rpm. AC alternators driven by these APUs are typically driven directly by the engine, and there is no CSD required. For these units, the APU engine controls monitor the alternator output frequency. If the alternator

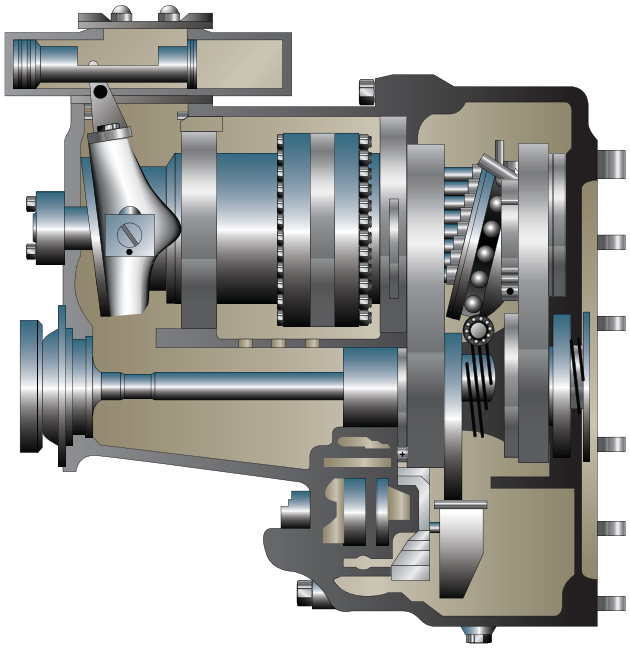


Figure 5-16. A hydraulic constant speed drive for an AC alternator.

output frequency varies from 400 Hz, the APU speed control adjusts the engine rpm accordingly to keep the alternator output within limits.

AC ALTERNATORS CONTROL SYSTEMS

Modern aircraft that employ AC alternators use several computerized control units, typically located in the aircraft's equipment bay for the regulation of AC power throughout the aircraft. ***Figure 5-18*** shows a photo of a typical equipment bay and computerized control units.

Since AC alternators are found on large transport category aircraft designed to carry hundreds of passengers, their control systems always have redundant computers that provide safety in the event of a system failure. Unlike DC systems, AC systems must ensure that the output frequency of the alternator stays within limits.

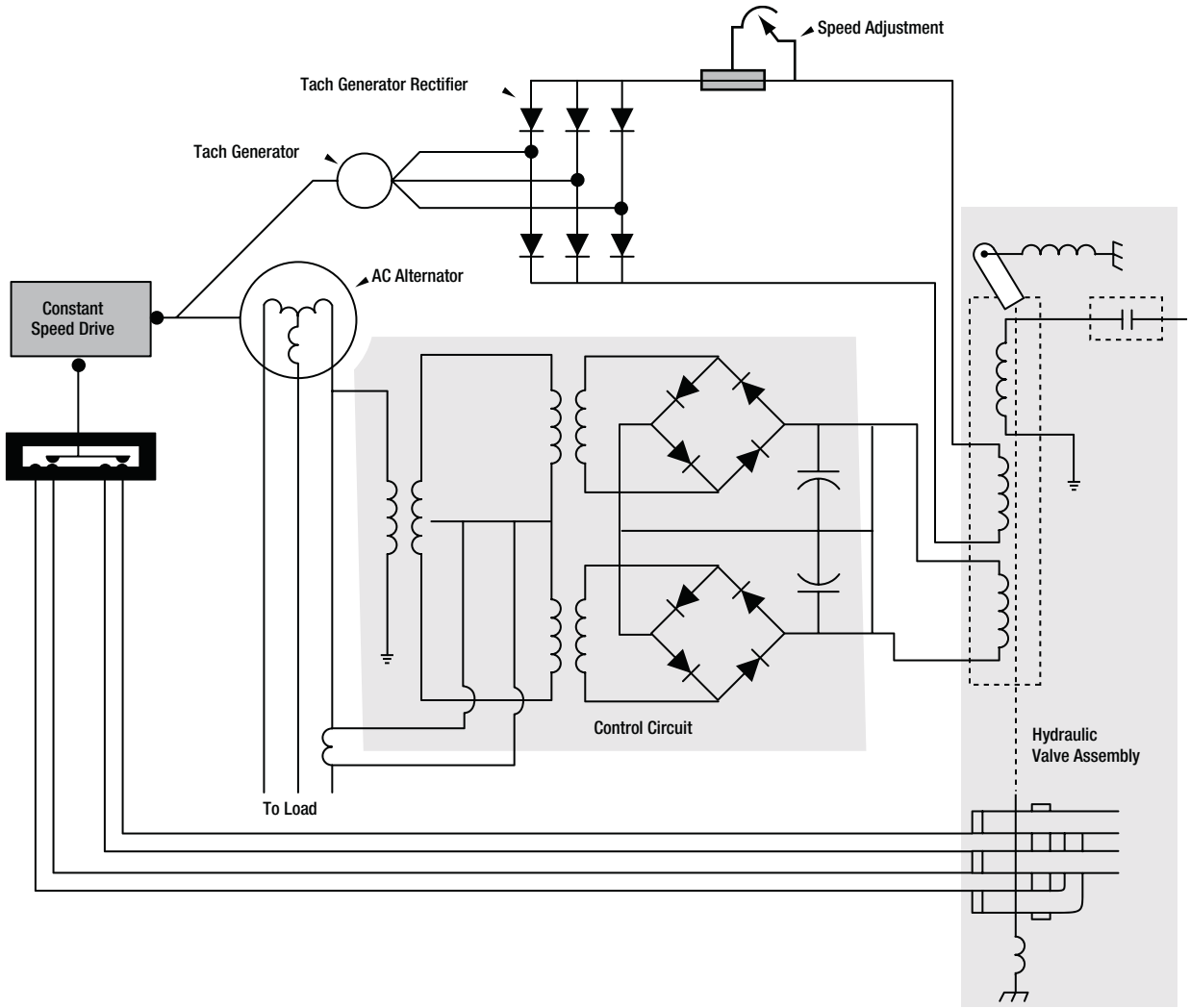


Figure 5-17. Speed control circuit.

If the frequency of an alternator varies from 400 Hz, or if two or more alternators connected to the same bus are out of phase, damage occurs to the system. All AC alternator control units contain circuitry that regulates both voltage and frequency. These control units also monitor a variety of factors to detect any system failures and take protective measures to ensure the integrity of the electrical system. The two most common units used to control AC alternators are the bus power control unit (BPCU) and the GCU. In this case, the term "generator" is used, and not alternator, although the meaning is the same. The GCU is the main computer that controls alternator functions.

The BPCU is the computer that controls the distribution of AC power to the power distribution busses located throughout the aircraft. There is typically one GCU used to monitor and control each AC alternator, and there can be one or more BPCUs on the aircraft. BPCUs are described later in this chapter; however, please note that the BPCU works in conjunction with the GCUs to control AC on modern aircraft.

A typical GCU ensures the AC alternator maintains a constant voltage, typically between 115 to 120 volts. The GCU ensures the maximum power output of the alternator is never exceeded. The GCU provides fault detection and circuit protection in the event of an alternator failure. The GCU monitors AC frequency and ensures the output if the alternator remains 400 Hz. The basic method of voltage regulation is similar to that found in all alternator systems; the output of the alternator is controlled by changing the strength of a magnetic field. As shown in ***Figure 5-19***, the GCU controls the exciter field magnetism within the brushless alternator to control alternator output voltage. The frequency is controlled by the CDS hydraulic unit in conjunction with signals monitored by the GCU.



Figure 5-18. Line replaceable units in an equipment rack.

The GCU is also used to turn the AC alternator on or off. When the pilot selects the operation of an AC alternator, the GCU monitors the alternator's output to ensure voltage and frequency are within limits. If the GCU is satisfied with the alternator's output, the GCU sends a signal to an electrical contactor that connects the alternator to the appropriate AC distribution bus. The contactor, often call the generator breaker, is basically an electromagnetic solenoid that controls a set of large contact points. The large contact points are necessary in order to handle the large amounts of current produced by most AC alternators. This same contactor is activated in the event the GCU detects a fault in the alternator output; however, in this case the contactor would disconnect the alternator from the bus.

DC POWER GENERATION

Most modern airliners use AC generators for the primary source of power. However, there are still aircraft flying that may use DC generators for power. Smaller aircraft may also have DC generators or DC alternators. When present, DC generators normally have output controlled to 14 or 28V DC. The output is controlled by controlling field current strength. It is more common on large aircraft to produce DC power from AC generator power. Transformer rectifiers (TR) are used to convert the 115V AC to any DC voltage required - normally 28V DC. This DC powers various buses. Transformer rectifiers are also found in battery chargers.

An engine-driven DC generator requires a control circuit in order to ensure the generator maintains the correct voltage and current for the current electrical conditions of the aircraft. All aircraft are designed to operate within a specific voltage range (for example 13.5–14.5 volts). And since aircraft operate at a variety of engine speeds (remember, the engine drives the

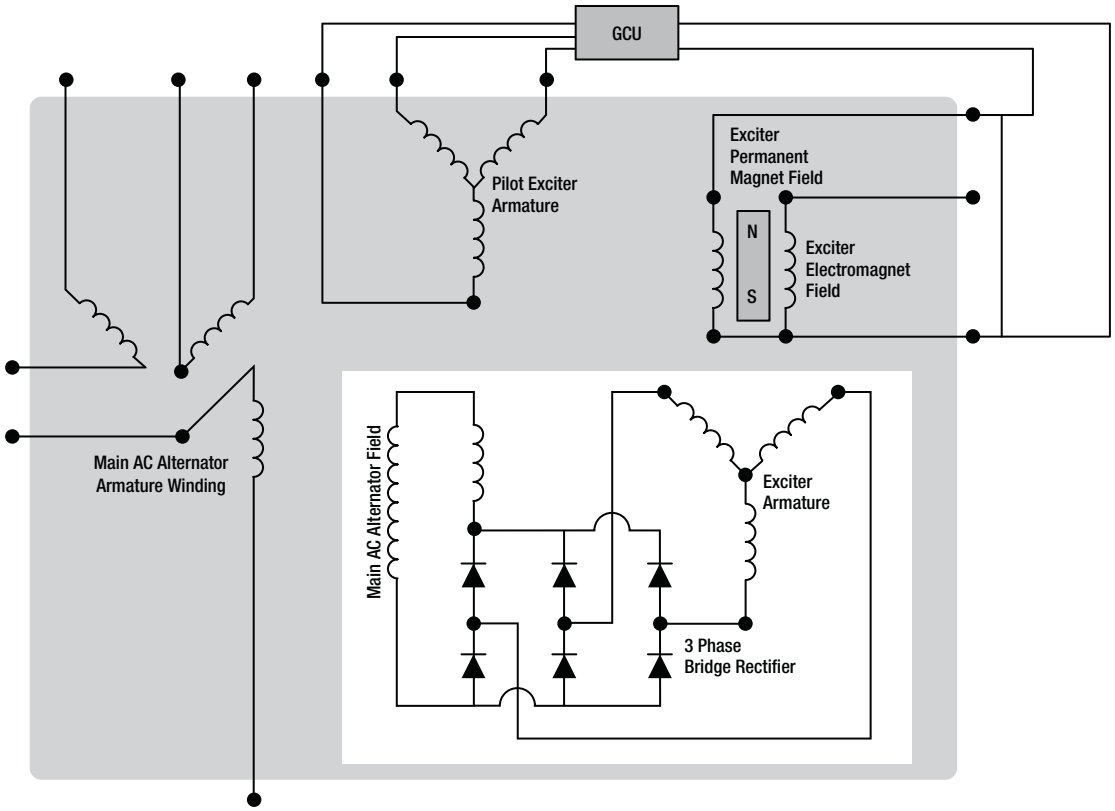


Figure 5-19. Schematic GCU control of the exciter field magnetism.

generator) and with a variety of electrical demands, all generators must be regulated by some control system. The generator control system is designed to keep the generator output within limits for all flight variables. Generator control systems are often referred to as voltage regulators or generator control units (GCU).

Aircraft generator output can easily be adjusted through control of the generator's magnetic field strength. Remember, the strength of the magnetic field has a direct effect on generator output. More field current means more generator output and vice versa. ***Figure 5-20*** shows a simple generator control used to adjust field current. When field current is controlled, generator output is controlled. Keep in mind, this system is manually adjusted and would not be suitable for aircraft. Aircraft systems must be automatic and are therefore a bit more complex.

There are two basic types of generator controls: electro-mechanical and solid-state (transistorized). The electromechanical type controls are found on older aircraft and tend to require regular inspection and maintenance. Solid-state systems are more modern and typically considered to have better reliability and more accurate generator output control.

FUNCTIONS OF GENERATOR CONTROL SYSTEMS

Most generator control systems perform a number of functions related to the regulation, sensing, and protection of the DC generation system.

VOLTAGE REGULATION

The most basic of the GCU functions is that of voltage regulation. Regulation of any kind requires the regulation unit to take a sample of a generator output and compare that sample to a known reference. If the generator's output voltage falls outside of the set limits, then the regulation unit must provide an adjustment to the generator field current. Adjusting field current controls generator output.

OVERVOLTAGE PROTECTION

The over-voltage protection system compares the sampled voltage to a reference voltage. The over-voltage protection circuit is used to open the relay that controls the field excitation current. It is typically found on more complex generator control systems.

PARALLEL GENERATOR OPERATIONS

On multi-engine aircraft, a paralleling feature must be employed to ensure all generators operate within limits. In general, paralleling systems compare the voltages between two or more generators and adjust the voltage regulation circuit accordingly.

OVER-EXCITATION PROTECTION

When one generator in a paralleled system fails, one of the generators can become overexcited and tends to carry more than its share of the load, if not all of the loads. Basically, this condition causes the generator to produce too much current. If this condition is sensed, the overexcited generator must be brought back within limits, or damage occurs. The over-excitation circuit often works in conjunction with the overvoltage circuit to control the generator.

DIFFERENTIAL VOLTAGE

This function of a control system is designed to ensure all generator voltage values are within a close tolerance before being connected to the load bus. If the output is not within the specified tolerance, then the generator contactor is not allowed to connect the generator to the load bus.

REVERSE CURRENT SENSING

If the generator cannot maintain the required voltage level, it eventually begins to draw current instead of providing it. This situation occurs, for example, if a generator fails. When a generator fails, it becomes a load to the other operating generators or the battery. The defective generator must be removed from the bus. The reverse current sensing function monitors the system for a reverse current. Reverse current indicates that current is flowing to the generator not from the generator. If this occurs, the system opens the generator relay and disconnects the generator from the bus.

GENERATOR CONTROLS FOR HIGH OUTPUT GENERATORS

Most modern high output generators are found on turbine powered corporate-type aircraft. These small business jets and turboprop aircraft employ a generator and starter combined into one unit. This unit is referred to as a starter-generator. A starter-generator has the advantage of combining two units into one housing, saving space and weight. Since the starter-generator performs two tasks, engine starting and generation of electrical power, the control system for this unit is relatively complex.

A simple explanation of a starter-generator shows that the unit contains two sets of field windings. One field is used to start the engine and one used for the generation of electrical power. (***Figure 5-21***)

During the start function, the GCU must energize the series field and the armature causes the unit to act like a motor. During the generating mode, the GCU must disconnect the series field, energize the parallel field, and control the current produced by the armature. At this time, the starter-generator acts like a typical generator. Of course, the GCU must perform all the functions described earlier to control voltage and protect the system. These functions include voltage regulation, reverse current sensing, differential voltage, over-excitation protection, over-voltage protection, and parallel generator operations. A typical GCU is shown in ***Figure 5-22***.

In general, modern GCUs for high-output generators employ solid-state electronic circuits to sense the operations of the generator or starter-generator. The

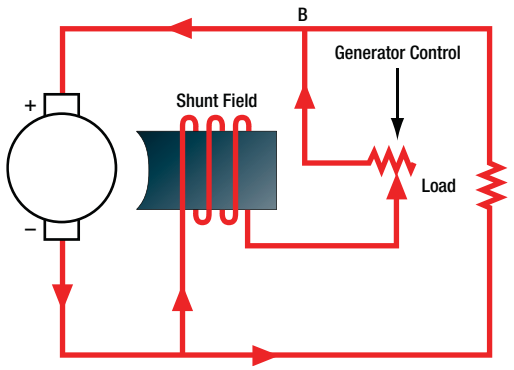


Figure 5-20. Regulation of generator voltage by field rheostat.

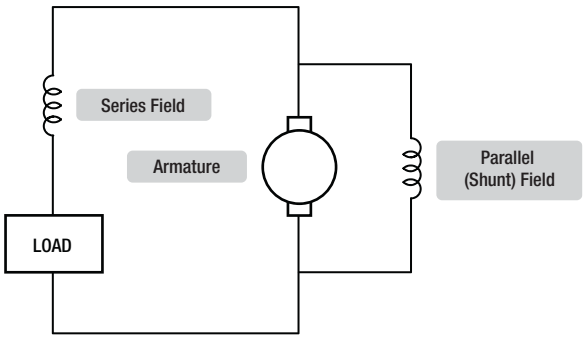


Figure 5-21. Starter-generator.

circuitry then controls a series of relays and/or solenoids to connect and disconnect the unit to various distribution busses. One unit found in almost all voltage regulation circuitry is the zener diode. The zener diode is a voltage sensitive device that is used to monitor system voltage. The zener diode, connected in conjunction to the GCU circuitry, then controls the field current, which in turn controls the generator output.

OTHER VOLTAGE REGULATION

Small aircraft and older large aircraft that use DC generators for primary power must have a means for voltage regulation. The typical device for this is a generator control unit or voltage regulator that modifiers current to the generator field to control generator output power. As flight variables and electrical loads change, the voltage regulator monitors the electrical system and make the appropriate adjustments to ensure proper system voltage and current.

Voltage regulators found on older aircraft and small aircraft are for the low-output generators used on these types of aircraft. They are typically electromechanical devices. Solid-state units are found on more modern aircraft that employ DC alternators and not DC generators. The two most common types of voltage regulator are the carbon pile regulator and the three-unit regulator. Each of these units controls field current using a type of variable resistor. Controlling field current then controls generator output. A simplified generator control circuit is shown in ***Figure 5-23***.

CARBON PILE REGULATORS

The carbon pile regulator controls DC generator output by sending the field current through a stack of carbon disks (the carbon pile). The carbon disks are in series with the generator field. If the resistance of the disks increases, the field current decreases and the generator output goes down. If the resistance of the disks decreases, the field current increases and generator output goes up. As seen in ***Figure 5-24***, a voltage coil is installed in parallel with the generator output leads. The voltage coil acts like an electromagnet that increases or decrease strength as generator output voltage changes. The magnetism of the voltage coil controls the pressure on the carbon stack. The pressure on the carbon stack controls the resistance of the carbon; the resistance of the carbon controls field current and the field current controls generator output.

Carbon pile regulators require regular maintenance to ensure accurate voltage regulation; therefore, most have been replaced on aircraft with more modern systems.



Figure 5-22. Generator control unit (GCU).

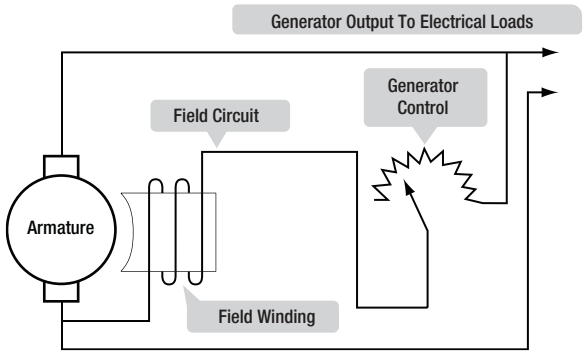


Figure 5-23. Voltage regulator for low-output generator.

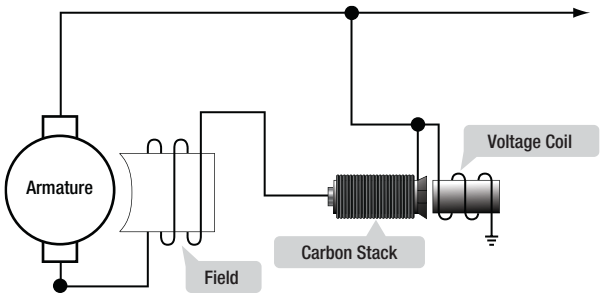


Figure 5-24. Carbon pile regulator.

THREE-UNIT REGULATORS

The three-unit regulator used with DC generator systems is made of three distinct units. Each of these units performs a specific function vital to correct electrical system operation. A typical three-unit regulator consists of three relays mounted in a single housing. Each of the three relays monitors generator outputs and opens or closes the relay contact points according to system needs. A typical three-unit regulator is shown in ***Figure 5-25***.

The voltage regulator section of the three-unit regulator is used to control generator output voltage. The voltage regulator monitors generator output and controls the generator field current as needed. If the regulator senses that system voltage is too high, the relay points open and the current in the field circuit must travel through a resistor. This resistor lowers field current and therefore lowers generator output. Remember, generator output goes down whenever generator field current goes down.

As seen in ***Figure 5-26***, the voltage coil is connected in parallel with the generator output, and it therefore measures the voltage of the system. If voltage gets beyond a predetermined limit, the voltage coil becomes a strong magnet and opens the contact points. If the contact points are open, field current must travel through a resistor and therefore field current goes down. The dotted arrow shows the current flow through the voltage regulator when the relay points are open.

Since this voltage regulator has only two positions (points open and points closed), the unit must constantly be in adjustment to maintain accurate voltage control. During normal system operation, the points are opening and closing at regular intervals. The points are in effect vibrating. This type of regulator is sometimes referred to as a vibrating type regulator. As the points vibrate, the field current raises and lowers and the field magnetism averages to a level that maintains the correct generator output voltage. If the system requires more generator output, the points remain closed longer and vice versa.

There is a current limiter section of the three-unit regulator. It is designed to limit generator output current. This unit contains a relay with a coil wired in series with respect to the generator output. As seen in ***Figure 5-27***, all the generator output current must travel through the current coil of the relay. This creates a relay that is sensitive to the current output of the generator. That is, if generator output current increases, the relay points open and vice versa. The dotted line shows the current flow to the generator field when the current limiter points are open.

It should be noted that, unlike the voltage regulator relay, the current limiter is typically closed during normal flight. Only during extreme current loads must the current limiter points open; at that time, field current is lowered and generator output is kept within limits.

The third unit of a three-unit regulator is used to prevent current from leaving the battery and feeding the generator. This type of current flow would discharge the battery and is opposite of normal operation. It can be thought of as a reverse current situation and the third unit is known as reverse current relay. The simple reverse current relay shown in ***Figure 5-28*** contains both a voltage coil and a current coil.

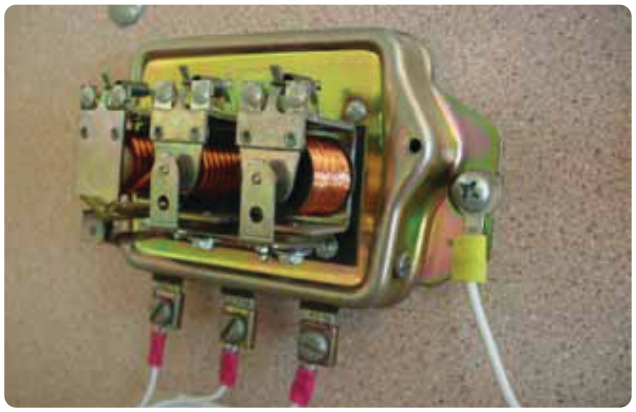


Figure 5-25. The three relays found on this regulator are used to regulate voltage, limit current, and prevent reverse current flow.

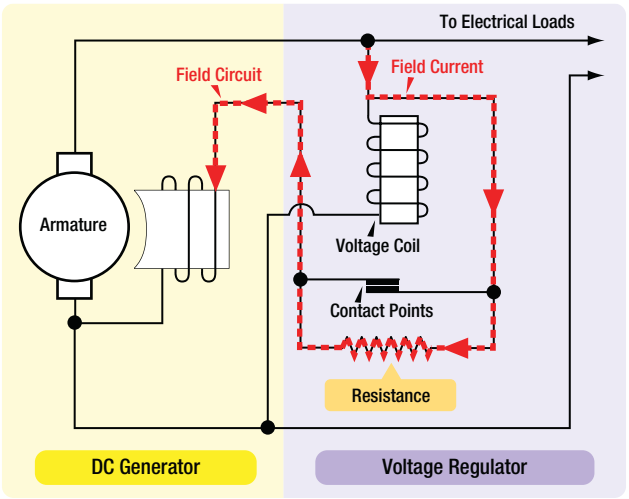


Figure 5-26. Voltage regulator.

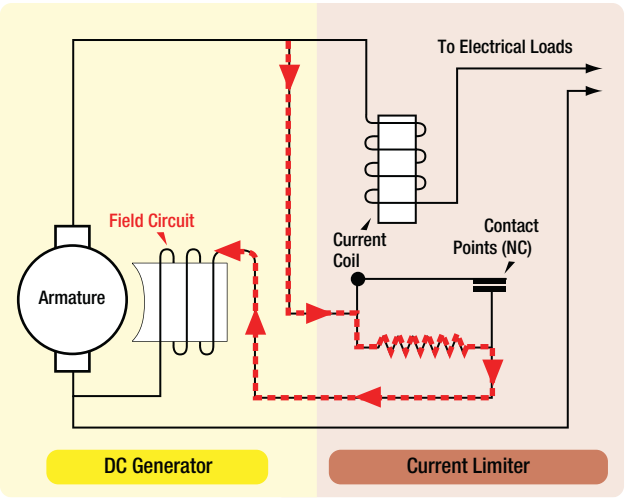


Figure 5-27. Current limiter.

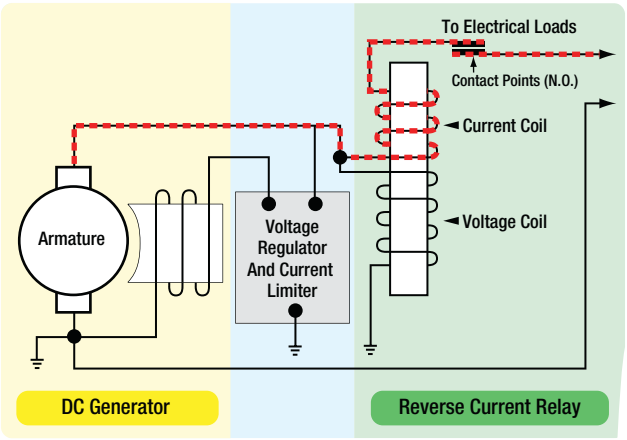


Figure 5-28. Reverse-current relay.

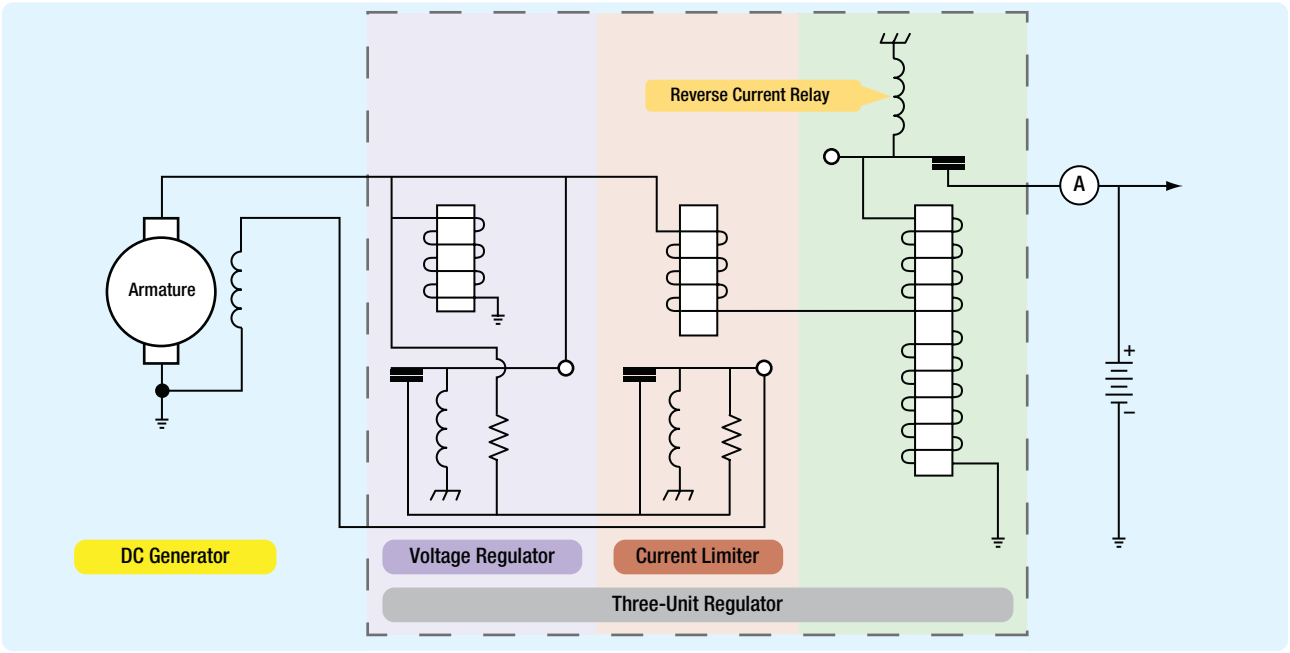


Figure 5-29. Three-unit regulator for variable speed generators.

The voltage coil is wired in parallel to the generator output and is energized any time the generator output reaches its operational voltage. As the voltage coil is energized, the contact points close and the current is then allowed to flow to the aircraft electrical loads, as shown by the dotted lines. The diagram shows the reverse current relay in its normal operating position; the points are closed and current is flowing from the generator to the aircraft electrical loads. As current flows to the loads, the current coil is energized and the points remain closed. If there is no generator output due to a system failure, the contact points open because magnetism in the relay is lost. With the contact points open, the generator is automatically disconnected from the aircraft electrical system, which prevents reverse flow from the load bus to the generator. A typical three-unit regulator for aircraft generators is shown in ***Figure 5-29***.

As seen in ***Figure 5-29***, all three units of the regulator work together to control generator output. The regulator monitors generator output and controls power to the aircraft loads as needed for flight variables. Note that the vibrating regulator just described was simplified for explanation purposes. A typical vibrating regulator found on an aircraft would probably be more complex.

DC ALTERNATORS AND CONTROLS

DC alternators (like generators) change mechanical energy into electrical energy by the process of electromagnetic induction. In general, DC alternators are lighter and more efficient than DC generators. DC alternators and their related controls are found on modern, light, piston-engine aircraft. The alternator is mounted in the engine compartment driven by a v-belt, or drive gear mechanism, which receives power from the aircraft engine. (***Figure 5-30***)

The control system of a DC alternator is used to automatically regulate alternator output power and ensure the correct system voltage for various flight parameters.

DC ALTERNATORS

DC alternators contain two major components: the armature winding and the field winding. The field winding (which produces a magnetic field) rotates inside the armature and, using the process of electromagnetic induction, the armature produces a voltage. This voltage produced by the armature is fed to the aircraft electrical bus and produces a current to power the electrical loads. ***Figure 5-31*** shows a basic diagram of a typical alternator.

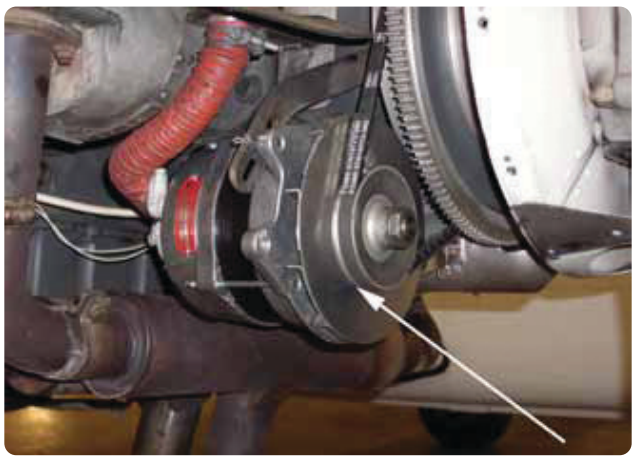


Figure 5-30. DC alternator installation.

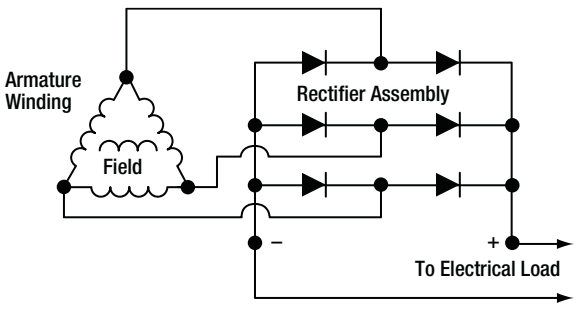


Figure 5-31. Diagram of a typical alternator.

The armature used in DC alternators actually contains three coils of wire. Each coil receives current as the magnetic field rotates inside the armature. The resulting output voltage consists of three distinct AC sine waves, as shown in ***Figure 5-32***. The armature winding is known as a three-phase armature, named after the three different voltage waveforms produced.

Figure 5-33 shows the two common methods used to connect the three phase armature windings: the delta winding and the Y winding. For all practical purposes, the two windings produce the same results in aircraft DC alternators.

Since the three-phase voltage produced by the alternators armature is AC, it is not compatible with typical DC electrical loads and must be rectified (changed to DC). Therefore, the armature output current is sent through a rectifier assembly that changes the three-phase AC to DC. (***Figure 5-33***) Each phase of the three-phase armature overlaps when rectified, and the output becomes a relatively smooth ripple DC. (***Figure 5-34***)

The invention of the diode has made the development of the alternator possible. The rectifier assembly is comprised of six diodes. This rectifier assembly replaces the commutator and brushes found on DC generators and helps to make the alternator more efficient. ***Figure 5-35*** shows the inside of a typical alternator; the

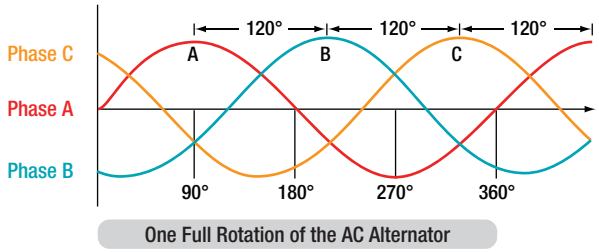


Figure 5-32. Sine waves.

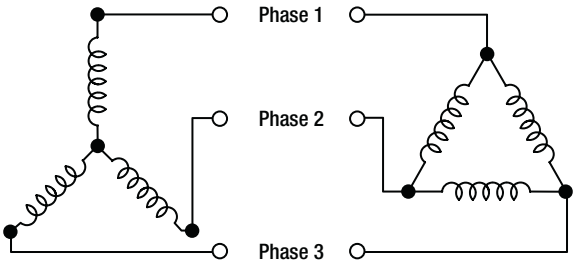


Figure 5-33. Three-phase armature windings: Y on the left and delta winding on the right.

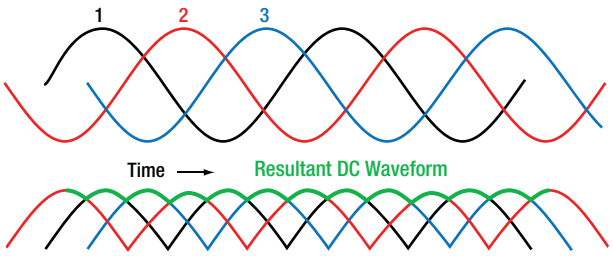


Figure 5-34. Relatively smooth ripple DC.

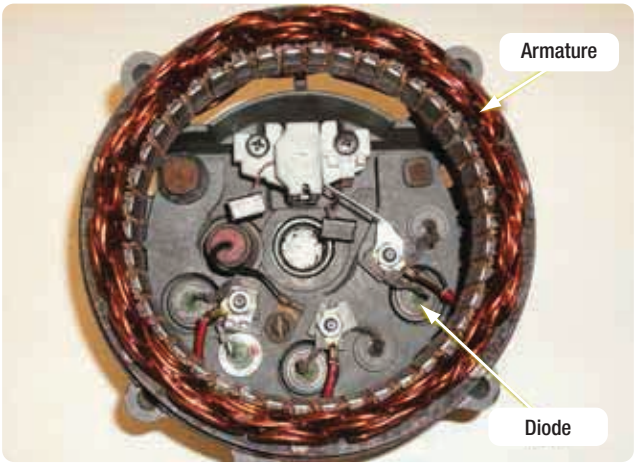


Figure 5-35. Diode assembly.

armature assembly is located on the outer edges of the alternator and the diodes are mounted to the case. The field winding, shown in ***Figure 5-36***, is mounted to a rotor shaft so it can spin inside of the armature assembly.

The field winding must receive current from an aircraft battery in order to produce an electromagnet. Since the field rotates, a set of brushes must be used to send power to the rotating field. Two slip rings are mounted to the rotor and connect the field winding to electrical contacts called brushes. Since the brushes carry relatively low current, the brushes of an alternator are typically smaller than those found inside a DC generator. (***Figure 5-37***) DC alternator brushes last longer and require less maintenance than those found in a DC generator. The alternator case holds the alternator components inside a compact housing that mounts to the engine. Aircraft alternators either produce a nominal 14-volt output or a 25-volt output. The physical size of the alternator is typically a function of the alternator's amperage output. Common alternators for light aircraft range in output form 60–120 amps.

DC ALTERNATOR VOLTAGE REGULATION

Voltage regulators for DC alternators are similar to those found on DC generators. The general concepts are the same in that adjusting alternator field current controls alternator output. Regulators for most DC alternators are either the vibrating-relay type or the solid-state type found on most modern aircraft. Vibrating-relay regulators are similar to those discussed in the section on generator regulators. As the points of the relay open, the field current is lowered and alternator output is lowered and vice versa.

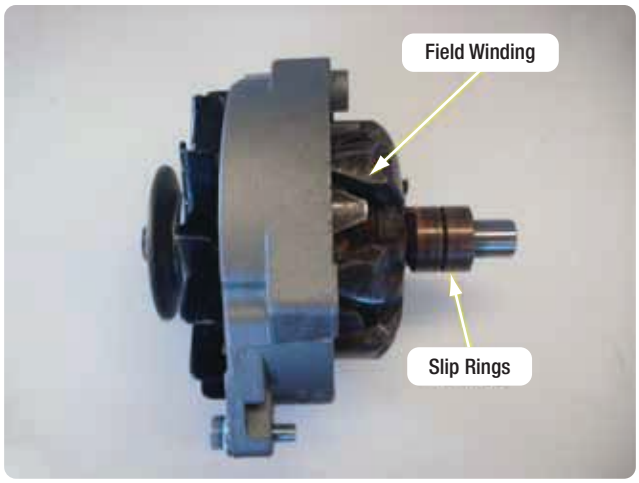


Figure 5-36. Alternator field winding.


Figure 5-37. Alternator brushes.

Solid State Voltage Regulators

Solid-state regulators for modern light aircraft are often referred to as alternator control units (ACUs). These units contain no moving parts and are generally considered to be more reliable and provide better system regulation than vibrating-type regulators. Solid-state regulators rely on transistor circuitry to control alternator field current and alternator output. The regulator monitors alternator output voltage/current and controls alternator field current accordingly. Solid-state regulators typically provide additional protection circuitry not found in vibrating-type regulators. Protection may include over-

voltage and under-voltage protection, over-current protection, and monitoring the alternator for internal defects, such as a defective diode. In many cases, the ACU also provides a warning indication to the pilot if a system malfunction occurs. A key component of any solid-state voltage regulator is known as the zener diode. ***Figure 5-38*** shows the schematic diagram symbol of a zener diode, as well as one installed in an ACU.

The operation of a zener diode is similar to a common diode in that the zener only permits current flow in one direction. This is true until the voltage applied to the zener reaches a certain level. At that predetermined voltage level, the zener then permits current flow with either polarity. This is known as the breakdown or zener voltage.

As an ACU monitors alternator output, the zener diode is connected to system voltage. When the alternator output reaches the specific zener voltage, the diode controls a transistor in the circuit, which in turn controls the alternator field current. This is a simplified explanation of the complete circuitry of an ACU. (***Figure 5-39***) However, it is easy to see how the zener diode and transistor circuit are used in place of an electromechanical relay in a vibrating-type regulator. The use of solid-state components creates a more accurate regulator that requires very little maintenance. The solid state ACU is, therefore, the control unit of choice for modern aircraft with DC alternators.

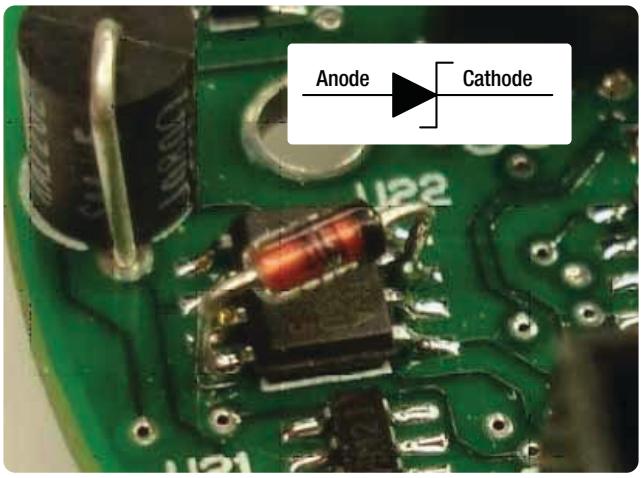


Figure 5-38. Zener diode.

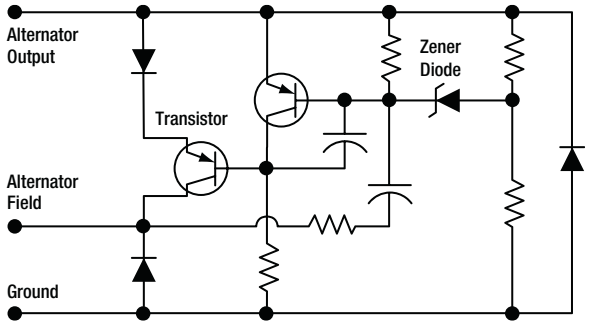


Figure 5-39. ACU circuitry.

POWER DISTRIBUTION

POWER DISTRIBUTION ON SMALL MULTI-ENGINE AIRCRAFT

The power distribution systems found on modern multi-engine aircraft contain several distribution points (busses) and a variety of control and protection components to ensure the reliability of electrical power. As aircraft employ more electronics to perform various tasks, the electrical power systems becomes more complex and more reliable. One means to increase reliability is to ensure more than one power source can be used to power any given load. Another important design concept is to supply critical electrical loads from more than one bus. Twin-engine aircraft, such as a typical corporate jet or commuter aircraft, have two DC generators; they also have multiple distribution busses fed from each generator. ***Figure 5-40*** shows a simplified diagram of the power distribution system for a twin-engine turboprop aircraft. This aircraft contains two starter generator units used to start the engines and generate DC electrical power. The system is typically defined as a split-bus power distribution system since there is a left and right generator bus that splits (shares) the electrical loads by connecting to each sub-bus through a diode and current limiter. The generators are operated in parallel and equally carry the loads.

The primary power supplied for this aircraft is DC, although small amounts of AC are supplied by two inverters. The aircraft diagram shows the AC power distribution at the top and mid left side of the diagram. One inverter is used for main AC power and the second operated in standby and ready as a backup. Both inverters produce 25-volt AC and 115-volt AC. There is an inverter select relay operated by a pilot controlled switch used to choose which inverter is active.

The hot battery bus (right side of ***Figure 5-40***) shows a direct connection to the aircraft battery. This bus is always hot if there is a charged battery in the aircraft. Items powered by this bus may include some basics like the entry door lighting and the aircraft clock, which should always have power available. Other items on this bus would be critical to flight safety, such as fire extinguishers, fuel shut offs, and fuel pumps. During a massive system failure, the hot battery bus is the last bus on the aircraft that should fail.

If the battery switch is closed and the battery relay activated, battery power is connected to the main battery bus and the isolation bus. The main battery

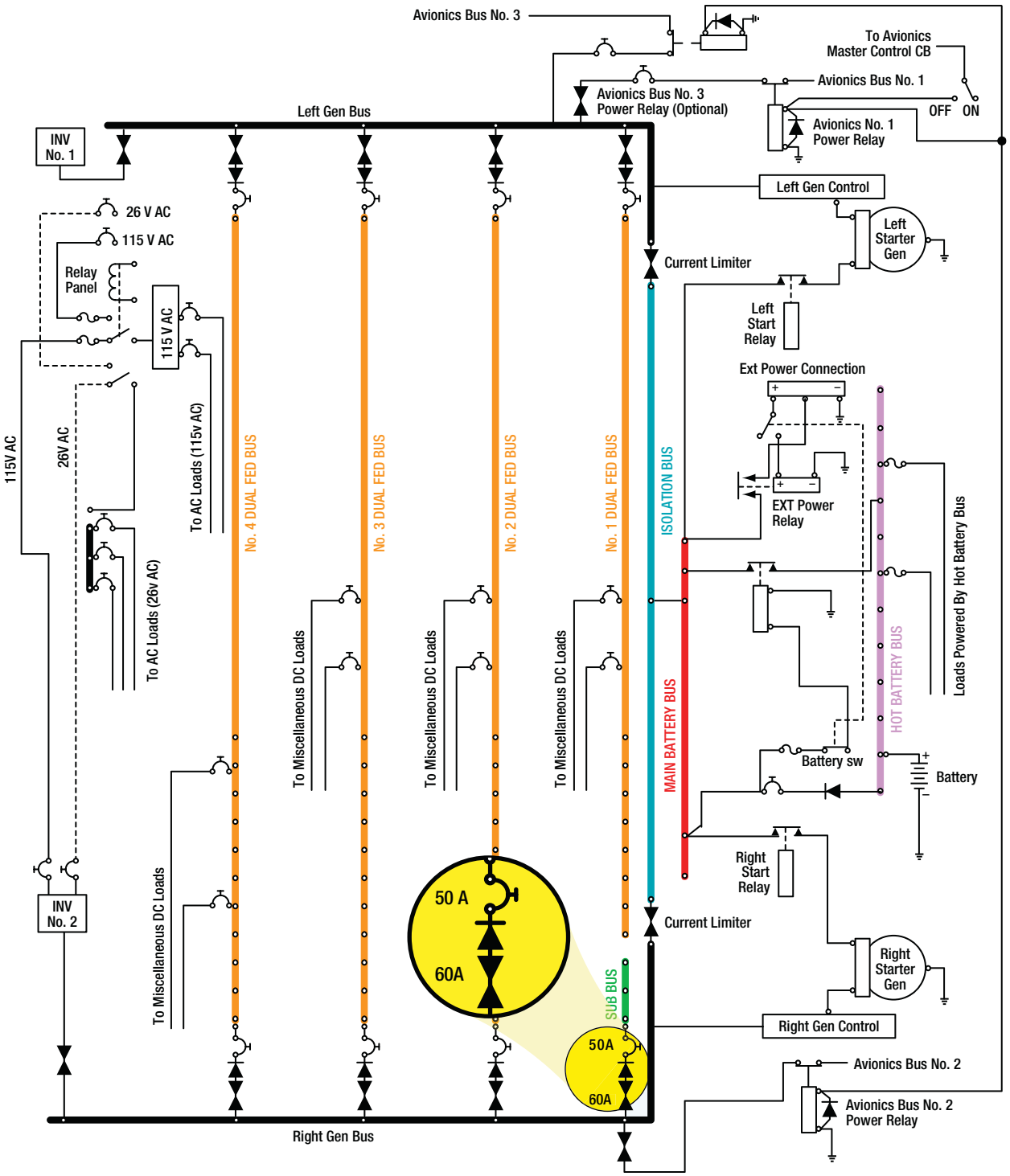


Figure 5-40. Diagram of the power distribution system for a twin-engine turboprop aircraft.

bus carries current for engine starts and external power. So the main battery bus must be large enough to carry the heaviest current loads of the aircraft. It is logical to place this bus as close as practical to the battery and starters and to ensure the bus is well protected from shorts to ground.

The isolation bus connects to the left and right busses and receives power whenever the main battery bus is energized. The isolation bus connects output of the left and right generators in parallel. The output of the two generators is then sent to the loads through additional busses. The generator busses are connected to the isolation bus through a fuse known as a current limiter. Current limiters are high amperage fuses that isolate busses if a short circuit occurs. There are several current limiters used in this system for protection between busses. As can be seen in ***Figure 5-40***, a current limiter symbol looks like two triangles pointed toward each other. The current limiter between the isolation bus and the main generator busses are rated at 325 amps and can only be replaced on the ground. Most current limiters are designed for ground replacement only and only after the malfunction that caused the excess current draw is repaired.

The left and right DC generators are connected to their respective main generator busses. Each generator feeds its respective bus, and since the busses are connected under normal circumstances, the generators operate in parallel. Both generators feed all loads together. If one generator fails or a current limiter opens, the generators can operate independently. This design allows for redundancy in the event of failure and provides battery backup in the event of a dual generator failure.

In the center of ***Figure 5-40*** are four dual-feed electrical busses. These busses are considered dual-feed since they receive power from both the left and right generator busses. If a fault occurs, either generator bus can power any or all loads on a dual-feed bus. During the design phase of the aircraft, the electrical loads must be evenly distributed between each of the dual-feed busses. It is also important to power redundant systems from different busses. For example, the pilot's windshield heat would be powered by a different bus from the one that powers the copilot's windshield heat. If one bus fails, at least one windshield heat continues to work properly, and the aircraft can be landed safely in icing conditions.

Notice that the dual-feed busses are connected to the main generator busses through both a current limiter and a diode. Remember, a diode allows current flow in only one direction. (***Figure 5-41***)

The current can flow from the generator bus to the dual-feed bus, but the current cannot flow from the dual fed bus to the main generator bus. The diode is placed in the circuit so the main bus must be more positive than the sub bus for current flow. This circuit also contains a current limiter and a circuit breaker. The circuit breaker is located on the flight deck and can be reset by the pilot. The current limiter can only be replaced on the ground by a technician. The circuit breaker is rated at a slightly lower current value than the current limiter; therefore, the circuit breaker should open if a current overload exists. If the circuit breaker fails to open, the current limiter provides backup protection and disconnects the circuit.

POWER DISTRIBUTION ON LARGE AIRCRAFT

SPLIT BUS SYSTEMS

Modern twin-engine aircraft, such as the Boeing 737, 757, 777, Airbus A-300, A-320, and A-310, employ a split-bus power distribution system. During normal conditions, each engine-driven AC generator powers only one main AC bus. The busses are kept split from each other, and two generators can never power the same bus simultaneously. This is very important since the generator output current is not phase regulated. (If two out-of-phase generators were connected to the same bus, damage to the system would occur.) The split-bus system does allow both engine-driven

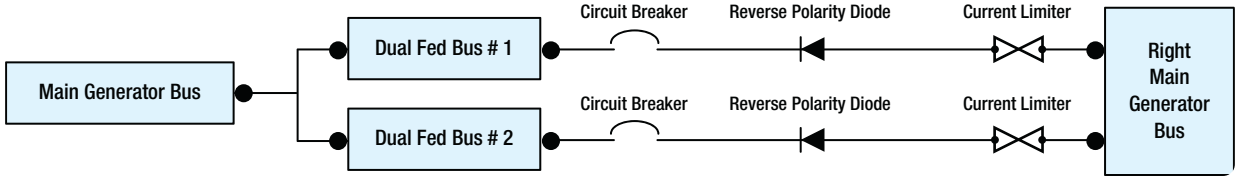


Figure 5-41. Dual-feed bus system.

generators to power any given bus, but not at the same time. Generators must remain isolated from each other to avoid damage. The GCUs and BPCU ensures proper generator operation and power distribution.

On all modern split bus systems, the APU can be started and operated during flight. This allows the APU generator to provide back-up power in the event of a main generator failure. A fourth emergency generator powered by the ram air turbine is also available if the other generators fail.

The four AC generators are shown at the bottom of ***Figure 5-42***. These generators are connected to their respective busses through the generator breakers (GB's). For example, generator 1 sends current through GB1 to AC bus 1. AC bus 1 feeds a variety of primary electrical loads, and also feeds sub-busses that in turn power additional loads. With both generators operating and all systems normal, AC bus 1 and AC bus 2 are kept isolated. Typically during flight, the auxiliary power breaker (APB) (bottom center of ***Figure 5-42***) would be open and the APU generator off; the emergency generator (bottom right) would also be off and disconnected. If generator one should fail, the following happens:

1. The GB 1 is opened by the GCU to disconnect the failed generator.
2. The BPCU closes BTB 1 and BTB 2. This supplies AC power to AC bus 1 from generator 2.
3. The pilots start the APU and connect the APU generator. At that time, the BPCU and GCUs move the appropriate BTBs to correctly configure the system so the APU powers bus 1 and generator 2 powers bus 2. Once again, two AC generators operate independently to power AC bus 1 and 2.

If all generators fail, AC is also available through the static inverter (center of ***Figure 5-42***). The inverter is powered from the hot battery bus and used for essential AC loads if all AC generators fail. Of course, the GCUs and BPCU take the appropriate actions to disconnect defective units and continue to feed essential AC loads using inverter power.

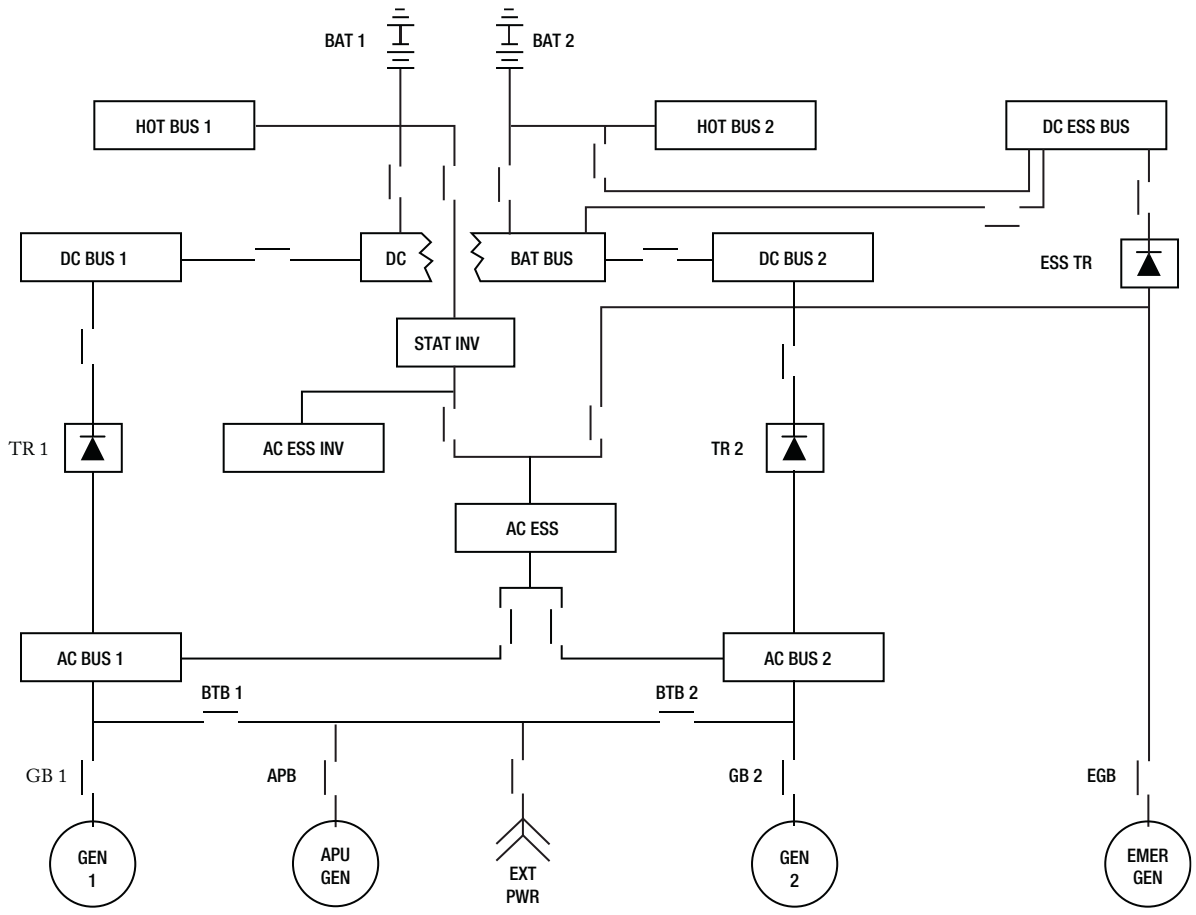


Figure 5-42. Schematic of split-bus power distribution system.

To produce DC power, AC bus 1 sends current to its transformer rectifier (TR), TR 1 (center left of ***Figure 5-42***). The TR unit is used to change AC to DC. The TR contains a transformer to step down the voltage from 115-volt AC to 25-volt AC and a rectifier to change the 25-volt AC to 25- volt DC. The output of the TR is therefore compatible with the aircraft battery at 25-volt DC. Since DC power is not phase sensitive, the DC busses are connected during normal operation. In the event of a bus problem, the BPCU may isolate one or more DC busses to ensure correct distribution of DC power. This aircraft contains two batteries that are used to supply emergency DC power.

PARALLEL BUS SYSTEMS

Multi-engine aircraft, such as the Boeing 727, MD-11, and the early Boeing 747, employ a parallel power distribution system. During normal flight conditions, all engine-driven generators connect together and power the AC loads. In this configuration, the generators are operated in parallel; hence the name parallel power distribution system. In a parallel system, all generator output current must be phase regulated. Before generators are connected to the same bus, their output frequency must be adjusted to ensure the AC output reaches the positive and negative peaks simultaneously. During the flight, generators must maintain this in-phase condition for proper operation.

One advantage of parallel systems is that in the event of a generator failure, the busses are already connected and the defective generator need only be isolated from the system. A paralleling bus, or synchronizing bus, is used to connect the generators during flight. The synchronizing bus is often referred to as the sync bus. Most of these systems are less automated and require that flight crew monitor systems and manually control bus contactors. BTBs are operated by the flight crew through the electrical control panel and used to connect all necessary busses. GBs are used to connect and disconnect the generators.

Figure 5-43 shows a simplified parallel power distribution system. This aircraft employs three main-engine driven generators and one APU generator. The APU (bottom

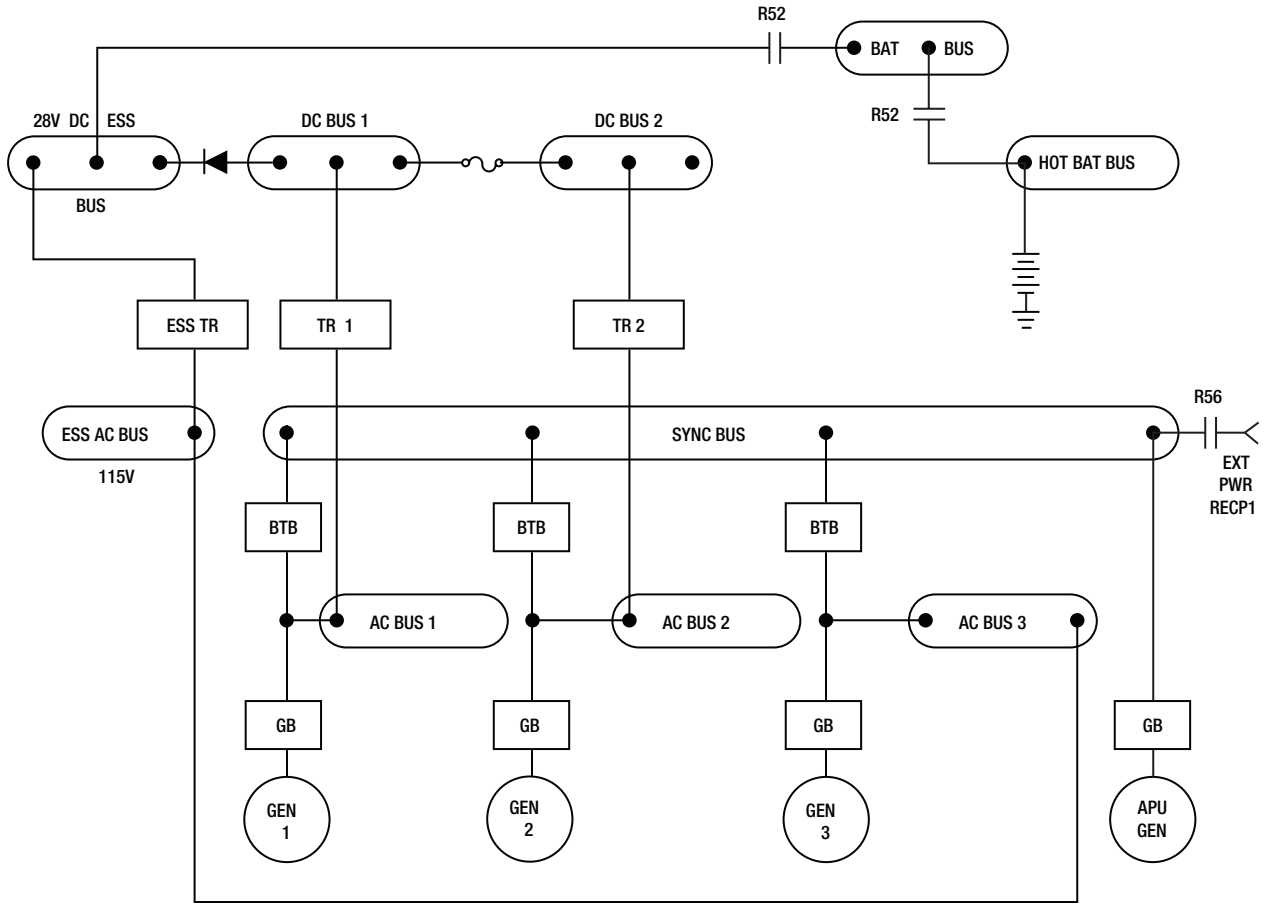


Figure 5-43. Parallel power distribution system.

right) is not operational in flight and cannot provide backup power. The APU generator is for ground operations only. The three main generators (bottom of ***Figure 5-43***) are connected to their respective AC bus through GBs one, two, and three. The AC busses are connected to the sync bus through three BTBs. In this manner, all three generators share the entire AC electrical loads. Keep in mind, all generators connected to the sync bus must be in phase. If a generator fails, the flight crew would simply isolate the defective generator and the flight would continue without interruption.

The number one and two DC busses (***Figure 5-43*** top left) are used to feed the DC electrical loads of the aircraft. DC bus 1 receives power form AC bus 1 though TR1. DC bus 2 is fed in a similar manner from AC bus 2. The DC busses also connect to the battery bus and eventually to the battery. The essential DC bus (top left) can be fed from DC bus 1 or the essential TR. A diode prevents the essential DC bus from powering DC bus 1. The essential DC bus receives power from the essential TR, which receives power from the essential AC bus. This provides an extra layer of redundancy since the essential AC bus can be isolated and fed from any main generator. ***Figure 5-43*** shows generator 3 powering the essential AC bus.

SPLIT-PARALLEL SYSTEMS

A split-parallel bus basically employs the best of both split-bus and the parallel-bus systems. The split-parallel system is found on the Boeing 747-400 and contains four generators driven by the main engines and two APU-driven generators. The system can operate with all generators in parallel, or the generators can be operated independently as in a split-bus system. During a normal flight, all four engine-driven generators are operated in parallel. The system is operated in split-bus mode only under certain failure conditions or when using external power. The Boeing 747-400 split-parallel system is computer controlled using four GCU and two BPCU. There is one GCU controlling each generator; BPCU 1 controls the left side bus power distribution, and BPCU 2 controls the right side bus power. The GCUs and BPCUs operate similarly to those previously discussed under the split-bus system.

Figure 5-44 shows a simplified split-parallel power distribution system. The main generators (top of ***Figure 5-44***) are driven by the main turbine engines. Each generator is connected to its load bus through a generator control breaker (GCB). The generator control unit closes the GCB when the pilot calls for generator power and all systems are operating normally. Each load bus is connected to various electrical systems and additional sub-busses. The BTB are controlled by the BPCU and connect each load bus to the left and right sync bus. A

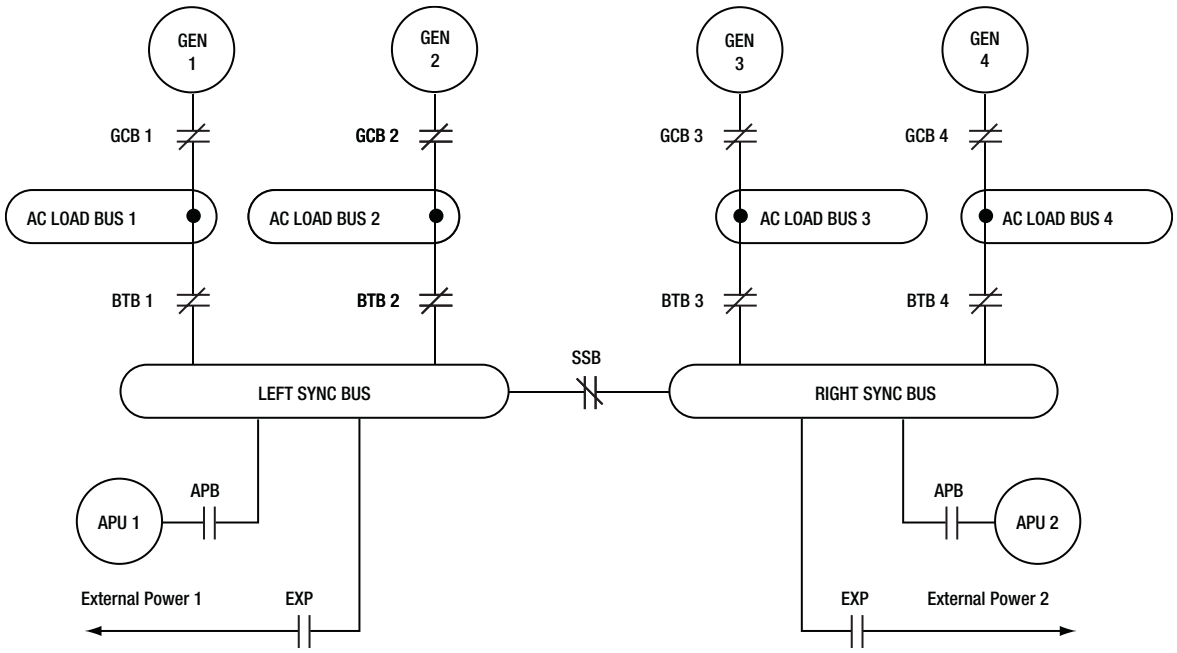


Figure 5-44. Split-parallel distribution system.

split systems breaker (SSB) is used to connect the left and right sync busses and is closed during a normal flight. With the SSB, GCBs, and BTBs, in the closed position the generators operate in parallel. When operating in parallel, all generators must be in phase.

If the aircraft electrical system experiences a malfunction, the control units make the appropriate adjustments to ensure all necessary loads receive electrical power. For example, if generator 1 fails, GCU 1 detects the fault and command GCB 1 to open. With GCB 1 open, load bus 1 now feeds from the sync bus and the three operating generators. In another example, if load bus 4 should short to ground, BPCU 4 opens the GCB 4 and BTB 4. This isolates the shorted bus (load bus 4). All loads on the shorted bus are no longer powered, and generator 4 is no longer available. However, with three remaining generators operational, the flight continues safely.

As with all large aircraft, the Boeing 747-400 contains a DC power distribution system. The DC system is used for battery and emergency operations. The DC system is similar to those previously discussed, powered by TR units. The TRs are connected to the AC busses and convert AC into 25-volt DC.

The DC power systems are the final backups in the event of a catastrophic electrical failure. The systems most critical to fly the aircraft can typically receive power from the battery. This aircraft also contains two static inverters to provide emergency AC power when needed.

INVERTERS, TRANSFORMERS AND RECTIFIERS

INVERTERS

A modern inverter is a solid-state device that converts DC power into AC power. The electronic circuitry within an inverter is quite complex; however, for an aircraft technician's purposes, the inverter is simply a device that uses DC power, then feeds power to an AC distribution bus. Many inverters supply both 25-volt AC, as well as 115-volt AC. The aircraft can be designed to use either voltage or both simultaneously. If both voltages are used, the power must be distributed on separate 25-and 115-volt AC busses. Typical AC inverter output frequency is 400 cycles per second (cps).

There are two basic types of inverters: the rotary and the static. Either type can be single phase or multiphase. The multiphase inverter is lighter for the same power rating than the single phase, but there are complications in distributing multiphase power and in keeping the loads balanced. Most modern aircraft use solid state static inverters rather than the rotary type.

ROTARY INVERTERS

There are many sizes, types, and configurations of rotary inverters. Such inverters are essentially AC generators and DC motors in one housing. The generator field, or armature, and the motor field, or armature, are mounted on a common shaft that will rotate within the housing. One common type of rotary inverter is the permanent magnet inverter.

Permanent Magnet Rotary Inverter

A permanent magnet inverter is composed of a DC motor and a permanent magnet AC generator assembly. Each has a separate stator mounted within a common housing. The motor armature is mounted on a rotor and connected to the DC supply through a commutator and brush assembly. The motor field windings are mounted on the housing and connected directly to the DC supply. A permanent magnet rotor is mounted at the opposite end of the same shaft as the motor armature, and the stator windings are mounted on the housing, allowing AC to be taken from the inverter without the use of brushes. ***Figure 5-45*** shows an internal wiring diagram for this type of rotary inverter. The generator rotor has six poles, magnetized to provide alternate north and south poles about its circumference.

When the motor field and armature are excited, the rotor will begin to turn. As the rotor turns, the permanent magnet will rotate within the AC stator coils, and the magnetic flux developed by the permanent magnets will be cut by the conductors in the AC stator coils. An AC voltage will be produced in the windings whose polarity will change as each pole passes the windings. This type inverter may be made multiphase by placing more AC stator coils in the housing in order to shift the phase the proper amount in each coil.

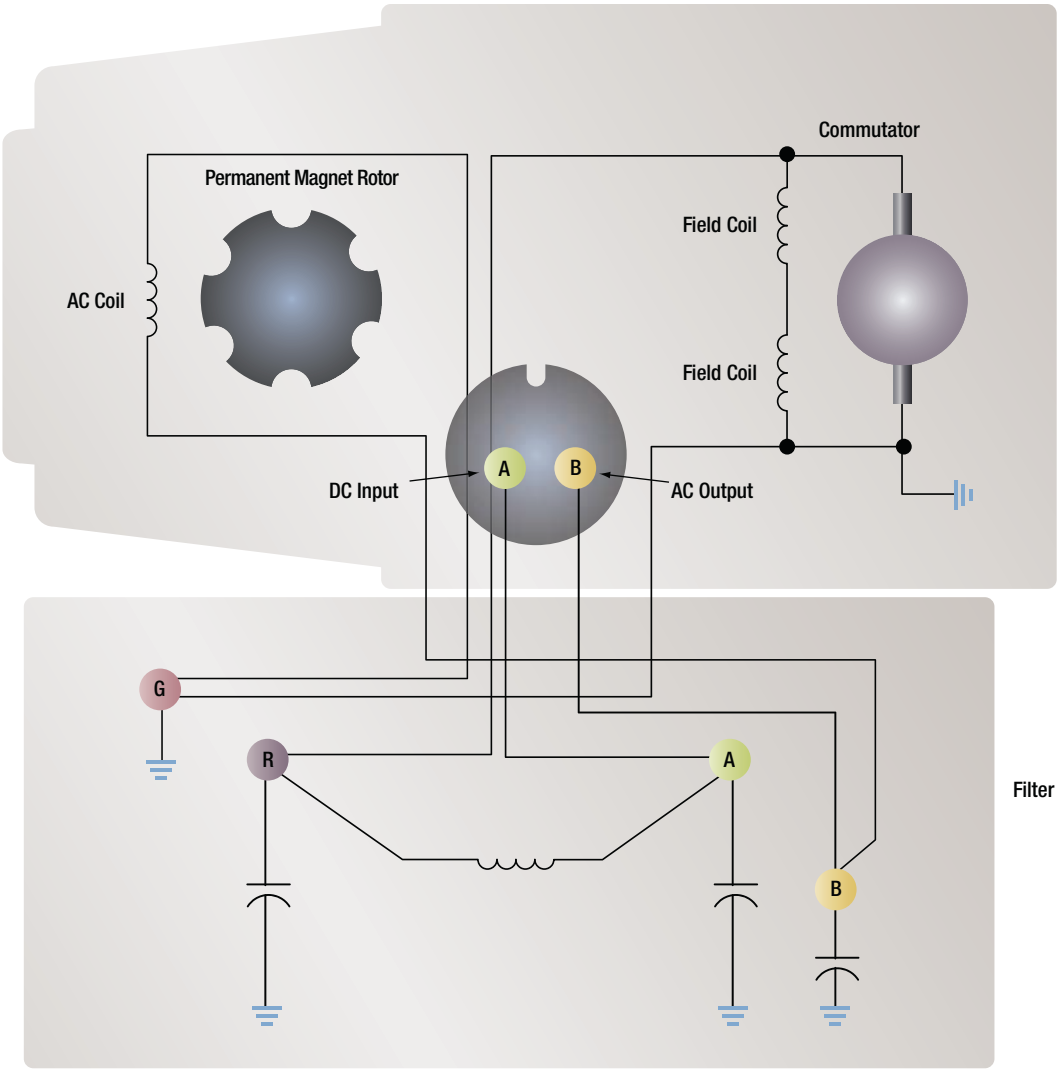


Figure 5-45. Internal wiring diagram of single-phase permanent magnet rotary inverter.

As the name of the rotary inverter indicates, it has a revolving armature in the AC generator section. The illustration in ***Figure 5-46*** shows the diagram of a revolving armature, three phase inverter.

The DC motor in this inverter is a four pole, compound wound motor. The four field coils consist of many turns of fine wire, with a few turns of heavy wire placed on top. The fine wire is the shunt field, connected to the DC source through a filter and to ground through a centrifugal governor. The heavy wire is the series field, which is connected in series with the motor armature. The centrifugal governor controls the speed by shunting a resistor that is in series with the shunt field when the motor reaches a certain speed.

The alternator is a three-phase, four-pole, star-connected AC generator. The DC input is supplied to the generator field coils and connected to ground through a voltage regulator. The output is taken off the armature through three slip rings to provide three-phase power.

The inverter would be a single-phase inverter if it had a single armature winding and one slip ring. The frequency of this type unit is determined by the speed of the motor and the number of generator poles.

Inductor-Type Rotary Inverter

Inductor-type inverters use a rotor made of soft iron laminations with grooves cut laterally across the surface to provide poles that correspond to the number of stator poles, as illustrated in ***Figure 5-47***. The field coils are wound on one set of stationary poles and the AC armature coils on the other set of stationary poles.

When DC is applied to the field coils, a magnetic field is produced. The rotor turns within the field coils and, as the poles on the rotor align with the stationary poles, a low reluctance path for flux is established from the field pole through the rotor poles to the AC armature pole and through the housing back to the field pole. In this circumstance, there will be a large amount of magnetic flux linking the AC coils.

When the rotor poles are between the stationary poles, there is a high reluctance path for flux, consisting mainly of air; then, there will be a small amount of magnetic flux linking the AC coils. This increase and decrease in flux density in the stator induces an alternating current in the AC coils.

The number of poles and the speed of the motor determine the frequency of this type of inverter. The DC stator field current controls the voltage. A cutaway view of an inductor-type rotary inverter is shown in ***Figure 5-48***.

Figure 5-49 is a simplified diagram of a typical aircraft AC power distribution system, utilizing a main and a standby rotary inverter system.

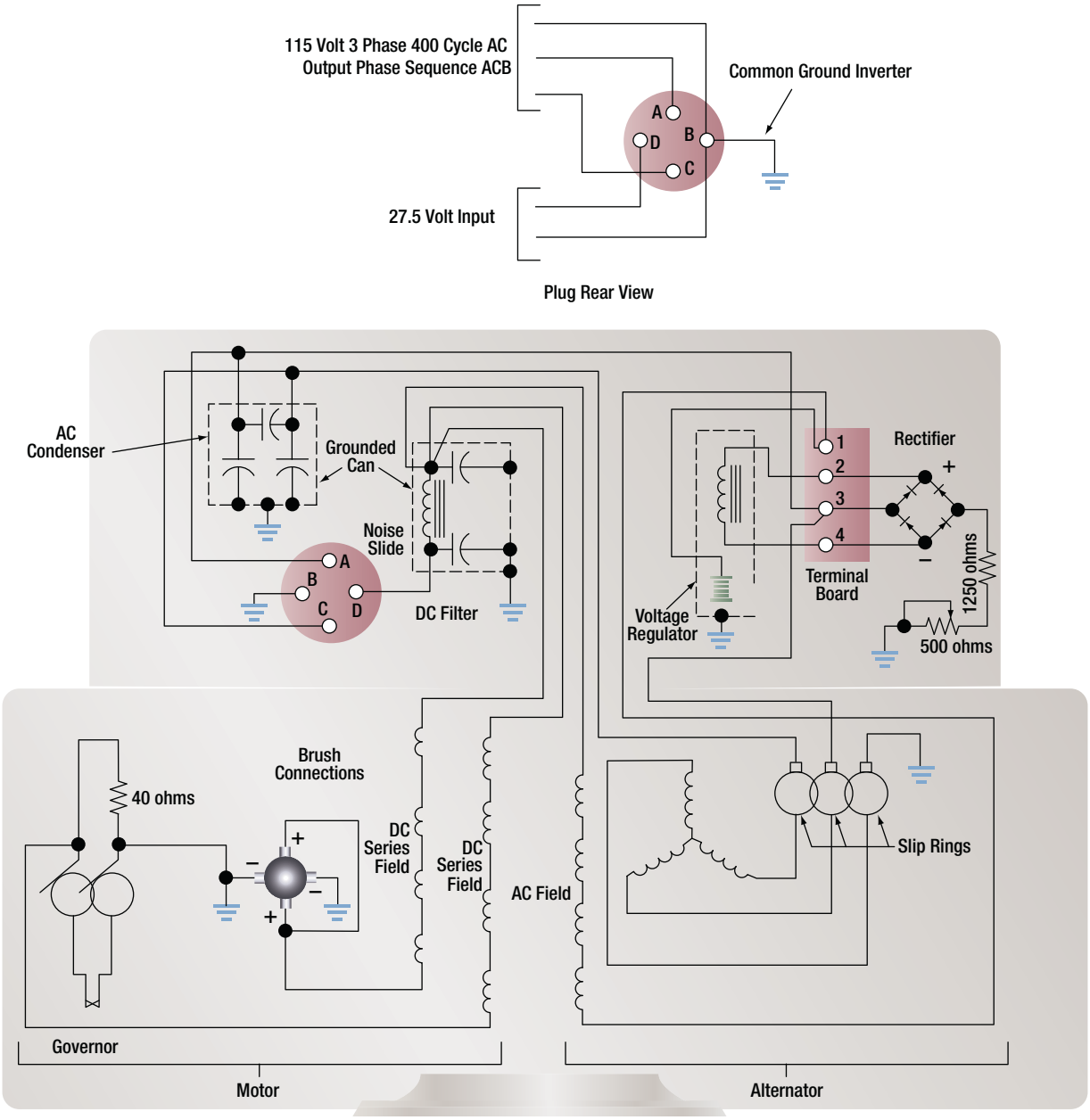


Figure 5-46. Internal wiring diagram of three-phase, revolving armature.

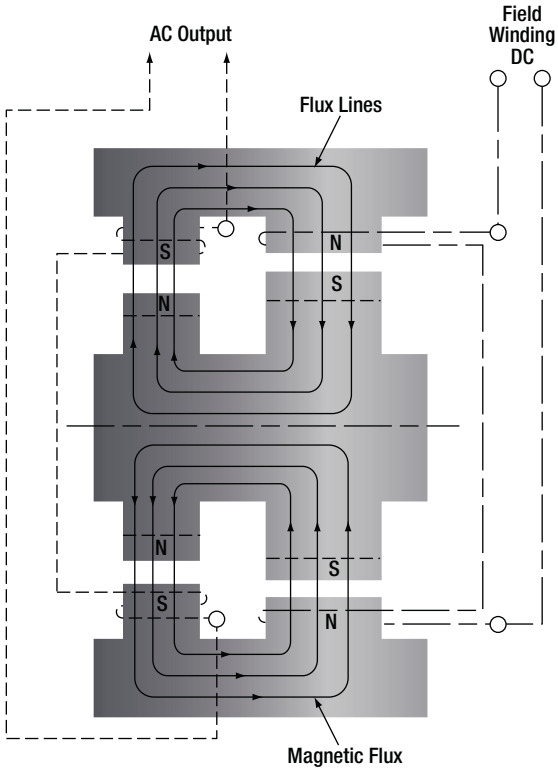


Figure 5-47. Diagram of basic inductor-type inverter.

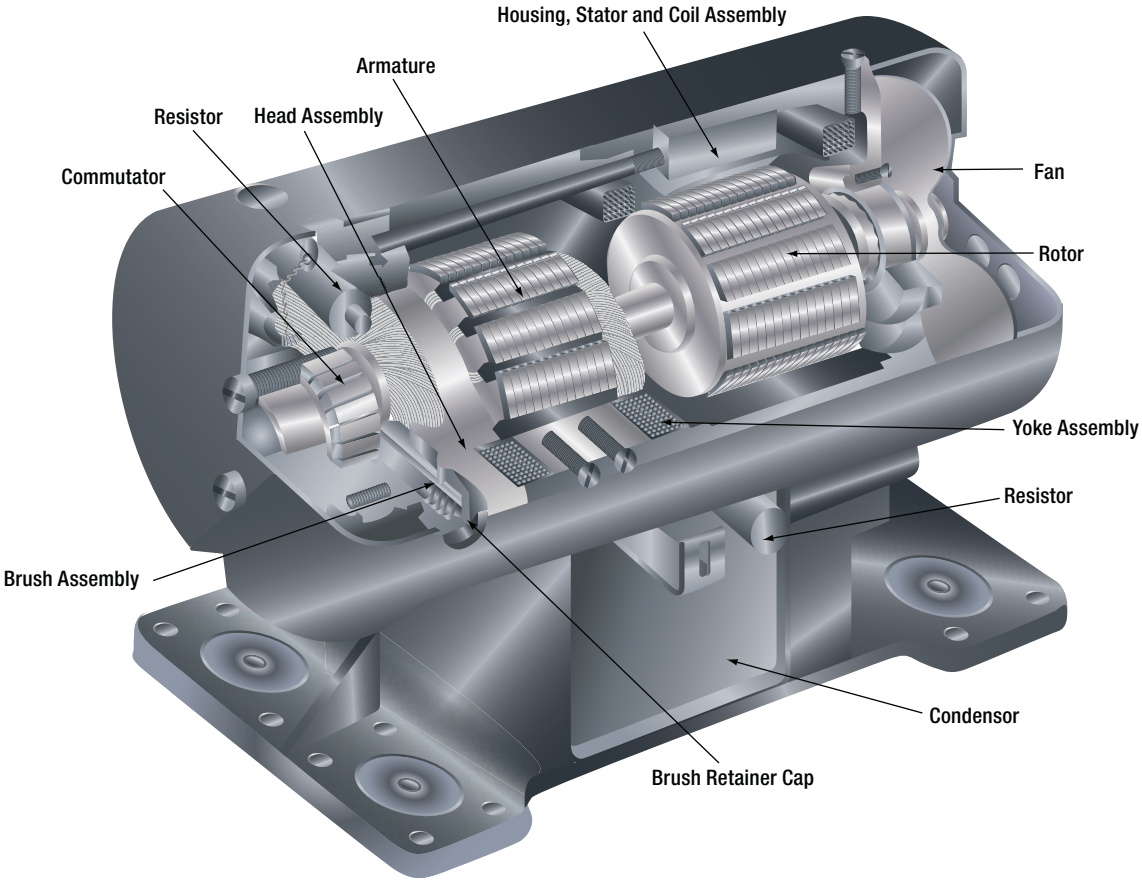


Figure 5-48. Cutaway view of inductor-type rotary inverter.

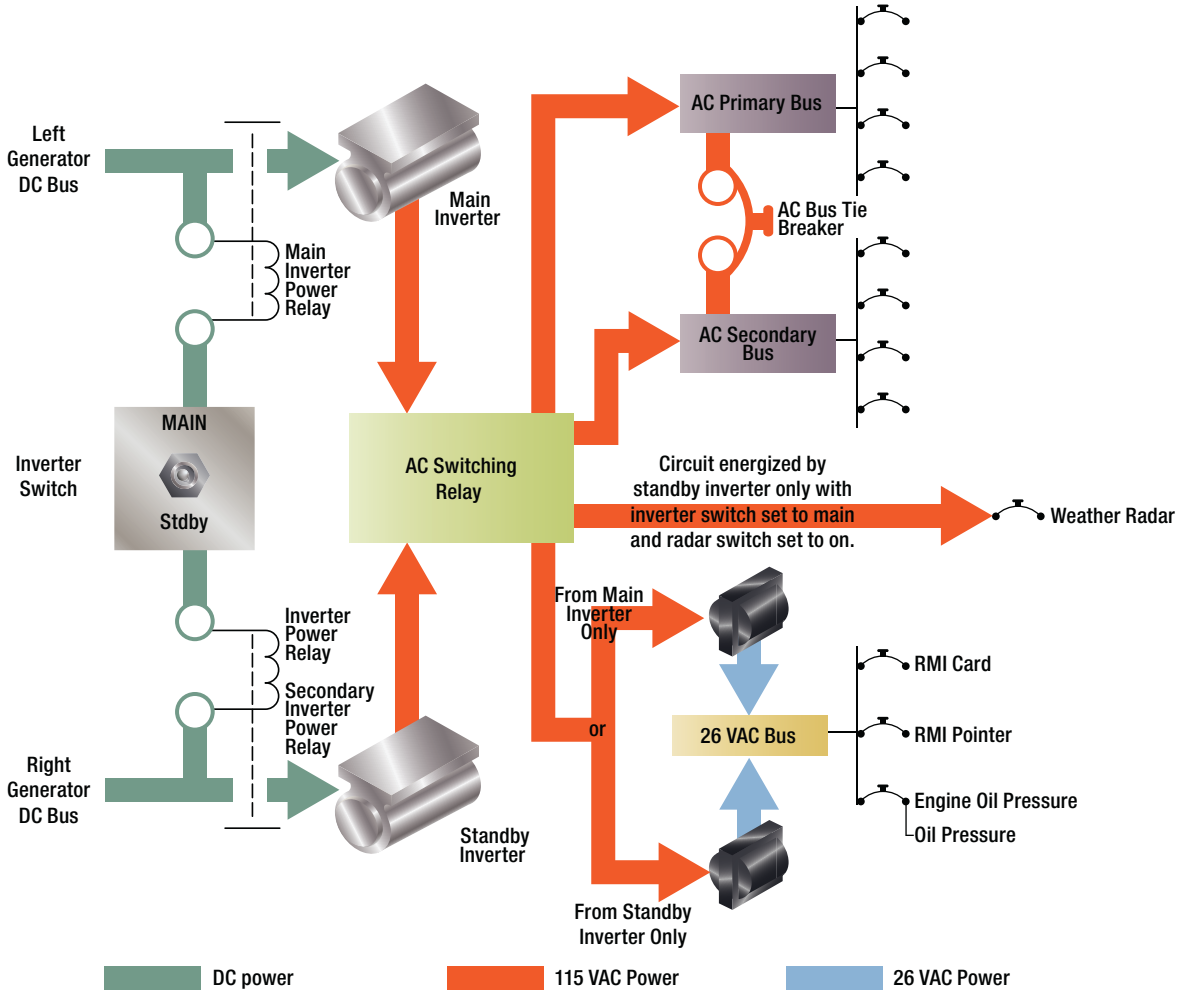


Figure 5-49. A typical aircraft AC power distribution system using main and standby rotary inverters.

STATIC INVERTERS

In many applications where continuous DC voltage must be converted to alternating voltage, static inverters are used in place of rotary inverters or motor generator sets. The rapid progress made by the semiconductor industry is extending the range of applications of such equipment into voltage and power ranges that would have been impractical a few years ago. Some such applications are power supplies for frequency sensitive military and commercial AC equipment, aircraft emergency AC systems, and conversion of wide frequency range power to precise frequency power.

The use of static inverters in small aircraft also has increased rapidly in the last few years, and the technology has advanced to the point that static inverters are available for any requirement filled by rotary inverters. For example, 250 VA emergency AC supplies operated from aircraft batteries are in production, as are 2 500 VA main AC supplies operated from a varying frequency generator supply. This type of equipment has certain advantages for aircraft applications, particularly the absence of moving parts and the adaptability to conduction cooling.

Static inverters, referred to as solid-state inverters, are manufactured in a wide range of types and models, which can be classified by the shape of the AC output waveform and the power output capabilities. One of the most commonly used static inverters produces a regulated sine wave output. A block diagram of a typical regulated sine wave static inverter is shown in ***Figure 5-50***. This inverter converts a low DC voltage into higher AC voltage. The AC output voltage is held to a very small voltage tolerance, a typical variation of less than 1 percent with a full input load change. Output taps are normally provided to permit selection of various voltages; for example, taps may be provided for a 105, 115, and 125 volt AC outputs. Frequency regulation is typically within a range of one cycle for a 0 – 100 percent load change.

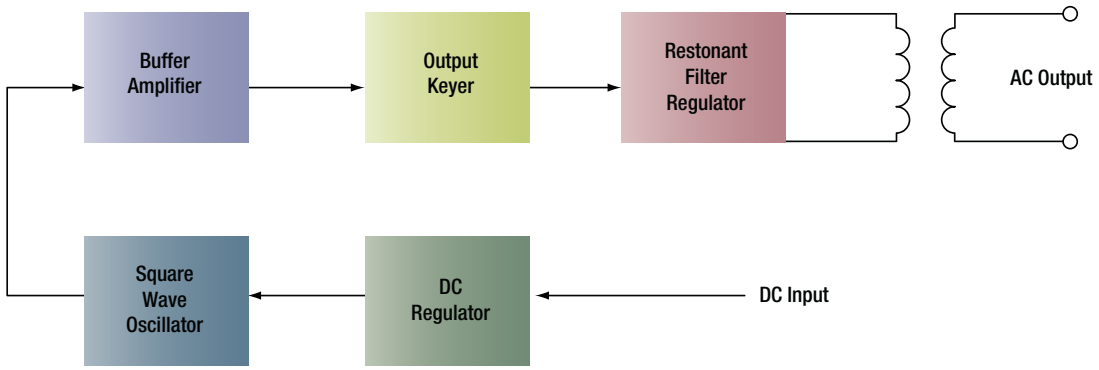


Figure 5-50. Regulated sine wave static inverter.

Variations of this type of static inverter are available, many of which provide a square wave output. Since static inverters use solid-state components, they are considerably smaller, more compact, and much lighter in weight than rotary inverters. Depending on the output power rating required, static inverters that are no larger than a typical airspeed indicator can be used in aircraft systems.

Some of the features of static inverters are:

1. High efficiency.
2. Low maintenance, long life.
3. No warmup period required.
4. Capable of starting under load.
5. Extremely quiet operation.
6. Fast response to load changes.

Static inverters are commonly used to provide power for such frequency sensitive instruments as the attitude gyro and directional gyro. They also provide power for autosyn and magnesyn indicators and transmitters, rate gyros, radar, and other airborne applications. ***Figure 5-51*** is a schematic of a typical small jet aircraft auxiliary battery system. It shows the battery as input to the inverter, and the output inverter circuits to various subsystems.

TRANSFORMERS

A transformer changes electrical energy of a given voltage into electrical energy at a different voltage level. It consists of two coils that are not electrically connected, but are arranged so that the magnetic field surrounding one coil cuts through the other coil. When an alternating voltage is applied to (across) one coil, the varying magnetic field set up around that coil creates an alternating voltage in the other coil by mutual induction.

A transformer can also be used with pulsating DC, but a pure DC voltage cannot be used, since only a varying voltage creates the varying magnetic field that is the basis of the mutual induction process.

A transformer consists of three basic parts. (***Figure 5-52***) These are an iron core which provides a circuit of low reluctance for magnetic lines of force, a primary winding which receives the electrical energy from the source of applied voltage, and a secondary winding which receives electrical energy by induction from the primary coil.

The primary and secondary of this closed core transformer are wound on a closed core to obtain maximum inductive effect between the two coils. There are two classes of transformers: (1) voltage transformers used for stepping up or stepping down voltages, and (2) current transformers used in instrument circuits.

In voltage transformers, the primary coils are connected in parallel across the supply voltage as shown in ***Figure 5-53A***. The primary windings of current transformers are connected in series in the primary circuit (***Figure 5-53B***). Of the two types, the voltage transformer is the more common.

There are many types of voltage transformers. Most of these are either step-up or step-down transformers. The factor that determines whether a transformer is a step-up, or step-down type is the "turns" ratio. The turns ratio is the ratio of the number of turns in the primary winding to the number of turns in the secondary winding.

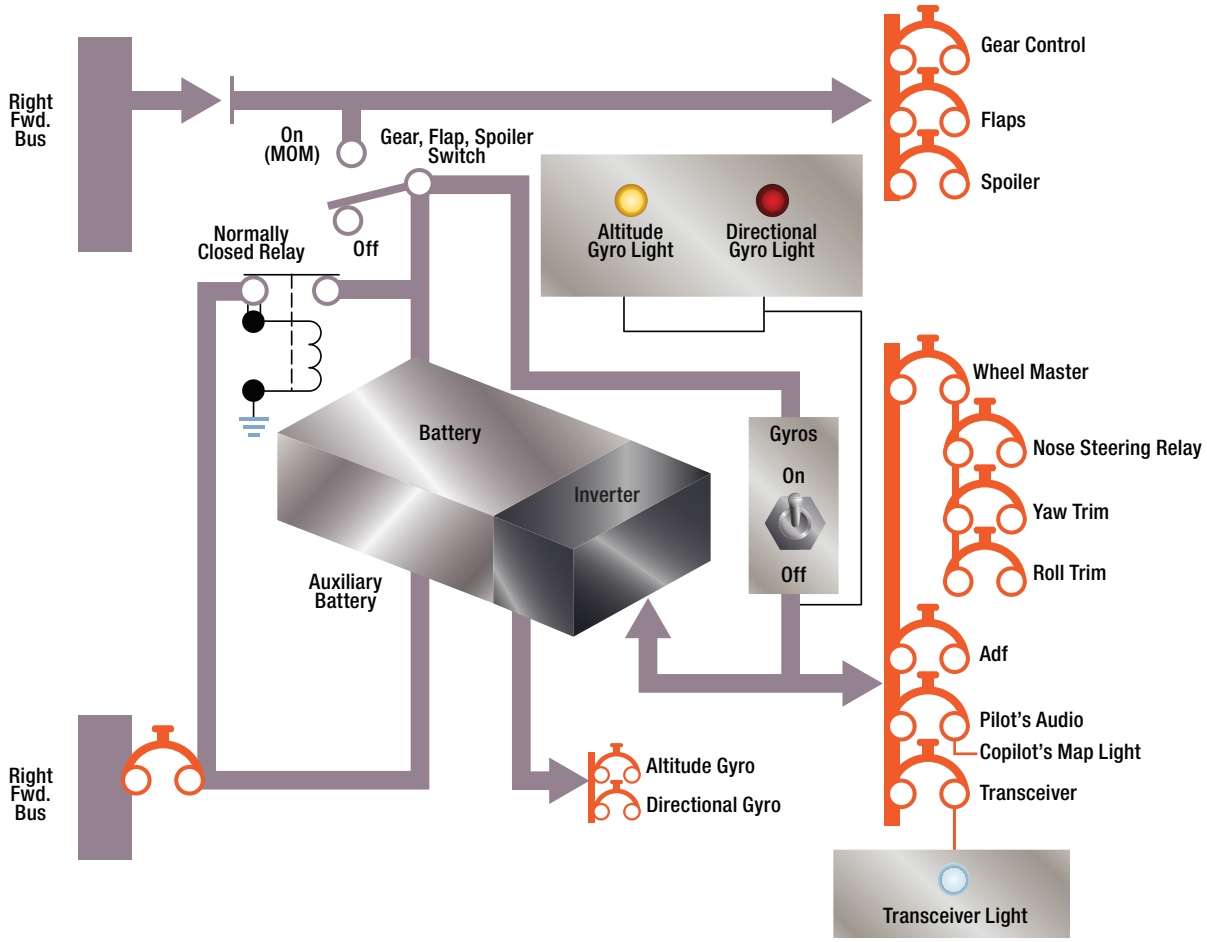


Figure 5-51. Auxiliary battery system using static inverter.

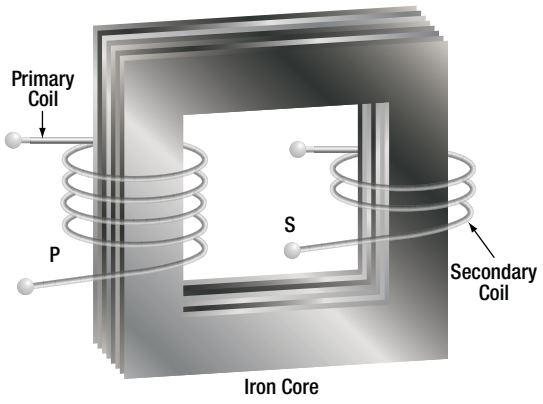


Figure 5-52. An iron-core transformer.

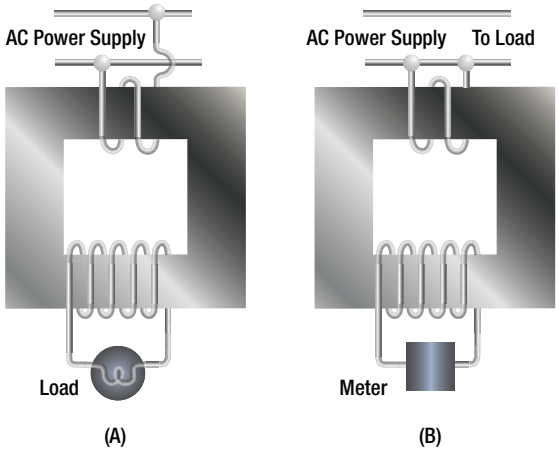


Figure 5-53. Voltage and current transformers.

For example, the turns ratio of the step-down transformer shown in ***Figure 5-54A*** is 5 to 1, since there are five times as many turns in the primary as in the secondary. The step-up transformer shown in ***Figure 5-54B*** has a 1 to 4 turns ratio.

The ratio of the transformer input voltage to the output voltage is the same as the turns ratio if the transformer is 100 percent efficient. Thus, when 10 volts are applied to the primary of the transformer shown in ***Figure 5-54A***, two volts are induced in the secondary. If 10 volts are applied to the primary of the transformer in ***Figure 5-54B***, the output voltage across the terminals of the secondary will be 40 volts.

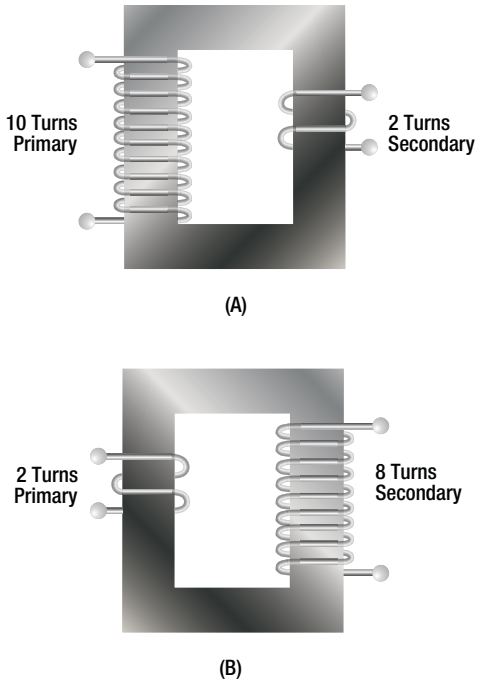


Figure 5-54. A step-down and a step-up transformer.

No transformer can be constructed that is 100 percent efficient, although iron core transformers can approach this figure. This is because all the magnetic lines of force set up in the primary do not cut across the turns of the secondary coil.

A certain amount of the magnetic flux, called leakage flux, leaks out of the magnetic circuit. The measure of how well the flux of the primary is coupled into the secondary is called the "coefficient of coupling." For example, if it is assumed that the primary of a transformer develops 10 000 lines of force and only 9 000 cut across the secondary, the coefficient of coupling would be 0.9 or, stated another way, the transformer would be 90 percent efficient.

When an AC voltage is connected across the primary terminals of a transformer, an alternating current will flow and self induce a voltage in the primary coil that is opposite and nearly equal to the applied voltage. The difference between these two voltages allows just enough current in the primary to magnetize its core. This is called the exciting, or magnetizing, current. The magnetic field caused by this exciting current cuts across the secondary coil and induces a voltage by mutual induction.

If a load is connected across the secondary coil, the load current flowing through the secondary coil will produce a magnetic field which will tend to neutralize the magnetic field produced by the primary current. This will reduce the self-induced (opposition) voltage in the primary coil and allow more primary current to flow. The primary current increases as the secondary load current increases, and decreases as the secondary load current decreases. When the secondary load is removed, the primary current is again reduced to the small exciting current sufficient only to magnetize the iron core of the transformer.

If a transformer steps up the voltage, it will step down the current by the same ratio. This should be evident if the power formula is considered, for the power (I × E) of the output (secondary) electrical energy is the same as the input (primary) power minus that energy loss in the transforming process. Thus, if 10 volts and 4 amps (40 watts of power) are used in the primary to produce a magnetic field, there will be 40 watts of power developed in the secondary (disregarding any loss).

If the transformer has a step-up ratio of 4 to 1, the voltage across the secondary will be 40 volts and the current will be 1 amp. The voltage is 4 times greater and the current is one-fourth the primary circuit value, but the power (I × E value) is the same.

When the turns ratio and the input voltage are known, the output voltage can be determined as follows:

$$\frac{E_2}{E_1} = \frac{N_2}{N_1}$$

Where E is the voltage of the primary, E_2 is the output voltage of the secondary, and N_1 and N_2 are the number of turns of the primary and secondary, respectively. Transposing the equation to find the output voltage gives:

$$E_2 = \frac{E_1 N_2}{N_1}$$

The most commonly used types of voltage transformers are as follows:

1. Power transformers are used to step up or step down voltages and current in many types of power supplies. They range in size from the small power transformer shown in ***Figure 5-55*** used in a radio receiver to the large transformers used to step down high power line voltage to the 110 – 120 volt level used in homes.

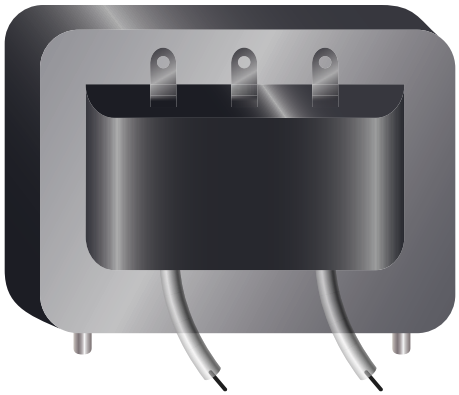


Figure 5-55. Power supply transformer.

Figure 5-56 shows the schematic symbol for an iron core transformer. In this case, the secondary is made up of three separate windings. Each winding supplies a different circuit with a specific voltage, which saves the weight, space, and expense of three separate transformers. Each secondary has a midpoint connection, called a "center tap," which provides a selection of half the voltage across the whole winding.

The leads from the various windings are color coded by the manufacturer, as labeled in ***Figure 5-56***. This is a standard color code, but other codes or numbers may be used.

2. Audio transformers resemble power transformers. They have only one secondary and are designed to operate over the range of audio frequencies (20 to 20 000 cps).
3. RF transformers are designed to operate in equipment that functions in the radio range of frequencies. The symbol for the RF transformer is the same as for an RF choke coil. It has an air core as shown in ***Figure 5-57***.
4. Autotransformers are normally used in power circuits; however, they may be designed for other uses. Two different symbols for autotransformers used in power or audio circuits are shown in ***Figure 5-58***. If used in an RF communication or navigation circuit (***Figure 5-58B***), it is the same, except there is no symbol for an iron core. The autotransformer uses part of a winding as a primary; and, depending on whether it is step up or step down, it uses all or part of the same winding as the secondary. For example, the autotransformer shown in ***Figure 5-58A*** could use the following the choices for primary and secondary terminals as shown.

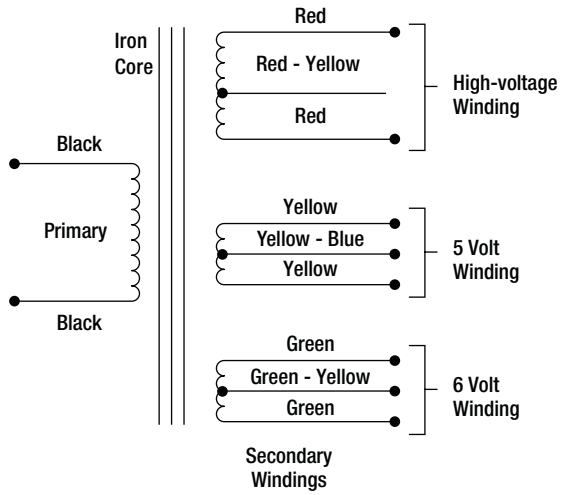


Figure 5-56. Schematic symbol for an iron-core power transformer.

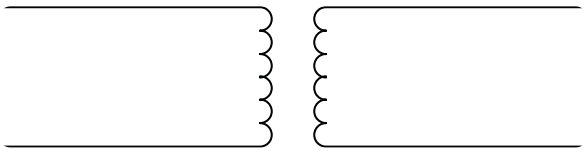


Figure 5-57. An air-core transformer.

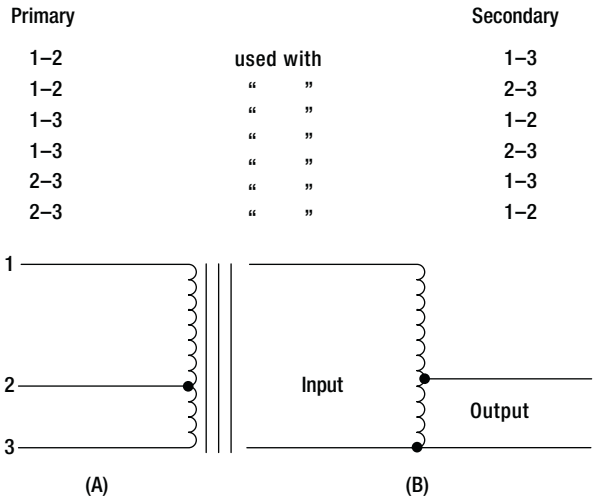


Figure 5-58. Autotransformers.

CURRENT TRANSFORMERS

Current transformers are used in AC power supply systems to sense generator line current and to provide a current, proportional to the line current, for circuit protection and control devices.

The current transformer is a ring-type transformer using a current carrying power lead as a primary (either the power lead or the ground lead of the AC generator). The current in the primary induces a current in the secondary by magnetic induction.

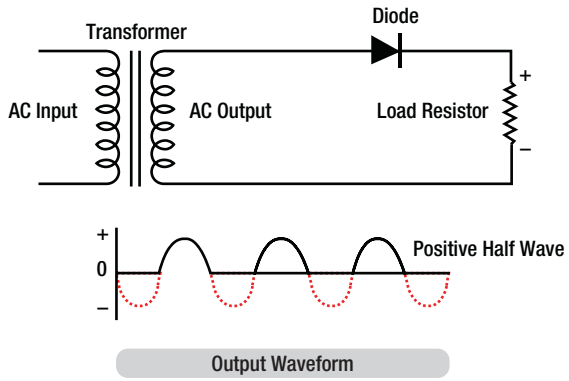


Figure 5-59. A half wave rectifier uses one diode to produce pulsating DC current from AC. Half of the AC cycle is wasted when the diode blocks the current flow as the AC cycles below zero.

The sides of all current transformers are marked "H1" and "H2" on the unit base. The transformers must be installed with the "H1" side toward the generator in the circuit in order to have proper polarity. The secondary of the transformer should never be left open while the system is being operated; to do so could cause dangerously high voltages, and could overheat the transformer. Therefore, the transformer output connections should always be connected with a jumper when the transformer is not being used but is left in the system.

TRANSFORMER LOSSES

In addition to the power loss caused by imperfect coupling, transformers are subject to "copper" and "iron" losses. The resistance of the conductor comprising the turns of the coil causes copper loss. The iron losses are of two types called hysteresis loss and eddy current loss. Hysteresis loss is the electrical energy required to magnetize the transformer core, first in one direction and then in the other, in step with the applied alternating voltage. Eddy current loss is caused by electric currents (eddy currents) induced in the transformer core by the varying magnetic fields. To reduce eddy current losses, cores are made of laminations coated with an insulation, which reduces the circulation of induced currents.

POWER IN TRANSFORMERS

Since a transformer does not add any electricity to the circuit but merely changes or transforms the electricity that already exists in the circuit from one voltage to another, the total amount of energy in a circuit must remain the same. If it were possible to construct a perfect transformer, there would be no loss of power in it; power would be transferred undiminished from one voltage to another.

Since power is the product of volts times amperes, an increase in voltage by the transformer must result in a decrease in current and vice versa. There cannot be more power in the secondary side of a transformer than there is in the primary. The product of amperes times volts remains the same.

The transmission of power over long distances is accomplished by using transformers. At the power source, the voltage is stepped up in order to reduce the line loss during transmission. At the point of utilization, the voltage is stepped down, since it is not feasible to use high voltage to operate motors, lights, or other electrical appliances.

RECTIFIERS

Rectifier circuits change AC voltage into DC voltage and are one of the most commonly used type of circuits in aircraft electronics. (***Figure 5-59***) The resulting DC waveform output is also shown. The circuit has a single semiconductor diode and a load resistor. When the AC voltage cycles below zero, the diode shuts off and does not allow current flow until the AC cycles through zero voltage again. The result is pronounced pulsating DC. While this can be useful, half of the original AC voltage is not being used.

A full wave rectifier creates pulsating DC from AC while using the full AC cycle. One way to do this is to tap the secondary coil at its midpoint and construct two circuits with the load resistor and a diode in each circuit. (***Figure 5-60***) The diodes are arranged so that when current is flowing through one, the other blocks current.

When the AC cycles so the top of the secondary coil of the transformer is positive, current flows from ground, through the load resistor (VRL), Diode 1, and the upper half of the coil. Current cannot flow through Diode 2 because it is blocked. (***Figure 5-60A***) As the AC cycles through zero, the polarity of the secondary coil changes. (***Figure 5-60B***)

Current then flows from ground, through the load resistor, Diode 2, and the bottom half of the secondary coil. Current flow through Diode 1 is blocked. This arrangement yields positive DC from cycling AC with no wasted current.

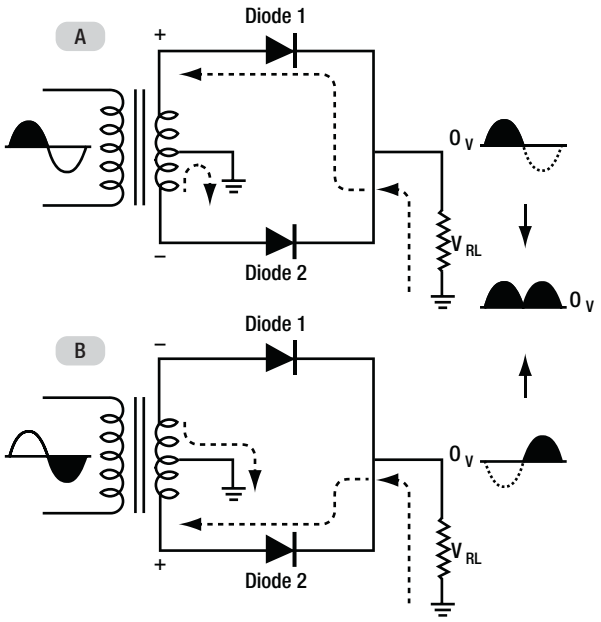


Figure 5-60. A full wave rectifier can be built by center tapping the secondary coil of the transformer and using two diodes in separate circuits. This rectifies the entire AC input into a pulsating DC with twice the frequency of a half wave rectifier.

Another way to construct a full wave rectifier uses four semiconductor diodes in a bridge circuit. Because the secondary coil of the transformer is not tapped at the center, the resultant DC voltage output is twice that of the two-diode full wave rectifier. (***Figure 5-61***) During the first half of the AC cycle, the bottom of the secondary coil is negative.

Current flows from it through diode (D1), then through the load resistor, and through diode (D2) on its way back to the top of the secondary coil. When the AC reverses its cycle, the polarity of the secondary coil changes. Current flows from the top of the coil through diode (D3), then through the load resistor, and through diode (D4) on its way back to the bottom of the secondary coil. The output waveform reflects the higher voltage achieved by rectifying the full AC cycle through the entire length of the secondary coil.

Use and rectification of three-phase AC is also possible on aircraft with a specific benefit. The output DC is very smooth and does not drop to zero. A six diode circuit is built to rectify the typical three-phase AC produced by an aircraft alternator. (***Figure 5-62***)

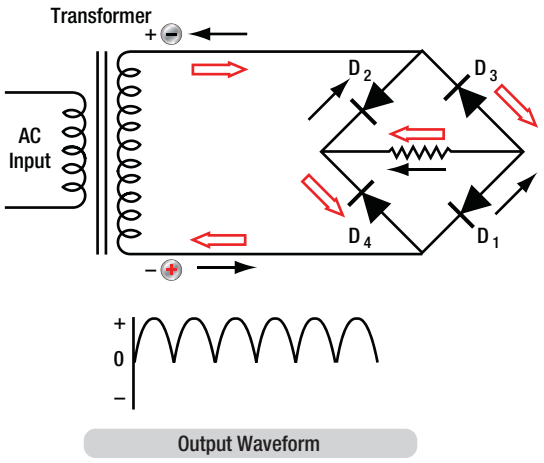


Figure 5-61. The bridge-type four-diode full wave rectifier circuit is most commonly used to rectify single-phase AC into DC avionics.

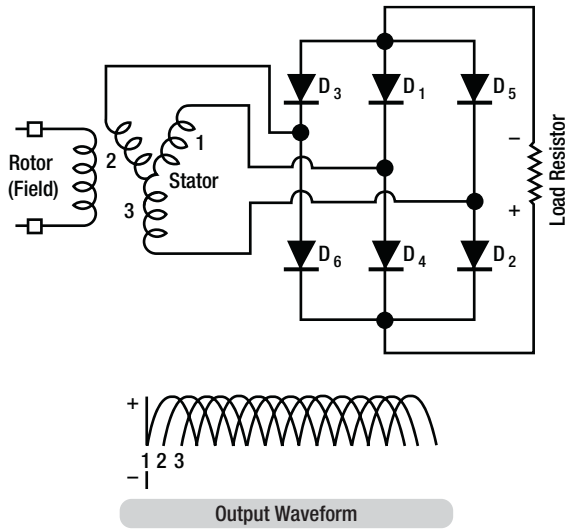


Figure 5-62. A six-diode three-phase AC rectifier.

Each stator coil corresponds to a phase of AC and becomes negative for 120° of rotation of the rotor. When stator 1 or the first phase is negative, current flows from it through diode (D1), then through the load resistor and through diode (D2) on its way back to the third phase coil.

Next, the second phase coil becomes negative and current flows through diode (D3). It continues to flow through the load resistor and diode (D4) on its way back to the first phase coil. Finally, the third stage coil becomes negative causing current to flow through diode (D5), then the load resistor and diode (D6) on its way back to the second phase coil. The output waveform of this three-phase rectifier depicts the DC produced. It is a relatively steady, non-pulsing flow equivalent to just the tops of the individual curves. The phase overlap prevents voltage from falling to zero producing smooth DC from AC.

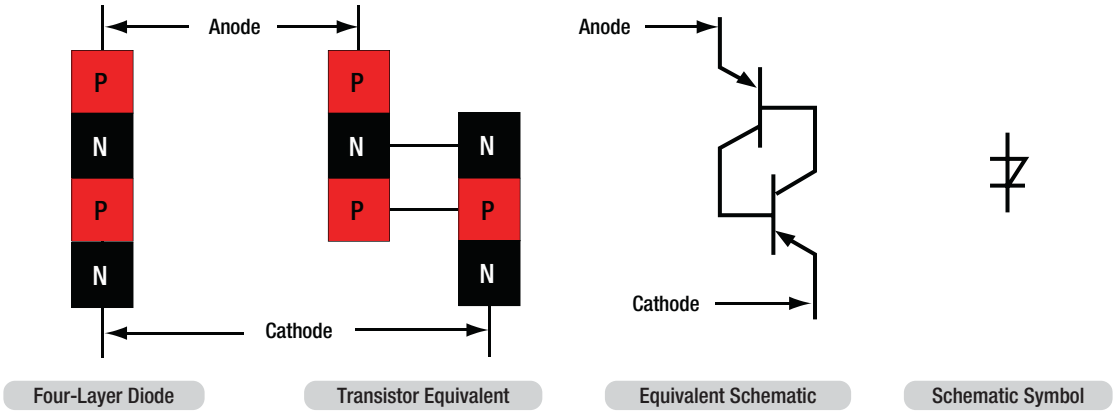


Figure 5-63. A four-layer semiconductor diode behaves like two transistors. When break-over voltage is reached, the device conducts current until the voltage is removed.

SILICON CONTROLLED RECTIFIERS

Combination of semiconductor materials is not limited to a two-type, three-layer sandwich transistor. By creating a four-layer sandwich of alternating types of semiconductor material (i.e., PNPN or NPNP), a slightly different semiconductor diode is created. As is the case in a two-layer diode, circuit current is either blocked or permitted to flow through the diode in a single direction. Within a four-layer diode, sometimes known as a Shockley diode, there are three junctions. The behavior of the junctions and the entire four-layer diode can be understood by considering it to be two interconnected three-layer transistors. (***Figure 5-63***)

Transistor behavior includes no current flow until the base material receives an applied voltage to narrow the depletion area at the base-emitter junction. The base materials in the four-layer diode transistor model receive charge from the other transistor's collector. With no other means of reducing any of the depletion areas at the junctions, it appears that current does not flow in either direction in this device. However, if a large voltage is applied to forward bias the anode or cathode, at some point the ability to block flow breaks down. Current flows through whichever transistor is charged. Collector current then charges the base of the other transistor and current flows through the entire device.

Some caveats are necessary with this explanation. The transistors that comprise this four-layer diode must be constructed of material similar to that described in a zener diode. That is, it must be able to endure the current flow without burning out. In this case, the voltage that causes the diode to conduct is known as breakover voltage rather than breakdown voltage.

Additionally, this diode has the unique characteristic of allowing current flow to continue until the applied voltage is reduced significantly, in most cases, until it is reduced to zero. In AC circuits, this would occur when the AC cycles.

While the four-layer, Shockley diode is useful as a switching device, a slight modification to its design creates a silicon controlled rectifier (SCR). To construct a SCR, an additional terminal known as a gate is added. It provides more control and utility. In the four-layer semiconductor construction, there are always two junctions forward biased and one junction reversed biased. The added terminal allows the momentary application of voltage to the reversed biased junction. All three junctions then become forward biased and current at the anode flows through the device. Once voltage is applied to the gate, the SCR become latched or locked on. Current continues to flow through it until the level drops off significantly, usually to zero. Then, another applied voltage through the gate is needed to reactivate the current flow. (***Figures 5-64 and 5-65***)

Figure 5-66. Phase control is a key application for SCR. By limiting the percentage of a full cycle of AC voltage that is applied to a load, a reduced voltage results. The firing angle or timing of a positive voltage pulse through the SCR's gate latches the device open allowing current flow until it drops below the holding current, which is usually at or near zero voltage as the AC cycle reverses.

SCRs are often used in high voltage situations, such as power switching, phase controls, battery chargers, and inverter circuits. They can be used to produce variable DC voltages for motors and are found in welding power

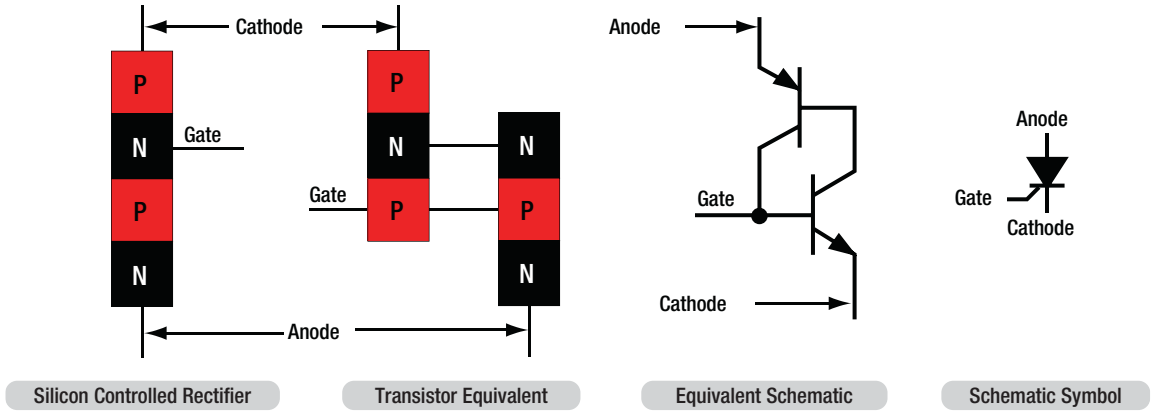


Figure 5-64. A silicon controlled rectifier (SCR) allows current to pass in one direction when the gate receives a positive pulse to latch the device in the on position. Current ceases to flow when it drops below holding current, such as when AC current reverses cycle.

supplies. Often, lighting dimmer systems use SCRs to reduce the average voltage applied to the lights by only allowing current flow during part of the AC cycle. This is controlled by controlling the pulses to the SCR gate and eliminating the massive heat dissipation caused when using resistors to reduce voltage. ***Figure 5-66*** graphically depicts the timing of the gate pulse that limits full cycle voltage to the load. By controlling the phase during which time the SCR is latched, a reduced average voltage is applied.

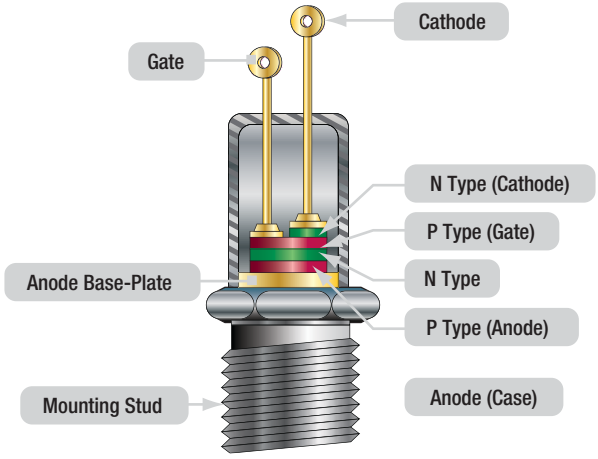


Figure 5-65. Cross-section of a medium-power SCR.

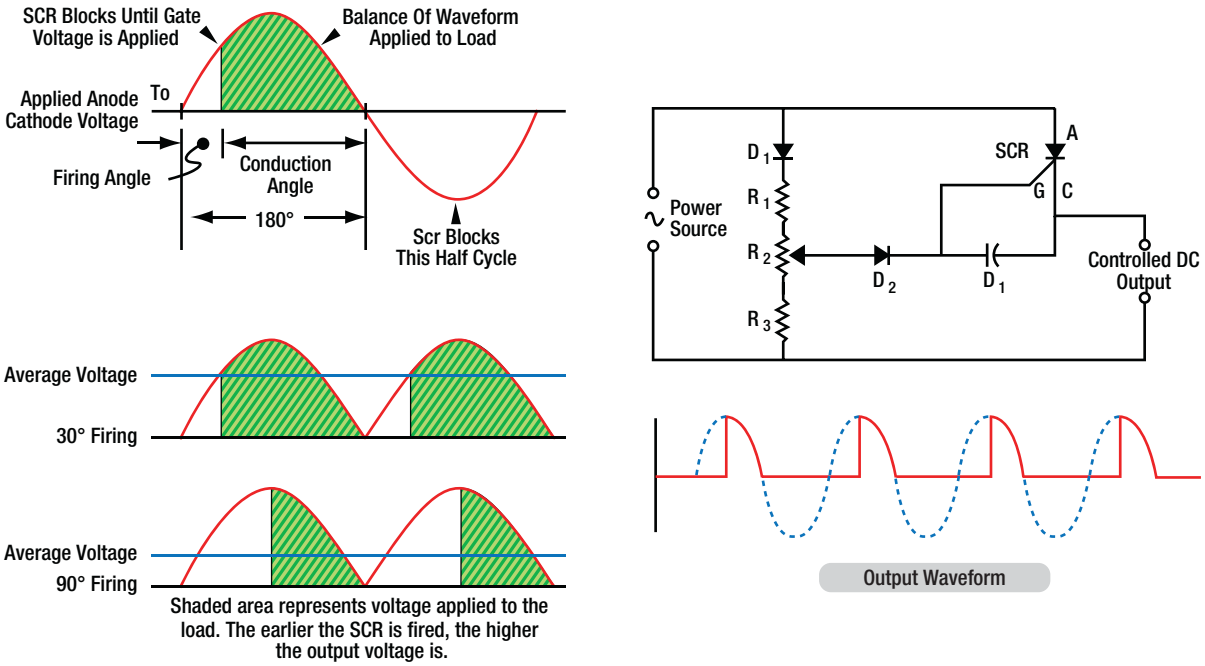


Figure 5-66. Phase control is a key application for SCR. By limiting the percentage of a full cycle of AC voltage that is applied to a load, a reduced voltage results. The firing angle or timing of a positive voltage pulse through the SCR's gate latches the device open allowing current flow until it drops below the holding current, which is usually at or near zero voltage as the AC cycle reverses.

TRANSFORMER RECTIFIERS

Control of the available electric power to numerous electrical devices on an aircraft in any and all situations makes the conversion from AC power to DC power common on modern airliners. Often, transformer rectifiers (TRs) are used for this purpose. They are typically found between an AC power source and a DC bus and inside battery chargers. The TR not only rectifies AC to produce DC current. It also contains a transformer to adjust the DC output to the precise voltage required.

CIRCUIT PROTECTION

CURRENT LIMITING DEVICES

In addition to the protection proved by the GCUs and the BPCU, individual conductive circuits on an aircraft are protected with current limiting devices.

Conductors should be protected with circuit breakers or fuses located as close as possible to the electrical power source bus. Normally, the manufacturer of the electrical equipment specifies the fuse or circuit breaker to be used when installing equipment. The circuit breaker or fuse should open the circuit before the conductor emits smoke. To accomplish this, the time current characteristic of the protection device must fall below that of the associated conductor. Circuit protector characteristics should be matched to obtain the maximum utilization of the connected equipment. ***Figure 5-67*** shows a chart used in selecting the circuit breaker and fuse protection for copper conductors.

This limited chart is applicable to a specific set of ambient temperatures and wire bundle sizes and is presented as typical only. It is important to consult such guides before selecting a conductor for a specific purpose. For example, a wire run individually in the open air may be protected by the circuit breaker of the next higher rating to that shown on the chart.

Wire AN Guage Copper	Circuit Breaker Amperage	Fuse Amperage
22	5	5
20	7.5	5
18	10	10
16	15	10
14	20	15
12	30	20
10	40	30
8	50	50
6	80	70
4	100	70
2	125	100
1		150
0		150

Figure 5-67. Wired and circuit protection chart.

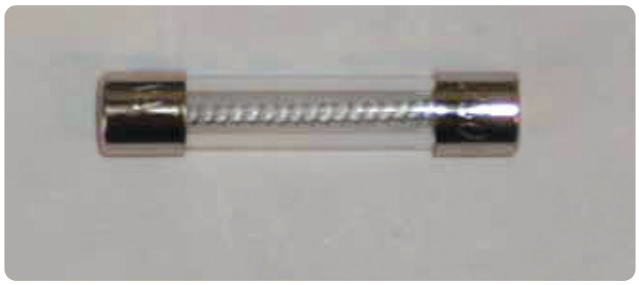


Figure 5-68. A fuse.

FUSES

A fuse is placed in series with the voltage source and all current must flow through it. (***Figure 5-68***) The fuse consists of a strip of metal that is enclosed in a glass or plastic housing. The metal strip has a low melting point and is usually made of lead, tin, or copper. When the current exceeds the capacity of the fuse the metal strip heats up and breaks. As a result of this, the flow of current in the circuit stops.

There are two basic types of fuses: fast acting and slow blow. The fast-acting type opens very quickly when their particular current rating is exceeded. This is important for electric devices that can quickly be destroyed when too much current flows through them for even a very small amount of time. Slow blow fuses have a coiled construction inside. They are designed to open only on a continued overload, such as a short circuit.

CIRCUIT BREAKERS

A circuit breaker is an automatically operated electrical switch designed to protect an electrical circuit from damage caused by an overload or short circuit. Its basic function is to detect a fault condition and immediately discontinue electrical flow. Unlike a fuse that operates once and then has to be replaced, a circuit breaker can

be reset to resume normal operation. All resettable circuit breakers should open the circuit in which they are installed regardless of the position of the operating control when an overload or circuit fault exists. Such circuit breakers are referred to as trip-free.

Note that automatic reset circuit breakers automatically reset themselves. They should not be used as circuit protection devices in aircraft. When a circuit breaker trips, the electrical circuit should be checked and the fault removed before the circuit breaker is reset.

Sometimes circuit breakers trip for no apparent reason, and the circuit breaker can be reset one time. If the circuit breaker trips again, there exists a circuit fault and the technician must troubleshoot the circuit before resetting the circuit breaker. (***Figure 5-69***)



Figure 5-69. Circuit breaker panel.

New aircraft designs use a digital circuit protection architecture. This system monitors the amperage through a particular circuit. When the maximum amperage for that circuit is reached, the power is rerouted away from the circuit. This system reduces the use of mechanical circuit breakers. The advantages are weight savings and the reduction of mechanical parts.

EMERGENCY POWER GENERATION

Power in an aircraft is distributed through various buses to permit control over small groups of electrical loads. This also protects against a single electrical power failure causing a severe loss of power. The main AC generators power the various buses including the DC buses which use transformer rectifiers to convert the AC to DC.

Modern aircraft typically have buses and loads divided so that the failure of a single generator is controlled by isolating the failed unit and use of the other main generator or back-up generators to power the buses of the failed unit. Automatic monitoring and switching of power sources is normal.

STANDBY POWER

When both of the main AC generators fail simultaneously, back-up or standby generators may still be used to power AC and DC buses. The failure of the main and back-up generators is rare. It presenters the crew with a situation in which emergency power must be used. On most large turbine powered aircraft, a standby power bus is used for emergency power when the main sources of power fail. The standby bus is usually a hot bus directly connected to the main aircraft battery. It is powered by the APU generator, the ram air turbine generator or the aircraft battery.

Standby power from the standby power bus is DC. However, inverters are used to create AC power from the DC standby bus to power vital AC circuits. A limited number of components and circuits are powered off the standby bus. The idea is to be able to maintain vital systems for flight and not deplete the power and sources of power that remain in the emergency situation.

The selection of standby power in an emergency is typically made by the crew with a switch on the flight deck. Automatic switching is more common on the most modern aircraft.

EXTERNAL/GROUND POWER

Most aircraft employ an external power circuit that provides a means of connecting electrical power from a ground source to the aircraft. External power is often used for starting the engine or maintenance activities on the aircraft. This type of system allows operation of various electrical systems without discharging the battery. The external power systems typically consists of an electrical plug located in a convenient area of the fuselage, an electrical solenoid used to connect external power to the bus, and the related wiring for the system. A common external power receptacle is shown in ***Figure 5-70***.

Figure 5-71 shows how the external power receptacle connects to the external power solenoid through a reverse polarity diode. This diode is used to prevent any accidental connection in the event the external power supply has the incorrect polarity (i.e., a reverse of the positive and negative electrical connections).



Figure 5-70. External power receptacle.

A reverse polarity connection could be catastrophic to the aircraft's electrical system. If a ground power source with a reverse polarity is connected, the diode blocks current and the external power solenoid does not close.

This diagram also shows that external power can be used to charge the aircraft battery or power the aircraft electrical loads. For external power to start the aircraft engine or power electrical loads, the battery master switch must be closed.

On most airliners, a separate ground handling bus and an APU battery bus are used. Many of these buses divide to power additional buses which distribute power strategically to sub-systems of electrical power.

On large aircraft, when the aircraft engines are running, AC generators mounted on and driven by the engines supply the DC buses through the use of transformer rectifiers units (TRU's). The TRU's convert the 115V AC generated into 28V DC power. When the aircraft is on the ground with external power connected, a separate TRU converts the AC from the ground power source to 28V DC to power the distribution buses. When the external power source is not being used, the main aircraft battery supplies the DC power buses.

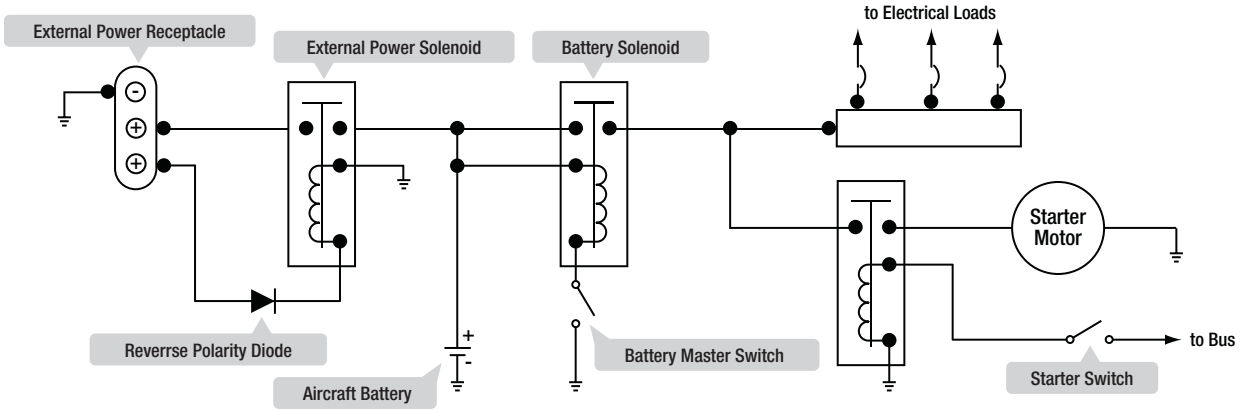


Figure 5-71. A simple external power circuit diagram.

QUESTIONS

Question: 5-1
Name 3 advantages and 1 disadvantage of $LiCoO_2$ batteries.

Question: 5-2
What is the single most important factor in aircraft battery installations?

Question: 5-3
An aircraft in flight has four potential sources of electrical power. Name them in order of priority.

Question: 5-4
A typical aircraft AC generator produces ____ phase, _____ volt power at _____Hz.

Question: 5-5
What regulates the frequency of an AC alternator?

Question: 5-6
A typical generator in a small jet includes two field windings. The purpose of each of these windings is to?

Question: 5-7
In large aircraft, why does each engine power lust one of two main electrical power distribution buses?

Question: 5-8
What is the advantage of a split bus power generation system?

ANSWERS

Answer: 5-1
Advantages – light weight, greater capacity less discharge when not in use.
Disadvantages – requires continuous temperature monitoring.

Answer: 5-2
redundancy.

Answer: 5-3
Engine powered generators; APU powered generators; batteries, ram air turbine (RAT) system.

Answer: 5-4
Three, 115, 400.

Answer: 5-5
Its RPM, as regulated by a CSD (Constant Speed Drive) unit.

Answer: 5-6
One for in-flight electrical power; one for engine starting.

Answer: 5-7
To prevent phase differences from entering each bus.

Answer: 5-8
Allows either engine generator to power either distribution bus (but not at the same time).



PART-66 SYLLABUS **LEVELS**

CERTIFICATION CATEGORY → **B2**

Sub-Module 06

EQUIPMENT AND FURNISHINGS (ATA 25)

Knowledge Requirements

13.6 - Equipment and Furnishings (ATA 25)	
Electronic emergency equipment requirements.	3
Cabin entertainment equipment.	

EMERGENCY EQUIPMENT REQUIREMENTS

Since aircraft leave the surface of the planet and fly in the sky, emergency equipment such as fire extinguishers, life rafts, escape slides, first aid kits, etc. may be required by authorities to be on board for the protection of passengers and crew. Different size aircraft designed for different uses have emergency equipment requirements suitable for the intended purpose of the aircraft. For example, a small single engine aircraft has fewer required fire extinguishers on board than a 300 seat airliner.

The specifications for emergency equipment requirements on any given aircraft are found in EASA-issued airworthiness codes called "certification specifications" (CS), i.e. CS-23, CS-25, etc. These certification standards are divided with separate sections devoted to different types of aircraft. CS-25 are those pertaining to large commercial aircraft. When an airliner is certified, it meets the requirements for the installation of emergency equipment in accordance with CS-25 specifications. For example, CS.25.851(a)(1) specifies the minimum number of portable fire extinguishers to be on board the aircraft. (***Figure 6-1***) To receive an airworthiness certificate, the manufacturer is required to make provisions for 4 extinguishers on a 250-seat aircraft. The aircraft operator must ensure the extinguishers are present and airworthy for each flight.

Many other specifications for emergency equipment are covered in CS-25 such as the number, type and location of emergency doors, seat belt strength, exit sign type, emergency lighting parameters and life raft locations. Details pertaining to the required emergency equipment itself may be found by reference to a European Technical Standard Order (ETSO). Acceptable Means for Compliance (AMC) for CS-25 (CS-25, Book 2) also contains details concerning components and procedures referenced in CS-25. Other documentation may also be referenced.

Passenger Capacity	Number of Extinguishers
7 to 30	1
31 to 60	2
61 to 200	3
201 to 300	4
301 to 400	5
401 to 500	6
501 to 600	7
601 to 700	8

Figure 6-1. Required minimum number of hand fire extinguishers.

Thus, the required emergency equipment is delivered and stowed in place when an aircraft is placed into service. A primary function of the technician is to ensure all of the required equipment is in its specified location and serviceable. Note that these locations are specified in the certification standards. Furthermore, evacuation slides, life rafts, fire extinguishers and oxygen bottles all have inspection requirements that include a pressure checked before each flight. These and other security and condition inspection items related to emergency equipment are written and require the signature of the technician who performs the check. This type of equipment is installed for a limited time period after which, it must be removed for in-depth inspection and recharging. Installation and removal procedures are detailed in the aircraft maintenance manual. (***Figure 6-2***)

Any batteries fitted as part of emergency equipment are serviced, and replaced, on a calendar basis. Some periodic checks require inspections and function checking of the associated equipment. There are many items on board an aircraft that contribute to safety via designs that are purposeful in an emergency.



Figure 6-2. Emergency slide inflation canister pressure is checked regularly on the gauge in the lower window (mostly yellow). A life-limit removal date sticker partially obscures the gauge in the photo.

Some items regularly considered emergency equipment are:

- Fire extinguishers
- First aid kits
- Life vests
- Megaphones
- Flashlights
- Fire axes
- Defibrillator's
- Life rafts
- Evacuation slides
- Emergency lights
- Emergency exit signs
- Escape ropes
- Smoke goggles and hoods

Technicians must be aware that when working with pressurized raft and door-slide installations, specific instructions must be followed to prevent inadvertent deployment and injury. Typically, the devices must armed after installation to operate as designed.

EMERGENCY LIGHTING

All emergency lighting have their own independent power supplies. These are commonly rechargeable batteries that are connected to the aircraft's power system under normal conditions. This provides a continuous trickle charge for the batteries to keep them fully charged, and a power source for the lights operation without a drain on the battery. (***Figure 6-3***)

- If the entire aircraft's power buses fail, then the emergency lights, such as EXIT lights and floor path lighting are provided with their own battery power for luminance.
- EXIT lights are permanently switched on under normal flight and ground operations as a standard safety requirement.
- All emergency electrically powered equipment have calendar servicing for capacity testing of batteries and maintenance.



Figure 6-3. Typical Exit lights in cabins.

SEATS, HARNESSES AND BELTS

SEATS

Aircraft seats are constructed to be very strong yet lightweight. They are typically mounted in groups of two or three seats in a frame. The seat assembly is fitted with quick-disconnect fittings for secure attachment into seat track mounting strips on the cabin floor. The seat tracks are bolted to structural fuselage frame members. The tracks allow adjustment of seats forward and aft in 1 inch increments before enabling a lockdown of the seat in the track. The result is that seats may be arranged in different configurations (spacing). (***Figure 6-4***)

It is typical for seats used in 1st class and business class to be larger than coach seats. However, the seat frame for the larger seats, which may be more elaborate with full-recline capability, still connects into seat tracks in the same manner as the coach class seats. Passenger seat assemblies are fitted with arm rests and a reclining mechanism for the seat back. A formed metal luggage restraint rail is part of the seat assembly. It prevents luggage stowed under the seat from sliding forward during abrupt deceleration. Each seat has retractable seat arms and a seat back tray table that unhinges from its stowage location in the seat back.

Seats on the flight deck for the captain and first officer are fully adjustable. Captain and first officer seats engage in floor-mounted tracks similar to the passenger seat assemblies. Five-point seat belts are attached to the seat frame assembly. (***Figure 6-5***) Observer seats typically fold down out of a wall or bulkhead and consist of a padded slab-style seats with safety harnesses.

The most modern aircraft include an LCD viewing screen mounted in the back of each upper seat back for use by the passenger in the seat directly aft. Safety briefing announcements, movies and internet may be accessed depending on the in flight entertainment offered by the airline. A seat electronics unit (SEU) is typically mounted under each seat for facilitation of the inflight entertainment (IFE). A digital data bus runs the length of the cabin so that each individual SEU can connect to the IFE computer.

SEAT BELTS AND HARNESSES

Seat belts for all passengers and seat harnesses for crew members are required on passenger service aircraft. If a seat belt is not functional and cannot be changed before flight, the seat must be placarded and not used. Seat belts are attached to the seat assembly/frame in most cases. Flight attendant and fold-down jump seats may attach the seat belts to a bulkhead. Typically passengers seats have a lap belt only. Seat belts that include shoulder harnesses are used by the crew. The captain and first officer's seat belts typically include both shoulder harness and crotch strap that fasten into a rotary buckle along with the lap belts. Seat belts and harnesses are constructed from nylon webbing which are woven to be extremely strong. Test procedures for seat belt and the forces they must withstand are also in CS-25.



Figure 6-4. Passenger seats are locked into a seat track on the floor of the cabin.

CABIN LAY-OUT

Operators may configure seats, galleys, bulkheads and lavatories in a variety of ways as long as CS-25 specifications are met. This does tend to limit configurations to those commonly found on airlines. Pragmatic location of galleys and lavatories also limits the cabin lay-out. Furthermore a lavatory for each seating zone (i.e. first class, coach, etc.) is required. Flight attendants are charged with assisting passengers at exits in case of an emergency so their seat locations and control stations are generally limited to be near the exits. Needless to say, no exit or emergency escape path may be blocked. Thus, cabin layout is a result of considerations for safety requirements and practical functional arrangements to serve the passengers.

Overhead storage bins run the length of the cabin. They are fastened to support structure attached to fuselage members . The exact location of the bins, however, is able to be modified to position them correctly in relation to the configuration of seats and other installations. Passenger service unit (PSU) locations are the same. Berths for crew changing or sleeping are located by design and access/egress requirements found in CS-25. These are usually at the forward or aft ends of the cabin or above the main flight and passenger cabins. The flight control cabin is always separated from the passenger cabin by a door. Closets, dividers, LCD monitors, video projectors and flight attendant locations are all slightly adjustable to accommodate different seating configurations.

Figure 6-6, illustrates the "flexibility zones" for lavatory location that Boeing provides on its 777-200 aircraft. Operators may configure the aircraft in numerous way as long as lavatories are installed in the zones illustrated. To facilitate this flexibility, the manufacturer makes available all water, air and electrical connections within the zones as well as engineered installation fittings and hardware. Galley installation is similarly flexible. A common adjustment to cabin layout and configuration is seat pitch. Pitch is the fore and aft distance between two rows of seating. A small seat pitch allows more seats to be installed on an aircraft. A large pitch is typically more comfortable. The seat pitch used is up to the operator within the confines of the engineering and certification specifications for the aircraft. The number of seats desired on a certain aircraft greatly influence the overall cabin lay-out.

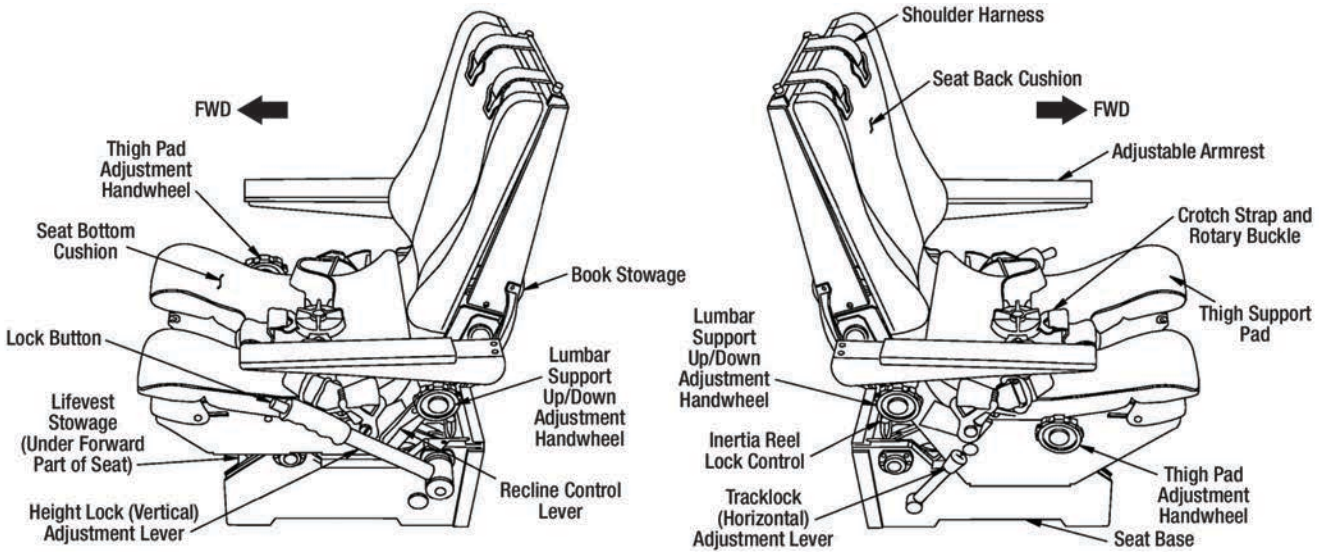


Figure 6-5. Typical captain's seat.



Lavatory Flexibility Zone (-200)

Figure 6-6. Flexibility zones for lavatory location.

EQUIPMENT LAY-OUT

The location of various equipment on a airliner is not always left up to the operator. As stated, CS-25 specifies many locations for emergency features and equipment. Some of the requirements for equipment location on the flight deck are described as "within arm's reach of the pilot while sitting". Obviously, this leaves the exact location for certain equipment open. However, when all location specifications are taken into account along with the practicality of the location of other equipment in the area, manufacturer's produce aircraft with fairly consistent locations for everything. It is typically the technicians job to identify the required equipment stowed in the chosen location and to inspect it for airworthiness. ***Figure 6-7*** illustrates an example of the equipment stowed on the sidewall next to the first officer seat on the flight deck of a 737.

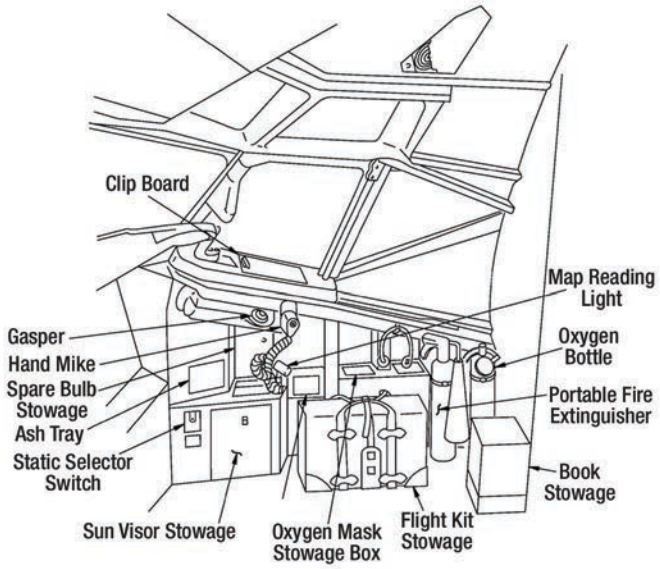


Figure 6-7. Equipment location on the flight deck.

Earlier models, or types, of large aircraft had their equipment layouts, placed in logical common areas. For example, the VHF's (which for a wide-bodied aircraft there could be four in total) could be secured on one dedicated rack, which if needed can be removed in its entirety from the aircraft.

Similarly all the ATC Transponders and associated equipment, such as TCAS, could be mounted on a common removable rack. With modern digital designed Avionics systems, all related Avionics is housed in

common electronic modules. Each specific component, such as a VHF radio, for example, is a sub-module, and can be removed from it's housed module. Separate systems, such as System1; System 2; and so forth are located in their specific housed Avionics Module.

Flight Data and Cockpit Voice Recorders, are positioned near the rear of aircraft. This is to ensure the high probability of them being least damaged after any mishaps. Large aircraft provide the ease of installation and accessibility having more available space. It should be mentioned that much of an airliner's electronic equipment is installed in dedicated areas for avionics below the passenger cabin floor. Often, just the control interface is accessible in the passenger compartment or on the flight deck while the "black box" containing the electronics and computer(s) associated with a piece of electronic equipment is located on a rack of an equipment bay.

CABIN FURNISHING INSTALLATION

Cabin furnishings and installations are considered for safety attributes in addition to their just being able to function. In particular, materials are closely scrutinized and tested to ensure that specifications for flammability and smoke production are low.

Sidewall veneers, seat covers, curtains, partitions, carpet and nearly every other material from which a furnishing is constructed must adhere to certification standards specified in CS-25. Elaboration on CS-25 regulations appear in the acceptable means of compliance (AMC) for CS-25.

CABIN ENTERTAINMENT EQUIPMENT

IN-FLIGHT ENTERTAINMENT

Cabin entertainment is a growing area of attention for airline operators. Many independent vendors compete to sell airlines particular in flight entertainment packages. Individual selection of music and movies at each seat location is common. Internet service is also common. Use of digital data busing for cabin entertainment equipment is the normal standard for modern designed IFE equipment. As mentioned, individual seat electronics units (SEUs) are typically mounted under the seats in each seat row. These are part of a computerized network that allow passengers access to and control of the entertainment that the airline offers.

Cabin entertainment systems with several LCD screens mounted in the ceiling throughout the cabin to be shared by all, are rarely seen in service in modern aircraft. Personal Video Screens (PVS's) are the standard now for IFE.

Passengers can select various Audio or Video, "on demand", at their convenience. These channels can be paused, fast forwarded, rewound etc., at will. Various selection includes: recorded music, recorded commercial radio and television, movies and games. A moving Map display with real time data of the aircraft's flight is also standard, with altitude, airspeed, ETA, distance to go and gone, etc., readily available for passengers to view. Forward and down looking cameras are also a common feature.

The IFE data is stored on a central server – latest media television programs, such as latest news for example, can be loaded onto the system prior to departure. Channels are streamed from the Central Server to individual seats via the SEU's.

CURRENT GENERATION IFE

Bandwidth and data capacity of current systems had been the main design considerations and obstacles for latest generation IFE. Some of these obstacles have been overcome, and such services as live commercial television on-board has become a reality.

A connectivity platform, for an on-board mobile phone system, was first introduced by Airbus Industries in 2007. Since 2010, Airbus have had a wireless broadband connectivity facility available with internet and mobile phone services. Airbus calls it: ALNA (airline network architecture). ALNA enables passengers to use their personal wireless communication devices for any internet application, giving them more freedom and connection opportunities than previous generation IFE systems. The devices are also able to be re-charged.

Management of this IFE system is carried out by the Cabin Crew. They are able to, for example, restrict cell-phone usage to data services only (SMS, email, mobile internet). This enables the system to operate within the capacity of it's design, without any temporary outages.

Airbus' A350 XWB is the first aircraft to integrate fourth-generation IFE systems. This uses fibre optics, enabling high-definition video to become a reality, using latest standard resolution seat screens. This fourth generation system does not require the SEU's, so more leg space is available for passengers. The fiber optic, and other cables run under the cabin floor.

Regardless whether individual or community systems are in place, maintenance information for cabin entertainment equipment is commonly the responsibility of the equipment vendor. Panasonic Corporation, Thales, and ALNA are the leading world manufacturers of IFE equipment.

SAFETY CONSIDERATIONS FOR IFE

Design and installation of IFE equipment requires strict adherence to relevant regulations. Overheating, arcing and insulation breakdown are the main factors to consider when carrying out the installation process. This is primarily because of the amount of additional wiring required.

IFE systems are isolated from all the other aircraft's systems, having their own designated power bus. They are independent of all other aircraft's power and digital systems. This ensures that any IFE failure cannot affect the function and safety of the aircraft. The integrity of the aircraft's flight systems is thus maintained. Some IFE systems have a Master Switch in the cockpit.

MULTIPLEXING.

Multiplexing techniques have greatly reduced the amount of wiring required for IFE, especially as more channels available for entertainment have been required in modern designs. Various multiplexing techniques are used, depending on the system, this topic is covered in another section.

FIBRE OPTICS

Fibre Optics are now found in most modern airliners, not only for aircraft systems, but for fourth generation IFE too. All the advantages, in comparison to conventional copper wiring, for fibre optic apply: less susceptible to interference; large BW; less noise; low attenuation; high reliability; isolation; and no ground loops. These far out-weigh any disadvantages such as: care of cable installation and fitting of connectors; and low mechanical strength of the cable.

GALLEY INSTALLATION

Galley installations on airliners are similar to lavatory installations. Areas of the aircraft are engineered to permit installation with fittings for connection into water and electrical systems. The galleys themselves are preconstructed modular units. One or more units are used on any given aircraft and configuration. Galleys typically install into the same floor tracks as the passenger seats but may also have specially dedicated fittings for installation. Tie rods to structure at the top of the fuselage further secure the installation. Most galleys are installed at the aft or forward end of the passenger cabin. Between classes of the cabin (i.e. coach and first class) is also a common galley location. (***Figure 6-8***) Large aircraft may have the galley installation on the lower level below the passenger cabin. An elevator is used to transport items to the passenger cabin. The specially designed lift system typically engages service carts securely to raise and lower food to and from the galley.

Most galleys include one or more ovens to heat meals, a refrigerated area, coffee makers, potable water and drain lines, a trash collection area and lighting.



Figure 6-8. Airbus A380 galley.

A control panel is installed with switches and circuit breakers for the electrical components. Often, flight attendant service carts are made to store in the galley module. Other features of a galley may include a fume extraction system, a trash compactor, an ice maker and an interphone system user's interface. Galley ovens and heating devices are electric and put a high demand on the aircraft electrical system.

Buses and power supplied to galley are engineered accordingly. On some aircraft, fully galley usage is restricted during periods of high electrical demand by critical aircraft systems. Typical galley power is 115 VAC three phase. A control panel with it's own switching and protection devices, such as circuit breakers, forms part of the dedicated galley bus from the aircraft's non-essential buses.

An average narrow bodied aircraft may have up to:

- (10) ovens rated at approximately 2.8 kW
- (5) ovens rated at approximately 1kW
- (5) refrigeration units rated at approximately 1kW

Galley power is the first to be shed when aircraft partial power generation is lost, and load shedding occurs. In recent years, aircraft manufacturers, especially Airbus, and vendors have redesigned the galley and its components to be lighter and more functional for flight attendants.

In recent years there has also been development of portable power supplies for galleys using approved liquid fuels, which is converted to electricity in a dedicated unit. This concept utilizes a light weight low volume unit that fits in part of existing galley modules. It will mean less demand made on aircraft power supplies, as more galley equipment is needed for future requirements.

CARGO HANDLING AND RETENTION EQUIPMENT

Cargo on passenger airliners is handled in different ways depending on the aircraft and the options installed by the aircraft operator. Cargo compartments are located below the passenger cabin floor with doors to the exterior of the aircraft so that cargo is loaded into the compartments from the airport ramp. They are lined with fire resistant sidewall and ceiling panels.

The cargo hold will always be colder than the main passenger cabin at cruise altitudes; the exact temperature determined by the aircraft type and configuration. Leakage of air-conditioned air from the passenger cabin is insufficient to achieve or maintain a comfortable temperature in the cargo hold.

Some aircraft have the lower cargo deck heated, or parts of it with heating devices such as: blankets or electric heating panels. This would be required for the comfort of pets or other cargo that is sensitive to cold temperatures. Typical temperatures achieved in the hold would be about 20 °C. Without these heating devices the temperature would be, on average, just below 10°C.

Small passenger transports and many larger ones may simply have an empty compartment finished to accept loose cargo on a piece-by-piece basis. A major concern is the shift of cargo during flight maneuvers which could cause a partial loss of control or inhibit performance. Cargo nets and barriers are used to keep cargo from shifting in these wide open berths. Numerous fittings are installed on the floor, walls and ceilings of the cargo compartment into which nets and barriers attach with quick release fittings. Cargo is loaded into position with consideration to weight and balance requirements on the aircraft and then the barriers are installed to hold it in place.

Medium to large sized airliners and cargo-dedicated transports commonly use cargo loading systems to place and secure cargo in the cargo compartments. These systems are electrically operated. The most modern systems permit loading with only one or two people. Certified containers are used into which cargo is first loaded before the containers are loaded into the cargo compartment. These containers are purpose built in different sizes. Their shapes and dimensions are made to follow the contour of the aircraft fuselage so as to maximize the use of space in the cargo compartment. Pallets of different sizes may also be loaded into a cargo compartment as well as uncertified containers of various design. Cargo containers are known as unit load devices (ULDs). They are given different names depending on size. For

example an LD-3 container is smaller than an LD-9. Certain foods, medical supplies and flora, such as flowers, do require a constant environment that has a stable low temperature. Transport of these sensitive cargo requires special ULD's (Unit Load Devices) that have their own temperature control. These are known as Insulated Load Devices, with some models specifying typical temperature controlled ranges from -20 °C to +20 °C.

Figure 6-9 shows an temperature controlled container produced by Envirotainer AB (Model RKN e 1) that has electric heating or cooling facilities, powered by rechargeable batteries. It's easy to use features includes redundancy back-up and is able to maintain cargo from within the temperature range +2 to +8 degrees. Specifications include more than 100 hours operation, under typical conditions, (+15 to +25 °C) without battery re-charge, or at least 30 hours under extreme conditions (down to -20 °C). It's batteries are able to be fully charged within 8 hours from any AC outlet.

To enable loading of such heavy cargo containers, some type of automatic electric cargo system is installed in the aircraft. The controls for the system are typically located at the entrance to the cargo compartment.

Usually, the cargo loading system has a series of floor panels that install to cover the cargo compartment floor. These panels are equipped with a spread field of ball rollers upon which containers can slide. Cylindrical and wheel-type rollers are also used. To move containers from side to side and fore and aft in the compartment, electric power drive units (PDUs) are installed in the floor panels and controlled from the operators control panel. These are LRU's consisting of an electric motor in a robust housing driving a roller wheel capable of maneuvering containers over the balls and rollers on the floor panels.

The PDU's come in various models depending on the type of aircraft. They are conveniently placed for easy access for maintenance. ***Figure 6-10*** shows a typical PDU manufactured by Aero Maintenance Group.

The LRU's motor includes differential gearing. Torque control couples the motor to the roller shaft. An erection system lifts the unit to engage the ULD so the roller can move the cargo. Any brushing, or scrubbing of the roller on the ULD is detected by a power sensor which results in modulation of the power to the motor to produce a smooth motion-a similar principle to anti-skidding.

Guides, restraints, stops and locks are all built in to assist in positioning and securing the loads. (***Figure 6-11***) Freight aircraft have larger and often very flexible cargo loading systems on the main cabin deck as well as in the lower cargo compartments. Containerized, palleted, and free cargo are always secured and loaded with respect to weight and balance requirements of the given aircraft.

AIRSTAIRS

Many airliners have an onboard set of stairs that can be deployed for use should an airport jetway malfunction or when no jetway service or portable stairs are available. These are called airstairs. They are typically an electrically deployed set of stairs stowed under the main cabin floor which extend out the fuselage from below



Figure 6-9. A temperature controlled container produced by Envirotainer AB.



Figure 6-10. Shows a typical PDU manufactured by Aero Maintenance Group.

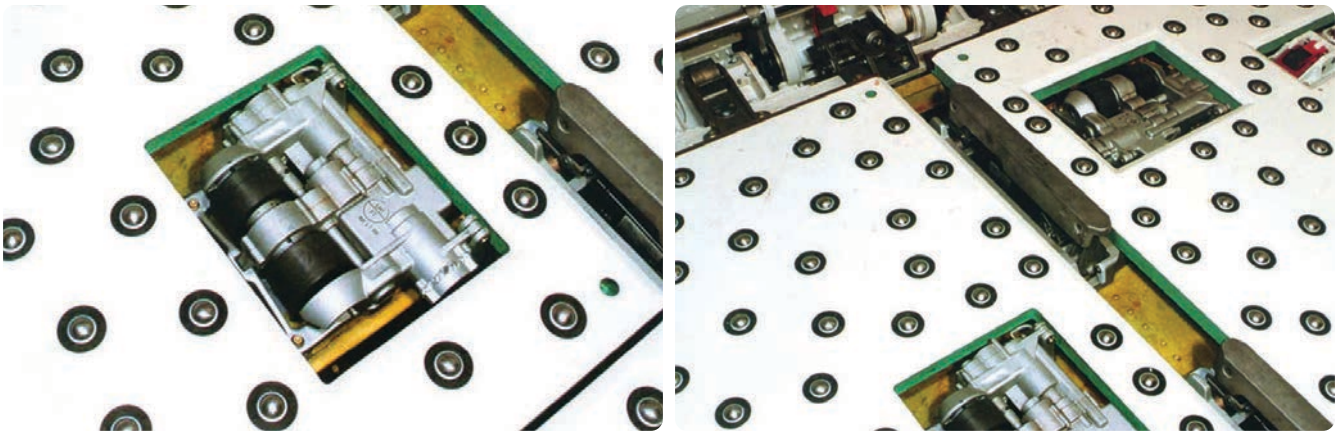


Figure 6-11. Cargo loading system floor panels with ball-rollers, electric power drive units and stops.

a cabin door to the ground. (***Figure 6-12***) Boeing 727 aircraft have air stairs that extend from the aft end of the fuselage. Air stairs are rarely used on large wide-body aircraft due to the height above the ground of the main passenger cabin. The size and weight of stairs to reach this level makes them too cumbersome.



Figure 6-12. Airstairs on an airliner stow under the main cabin floor.

Business aircraft make extensive use of a type of stair that attaches to and folds down from the main cabin entry doors. These are generally not considered air stairs but they perform the same function. Incorporation and use of air stairs has become rare, primarily to save the weight, as they are relatively heavy, and the extra space that the stowed air-stairs occupied. They are also quite costly to install and obviously require additional routine maintenance. Some aircraft, that still have them fitted, and have provision so that they can be operated from either inside the cabin, by cabin crew, or from the outside.

Early model Boeing 737 aircraft were fitted with forward airstairs and some with aft airstairs. The 737 system had two modes - NORMAL (requiring both AC and DC power) and STAND-BY (requiring only DC power from the external standby system, which came from a battery bus, so not requiring the battery Switch to be switched on). A safety feature built into the system requires the cabin entry door to be partially open before the air-stairs can extend. The 737 airstairs also has a maximum wind restriction speed, maximum 40 knots for operation, and a maximum 60 knots once extended. A duty cycle for the extension and retraction motor of not more than three operations every 20 minutes is also stipulated for the system.

Question: 6-1
Specifications for emergency equipment requirements on any given aircraft are found in _____.

Question: 6-2
Passenger seats on an airliner are installed by attachment into _____.

Question: 6-3
If a seat belt is not functional and cannot be changed before flight, the seat must _____.

Question: 6-4
A _____ for each seating class is required.

Question: 6-5
The black boxes that contain avionics computers are located _____.

Question: 6-6
Cabin furnishings materials are tested for _____.

Question: 6-7
Cabin entertainment equipment may use a _____ to provides services in each seat row.

Question: 6-8
Aircraft galleys connect into _____ and _____ systems.

ANSWERS

Answer: 6-1
CS-25 (Certification Standard 25)

Answer: 6-2
seat track mounting strips.

Answer: 6-3
be placarded and not used.

Answer: 6-4
lavatory.

Answer: 6-5
on a rack in an equipment bay.

Answer: 6-6
flammability and smoke production.

Answer: 6-7
a digital data bus.

Answer: 6-8
water.
electrical.



PART-66 SYLLABUS **LEVELS**

CERTIFICATION CATEGORY → **B2**

Sub-Module 07

FLIGHT CONTROLS (ATA 27)

Knowledge Requirements

13.7 - Flight Controls (ATA 27)

(a)	Primary controls: aileron, elevator, rudder, spoiler; Trim control; Active load control; High lift devices; Lift dump, speed brakes; System operation: manual, hydraulic, pneumatic; Artificial feel, Yaw damper, Mach trim, rudder limiter, gust locks; Stall protection systems.	2
(b)	System operation: electrical, fly-by-wire.	3

FLIGHT CONTROLS (ATA 27)

PRIMARY FLIGHT CONTROLS

Since the dawn of heavier-than-air flight and the discovery of the three-axis flight control network, airplanes continue to employ the three primary controls: elevator, aileron, and rudder. Before the discovery of the three-axis control system, gliders and airplanes were very difficult to control during flight. It should be noted that the same control inputs used by the pilot to fly small airplanes are used to control large aircraft.

The primary flight controls provide the aerodynamic force necessary to make the aircraft follow a desired flight path. (***Figure 7-1***) The flight control surfaces are normally hinged or movable airfoils designed to change the attitude of the aircraft by changing the airflow over the aircraft's surfaces during flight. These surfaces are used for controlling the aircraft about its three axes.

Typically, the ailerons and elevators are operated from the flight deck by means of a control stick, a control wheel, or yoke assembly and on some of the newer design aircraft, a joystick. Longitudinal control is the climb and dive movement or pitch of an aircraft that is controlled by the elevator. To cause the airplane to ascend from a straight and level attitude, the pilot pulls back on the control yoke or stick. Pushing the control forward lowers the nose of the aircraft for making descents.

Lateral control is the banking movement or roll of an aircraft that is controlled by the ailerons. To roll the airplane around the longitudinal axis, the pilot rotates the control wheel or moves the stick to the left or right, as desired. When the control is moved to the left, the left aileron rises above the wing and the right aileron descends below the wing. This causes the left wing to drop and the right wing to ascend resulting in a left bank. Some aircraft may use multiple ailerons so that each wing includes an inboard and outboard aileron. In such instances, the control network may lock out the outboard ailerons during high-speed flight. The inboard ailerons may be designed to slightly droop when the trailing edge flaps are extended. In addition to ailerons, spoilers may also be incorporated into aileron system. Each aircraft may have specific features contained in the flight control system to enhance the operation of the airplane.

Directional control around the vertical axis is the left and right movement or yaw of an aircraft that is controlled by the rudder. Some aircraft may employ lower and upper rudder control surfaces where both rudders are deflected for control while flying at lower airspeeds and a single rudder is used for high-speed flight. Foot pedals normally control the position of the rudder. Stepping on the right rudder pedal deflects the rudder to make a right turn. Stepping on the left pedal causes the aircraft to turn left. Most often when making turns during flight, the application of the rudder is made in combination with the aileron control. When the proper proportion of rudder and ailerons are inputted into the control system for the purpose of banking through a turn, the airplane is in a coordinated turn.

SECONDARY FLIGHT CONTROLS

Large airplanes will often employ a series of secondary flight controls to augment the performance of the aircraft during takeoff and landing and to supplement the controllability of the airplane throughout the various flight parameters. Secondary flight controls include: spoilers, leading edge flaps, leading edge slats, trailing edge flaps, and speed brakes. The secondary flight controls may further be used for aerodynamic braking once the airplane has landed.

A common secondary flight control involves the use of spoilers to assist in controlling the bank of the airplane. The flight spoilers rise on the side of the airplane where the aileron is deflected up. They remained down on the wing where the aileron is deflected below the surface of the wing. (***Figure 7-2***)

TRIM CONTROLS

Trim systems are added to flight control members to assist the crew in controlling the aircraft. Trim systems may also be used to control the aircraft, to a degree, during emergencies when the primary flight control system(s) fail or develop a fault. Pilots learn early in their training how to trim an aircraft to relieve them of having to maintain physical pressure on the controls. Included in the trim controls are the trim tabs, servo tabs, balance tabs, and spring tabs. Trim tabs are small airfoils recessed into the trailing edges of the primary control surfaces. Trim tabs can be used to correct any tendency of the aircraft to move toward an undesirable

flight attitude. Their purpose is to enable the pilot to trim out any unbalanced condition that may exist during flight, without exerting any pressure on the primary flight controls.

Servo tabs, sometimes referred to as flight tabs, are used primarily on the large main control surfaces. They aid in moving the main control surface and holding it in the desired position. Only the servo tab moves in response to control movements inputted by the pilots.

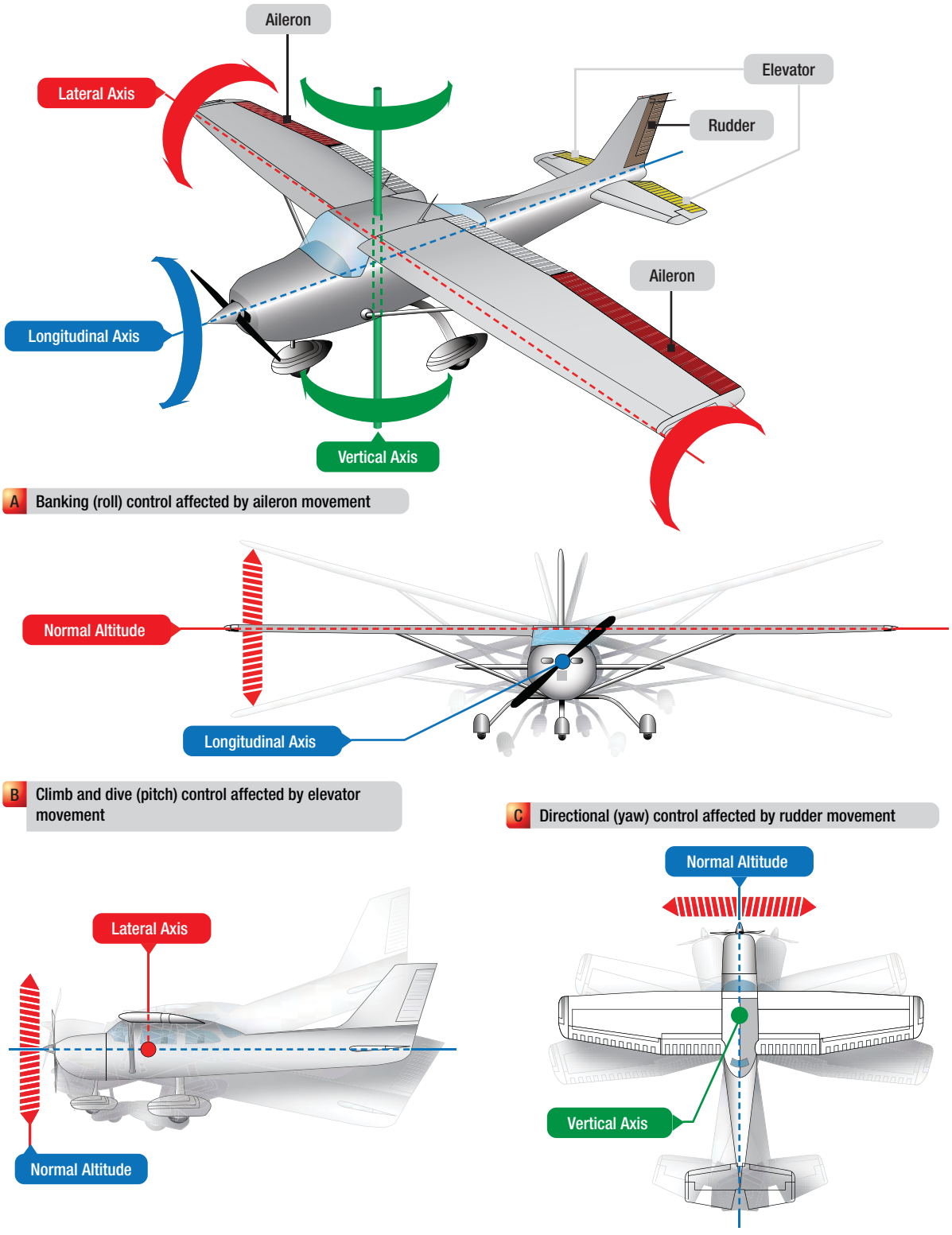


Figure 7-1. Aircraft 3-axis view with associated controls.



Figure 7-2. Flight Spoilers. Note how the flight spoilers, outboard of the engine, move up with the aileron (shown near the tip of the wing).

Balance tabs are designed to move in the opposite direction of the primary flight control. Thus, aerodynamic forces acting on the tab assist in moving the primary control surface. (***Figure 7-3***)

Spring tabs are similar in appearance to trim tabs, but serve an entirely different function. Spring tabs are used for the same purpose as hydraulic actuators–to aid the pilot in moving the primary control surface. In the ***Figure 7-4*** note how each trim tab is hinged to its parent primary control surface, but is operated by an independent control.

ACTIVE LOAD CONTROL

As the aircraft maneuvers through the atmosphere, numerous dynamic loads are generated. To counter these loads to a degree, engineers have developed active load control techniques. Active load control is a system whereby the stresses placed on the wings are redistributed to reduce the focus of the loads encountered during maneuvers involving rolls. During a roll, stresses on the wing increase as the g-load increases with bank angle. A standard approach to withstanding the additional loads is to make the wing stronger. Such reinforcement techniques add weight to the structure and empty weight of the airplane. Incorporating a system whereby the loads may be alleviated saves weight on the structural design of the aircraft. One example of an active load control system is the roll maneuver load alleviation (RMLA) approach. The RMLA network varies control surfaces deflections and extensions based roll command given by the pilot. Using the flexibility of the wing and torsional load placed on the wing structure, the RMLA system moderates or eliminates the deflection of the outboard aileron during rolls and relies on the deflection of the inboard ailerons and movement of leading edge flight control surfaces. A

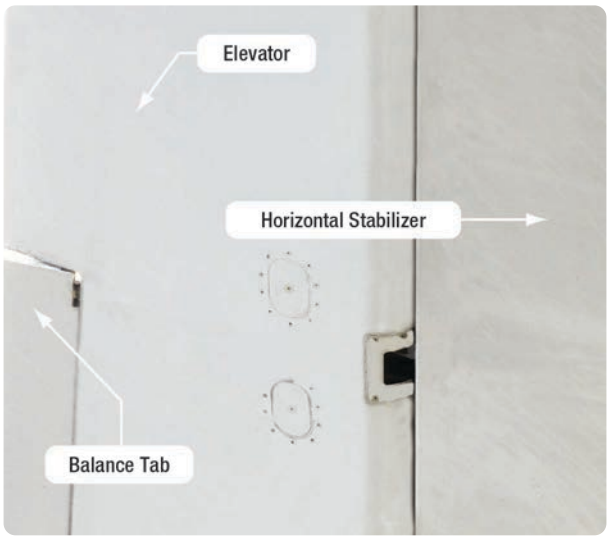


Figure 7-3. Balance Tab. Note that the balance tab moves in the opposite direction of the control surface. In this illustration the elevator is up while the balance tab is down.

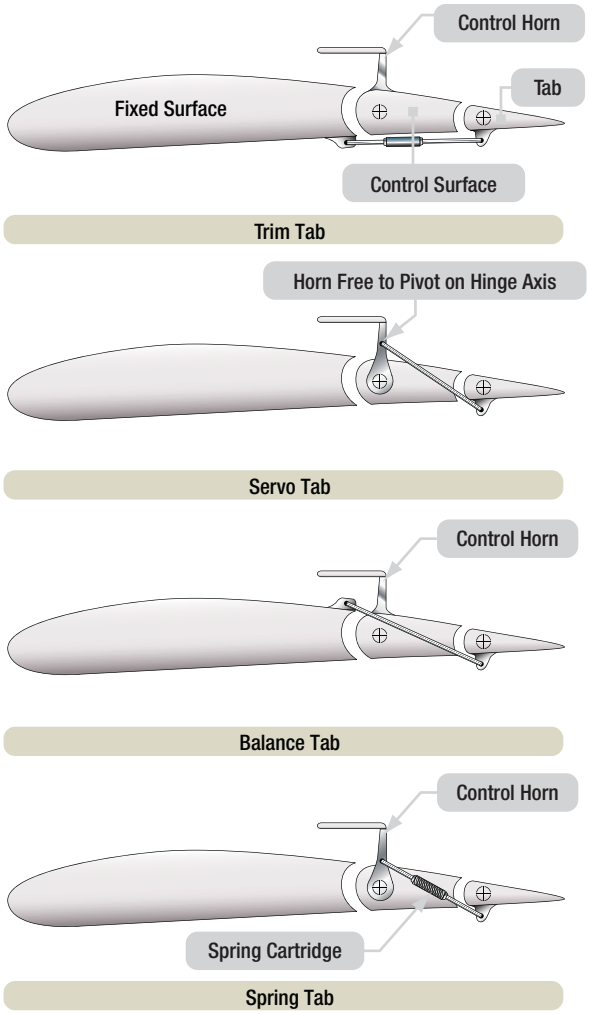


Figure 7-4. Types of trim tabs.

variety of scenarios must be met depending on the speed of the aircraft and aggressiveness of the control input. In the realm of fighter aircraft, active load control systems may be implemented to increase the roll rate of the aircraft at high speeds. Such increases in the maneuverability of the aircraft present distinct advantages. In the future, the aviation community will likely encounter a host of flight control systems available through the incorporation of advanced construction techniques and electronics.

HIGH LIFT DEVICES

Included in the high lift devices group of flight control surfaces are the wing trailing edge flaps, slats, leading edge flaps, and slots. They may be used independently or in combination to improve the performance of the aircraft. (***Figure 7-5 and Figure 7-6***)

The trailing edge airfoils (flaps) increase the wing surface area when extended, thereby increasing lift on takeoff, and decreasing the speed of the airplane during landing. These airfoils are retractable and fair into the wing contour. Other flaps are simply portions of the lower skin that extend into the airstream, thereby slowing the aircraft. Leading edge flaps are airfoils extended from and retracted into the leading edge of the wing. Some installations create a slot (an opening between the extended airfoil and the wing leading edge). The flap (termed slat by some manufacturers) and slot create additional lift at the lower speeds used during takeoff and landing. (***Figure 7-7 and Figure 7-8***)

Other installations have permanent slots built in the leading edge of the wing. At cruising speeds, the trailing edge and leading edge flaps (slats) are retracted into the wing proper. Slats are movable control surfaces attached to the leading edges of the wings. When the slat is closed, it forms the leading edge of the wing. When in the open position (extended forward), a slot is created between the slat and the wing leading edge. At low airspeeds, this increases lift and improves handling characteristics, allowing the aircraft to be controlled at airspeeds below the normal landing speed.

LIFT DUMP AND SPEED BRAKES

Lift decreasing devices are the speed brakes and spoilers. In some installations, there are two types of spoilers. Ground spoilers are extended only after the aircraft is on the ground, thereby assisting in the braking

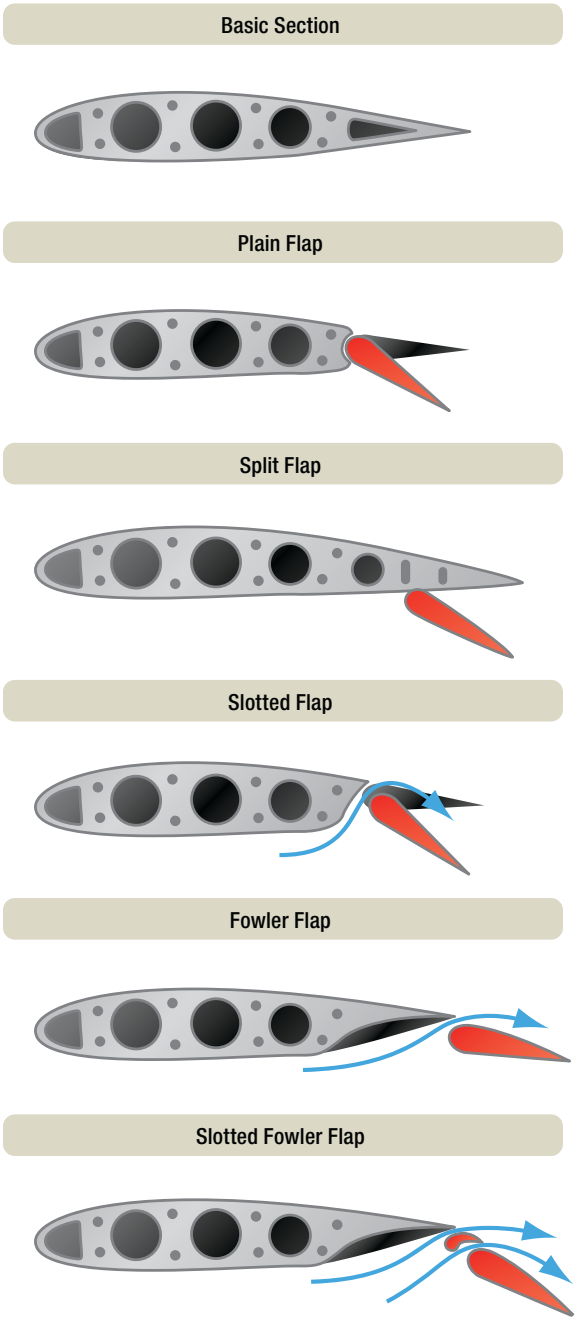


Figure 7-5. Types of flaps.

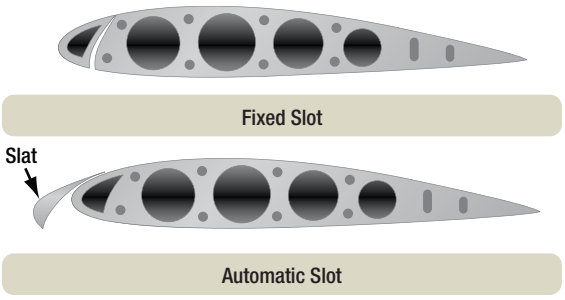


Figure 7-6. Slots.



Figure 7-7. Outboard Trailing Edge Flaps and Slats Extended. Note how the extension of the flaps and slats increase the surface area of the wing in addition to altering the shape of the wing to enhance low-speed performance and lift production.



Figure 7-8. Leading edge and trailing edge inboard flaps extended. In addition to the outboard trailing edge flaps and slats shown in the previous illustration, leading edge flaps and inboard trailing edge flaps, seen behind the landing gear, are extended to further enhance low-speed performance.

action. Flight spoilers assists in lateral control by being extended whenever the aileron on the same wing is deflected upward from neutral.

When actuated as speed brakes, the spoiler panels on both wings raise up. In-flight spoilers may also be located along the sides, underneath the fuselage, or back at the tail. In some aircraft designs, the wing panel on the up aileron side rises more than the wing panel on the down aileron side. This provides speed brake operation and lateral control simultaneously.
(***Figures 7-9 and 7-10***)



Figure 7-9. Speed Brakes for Aerodynamic Braking. The flight spoilers, shown as the two panels in the center, join the ground spoilers to maximize aerodynamic braking after landing. Note: the aileron near wing tip is not above the wing with flight spoilers acting as speed brakes.



Figure 7-10. Speed brake using the tail cone.

CONTROL SYSTEM OPERATION

MECHANICAL CONTROL

This is the basic type of system that was used to control early aircraft and is currently used in smaller aircraft where aerodynamic forces acting on the controls are not excessive. The controls are mechanical and manually operated by the pilot.

The mechanical system of controlling an aircraft can include cables, push-pull tubes, bell cranks, levers, jackscrews, cable drums, and torque tubes. The cable system is the most widely used because deflections of the structure to which it is attached do not affect its operation. Some aircraft incorporate control systems that are a combination of mechanical control mechanisms. These systems incorporate cable assemblies, cable guides, linkages, bell cranks, push-pull tubes, torque tubes, adjustable stops, and control surface snubbers or mechanical locking devices. Surface locking

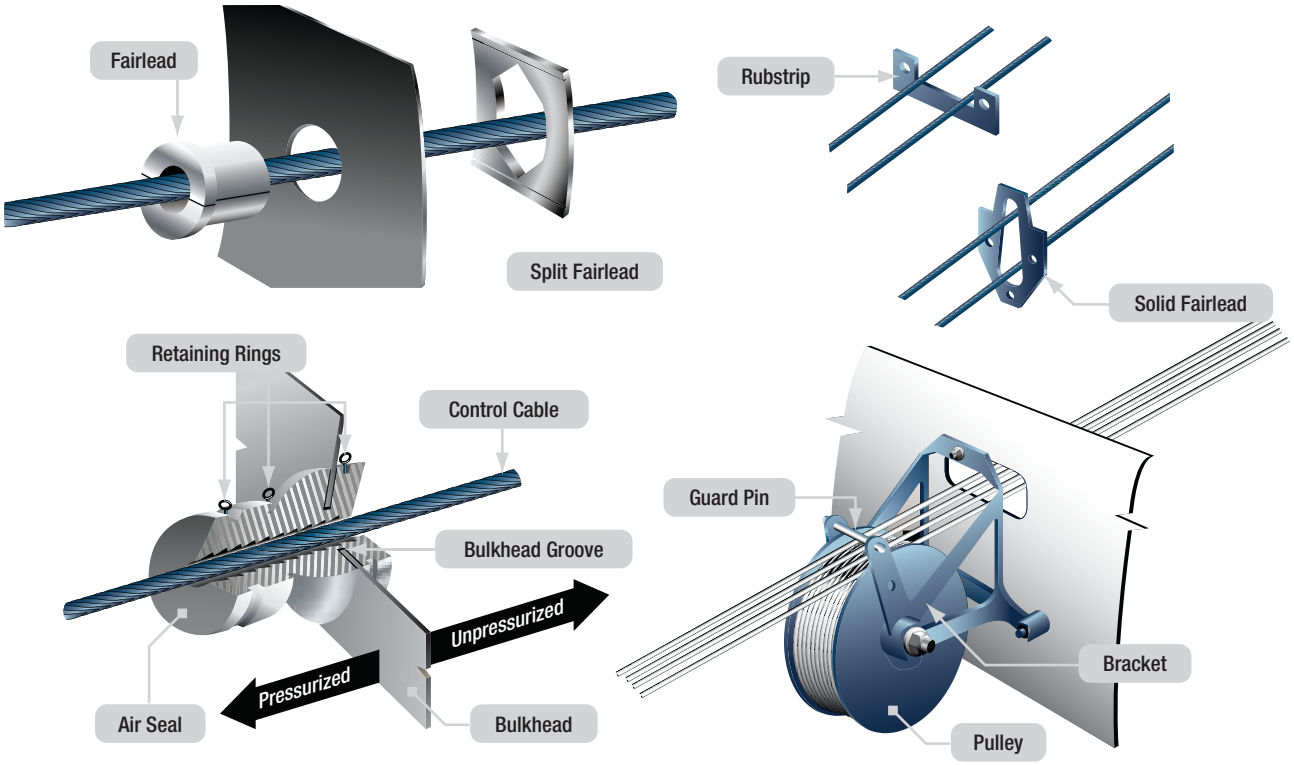


Figure 7-11. Fairleads, rubstrips, cable seals, and pulleys used to guide control cables.

devices, usually referred to as gust locks, limit external wind forces from damaging the control system while the aircraft is parked or tied down.

CONTROL CABLES

Control cables used in aviation are flexible steel wires. Cables are very strong when placed under a tensile or pulling load. Flexible cables do not have strength when pushed. Consequently, when cables are used for flight controls, they often employ multiple cables so that one cable is under tension when the control input is made in one direction and the other cable is under tension when the control input is made in the opposite direction. Control cables may run the entire length from the control mechanism manipulated by the crew to the control quadrant, cable drum, torque tube, bell crank, or lever that connects to the control surface. Other cables may run from the pilot's control mechanism to hydraulic valves or other devices that ultimately deflect the control surfaces. (***Figure 7-11***)

Most manufacturers of large aircraft will include some means whereby cables may be identified through labeling. Through the use of this naming system, technicians are able to identify the function of a cable or identify the unit in need of service operations.

Cables that extend from pressurized portions of the aircraft to unpressurized areas use seals to prevent loss of cabin pressure. Significant air leaks at such locations may affect the operation of the pressurization system.

Because airplanes stretch and contract with changes in temperatures, some airplanes use cable tension regulators to maintain proper cable tension throughout the range of conditions. Such devices are needed as the expansion and contraction of airplane structures made of aluminum are greater than that of the steel cables.

PUSH-PULL TUBES

Where cables only have strength when they are place under tension, or pulled, push-pull rods are able to transmit force in either direction. Push-pull rods may be solid or hollow. The ends attached to the push-pull rods may be fixed or adjustable. (***Figure 7-12***)

BELLCRANKS AND LEVERS

Bellcranks are constructed so that a series of levers are able to receive an input signal and deliver an output. The output from a lever or bellcrank may amplify the input or vice-versa. Frequently, bellcranks change the direction of movement. The input signal may come from a lateral direction and the output motion made in a longitudinal direction and vice-versa. (***Figure 7-13***)

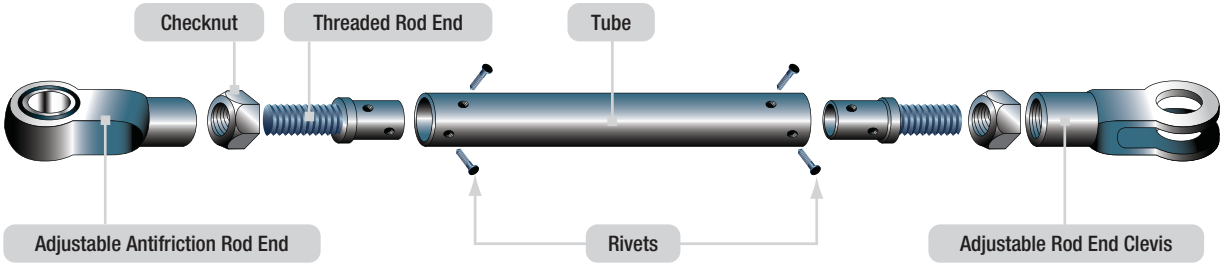


Figure 7-12. Exploded view of push-pull rod.

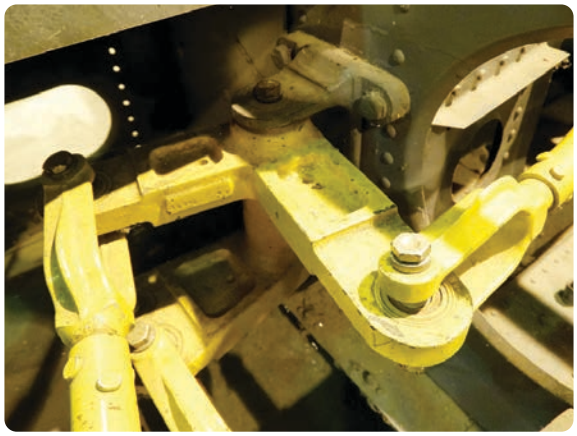


Figure 7-13. Bell crank with push-pull tubes.

JACKSCREWS

Jackscrews are commonly employed for moving surfaces that experience extreme aerodynamic loads, such as horizontal stabilizers and flaps. Jackscrews are threaded units that convert rotary motion into linear travel.

TORQUE TUBES

Torque tubes are used in many areas of the flight control system. Torque tubes apply torsional, or rotating, motion to a member of the control system. Often torque tubes receive their input from control cables or push-pull rods. (***Figure 7-14***)

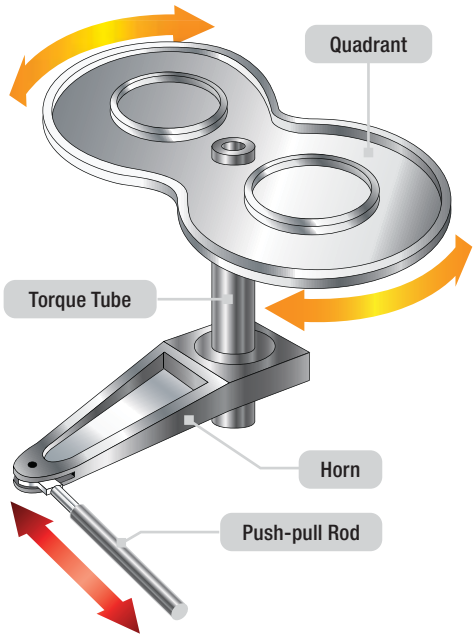


Figure 7-14. Torque tube transmitting input into rotary output.

HYDROMECHANICAL CONTROL

As the size, complexity, and speed of aircraft increased, actuation of controls in flight became more difficult to perform strictly using physical strength. It soon became apparent that the pilot needed assistance to overcome the aerodynamic forces encountered by the control surfaces in order to control the aircraft. Spring tabs, which were operated by the conventional control system, were moved so that the airflow over them actually moved the primary control surface. This was sufficient for the aircraft operating in the lowest of the high-speed ranges (250±300 mph). For higher speeds, a power-assisted (hydraulic) control system was designed and implemented.

Conventional cable or push-pull tube systems link the flight deck controls with the hydraulic system. With the system activated, the pilot's movement of a control causes the mechanical link to open and close servo valves, thereby directing hydraulic fluid to and from actuators, which convert hydraulic pressure into control surface movements. Because of the mechanical advantage of the hydromechanical flight control system, the pilot cannot feel the aerodynamic forces acting on the control surfaces, and there is a risk of overstressing the structure of the aircraft. To overcome this problem, aircraft designers incorporated artificial feel systems into the design that provides increased resistance to the controls at higher speeds. In essence, the artificial feel simulates what the pilot would sense in terms of control system input if the aircraft did not have a hydraulic

control network. Additionally, some aircraft with hydraulically powered control systems are fitted with a device called a stick shaker, which provides an artificial stall warning to the pilot.

Large aircraft often have the mechanical control network connected to the flight control as a back-up means of controlling the aircraft in the event of a hydraulic system failure or failure of the hydraulic control system. Often, aircraft are designed with multiple hydraulic actuation systems with the mechanical back-up to ensure that the crew is able to control the aircraft.

ELECTRICAL AND ELECTRONIC CONTROLS

Modern aircraft have widely adopted electronics in their flight control systems. Normally multiple computers are incorporated in the control network with computers interfacing with autopilots, auto-landing, auto-speed braking, flaps, stall warning, ground proximity system, and etc. Regardless of the intricacy of computers involved in the control of the aircraft, their main function is to translate the control inputs made by the crew into actual control surface deflections. Electric trim is often found to control the position of the horizontal stabilizer. To make the trimming operation convenient, the switches to operate the trim is located in the control yoke as shown in the figure below. (***Figure 7-15***) Pilots must activate both switches simultaneously to engage the trim motor(s). A mechanical means of elevator trim is also provided on most aircraft. This mechanism is commonly found on the pedestal and drives the trim transmission using flexible cables. On many airplanes, the mechanical trim system moves when the horizontal stabilizer is trimmed via the pilot controlled electrical switches or when the autopilot trims the stabilizer.

Because the horizontal trim system is able to pitch the airplane nose up or down in a commanding fashion, some aircraft are equipped with a horizontal stabilizer trim brake system. This mechanism arrests the motion of the stabilizer during trimming operations when the movement of the elevator control inputted by the pilot opposes the direction of trim. Electrical controls may further serve as a back-up system. In the figures below, (***Figure 7-16 to Figure 7-19***) the hydraulic motor normally drives the flap transmission. In the event of a complete hydraulic failure or fluid depletion, the crew may operate the flaps using an electric motor to power the flap transmission. The ability to extend flaps for landing enhances the safety of the operation.

PNEUMATIC

Figure 7-20 shows another technique for assisting in the movement of a control surface of a large aircraft. It is called balance panel. Not visible when approaching the aircraft, it is positioned in the linkage that hinges the control surface to the aircraft. Balance panels have been constructed typically of aluminum skin-covered frame assemblies or aluminum honeycomb structures. The



Figure 7-15. Trim Switches for the Horizontal Stabilizer. Both switches must be activated simultaneously to engage the trim motor. Upward movement of the switches commands a nose down reaction and downward movement of the switches provide a nose up change in flight attitude.

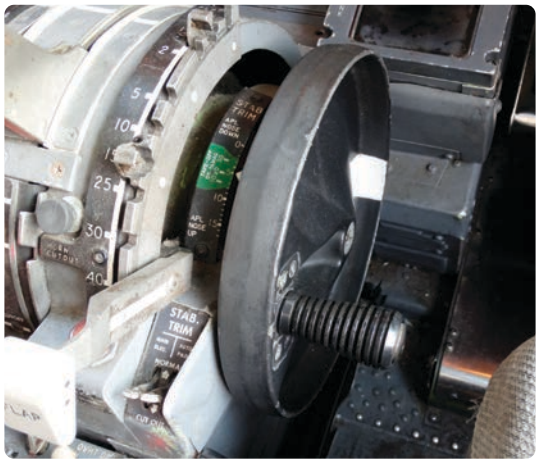


Figure 7-16. Mechanical Trim System and Trim Position Indicator for Horizontal Stabilizer. The protruding handle may be stowed within the trim wheel. The green band is used to signify the proper position of the horizontal stabilizer for takeoff. Trimming the horizontal stabilizer outside the green band provides an unsafe for takeoff warning scenario.



Figure 7-18. Horizontal stabilizer brake release knob.

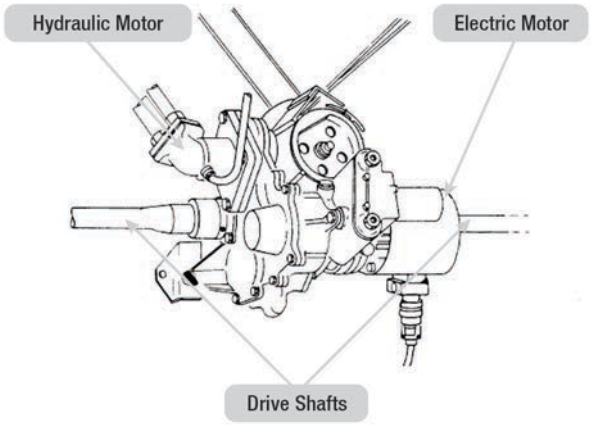


Figure 7-19. Flap Motor Drive. Flap system normally uses the hydraulic motor to extend and retract flaps. Electric motor serves as back-up system to ensure flaps are extended for landing.

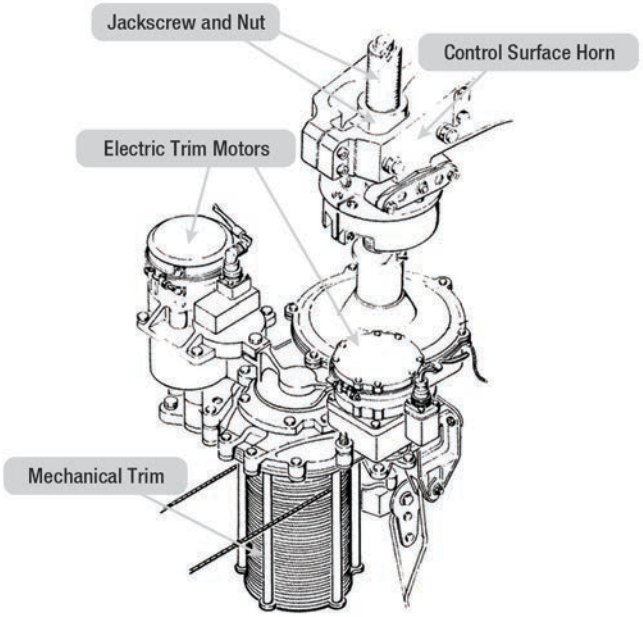


Figure 7-17. Electric and Mechanical Trim Systems for the Horizontal Stabilizer. The motor shown on the left is for manual trim operations from the switches on the control yokes, the motor on the right is used with the autopilot, and the cable drum is connected to the mechanical trim system on the pedestal.

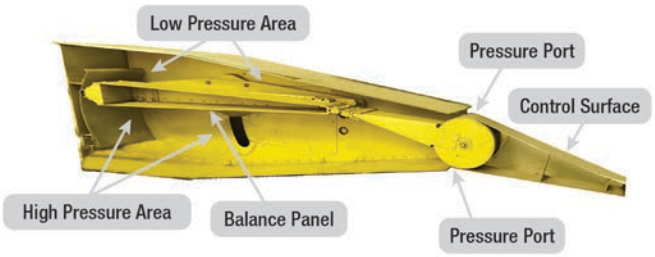


Figure 7-20. Balance Panel. In this illustration, the control surface is deflected down. The two pressure ports at the trailing edge of the wing have different pressures. The upper pressure port develops and transmits to the balance panel a low pressure while the lower pressure port develops a relatively higher pressure. The pressure differential acting on the balance panel assists in the movement of the flight control surface.

trailing edge of the location where the flight control is mounted is sealed to allow controlled airflow in and out of the hinge area where the balance panel is located. In essence, two chambers are established.

The pressure differential generated by the deflection of the control surface allows the balance panel to assist in the movement of the flight control. When the control surface is moved from the neutral position, differential pressure builds up across the balance panel. This differential pressure acts on the balance panel in a direction that assists in the control surface movement. For slight control surface movements, deflecting the control tab at the trailing edge of the surface is undemanding enough to not require significant assistance from the balance panel. But, as greater deflection is commanded, the force resisting control tab and control surface movement becomes greater and assistance from the balance panel is needed.

ARTIFICIAL FEEL

Aircraft that use purely mechanical flight control systems do not require artificial feel on the controls. The resistance transmitted through the control system provides the pilot with a natural feel regarding the magnitude of control input and associated stresses placed on the aircraft. (***Figure 7-21***) Aircraft that move control surfaces solely by hydromechanical and/or electromechanical means deprive the pilot of the feel of a mechanical control system. Consequently, the load

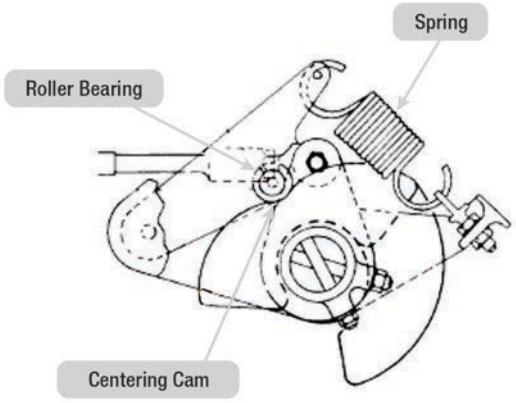


Figure 7-21. Artificial Feel Mechanism. As the pilot further deflects the control, the roller bearing rides up the side of the cam resulting in more spring opposition.

or resistance generated by the flight control surface as it is deflected into the airstream is not directly transmitted to the pilot. As a substitute, aircraft manufacturers have developed artificial feel systems to provide feedback regarding control input. Without artificial feel, pilots could generate high levels of loads on the aircraft structure without realizing it. Mechanisms used to produce artificial feel may be mechanical. A common approach is to use a spring-loaded roller that fits into the valley of a flattened v-shaped cam. As the control input is increased, the roller rides higher up the side of the cam, thereby increasing the spring resistance felt by the pilot.

YAW DAMPER

One common control system is the yaw damper used on many large aircraft. Typically associated with aircraft using swept wings that generate a motion referred to as a Dutch roll, the purpose of the yaw damper system is to counter the rolling tendency of the aircraft during flight. Yaw dampers work when the aircraft is controlled manually by the flight crew or during operations involving the autopilot. The yaw damper system provides inputs to the rudder in proportion to the yaw rate of the aircraft and in a direction that negates the oscillations that would otherwise take place during flight. Aside from increasing the stability of the aircraft, the yaw damper provides a smoother ride for the passengers.

MACH TRIM

Airfoils traveling at low subsonic speeds have a center of pressure acting on the wing that is approximately one-quarter the distance of the chord, aft of the leading edge. The center of pressure does not move much until the aircraft begins traveling at high speeds. When the aircraft passes through the air at speeds around Mach 0.7 and above, the center of pressure begins to move aft on the wing. As aircraft approach the speed of sound, their form may further accelerate the air flowing over the wings and other portions of the aircraft. When the aircraft reaches its critical Mach number, shock waves may develop over the wing. The area in front of the shock waves develops high lift. This action continues to travel aft as the aircraft gains more speed. The rearward movement of the lift production causes the aircraft to experience Mach tuck resulting in a nose down flight attitude.

To counter Mach tuck and keep the aircraft flying in a level attitude, Mach trim is incorporated in the control network. Mach, or the speed of sound, is not a constant value. The speed of sound varies largely with changes in temperature.

Another factor that enters the controllability of the airplane involves coffin corner. The operation of the airplane enters coffin corner when the stall speed of the aircraft flying at high altitudes for a given weight and load factor approaches the critical Mach number. Aircraft entering the coffin corner configuration may be very difficult to keep in stable flight. Any reduction of airspeed will cause the plane to stall and any increase of airspeed will generate a loss of lift due to entering critical Mach. Pilots strive to keep the airplane out of the portion of the flight envelope known as coffin corner.

Mach trim basically trims the nose of the aircraft up as Mach tuck begins to act on the aircraft. Most systems of Mach trim are automatic in that the flight crew does not have to manually change trim settings. The crew may notice changes in trim as the control network implements Mach trim input. To ensure the crew does not lose Mach trim during flight, airplanes will typically have redundant Mach trim systems.

RUDDER LIMITER

Airplanes that have a relatively low speed range (e.g., 200 knots) generally do not need flight control networks that limit control surface travel at higher speeds. The structure of such airplanes is capable of absorbing the loads generated by large control surface deflections. But airplanes that are capable of traveling at high speeds (e.g., in excess of 350 knots) would require an extensive amount of structural reinforcement to handle the loads

generated by large control deflections. Such addition to the structure results in extra weight. To combat the need for excess structure, many high-speed aircraft resort to limiting control surface deflection during high-speed operation. This is similar to operating an automobile. When traveling along a highway at high speeds, the driver does not apply large inputs to the steering wheel, but rather small inputs. The same automobile may need full steering deflection while traveling at low speeds as in the example of parking.

Some aircraft reduce the travel available to the rudder based on the speed of the aircraft. At low speeds the need for substantial rudder travel is required. At high speeds (e.g., above 250 knots) the effectiveness of the rudder is increased, thereby reducing the need for large deflections. For the same number of degrees of rudder deflection, the load placed on the structure increases with the speed of the aircraft. Consequently, aggressive rudder deflections at high speeds may exceed the structural limitation of the aircraft. To minimize the risk of exceeding structural limitations, aircraft may include rudder limiters that reduce rudder deflection at high speeds. In other words, full rudder deflection is only available at lower airspeeds. For example, an airplane may have 30° of rudder deflection in the left and right directions at low speeds, such as takeoff, landing, climb, etc., with full pedal travel. At cruise speeds the rudder limiter restricts the rudder deflection to 7° left and right with full pedal travel.

GUST LOCK SYSTEMS

Aircraft that use mechanical flight control systems will typically include a method for locking the controls when the aircraft is parked. Normally referred to as gust locks, these mechanism may either be separate from the control system or an integral part of the control. Separate gust locks may consist of a device that extends from a stationary part of the aircraft, such as the wing, and passes over and locks in place the flight control surface (e.g., the ailerons). Another technique is to lock the movement of the flight controls with pins and other devices. Rather than being on the exterior of the aircraft, such locking devices are installed in the flight compartment to keep the controls from moving. By physically locking the flight controls in place, damage to the structure or control network is eliminated during times when the aircraft is parked and the wind acts to deflect the flight control surfaces. Gust locks will typically include a warning streamer with the following or similarly worded phrase: "REMOVE BEFORE FLIGHT".

Large aircraft that have hydraulic assist systems to move flight control surfaces often include gust dampers in their power control units. By using hydraulic fluid contained within the power control units that drive the flight control surfaces during flight, movement of the control surfaces by wind feeds a force into the hydraulic units. These mechanisms provide gust snubbing by forcing hydraulic fluid through special bypass valves and other devices. The end result is that the flight control surfaces are protected from wind gust damage.

PROTECTION/WARNING SYSTEMS

Stall warning systems incorporated on modern day jetliners are far more advanced than those used on smaller general aviation airplanes. Stall warning systems typically involve multiple computers that monitor the configuration of the aircraft and flight data. Analyzing those bytes of information, the stall warning computers calculate when an airplane is nearing a stall condition. In such instances, a stick shaker that provides a violent shaking motion to the control yoke will give the crew warning that a stall is imminent. Some airplanes include a stick nudger or pusher that applies a nose down input to the elevator in an attempt to avoid the impending stall. The flight crew has the option of overcoming the input made by the stick nudger or pusher. The need for stick nudgers or pushers is due to the stall recovery characteristics of many larger airplanes. Where the stick nudger or pusher is designed to avoid a stall, the stick shaker is a stall warning mechanism. (***Figure 7-22***)



Figure 7-22. Stick shaker bolted to control yoke post. This motor violently shakes the control yoke.

The stall speed of the aircraft is affect by a number of variables. In calculating the potential stall, the computers look at the position of the flaps, slats, speed brakes, airspeed, angle of attack, and other parameters. Failure to take the proper corrective action during a stall may lead to serious consequences. Airplane stalls have claimed many lives over the history of flight. Angle of attack sensors commonly use a vane on the side of the fuselage that provides data regarding the angle that the aircraft is passing through the atmosphere. As the airplane changes its angle of attack, the vane reacts by rotating to a new position that is parallel to the airflow passing across it and sending a signal to the appropriate computer(s). As the data provided by the angle of attack sensor is critical to the safety of the aircraft during flight, the device is normally equipped with a heater to prevent the build-up of ice. (***Figure 7-23***)

Unsafe for takeoff configuration warning is typically provided on large aircraft. This warning, often an aural warning sound in conjunction with a visual warning light(s), is given when there is an unsafe condition prior to takeoff. Such conditions include the improper position of the flaps or slats, the horizontal stabilizer position, the extension of speed brakes, the parking brake set, and so on. The warning is normally triggered when the crew advances the throttle and a problem is present. The value of this system is difficult to assess as attempting a takeoff when the airplane is improperly configured is likely to result in a tragic incident.

Landing configuration warning is provided when the airplane is improperly set up for landing. One common warning occurs when the all members of the landing gear are not locked in the down position and a throttle lever is reduced to a low power setting. A warning is frequently given when the flaps are extended for landing and the landing gear is not down and locked.



Figure 7-23. Angle of attack vane.

The extension of spoilers at low altitudes is likely to provide an unsafe landing configuration warning. As with the unsafe for takeoff warning network, the crew receives an aural and visual warning when the landing configuration is improper.

SECTION B

FLY-BY-WIRE CONTROL

The fly-by-wire (FBW) control system employs electrical signals that transmit the pilot's actions from the flight deck through a computer to the various flight control actuators.

FBW evolved as a way to reduce the system weight of the hydromechanical system, reduce maintenance costs, and improve reliability. Electronic FBW control systems can respond to changing aerodynamic conditions by adjusting flight control movements so that the aircraft response is consistent for all flight conditions. Computers can be programmed to prevent undesirable and dangerous characteristics, such as stalling and spinning. In addition, many of the later generation military high-performance aircraft are not aerodynamically stable. This characteristic is designed into the aircraft for increased maneuverability and responsive performance. Without the flight control computers reacting to the instability, the pilot would experience great difficulties controlling the aircraft.

The cockpit controls are devices that convert the movements or pressures exerted by the pilot into electrical signals which are sent into a computer programmed with the flight characteristics of the airplane. This computer output is directed through wires to electro-hydraulic valves that convert the electrical signal into hydraulic fluid flow. This flow changes the position of a main control valve which directs hydraulic fluid to the appropriate control actuators. Within the actuators, linear variable displacement transducers complete the loop and send feedback signals to the computer, informing it of the amount of actuator movement. Rather than using a control wheel or stick that actually moves, some fly-by-wire equipped aeroplanes have sidestick controllers to fly the airplane. Pressures exerted on the controller mounted on the cockpit side console are converted into electrical signals just as are movements of conventional controls. Airbus fly-by-wire systems use such a process. (***Figure 7-24***)

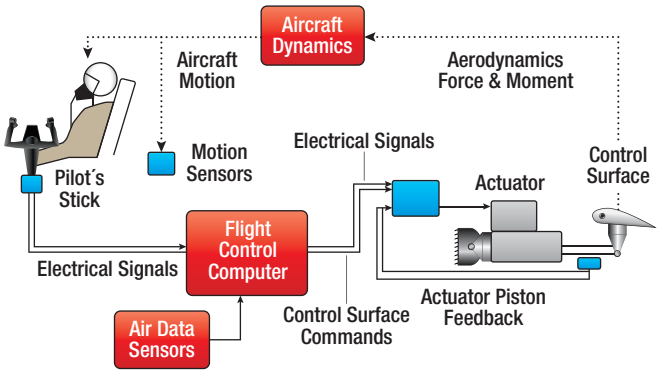


Figure 7-24. The various inputs affecting a fly-by-wire control system.

In normal configuration (airborne with all systems operational) the aircraft is said to be in "Normal Law". In this law, the computer supports the pilot by controlling elevators in a turn, lateral attitude hold, automatic pitch trim, turn coordination, dutch roll damping and engine failure compensation. It also prevents unsafe maneuvers. Even if a side stick is pulled fully back, which would normally lead to a stall, the maximum angle of attack and pitch attitude are limited by the computer. Information of many other on-board sources such as air data, inertial reference, slat and flap positions, etc. are sent to the main flight control computers to calculate and prevent dangerous situations like extreme attitudes, overspeed, excessive load factors, flutter, and stall.

FLY-BY-WIRE REDUNDANCY/FAIL SAFE

Hydraulically operated flight primary control systems are often designed with redundancy. This means that two or three independent actuators perform the same function from independent hydraulic systems. In addition, for safety reasons on most airplanes independent computers can perform the exact same functions. Typically, a primary computer is in active mode and a secondary is in stand by mode, monitoring the primary computer. In case of failure of the active primary system, an automatic changeover will take place to engage the secondary computer and place the faulty primary computer in an isolated state.

An additional aspect of fail safe operation is redundancy of the primary flight control actuators. In this case, each primary control surface is provided with a primary and secondary hydraulic system. The loss of a primary flight control hydraulic system would allow sufficient control to be maintained.

OPERATION OF A FLY-BY-WIRE HYDRAULIC AILERON ACTUATOR

The following describes in detail the operation of a fly-by-wire hydraulic aileron actuator on an Airbus A-320. ***Figure 7-25*** with the following references.

1. Solenoid Valve
2. Pressure Line Closing Valve
3. Return Line Closing Valve
4. Servovalve
5. Mode Selector Valve
6. Damping Orifice
7. Check Valve
8. Return Relief Valve
9. Fluid Reserve
10. Feedback Transducer
11. Linear Variable – Differential Transformer

P - Pressure Port
R - Return Port

Normal Operation With The Servo Control Pressurized

The pressurization of the servo control involves the opening of the pressure-line closing valve and of the return-line closing valve. Thus, the servo valve is supplied from the high pressure system and the return system of the servo control is connected to the low pressure system.

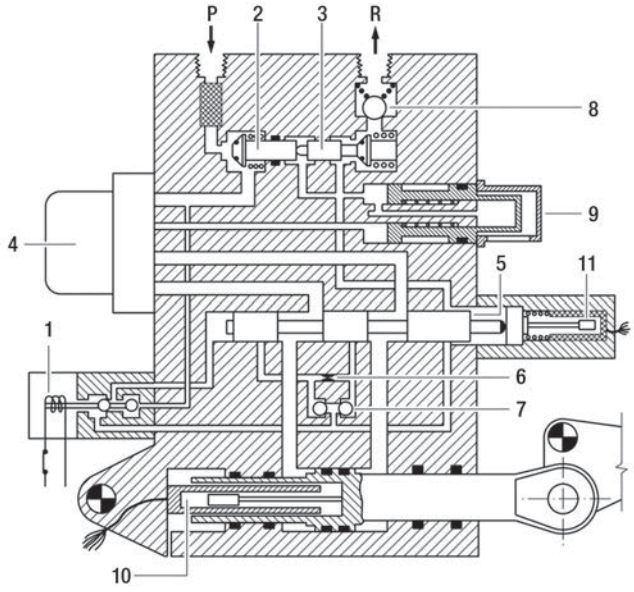


Figure 7-25. Fly-by-wire hydraulic aileron actuator diagram for an Airbus A-320.

Servocontrol In The Active Mode

In the active mode, the solenoid valve is energized and lets in the high pressure flow which then puts the mode selector valve in the active mode. The two chambers of the actuator are thus connected to the servo valve control lines. The servo control is then in the active mode. The Linear Variable-Differential Transducer supplies an electrical signal which identifies this change. The feedback transducer provides feedback on the servo loop.

Servocontrol In The Damping Mode:

In the dampening mode, the solenoid valve is de-energized and the mode selector valve moves under the action of its spring. This causes the two chambers of the actuator to be interconnected through a damping orifice. The linear variable-differential transformer identifies this change of state. The check valve and the fluid reserve holds the volume of fluid in the actuator chambers if the temperature of the hydraulic fluid changes or if there is a leak. The reserve of fluid is permanently connected to the return line of the servo valve. The return relief valve increases the pressure in the return line and permits it to fill the reserve of fluid.

Operation After A Hydraulic Failure

In the event of failure, the pressure line closing valve and return line closing valve close isolating the servo control from the aircraft hydraulic system. The return relief valve keeps the pressure in the low pressure line higher than the pressure in the aircraft return system and so permits it to fill the accumulator. If there is a rupture in the return system, the return relief valve holds the volume of fluid in the accumulator. The mode selector valve moves under the action of its spring and the servo valve then operates in the dampening mode.

Operation After An Electrical Failure

In the event of an electrical failure, the solenoid valve is de-energized. The mode selector valve moves under the action of its spring and the servo control valve operates in the dampening mode.

Maintenance And Rigging

Maintenance of the actuators are done in an "on condition". Primarily, the roller bearings are re-lubricated. Other components such as the filters, servolators, solenoid valves, mode selector valves and transducers and differential pressure transducers are Line Replaceable Units (LRUs).

After a replacement of the servo controls, it is necessary to adjust the feedback transducer. Adjustments may also be needed to re-establish zero voltage in the secondary windings when the aileron is in neutral position. This is done through an adjustment on the feedback transducer.

FLY-BY-OPTICS

Aircraft designers continue to enhance flight control systems. Where fly-by-wire systems are able to use computers to control the position of multiple flight control surfaces, fly-by-optics further improve the ability of the system to transfer data. Fly-by-optics networks are able to transfer data at higher speeds than wired systems. Fly-by-optics systems, also known as fly-by-light, are more immune to electrical interference that may affect fly-by-wire systems.

FLY-BY-WIRELESS

The next generation of flight controls include fly-by-wireless systems. Similar to fly-by-wire systems, fly-by-wireless networks offer a reduction in weight of the aircraft by removing the extensive bundle of wires used in fly-by-wire aircraft. The weight savings further translate into a measure of efficiency. Another benefit of the fly-by-wireless system involves reduced maintenance. Through years of service, the fly-by-wire harnesses will develop maintenance issues with connectors, corrosion, broken wires, etc. Every connection becomes a potential point of failure. Removing the wires from the flight control network, or reducing the number of associated wires, saves maintenance costs over the life of the aircraft.

QUESTIONS

Question: 7-1
Of the following, which are considered primary flight controls and which are considered secondary?
- Elevators
- Ailerons
- Speed Brakes
- Rudders
- Flaps
- Trim Tabs

Question: 7-2
What effect does flap extension have during takeoff and landing?

Question: 7-3
In which mode of operation (push or pull) do flexible cables have strength?

Question: 7-4
What is the purpose of a horizontal stabilizer trim brake system?

Question: 7-5
What is the advantage of fly-by-optics control systems versus fly-by-wire?

Question: 7-6
What are two benefits of a yaw damper system?

Question: 7-7
The fly-by-wire system utilizes _____ to control the actuators.

Question: 7-8
When landing configuration is improper the pilot receives a _____ and _____ warning.

ANSWERS

Answer: 7-1
Elevators, rudder, and ailerons are primary, flaps, trim tabs, and speed brakes are secondary.

Answer: 7-2
Increases lift allowing for lower takeoff, approach, and landing speeds.

Answer: 7-3
When being pulled.

Answer: 7-4
Arrests the motion of the stabilizer during trimming operations when the pilots elevator control movement opposes the trim direction.

Answer: 7-5
Eliminates electromagnetic interference.

Answer: 7-6
Increased stability for the pilot; smoother ride for the passengers.

Answer: 7-7
electrical signals.

Answer: 7-8
visual.
aural.



PART-66 SYLLABUS **LEVELS**

CERTIFICATION CATEGORY → **B2**

Sub-Module 08

INSTRUMENTS (ATA 31)

Knowledge Requirements

13.8 - Instruments (ATA 31) 3

Classification;
Atmosphere;
Terminology;
Pressure measuring devices and systems;
Pitot-static systems;
Altimeters;
Vertical speed indicators;
Airspeed indicators;
Machmeters;
Altitude reporting/alerting systems;
Air data computers;
Instrument pneumatic systems;
Direct reading pressure and temperature gauges;
Temperature indicating systems;
Fuel quantity indicating systems;
Gyroscopic principles;
Artificial horizons;
Slip indicators;
Directional gyros;
Ground Proximity Warning Systems;
Compass systems;
Flight Data Recording systems;
Electronic Flight Instrument Systems;
Instrument warning systems including master warning systems and centralized warning panels;
Stall warning systems and angle of attack indicating systems;
Vibration measurement and indication;
Glass cockpit.

AIRCRAFT INSTRUMENTATION

Aircraft require instruments in the cockpit, so that the pilot can control all of the systems and fly correctly in a desired manner. The overall aircraft cockpit layout and embedded instrumentation is very important. All instruments are installed inside the cockpit adhering to basic ergonomic principles to make it easier for the pilot to monitor and control an array of aircraft systems. A modern cockpit (flight deck) no longer resembles cockpits that were used 50 years ago. Modern aircraft fly faster, higher and farther, thereby requiring that the crew have better, more advanced control of aircraft speed, altitude, attitude etc. The crew must also manage large amounts of data: Navigation systems, engines, cabin pressurization and temperature, fuel consumption, etc. Therefore, on these very complex aircraft the crew must also manage a large number of parameters. Given all the various safety conditions that are required, this cannot be done unless the instrument layout on the control panels meet international standards.

This can only be done with a minimum of risk when all aircraft instrument panel layouts meet internationally agreed standards. These are determined by the International Civil Aviation Organization (ICAO). The ICAO has sent minimum acceptable standards for aircraft certification. These standards do not replace the national regulations for each country, but are the minimum standards required for international certification.

CLASSIFICATIONS OF AIRCRAFT INSTRUMENTS

There are three types of aircraft instruments; flight instruments, engine instruments, and navigation instruments. The flight instruments category is further sub-divided into static, gyroscopic, and magnetic instruments. Magnetic instruments, however, are increasingly becoming obsolete with advancements in digital avionics, where computers do necessary processing and display vital system information on one or several screens inside the "glass cockpit". Additionally, there are various other gauges and indicators that provide information that do not fall into these classifications, especially on large complex aircraft. Flight control position, cabin environmental systems, electrical power, and auxiliary power units (APUs), for example, are all monitored and controlled from the cockpit via the use of instrument systems. All may be regarded as position/condition instruments since they usually report the position of a certain movable component on the aircraft, or the condition of various aircraft components or systems not included in the first three groups.

FLIGHT INSTRUMENTS

Flight instruments are responsible for displaying aircraft's flight attitude. An Altimeter that displays aircraft altitude; the airspeed indicator; and the magnetic direction indicator (which is a form of compass), all fall under the category of flight instruments. Some aircraft additionally have an artificial horizon, turn coordinator, and vertical speed indicator as part of their flight instruments suite. Older aircraft had their flight instruments laid out in a classic 'T' arrangement (***Figure 8-1***). The top center position directly in front of the pilot and copilot is the basic display position for the artificial horizon even in modern glass cockpits (those with solid-state, flat-panel screen indicating systems). Original analog flight instruments are operated by air pressure and the use of gyroscopes. This avoids the use of electricity, which could put the pilot in a dangerous situation if the aircraft lost electrical power. Development of sensing and display techniques, combined with advanced aircraft electrical systems, has made it possible for reliable primary and secondary instrument systems that are electrically operated.



Figure 8-1. The basic "T" arrangement of analog flight instruments.



Figure 8-2. Analog instruments embedded in a glass cockpit for redundancy (Cessna Sky Hawk).



Figure 8-3. Engine instruments centered on the cockpit front instrument panel for easy viewing by both co-pilot and pilot (Boeing 737 aircraft).

Nonetheless, often a pneumatic altimeter, a gyro artificial horizon, and a magnetic direction indicator are retained somewhere in the instrument panel for redundancy (***Figure 8-2***).

ENGINE INSTRUMENTS

The center position of the cockpit front panel is generally occupied by engine instrumentation. The central location (***Figure 8-3***) facilitates both the pilot and copilot to monitor critical engine parameters. Multiengine aircraft generally use a single gauge for a particular engine parameter which displays information for all engines through the use of multiple pointers on the same dial face, whereas in some cases, a single gauge is used to display a certain engine parameter separately for every engine.

NAVIGATION INSTRUMENTS

The Navigation instruments are those that contribute information used by the pilot to guide the aircraft along a definite course. This group includes compasses of various kinds, some of which incorporate the use of radio signals to define a specific course while flying the aircraft en route from one airport to another. Other navigational instruments are designed specifically to direct the pilot's approach to landing at an airport (***Figure 8-4***). Traditional navigation instruments include a clock and a magnetic compass. Along with the airspeed indicator and wind information, these can be used to calculate navigational progress. Radios and instruments sending locating information via radio waves have replaced these manual efforts in modern aircraft. Global position systems (GPS) use satellites to pinpoint the location of the aircraft via geometric triangulation. This technology is built into some aircraft instrument packages for navigational purposes.

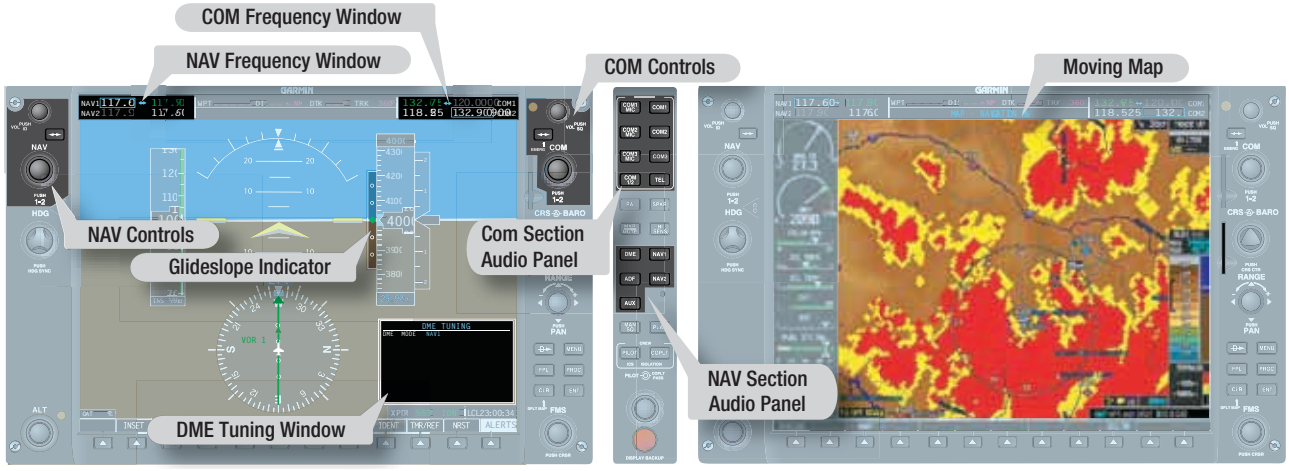


Figure 8-4. A modern navigational panel.

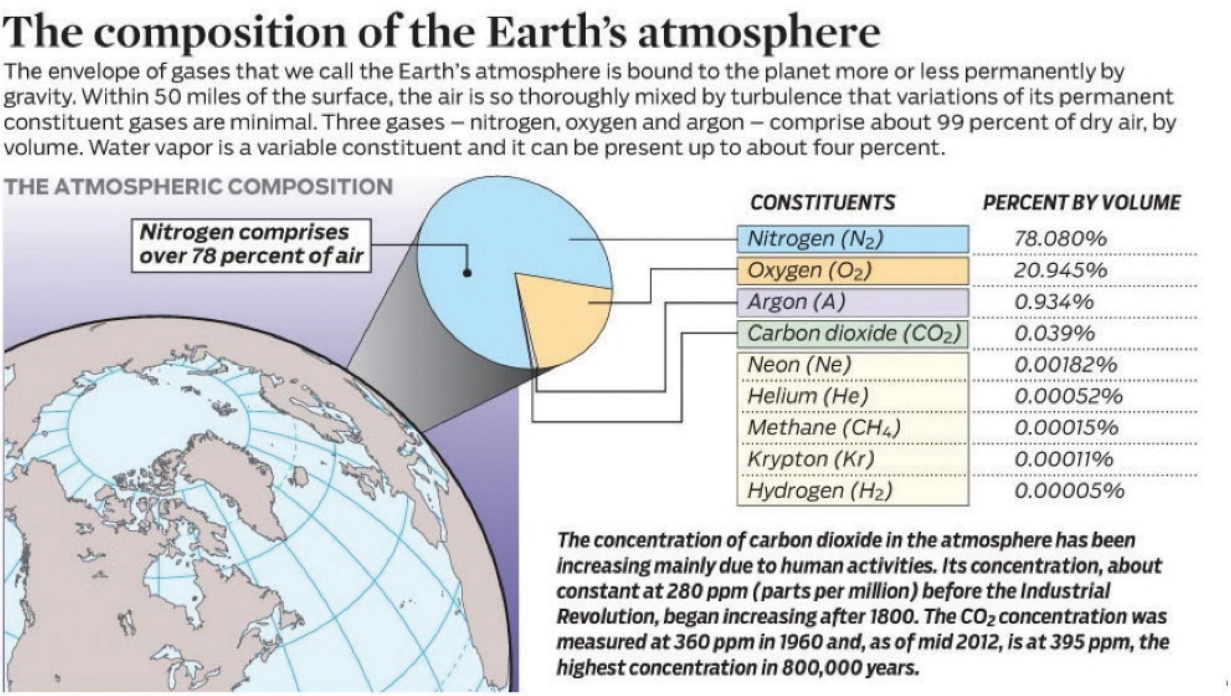


Figure 8-5. The composition of the Earth's atmosphere.

The Earth's atmosphere is a layer of gas that is held on by gravity, which prevents objects and particles flying off into space. The Earth atmosphere has a mass of about 5×10^{18} kg, three quarters of which is within about 11 km (6.8 miles; 36 000 ft) of the surface. From that point the atmosphere gets increasingly thin and eventually becomes space at an indistinct point around 100 km above sea level. (***Figure 8-5***)

CHARACTERISTICS OF THE ATMOSPHERE

Salient characteristics of the atmosphere are measured in pressure, density, humidity and temperature. These values are not fixed and vary immensely depending on time and geographical location.

PRESSURE

Atmospheric pressure is the weight of the column of gas directly above a certain point. Atmospheric pressure decreases, the higher the measurement is taken off the ground, due to the smaller column of gas remaining above. A column of air (one square inch) extending from sea level to the top of the atmosphere weighs approximately 14.7 pounds; therefore, atmospheric pressure is stated in pounds per square inch (psi). Thus, atmospheric pressure at sea level is 14.7 psi (***Figure 8-6***).

Atmospheric pressure is measured with an instrument shown in ***Figure 8-7***, called a barometer, composed of mercury in a tube that records atmospheric pressure in inches of mercury ("Hg). The standard measurement in aviation altimeters and U.S. weather reports has been "Hg. However, world-wide weather maps indicate pressure in millibars (mb), an SI metric unit. Aviators often interchange references to atmospheric pressure between linear displacement (e.g., inches of mercury) and units of force (e.g., psi).

Over the years, meteorology has shifted its use of linear displacement representation of atmospheric pressure to units of force. The unit of force nearly universally used today to represent atmospheric pressure in meteorology is the hectoPascal (hPa). A Pascal is a SI metric unit that expresses force in Newtons per square meter. A hectoPascal is 100 Pascals. 1 013.2 hPa is equal to 14.7 psi which is equal to 29.92 "Hg (***Figure 8-8***).

Pressure variation with increasing altitude is shown in ***Figure 8-9***. As depicted in the graph, the decrease in pressure is a rapid one and, at 50 000 feet, the atmospheric pressure has dropped to almost one-tenth of the sea level value. As an aircraft ascends, atmospheric pressure drops, the quantity of oxygen decreases, and

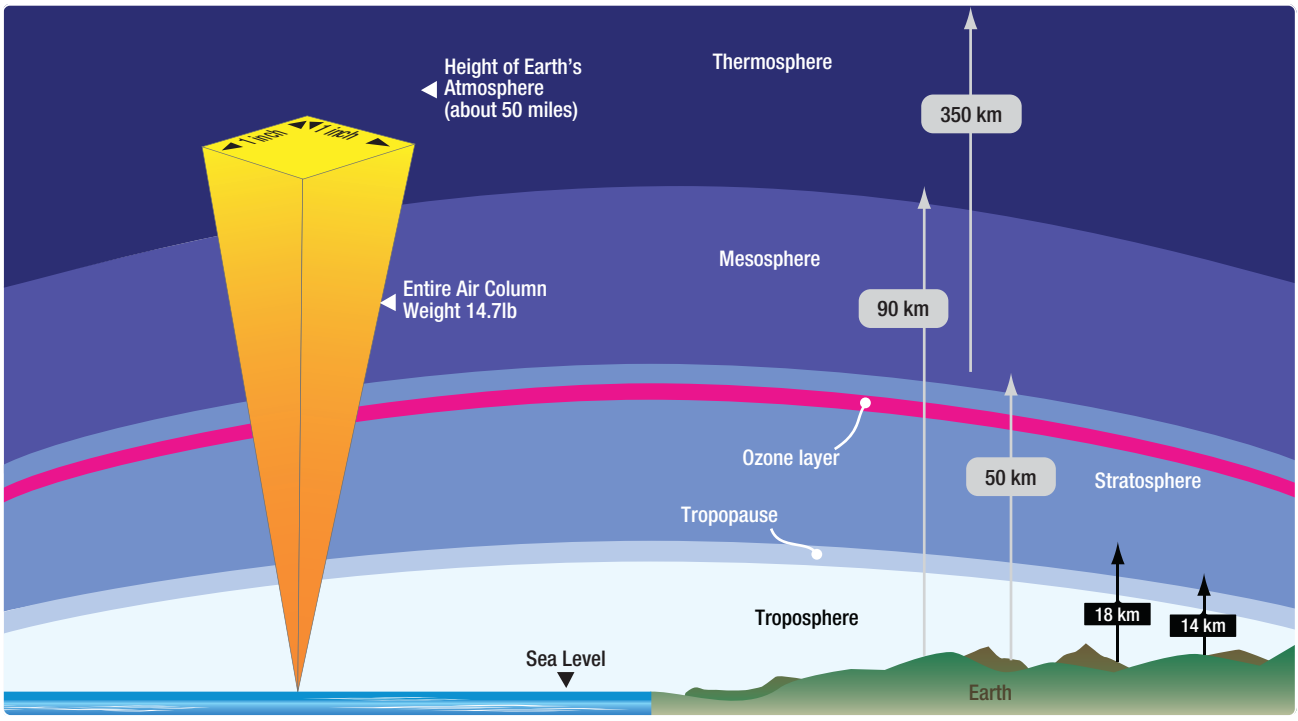


Figure 8-6. The weight exerted by a 1 square inch column of air stretching from sea level to the top of the atmosphere is what is measured when it is said that atmospheric pressure is equal to 14.7 psi.

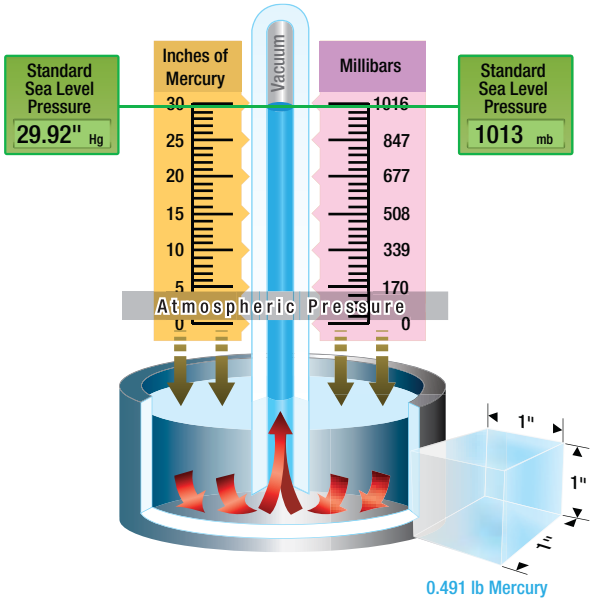


Figure 8-7. Temperature variation with increasing altitude in various atmospheric layers.

temperature drops. These changes in altitude affect an aircraft's performance in such areas as lift and engine horsepower. The effects of temperature, altitude, and density of air on aircraft performance are covered in the following paragraphs.

DENSITY

Density is weight per unit of volume. Since air is a mixture of gases, it can be compressed. If the air in one container is under half as much pressure as an equal amount of air in an identical container, the air under the greater pressure weighs twice as much as that in the container under lower pressure. The air under greater pressure is twice as dense as that in the other container. For the equal weight of air, that which is under the greater pressure occupies only half the volume of that under half the pressure.

The density of gases is governed by the following rules:

1. Density varies in direct proportion with the pressure.
2. Density varies inversely with the temperature. Thus, air at high altitudes is less dense than air at low altitudes, and a mass of hot air is less dense than a mass of cool air. Changes in density affect the aerodynamic performance of aircraft with the same horsepower. An aircraft can fly faster at a high altitude where the air density is low than at a low altitude where the density is greater. This is because air offers less resistance to the aircraft when it contains a smaller number of air particles per unit of volume.

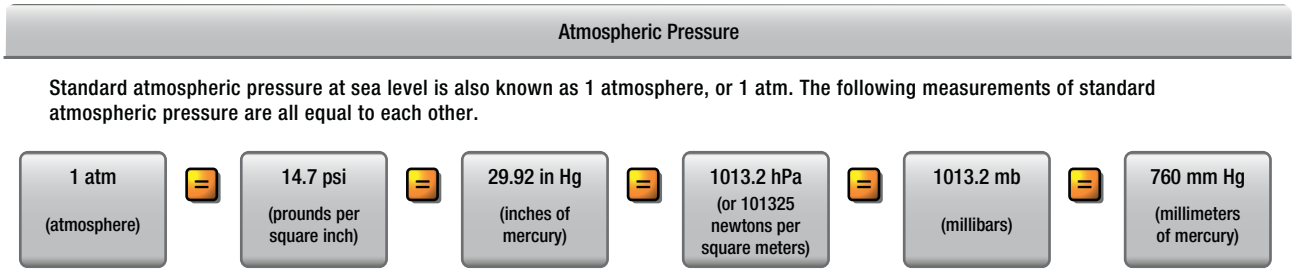


Figure 8-8. Various equivalent representations of atmospheric pressure at sea level.

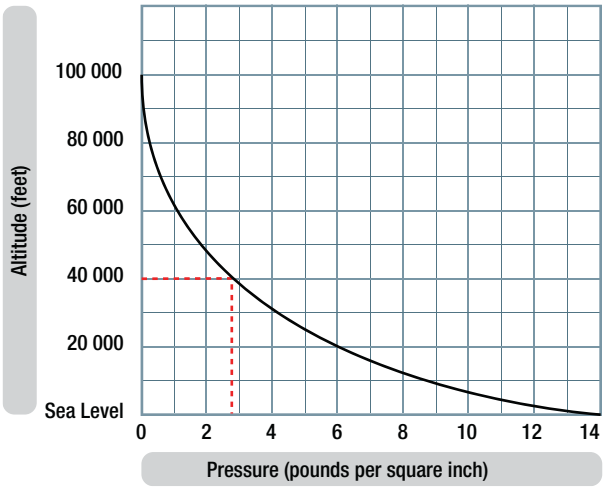


Figure 8-9. Atmospheric pressure decreasing with altitude. At sea level the pressure is 14.7 psi, while at 40 000 feet, as the dotted lines show, the pressure is only 2.72 psi.

HUMIDITY

Humidity is the amount of water vapor in the air. The maximum amount of water vapor that air can hold varies with the temperature. The higher the temperature of the air, the more water vapor it can absorb.

1. Absolute humidity is the weight of water vapor in a unit volume of air.
2. Relative humidity is the ratio, in percent, of the moisture actually in the air to the moisture it would hold if it were saturated at the same temperature and pressure.

Assuming that the temperature and pressure remain the same, the density of the air varies inversely with the humidity. On damp days, the air density is less than on dry days. For this reason, an aircraft requires a longer runway for takeoff on damp days than it does on dry days. By itself, water vapor weighs approximately five eighths as much as an equal amount of perfectly dry air. Therefore, when air contains water vapor, it is not as heavy as dry air containing no moisture.

TEMPERATURE AND ALTITUDE

Temperature variations in the atmosphere are of concern to aviators. Weather systems produce changes in temperature near the earth's surface. Temperature also changes as altitude is increased. The variation in temperature with increasing altitude in while crossing various layers of the atmosphere is illustrated in ***Figure 8-10***.

Most civilian aviation takes place in the troposphere in which temperature decreases as altitude increases. The rate of change is somewhat constant at about

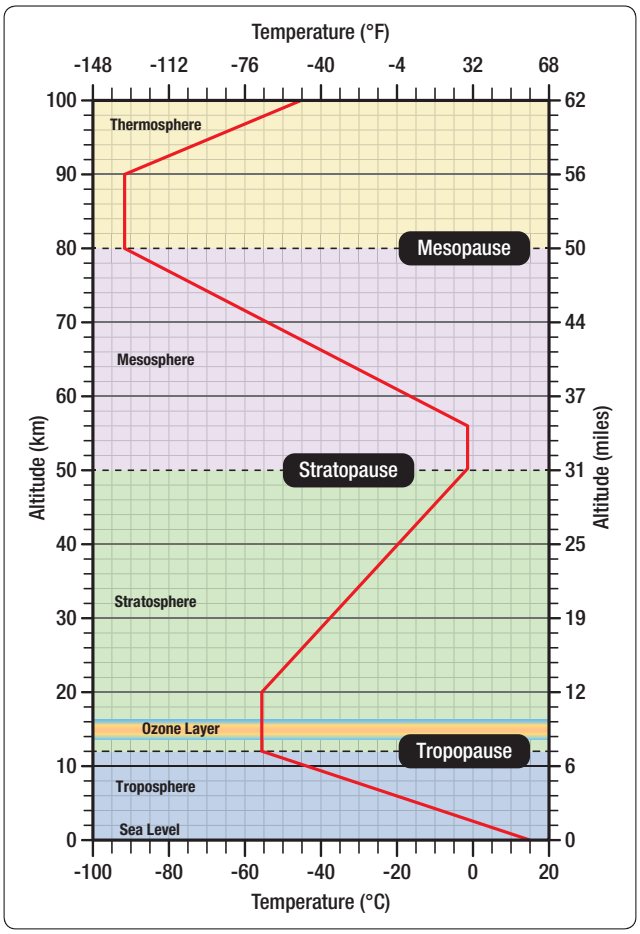


Figure 8-10. Temperature variation with increasing altitude in various atmospheric layers.

–2°C or –3.5°F for every 1 000 feet of increase in altitude. The upper boundary of the troposphere is the Tropopause. It is characterized as a zone of relatively constant temperature of –57°C or –69°F. Above the Tropopause lies the Stratosphere. Temperature increases with altitude in the Stratosphere to near 0°C before decreasing again in the Mesosphere, which lies above it. The stratosphere contains the ozone layer that protects the earth's inhabitants from harmful UV rays. Some civilian lights and numerous military flights occur in the stratosphere.

INTERNATIONAL STANDARD ATMOSPHERE

The atmosphere is never at rest. Pressure, temperature, humidity, and density of the air are continuously changing. To provide a basis for theoretical calculations, performance comparisons and instrumentation parity, standard values for these and other characteristic of the atmosphere have been developed. ICAO, ISO, and various governments establish and publish the values known as the International Standard Atmosphere (***Figure 8-11***).

In a theoretical example, where the real atmosphere was the same as ISA, the sea level temperature would be 15°C. Ascending 154m would drop the temperature by 1°C. The temperature drops 6.5° C per 1 000 m, up about to 11 000 m. Similarly, pressure would be 1 013 hPa at sea level. Every 8m of ascension would decrease the pressure by 1 hPa. This ratio of pressure difference to vertical distance is called the vertical 'baric gradient'. The ratio is constant in low atmospheric layers, but the baric gradient decreases higher up. While in the real world atmosphere surfaces are wavy, they are represented on maps as lines parallel to sea level. These lines are called isobars and are usually depicted 4 millibars apart.

ALTITUDE	TEMPERATURE		PRESSURE		DENSITY	
Feet	°F	°C	psi	hPa	slug/ft³	kg/m³
Sea Level	59	15	14.67	1013.53	0.002378	1.23
1 000	55.4	13	14.17	977.16	0.002309	1.19
2 000	51.9	11	13.66	941.82	0.002242	1.15
3 000	48.3	9.1	13.17	908.11	0.002176	1.12
4 000	44.7	7.1	12.69	874.94	0.002112	1.09
5 000	41.2	5.1	12.05	843.07	0.002049	1.06
6 000	37.6	3.1	11.78	812.2	0.001988	1.02
7 000	34	1.1	11.34	781.85	0.001928	0.99
8 000	30.5	-0.9	10.92	752.91	0.001869	0.96
9 000	26.9	-2.8	10.5	724.28	0.001812	0.93
10 000	23.3	-4.8	10.11	697.06	0.001756	0.9
15 000	5.5	-14.7	8.3	571.82	0.001496	0.77
20 000	-12.3	-24.6	6.75	465.4	0.001267	0.65
25 000	-30.2	-34.5	5.46	376.01	0.001066	0.55
30 000	-48	-44.4	4.37	301.3	0.000891	0.46
35 000	-65.8	-54.3	3.47	238.42	0.000738	0.38
40 000	-69.7	-56.5	2.72	187.54	0.000587	0.3
45 000	-69.7	-56.5	2.15	147.48	0.000462	0.24
50 000	-69.7	-56.5	1.68	115.83	0.000362	0.19

Figure 8-11. The International Standard Atmosphere.

PRESSURE MEASURING DEVICES AND SYSTEMS

A number of instruments inform the pilot of the aircraft's condition and flight situations through the measurement of pressure. Pressure measurements on an aircraft are related to control and operation of aircraft engines, environmental systems, hydraulic, pneumatic, air conditioning and oxygen, while other pressure instruments specialize in air data measurements or aerodynamic flight planning parameters (speeds, altitudes), a category known as Anemo-Barometric systems (***Figure 8-12***).

A pressure sensor is equipped with an element which is sensitive to changes in pressure. The sensing element is attached to a conversion device which either is linked directly to the indicator component itself (traditional direct-read manometers) or linked to a transmitter/transducer which allows information to be transmitted via an electrical signal. Depending on the amount of pressure to be measured and accuracy required, the sensor component can be one or more metal capsules, a flexible membrane, metal bellows or a Bourdon tube.

PRESSURE REFERENCES

ABSOLUTE PRESSURE

Absolute pressure sensors measure the pressure of a system relative to a perfect vacuum. These sensors incorporate sensing elements which are completely evacuated and sealed; the high pressure port is not present and input pressure is applied through the low port. The measurement is done in pounds per square inch absolute (psia).

DIFFERENTIAL PRESSURE

Differential pressure is measured by reading the difference between the inputs of two or more pressure levels. The sensor must have two separate pressure ports; the higher of the two pressures is applied through the high port and the lower through the low port. It is commonly measured in units of pounds per square inch (psi). An example of a differential pressure sensor is filter monitors; when the filter starts to clog the flow resistance and therefore the pressure drop across the filter will increase. Bidirectional sensors are able to measure positive and negative pressure differences. Unidirectional sensors only operate in the positive range and the highest pressure has to be applied to the pressure port defined as "high pressure".

GAUGE PRESSURE

Gauge sensors are the most common type of pressure sensors. The pressure is measured relative to ambient pressure which is the atmospheric pressure at a given location. The average atmospheric pressure at sea level is 1 013.25 mbar but changes in weather and altitude directly influence the output of the pressure sensor. In this device, the input pressure is through the high port and the ambient pressure is applied through the open low port. Vacuum sensors are gauge sensors used to measure the pressure lower than the localized atmospheric pressure. A vacuum is a volume of space that is essentially empty of matter. Vacuum sensors are divided into different ranges of low, high and ultra-high vacuum. Sealed gauged sensors measure pressure relative to one atmosphere at sea level (14.7 psi) regardless of local atmospheric pressure. ***Figure 8-13*** depicts a comparison between absolute, differential and gauge pressure.

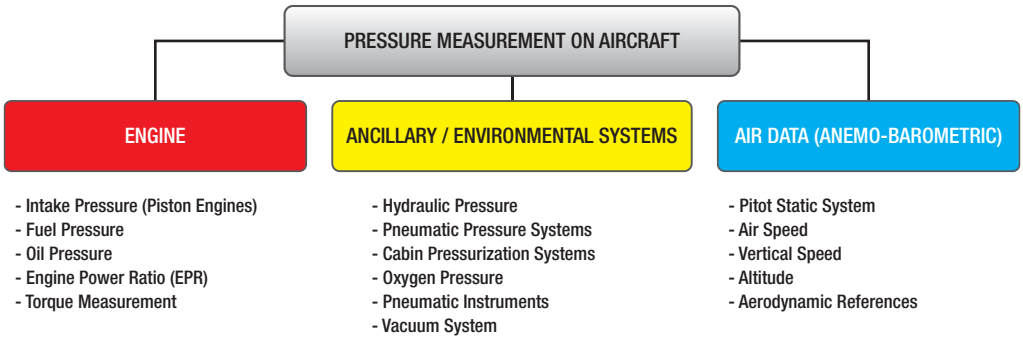


Figure 8-12. Types of pressure measurement on aircraft.

PRESSURE MEASUREMENTS (MECHANICAL)

Bourdon tubes, diaphragms, aneroids and bellows are mechanical pressure sensing devices often located inside a single instrument housing which contains the pointer and an instrument dial read by the pilot on the instrument panel. Many instruments make use of these sensitive and reliable mechanisms as direct reading gauges. But, many remote-sensing instrument systems also make use of the diaphragm and bellows. In this case, the sensing device containing the pressure sensitive diaphragm or bellows is located remotely on the engine or airframe. It is part of a transducer that converts the pressure into an electrical signal. The transducer, or transmitter, sends the signal to the gauge in the cockpit, or to a computer, for processing and subsequent display of the sensed condition. Examples of instruments that use a diaphragm or bellows in a direct reading or remote-sensing gauge are the altimeter, vertical speed indicator, cabin differential pressure gauge (in pressurized aircraft), and manifold pressure gauge.

BOURDON TUBE

A Bourdon tube is used for liquid and pneumatic pressure measurements. A Bourdon tube is illustrated in ***Figure 8-14***. It is based on the principal that a flattened tube regains more of its original circular shape the more pressure is put into it. A Phosphor Bronze, Beryllium Bronze, or a Beryllium Copper tube is typically used to mechanically move the hand of the instrument dial (***Figure 8-15***). Bourdon Tubes are known for its very high range of differential pressure measurement in the range of almost 100 000 psi (700 MPa). It is an elastic type pressure transducer.

The device was invented by Eugene Bourdon in the year 1849. The basic idea behind the device is that, cross-sectional tubing when deformed in any way will tend to regain its circular form under the action of pressure. The Bourdon pressure gauges used today have a slight elliptical cross-section and the tube is generally bent into a C-shape.

As the fluid pressure enters the Bourdon tube, it tries to be reformed and because of a free tip available, this action causes the tip to travel in free space and the tube unwinds. The simultaneous actions of bending and tension due to the internal pressure make a non-linear movement of the free tip. This travel is suitably guided

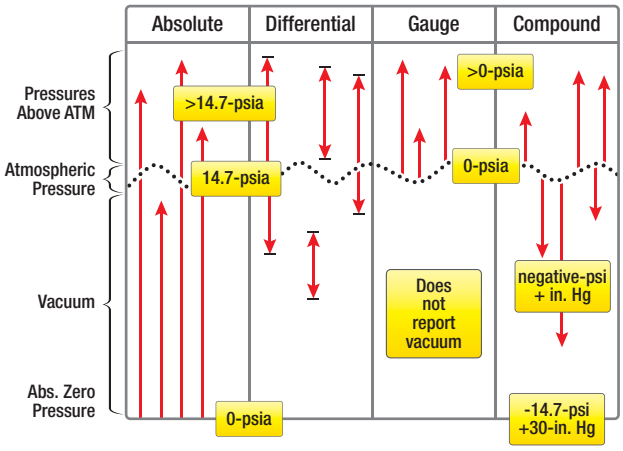


Figure 8-13. A comparison between Absolute, Differential and Gauge pressures.

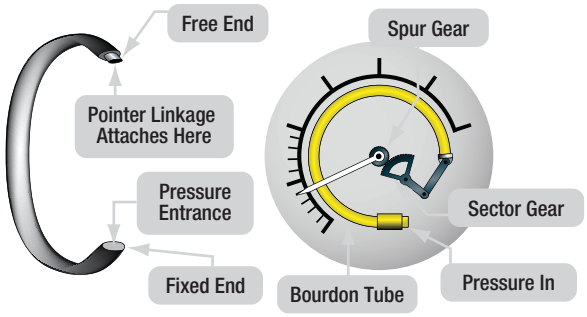


Figure 8-14. The Bourdon tube is one of the basic mechanisms for sensing pressure.

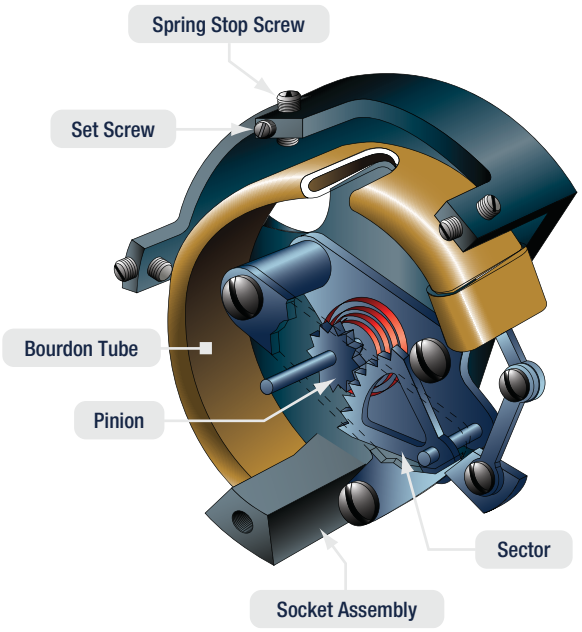


Figure 8-15. The Bourdon tube mechanism can be used to measure pressure or temperature by recalibrating the pointer's connecting linkage and scaling instrument face to read in degrees Celsius or Fahrenheit.

and amplified for the measurement of the internal pressure. But the main requirement of the device is that whenever the same pressure is applied, the movement of the tip should be the same and on withdrawal of the pressure the tip should return to the initial point.

Other than C-type, Bourdon gauges can also be constructed in the form of a helix or a spiral. The types are varied for specific uses and space accommodations, for better linearity and larger sensitivity. For thorough repeatability, the Bourdon tubes materials must have good elastic or spring characteristics. The surrounding in which the process is carried out is also important as corrosive atmosphere or fluid merit use of a material which is corrosion proof. Additionally, there is usually a restrictor of some kind to stop too much pressure from damaging the tube. Bourdon tubes can be calibrated with a mercury barometer. Bourdon tubes are used for static and quasi-static measurements and generally deployed in high pressure measurement applications. They may not be accurate enough for some aeronautical uses. In newer generations of aircraft, they have largely been replaced by modern electrical instruments that are much more responsive and accurate.

DIAPHRAGM

A diaphragm pressure transducer is used for low pressure measurement. They are commercially available in two types: metallic and non-metallic. Metallic diaphragms are known to have good spring characteristics and non-metallic types have no elastic characteristics. In a typical diaphragm assembly, when a force acts against a thin stretched diaphragm, it causes a deflection of the diaphragm with its center deflecting the most. (***Figure 8-16***).

An evacuated sealed diaphragm is called an aneroid. Since the elastic limit of a diaphragm has to be maintained, the deflection of the diaphragm must be kept in a restricted manner. This can be done by cascading many diaphragm capsules. A main capsule is designed by joining two diaphragms at the periphery. A pressure inlet line is provided at the central position. When the pressure enters the capsule, the deflection will be the sum of deflections of all the individual capsules. Mostly corrugated diaphragms are used instead of the conventional ones. Corrugated designs help in providing a linear deflection and also increase the member strength. The total amount of deflection for a given

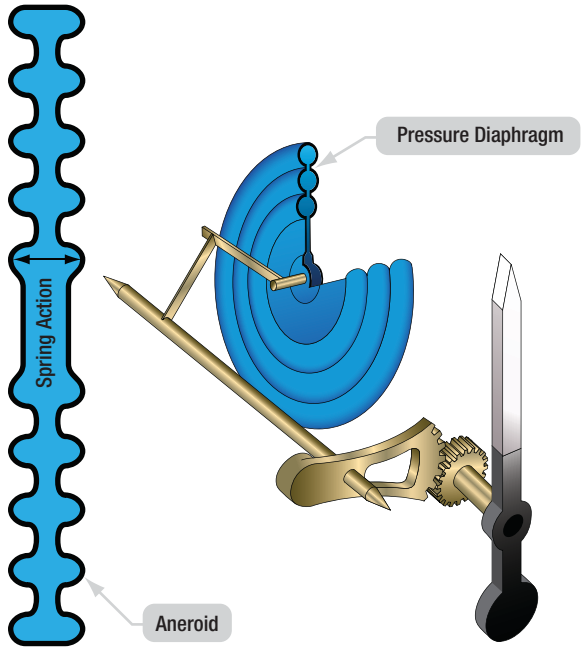


Figure 8-16. A diaphragm used for measuring pressure. An evacuated sealed diaphragm is called an aneroid.

pressure differential is known by the number and depth of corrugation, number of capsules, capsule diameter and shell thickness. Materials used in construction of metal diaphragms are the same as those used for Bourdon tube.

CAPSULES

Capsules are made of two diaphragms joined together at the edges to form a chamber which may be completely sealed or open to a source of pressure. They are generally employed, like diaphragms, as a low pressure measuring device, a capsule however being more sensitive. An aneroid barometer is a typical capsule operated device. It is simply a barometer without liquid that is used to measure the atmospheric pressure.

BELLOWS

When a number of diaphragm chambers (capsules) are connected together, the device is called Bellows. This accordion-like assembly of diaphragms (capsules) can be very useful when measuring the difference in pressure between two gases, called differential pressure. Just as with a single diaphragm, it is the movement of the side walls of the bellows assembly that correlates with changes in pressure and to which a pointer linkage and gearing is attached to inform the pilot (***Figure 8-17***).

PRESSURE SENSATION & TRANSMISSION (ELECTRICAL)

Pressure transducers are pressure sensing devices (***Figure 8-18***). It converts an applied pressure into an electrical signal. The output signal is generated by the primary sensing element and the device maintains the natural characteristics of the sensing technology. A transducer is also a sensor but a transducer always converts the non-electric pressure signal into an electrical signal. Therefore, a transducer is always a sensor but a sensor is not always a transducer. In industry the terms are often interchanged. There are several types of transducers including, strain gauge, thick film and thin film. Strain gauges are bonded to a larger structure that deforms as pressure changes. Piezoresistive devices sense shifts of electrical charges within a resistor.

Piezoelectric pressure instruments measure dynamic and quasi-static pressures. Their common modes of operation are charge mode, which generates a high impedance charge output; and voltage mode, which uses an amplifier to convert the high-impedance charge into a low-impedance output voltage. Thin film devices consist of an extremely thin layer of material, usually titanium nitride or polysilicon, deposited on a substrate. Pressure instruments that use micro-electro-mechanical systems (MEMS), variable capacitance, and vibrating elements are commonly available.

Pressure instruments can produce several types of electrical signals, including analog voltage and analog current. These output signals can be encoded via amplitude modulation (AM), frequency modulation (FM), or some other modulation scheme such as sine wave or pulse train. Common communication protocols include Ethernet, DeviceNet, Foundation Fieldbus, and highway addressable remote transmission. Several serial and parallel interfaces for pressure instruments are available. RS232 and RS485 are serial, digital protocols. Parallel protocols include the general-purpose interface bus, a standard which is also known as IEEE 488. Some pressure instruments output TTL signals. Others provide outputs that change the state of a switch or alarm.

PRESSURE SWITCHES

Pressure switches will usually use a diaphragm that works against a calibrated spring or other form of tension. When pressure is applied to the diaphragm

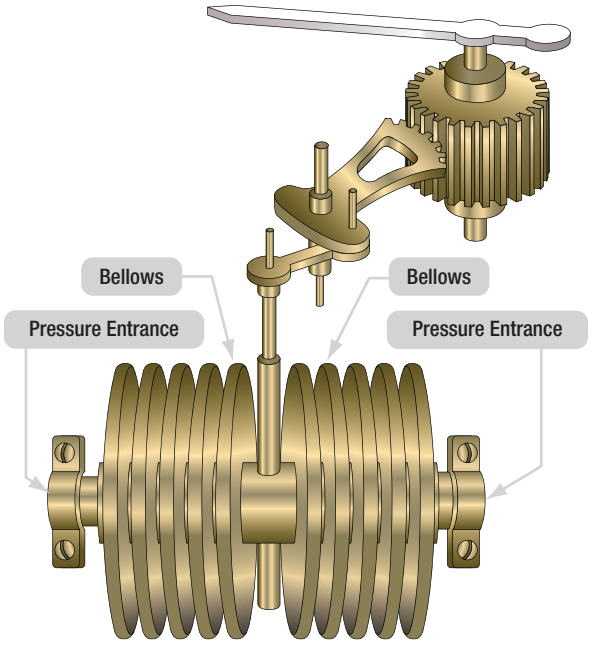


Figure 8-17. A bellows unit in a differential pressure gauge compares two different pressure values. End movement of the bellows away from the side with the highest pressure input occurs when the pressures in the bellows are not equal. The indicator linkage is calibrated to display the difference.



Figure 8-18. A typical pressure transducer converts the non-electric pressure signal into an electrical signal.

that is of a sufficient value to overcome the spring tension a switch is closed. Current that is supplied to the switch now has a completed path to ground. In a very simple warning light circuit, the closed pressure switch completed the circuit for the bulb and alerts the driver to an unacceptable condition. For example a simple oil pressure warning lamp circuit will use a pressure switch (***Figure 8-19***).

Computer monitored pressure switch circuits use the change in voltage as an indication of pressure. When the pressure change (either from low to high or high to

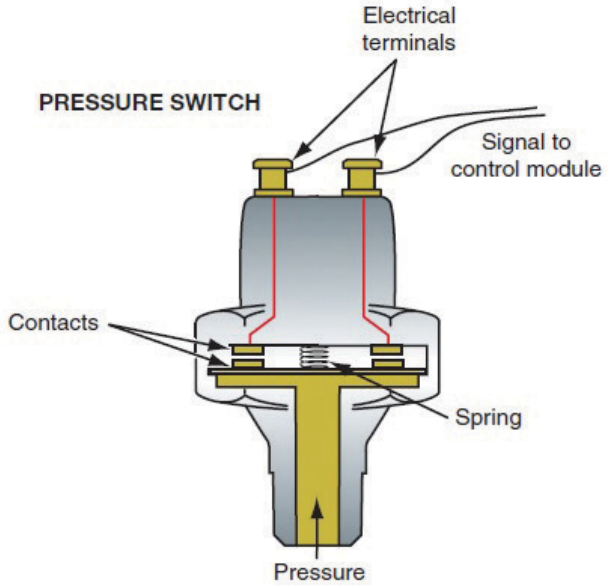


Figure 8-19. Simple pressure switch uses contracts to complete electrical circuit.

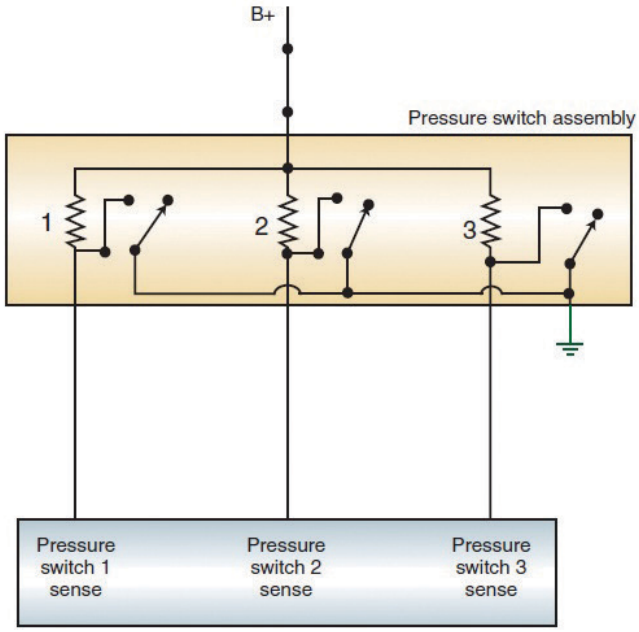


Figure 8-20. Computer monitored pressure switch circuit to indicate pressure above or below a set-point.

low) changes the state of the switch, the voltage change is interpreted by the computer. Pressure switches are used to monitor the presence of pressure that is above or below a set-point; they do not indicate the exact amount of pressure being applied. (***Figure 8-20***)

POTENTIOMETRIC PRESSURE SENSOR

One of the basic types of pressure sensor is the potentiometric pressure sensor. The potentiometric pressure sensors use a Bourdon tube, a capsule, or bellows to move a wiper arm on a resistive element. Using the principle of variable resistance, the movement of the wiper across the resistive element will record a different voltage reading to the computer. Although this type of sensor can be used as a computer input, a computer is not always involved. Some early analog instrument panels used this sensor unit with an air core gauge to display oil pressure.

PIEZORESISTIVE DEVICES

A strain gauge sensor determines the amount of applied pressure by measuring the strain a material experiences when subjected to the pressure. In their simplest form, a strain gauge sensor is a Piezoresistive device. A piezoresistive sensor behaves like a variable resistor because its resistance value changes as the pressure applied to the sensing material changes. This type of sensor changes resistance values as a function of pressure changes. A voltage regulator supplies a constant voltage to the sensor. Since the amount of voltage that the sensor drops will change with the change of resistance, the control module can determine the amount of pressure on the sensing material by measuring the voltage drop across the sensor. Piezoresistive sensors are commonly used as a gauge sending unit. (***Figure 8-21***)

WHEATSTONE BRIDGE

A Wheatstone bridge is commonly used to measure changes in pressure or strain. A Wheatstone bridge is nothing more than two simple series circuits connected in parallel across a power supply (***Figure 8-22***).

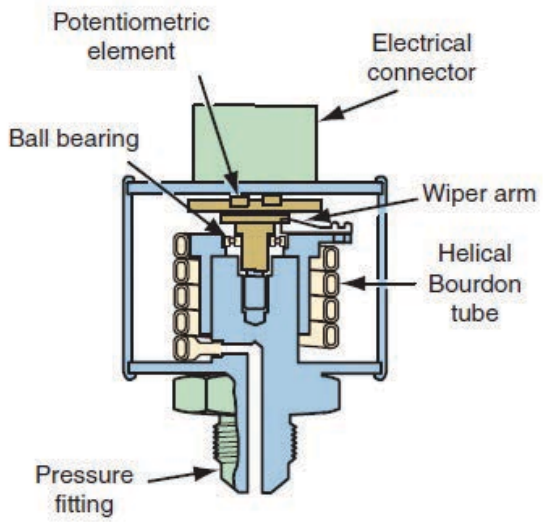


Figure 8-21. Potentiometric pressure sensors use a Bourdon tube, capsule, or bellows to drive a wiper arm on a resistive element.

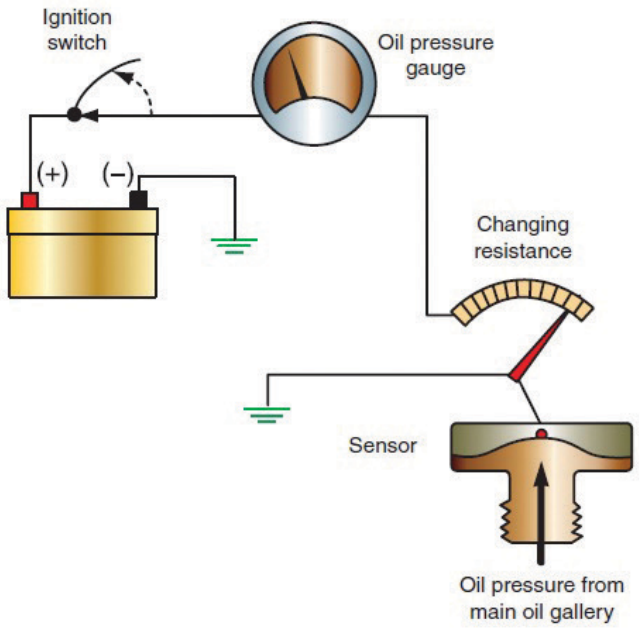


Figure 8-22. Oil pressure sensor used in gauge indicator circuit.

Usually three of the resistors are kept at exactly the same value and the fourth is the sensing resistor. When all four resistors have the same value, the bridge is balanced and the voltage sensor will indicate a value of 0 volts. The output from the amplifier acts as a voltmeter. Remember, since a voltmeter measures electrical pressure between two points, it will display this value. For example, if the reference voltage is 5 volts and the resistors have the same value, then the voltage drop over each resistor is 2.5 volts. Since the voltmeter is measuring the potential on the line between R_S and R_t and R_2 and R_3, it will read 0 volts because both of these lines have 2.5 volts on them (***Figure 8-23***). If there is a change in the resistance value of the sense resistor, a change will occur in the circuit's balance. The sensing circuit will receive a voltage reading that is proportional to the amount of resistance change.

PIEZOELECTRIC DEVICES

Piezoelectric devices are used to measure pressures by the generation of voltage. Piezoelectric sensors are constructed from alumina ceramics, metalized quartz, single crystals, or ultrasonic transducer materials that make up a bi-directional transducer capable of converting stress into an electric potential. The piezoelectric materials consist of polarized ions within the crystal. As pressure is applied on the piezoelectric material some mechanical deformation occurs in the polarized crystal, which produces a proportional output charge due to the displacement in the ions (***Figure 8-24***). Uses of this type of sensor in the aerospace industry include piezoelectric accelerometers, piezoelectric force sensors, and piezoelectric pressure sensors.

The sensor is a voltage generator and has a resistor connected in series with it. The resistor protects the sensor from excessive current flow in case the circuit becomes shorted. High voltage generator is a thin ceramic disc attached to a metal diaphragm. When pressure is applied on the diaphragm, it resultantly puts pressure on the piezoelectric crystals in the ceramic disc (***Figure 8-25***). The disc generates a voltage that is proportional to the amount of pressure. The voltage generated ranges from zero to one or more volts. Each time pressure is applied, the sensor generates a voltage spike.

CAPACITANCE DISCHARGE SENSOR

Another variation of the piezosensor uses capacitance discharge. Instead of using a silicon diaphragm, the capacitance discharge sensor uses a variable capacitor. In the capacitor capsule type sensor, two flexible alumina plates are separated by an insulating washer. A film electrode is deposited on the inside surface

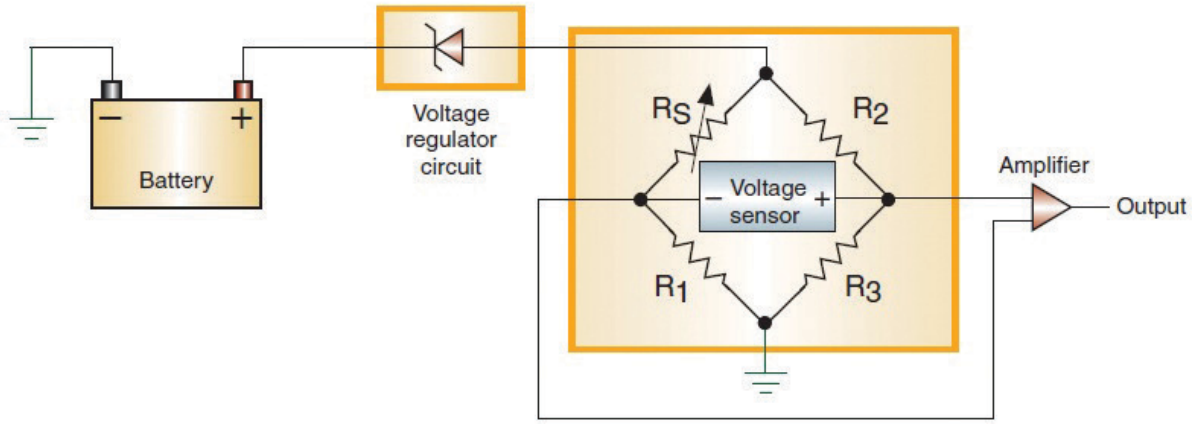


Figure 8-23. Wheatstone bridge.

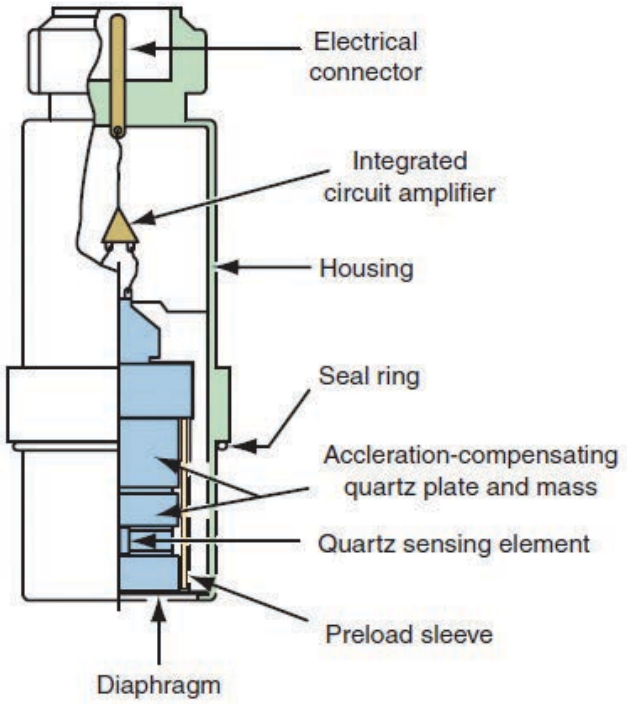


Figure 8-24. Piezoelectric sensors convert stress into an electric potential and vice versa. Sensors based on this technology are used to measure varying pressures.

of each plate and a connecting lead is extended for external connections. The result of this construction is a parallel plate capacitor with a vacuum between the plates. This capsule is placed inside a sealed housing that is connected to the sensed pressure. If constructed to measure vacuum, as the pressure increases (goes toward atmospheric), the alumina plates deflect inward, resulting in a decrease in the distance between the electrodes (***Figure 8-26***).

As the distance between the electrodes changes, so does the capacity of the capacitor. A measure of capacitance constitutes a measurement of pressure that is detected by a bridge circuit. The output from the bridge circuit can be either an analog DC voltage or applied to a chip that produces a frequency modulated digital signal.

PRESSURE MEASUREMENT ACCURACY

Accuracy is defined as the difference (error) between the true value and the indicated value expressed as percent of the span. Pressure transducers are mechanical structures made from more than one material. Because of this, they respond not only to changes in pressure, but to changes in temperatures as well. These changes can affect both the zero and full scale output (FSO) of the transducer, regardless of its type. The term temperature effect upon zero refers to the change in output at constant pressure as temperature is varied over a stated range. Extreme temperature fluctuations may change a transducer's output signal even though pressure remains constant. Accuracy of a pressure transducer is observed in three different areas; static, thermal, and total. Total accuracy is the combination of static and thermal accuracy.

STATIC

Static accuracy is the combined effects of linearity, hysteresis, and repeatability (***Figure 8-27***). It is expressed as +/- percentage of full scale output. The static error band is a good measure of the accuracy that can be expected at constant temperature.

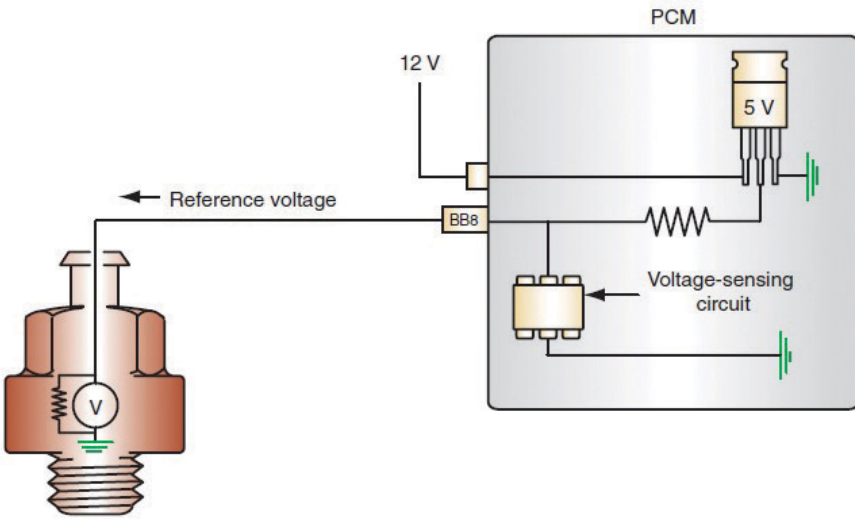


Figure 8-25. Piezoelectric sensor circuit.

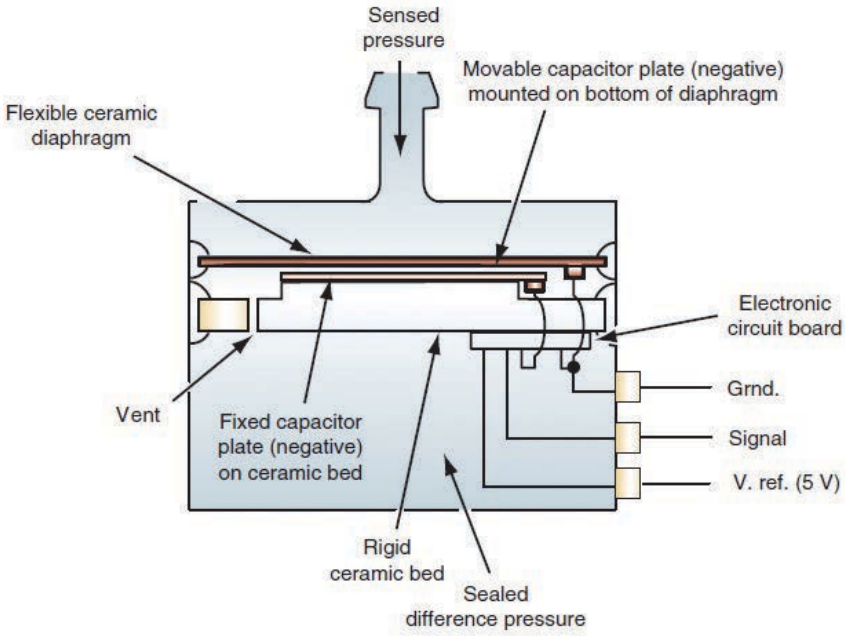


Figure 8-26. Capacitance discharge sensor.

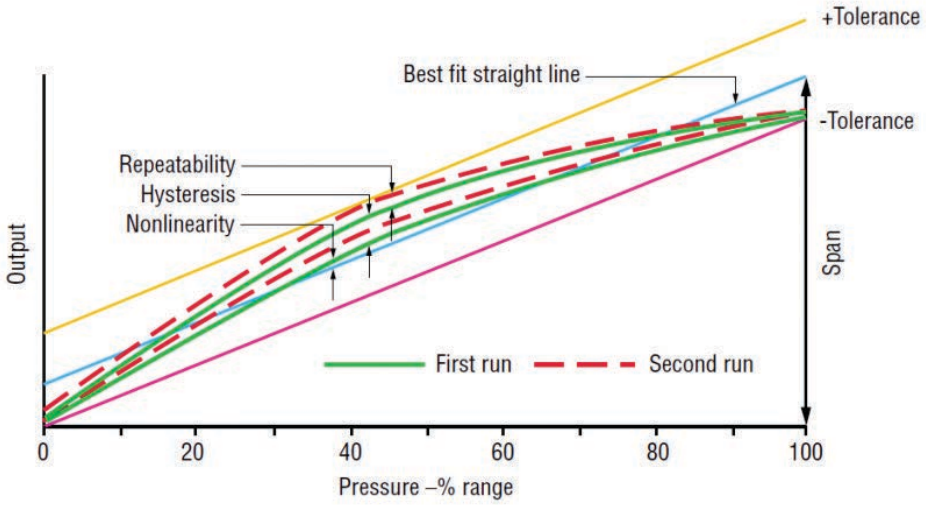


Figure 8-27. Components of Static Accuracy: Non-Linearity, Hysteresis and Repeatability.

LINEARITY/NON-LINEARITY

Linearity is the deviation of a calibration curve from a specified straight line. One way to measure linearity is to use the least squares method, which gives a best fit straight line. The best straight line is a line between two parallel lines that enclose all output vs. pressure values on the calibration curve.

HYSTERISIS

Hysteresis is the maximum difference in output at any pressure within the specified range, when the value is first approached with increasing and then with decreasing pressure. Temperatures hysteresis is the sensor's ability to give the same output at a given temperature before and after a temperature cycle. For example, in the case of a mechanical analog altimeter, its diaphragm pressure measuring apparatus has limitations. The diaphragm itself is only so elastic when responding to static air pressure changes. Hysteresis is the term for when the material from which the diaphragm is made takes a set during long periods of level flight. If followed by an abrupt altitude change, the indication lags or responds slowly while expanding or contracting during a rapid altitude change. While temporary, this limitation does cause an inaccurate altitude indication.

REPEATABILITY

Repeatability is the ability of a transducer to reproduce output readings when the same pressure is applied to the transducer repeatedly, under the same conditions and in the same direction.

THERMAL

Thermal accuracy observes how temperature affects the output. It is expressed as a percentage of full scale output or as a percentage of full scale per degree Celsius, degree Fahrenheit or Kelvin.

PITOT-STATIC SYSTEM

Some of the most important flight instruments derive their indications from measuring air pressure. Gathering and distributing various air pressures for flight instrumentation is the function of the pitot-static system. The altimeter, airspeed indicator, and vertical speed indicator are the three most common pitot-static instruments.

PITOT TUBE

On simple aircraft, this may consist of a pitot-static system head or pitot tube with impact and static air pressure ports and leak-free tubing connecting these air pressure pickup points to the instruments that require the air for their indications. A pitot tube is open and faces into the airstream to receive the full force of the impact air pressure as the aircraft moves forward. (***Figure 8-28***)

This air passes through a baffled plate designed to protect the system from moisture and dirt entering the tube. Below the baffle, a drain hole is provided, allowing moisture to escape. The ram air is directed aft to a chamber in the shark fin of the assembly. An upright tube, or riser, leads this pressurized air out of the pitot assemble to the airspeed indicator. The aft section of the pitot tube is equipped with small holes on the top and bottom surfaces that are designed to collect air pressure that is at atmospheric pressure in a static, or still, condition. The static section also contains a riser tube and the air is run out the pitot assembly through tubes and is connected to the altimeter, the airspeed indicator, and the vertical speed indicator. Many pitot-static tube heads contain heating elements to prevent icing during flight. The pilot can send electric current to the element with a switch in the cockpit when ice-forming conditions exist. Often, this switch is wired through the ignition switch so that when the aircraft is shut down, a pitot tube heater inadvertently left on does not continue to draw current and drain the battery. Caution should

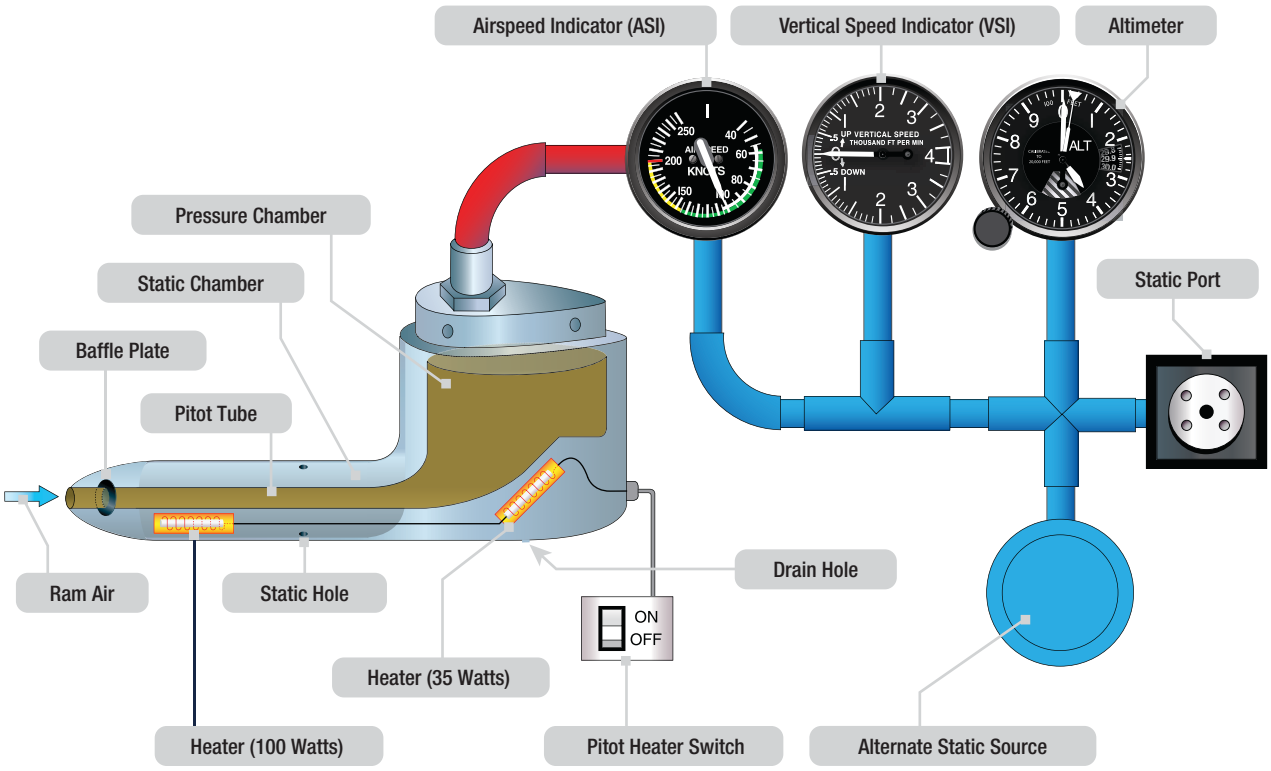


Figure 8-28. A typical pitot-static system head, or pitot tube, collects ram air and static pressure for use by the flight instruments.



Figure 8-29. Pitot-static tubes, can be of various designs and locations on airframes.

be exercised when near the pitot tube, as these heating elements make the tube too hot to be touched without receiving a burn.

The pitot-static tube is mounted on the outside of the aircraft at a point where the air is least likely to be turbulent. It is pointed in a forward direction parallel to the aircraft's line of flight. The location may vary. Some are on the nose of the fuselage and others may be located on a wing. A few may even be found on the empennage. Various designs exist but the function remains the same, to capture impact air pressure and static air pressure and direct them to the proper instruments (***Figure 8-29***).

Most aircraft equipped with a pitot-static tube have an alternate source of static air pressure provided for emergency use. The pilot may select the alternate with a switch in the cockpit should it appear the flight instruments are not providing accurate indications. On low-flying unpressurized aircraft, the alternate static source may simply be air from the cabin. On pressurized aircraft, cabin air pressure may be significantly different than the outside ambient air pressure.

Another type of pitot-static system provides for the location of the pitot and static sources at separate positions on the aircraft. The pitot tube in this arrangement is used only to gather ram air pressure.

STATIC PORTS

Static ports (vents) are used to collect static air pressure information. Usually, these are located flush on the side of the fuselage. There may be two or more vents. A primary and alternate source vent is typical, as well as separate dedicated vents for the pilot and first officer's instruments. Also, two primary vents may be located on opposite sides of the fuselage and connected with a "T" or "Y" tubing for input to the instruments (***Figure 8-30***). The "Y" arrangement helps to average out any measurement errors, a technique called 'Static balancing'. This is done to compensate for any variations in static air pressure on the vents due to the aircraft's attitude and correct any errors associated with asymmetrical flight (slip-skid).

Vent pipes are typically angled downwards to drain any water or contaminants which may block the airway. To avoid impairing thermal efficiency of static ports, they must not be painted, a warning to this affect is generally imprinted adjacent to static ports.

Regardless of the number and location of separate static vents, they may be heated as well as the separate ram air pitot tube to prevent icing. If failure of the primary pitot static pressure source should occur, for example icing up of a pitot or pressure head due to a failed heater circuit, errors may be introduced in the instrument readings and other areas dependent on such pressure. As a safeguard against partial failure, a standby system may be installed in some aeroplanes, whereby static pressure and/or pitot pressure from alternate sources can be selected and connected into the primary system. A blockage of the pitot source is not serious, as it will only affect the ASI. A blockage of the static source will however affect all of the instruments, and it is thus common practice to provide an alternate static supply.

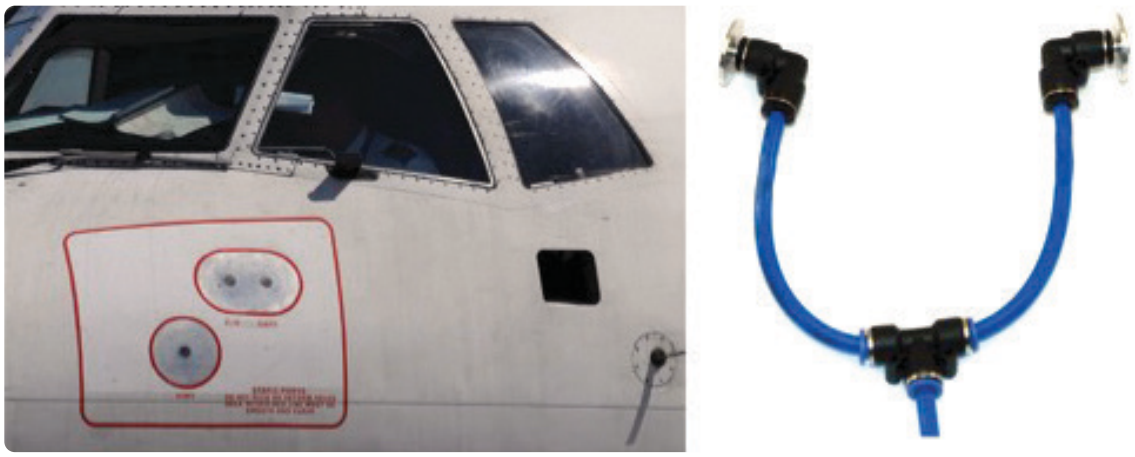


Figure 8-30. Static vents are located on both sides of the fuselage and connected with a "Y" fitting to provide an average pressure to the instruments.

MULTIFUNCTION PROBES

A multi-function air data sensing probe has a strut that is mounted on an aircraft and extends laterally from the aircraft skin. The strut is supported on a base plate, and has a pitot pressure sensing tube at the outer end thereof, with a pitot port facing upstream, and also includes a passageway for total air temperature sensor including a forwardly facing inlet scoop that leads to a chamber in the strut that is laterally offset from the inlet scoop so that flow changes direction as it enters the chamber. The surface defining the change of direction between the scoop and the chamber is provided with bleed holes for bleeding off boundary layer air. A vane type air data sensor is mounted on a shaft that rotates freely and is supported on the strut, and is positioned to sense the relative air flow past the strut to determine changes of relative angles of such air flow. In addition, the strut has static pressure sensing ports on lateral sides thereof leading to a separate chamber on the interior of the strut (***Figure 8-31***).



Figure 8-31. Multifunction Probes (MFP) installed on the Airbus A350 XWB aircraft.

The pitot-static systems of complex, multi-engine, and pressurized aircraft can be elaborate. Additional instruments, gauges, the autopilot system, and computers may need pitot and static air information. For example, the autopilot system also requires static pressure information, as does the cabin pressurization unit. Separate heated sources for static air pressure are taken from both sides of the airframe to feed independent static air pressure manifolds; one each for the pilot's flight instruments and the copilot's flight instruments. This is designed to ensure that there is always one set of flight instruments operable in case of a malfunction.

ALTIMETERS

An altimeter is an instrument that is used to indicate the height of the aircraft above a predetermined level, such as sea level or the terrain beneath the aircraft. The most common way to measure this distance is rooted in discoveries made by scientist's centuries ago. Seventeenth century work proving that the air in the atmosphere exerted pressure on the things around us led Evangelista Torricelli to the invention of the barometer. Also in that century, using the concept of this first atmospheric air pressure measuring instrument, Blaise Pascal was able to show that a relationship exists between altitude and air pressure. As altitude increases, air pressure decreases. The amount that it decreases is measurable and consistent for any given altitude change. Therefore, by measuring air pressure, altitude can be determined (***Figure 8-32***).

TRIPLE POINTER DISPLAY ALTIMETER

Altimeters that measure the aircraft's altitude by measuring the pressure of the atmospheric air are known as pressure altimeters. A pressure altimeter is

made to measure the ambient air pressure at any given location and altitude. In aircraft, it is connected to the static vent(s) via tubing in the pitot-static system.

Atmosphere Pressure	
Altitude (ft)	Pressure (psi)
Sea Level	14.69
2 000	13.66
4 000	12.69
6 000	11.77
8 000	10.91
10 000	10.10
12 000	9.34
14 000	8.63
16 000	7.96
18 000	7.34
20 000	6.75
22 000	6.20
24 000	5.69
26 000	5.22
28 000	4.77
30 000	4.36
32 000	3.98
34 000	3.62
36 000	3.29
38 000	2.99
40 000	2.72
42 000	2.47
44 000	2.24
46 000	2.04
48 000	1.85
50 000	1.68

Figure 8-32. Air pressure is inversely related to altitude. This consistent relationship is used to calibrate the pressure altimeter.

The relationship between the measured pressure and the altitude is indicated on the instrument face, which is calibrated in feet. These devises are direct-reading instruments that measure absolute pressure. An aneroid or aneroid bellows is at the core of the pressure altimeter's inner workings. Attached to this sealed diaphragm are the linkages and gears that connect it to the indicating pointer. Static air pressure enters the airtight instrument case and surrounds the aneroid. At sea level, the altimeter indicates zero when this pressure is exerted by the ambient air on the aneroid. As air pressure is reduced by moving the altimeter higher in the atmosphere, the aneroid expands and displays altitude on the instrument by rotating the pointer.

As the altimeter is lowered in the atmosphere, the air pressure around the aneroid increases and the pointer moves in the opposite direction (***Figure 8-33***). The face, or dial, of an analog altimeter is read similarly to a clock. As the longest pointer moves around the dial, it is registering the altitude in hundreds of feet. One complete revolution of this pointer indicates 1 000 feet of altitude. The second-longest point moves more slowly. Each time it reaches a numeral, it indicates 1 000 feet of altitude. Once around the dial for this pointer is equal to 10 000 feet. When the longest pointer travels completely around the dial one time, the second-longest point moves only the distance between two numerals indicating 1 000 feet of altitude has been attained. If so equipped, a third,

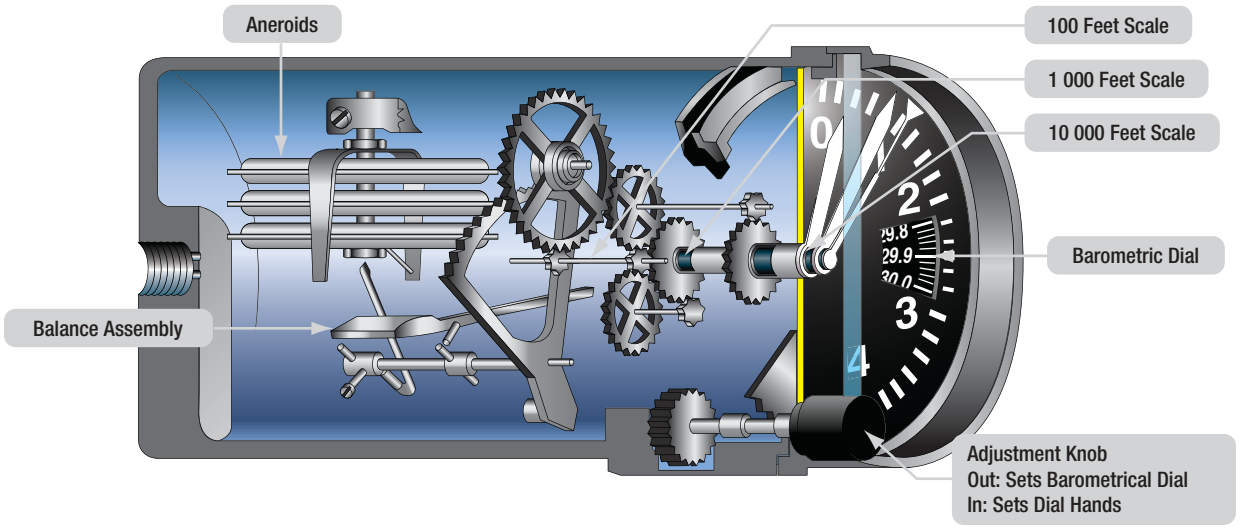


Figure 8-33. The internal arrangement of a sealed diaphragm pressure altimeter. At sea level and standard atmospheric conditions, the linkage attached to the expandable diaphragm produces an indication of zero. When altitude increases, static pressure on the outside of the diaphragm decreases and the aneroid expands, producing a positive indication of altitude. When altitude decreases, atmospheric pressure increases. The static air pressure on the outside of the diaphragm increases and the pointer moves in the opposite direction, indicating a decrease in altitude.

shortest or thinnest pointer registers altitude in 10 000 foot increments. When this pointer reaches a numeral, 10 000 feet of altitude has been attained. Sometimes a black-and white or red-and-white cross-hatched area is shown on the face on the instrument until the 10 000 foot level has been reached. (***Figure 8-34***)

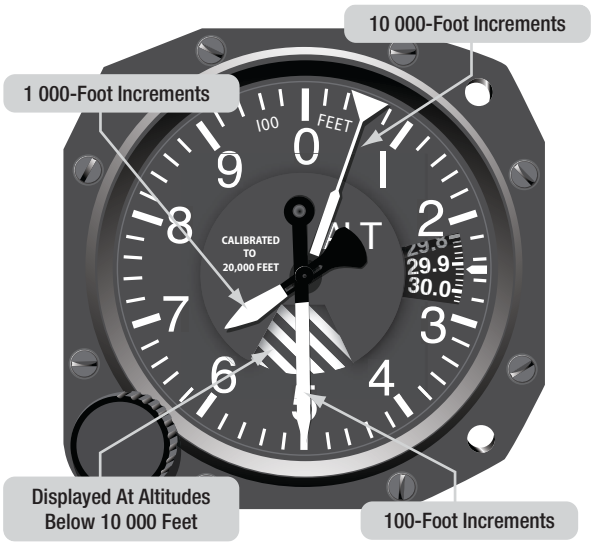


Figure 8-34. A sensitive altimeter with three pointers and a crosshatched area displayed during operation below 10 000 feet.

COUNTER (DRUM) TYPE ALTIMETER

Many altimeters also contain linkages that rotate a numerical counter in addition to moving pointers around the dial. This quick reference window allows the pilot to simply read the numerical altitude in feet. The motion of the rotating digits or drum-type counter during rapid climb or descent makes it difficult or impossible to read the numbers. Reference can then be directed to the classic clock-style indication. ***Figure 8-35*** illustrates the inner workings behind a typical counter (drum) type altimeter.

SERVO ASSISTED ALTIMETER

Servo assisted altimeters further increase the accuracy of the system by no longer relying on a direct mechanical link between the capsules and the height pointers. It uses an electromagnetic pick-off, the E and I-bar, which acts as a transducer, convert capsule movement into an electrical current. The I-bar is connected to the capsule stack, whilst the output of the secondary coils of the E-bar is amplified and used to drive the motor, which in turn drives the height pointers/height counters and, via a worm gear shaft/cam/cam follower, the E-bar itself. (***Figure 8-36***)

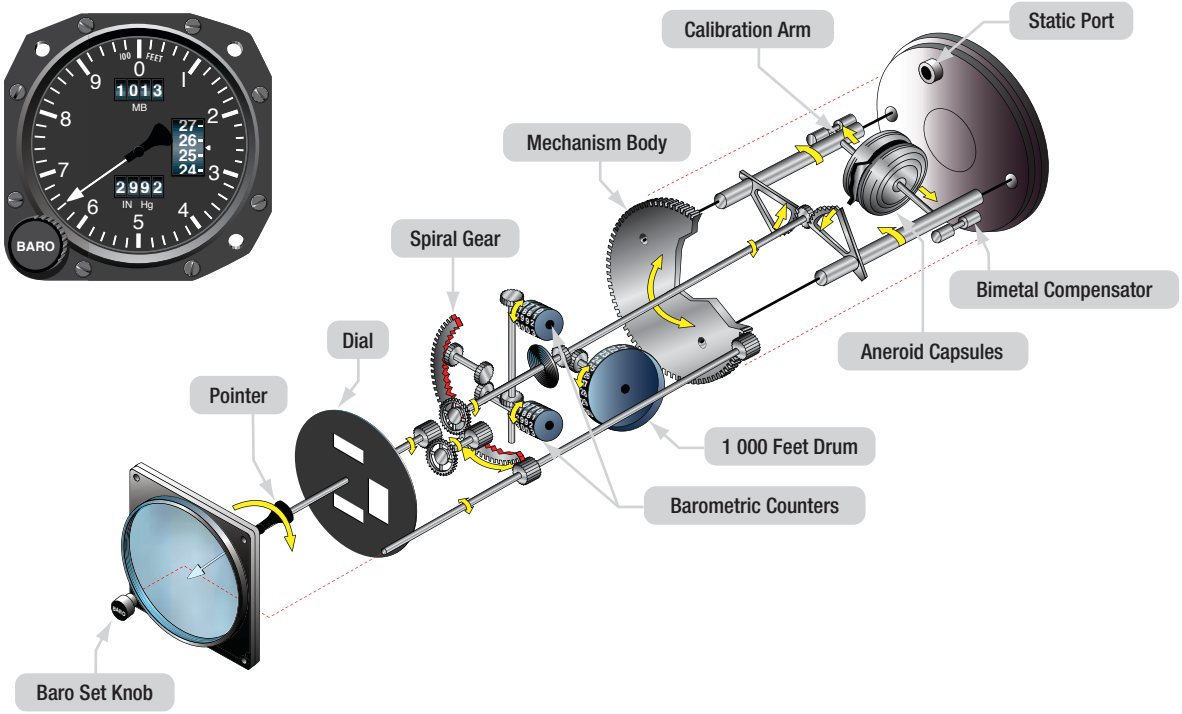


Figure 8-35. A drum-type counter can be driven by the altimeter's aneroid for numerical display of altitude. Drums can also be used for the altimeter's setting indications.

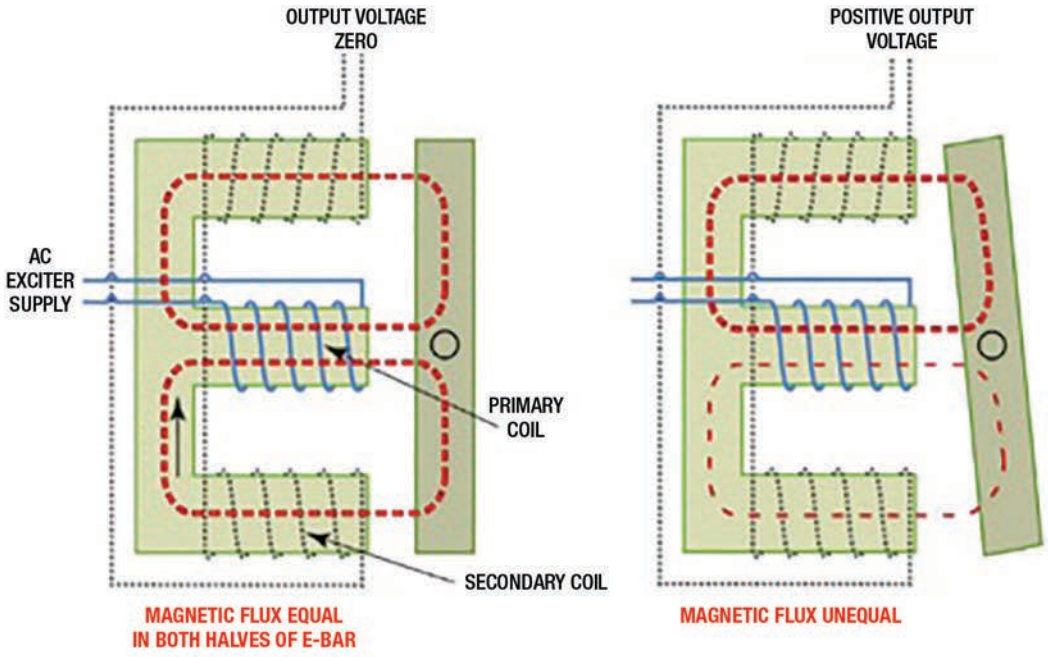


Figure 8-36. E and I bar transducer used in a servo-assisted altimeter.

An alternating current is fed to the primary winding on the center arm of the E-bar. If the air gaps between the E and I-bars are equidistant, alternating currents of equal magnitude will be induced into the two secondary coils on the upper and lower arms of the E-bar, and there will therefore be no output. As the aircraft climbs, the stack of capsules will expand, closing the gap at the lower arm of the E-bar and opening the gap at the upper arm of the E-bar (***Figure 8-37***).

Currents of differing magnitudes will now be induced into the secondary circuits and this imbalance in secondary currents is used, after amplification, to energize the two-phase servo motor. The motor drives a system of gears which drive the altimeter needles and digital readout. The same gearing drives a worm gear shaft which causes the cam to rotate. This movement of the cam is transferred via the cam follower to the E-bar, which moves until such time as the air gaps are again equidistant. With this electro-magnetic linkage, bearing friction is minimal. Consequently, minute movements of the capsule stack are sensed, whereas with a mechanical linkage, such small movements would be masked by friction and/or play in the bearings. The advantages of the servo-assisted altimeter apart from greater accuracy are that it is usable to greater altitudes and incorporates an easy to use digital read-out along with several warning indications/flags.

DIGITAL DISPLAY ALTIMETER

True digital instrument displays can show altitude in numerous ways. Use of a numerical display rather than a reproduction of the clock-type dial is most common. Often a digital numeric display of altitude is given on the electronic primary flight display near the artificial horizon depiction. A linear vertical scale may also be presented to put this hard numerical value in perspective. An example of this type of display of altitude information is shown in ***Figure 8-38***.

CABIN ALTIMETER/DIFFERENTIAL GAUGE

The instrument is typically used in pressurized aircraft to display cabin pressure in terms of altitude and/or differential. It enables the pilot or the flight engineer to effectively manage air conditioning and cabin pressurization systems. Due to inherent design inaccuracies, cabin altimeters are never used by the aircrew as a primary source of aircraft altitude reference. Nowadays, most cabin altimeters incorporate separate pointers for indicating atmosphere and cabin altitudes. The difference between the two pointers is used to indicate pressure differential in pounds per square inch (psi). There is no compensating devices or zero setting mechanism. The atmospheric altitude will only be approximate and can only be used as crude reference for aircraft altitude. The cabin altimeter pointers have a sector or arc associated with them. The outer scale shows cabin altitude graduated in 1 000 ft increments.

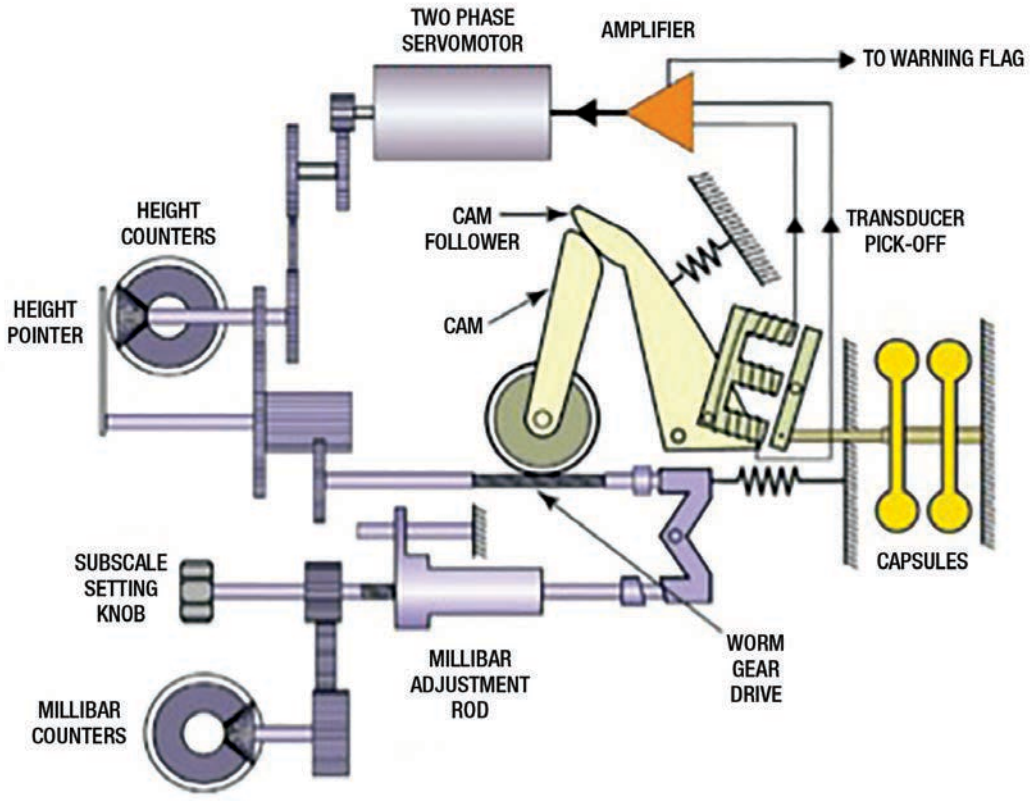


Figure 8-37. Schematic of a servo-assisted altimeter.

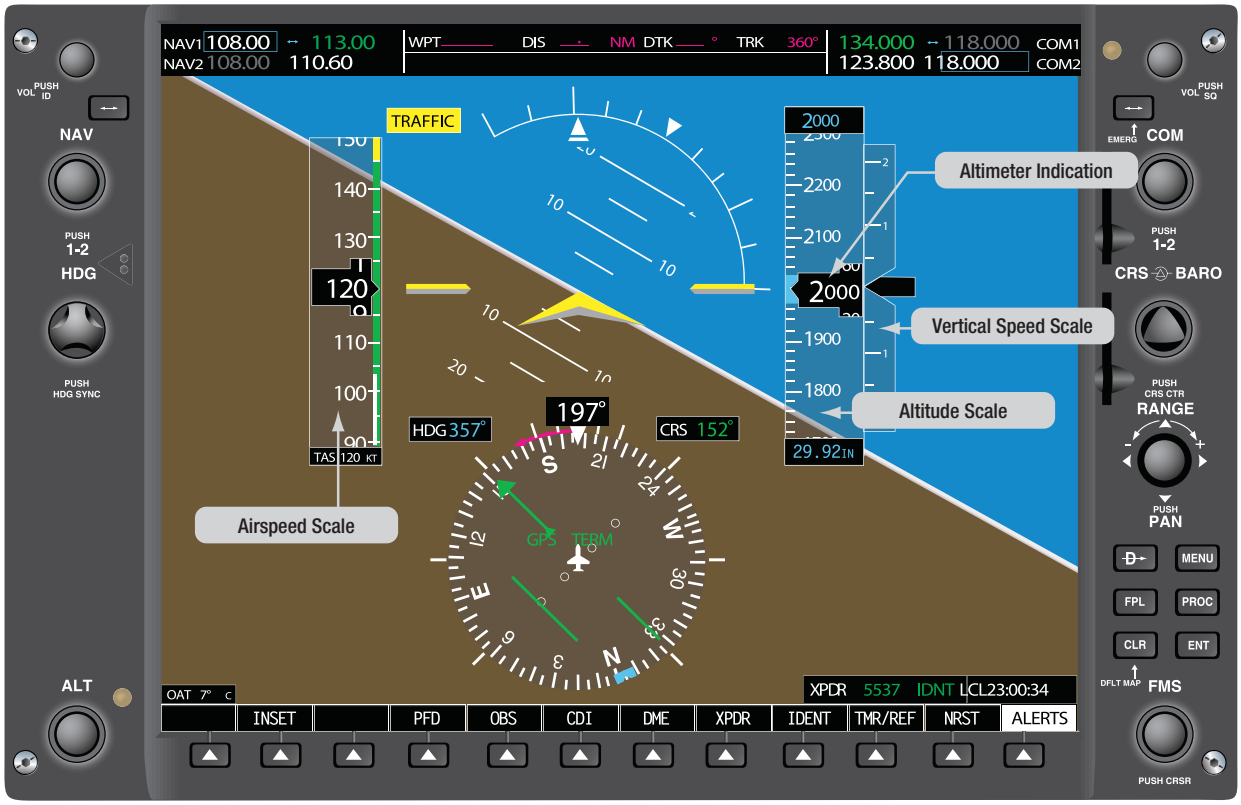


Figure 8-38. This primary flight display unit of a Garmin 1000 series glass cockpit instrumentation package for light aircraft indicates altitude using a vertical linear scale and a numerical counter. As the aircraft climbs or descends, the scale behind the black numerical altitude readout changes.

The inner scale is graduated in PSI and colored green, with a red mark to show the maximum cabin differential (***Figure 8-39***).

The cabin altimeter is mostly coupled with another gauge called the Cabin Rate of Climb Indicator which is similar to a vertical speed indicator (VSI). This gauge displays the rate at which cabin altitude is climbing or descending. Under normal circumstances, when the aircraft is climbing, it may be observable that the cabin rate of climb is less than aircraft rate of climb, cabin altitude is less than aircraft altitude and pressure differential is increasing. In case of aircraft descending, the cabin rate of climb shows a descent rate almost half of the aircraft rate of descent. Cabin altitude should decrease accordingly with pressure differential dropping slowly to zero. At about 500 ft AGL, cabin and aircraft altitude are equalized and pressure differential will be zero.



Figure 8-39. Cabin/differential pressure indicator and Cabin rate-of-climb indicator.

FACTORS AFFECTING ALTITUDE READINGS

AIR PRESSURE VARIATION

Air pressure is in a constant state of change due to variation in weather conditions, which could result in widely differing height readings being indicated. To overcome this problem, sensitive altimeters are provided with a 'Pressure Setting Scale' calibrated in millibars or inches of mercury. A setting knob adjusts the instrument mechanism so that the pointers are moved to give the correct reading when on the ground. It will also give the correct reading with reference to the ground, sea level, or other aircraft, when flying. To enable this adjustment to be made, three pressure values are used, which are normally referred to by their International Communication Code Letter 'Q'. (***Figure 8-40***).

QNE (AA/ALTITUDE AIRWAYS-FLIGHT LEVEL)

The altimeter's pressure reference is set to ICAO standard mean atmospheric pressure at sea level which is basically 1 013.25 mb. This pressure setting is referred to as 'standard' in aviation. The altimeter is set to QNE

QNE	QFE	QNH
I.C.A.O. standard mean sea level pressure of 1013.25 mb.	Ambient static pressure existing at mean airfield level at any particular time of the day i.e. the time the pressure value is required.	Local sea level pressure at any particular time of the day, of QFE converted to QNH by using I.C.A.O. standard atmosphere tables.

Figure 8-40. QNE, QFE and QNH altitude references.

from QNH, when climbing up through the "Transition Level". When flying above a transition height, the altimeter is set to QNE so that all aircraft altimeters will read the same at the same altitude. An altimeter, when set to QNE will read aircraft's Flight Level (FL). A reading of 25 000 ft is spelled out as FL250, 5 000 ft as FL050 and 13 500 ft as FL135.

QFE (Q-CODE FIELD ELEVATION)/AGL

QFE is mean sea level pressure corrected for temperature, adjusted for a specific site or datum like an airfield, being the most obvious example. When QFE is set on the

altimeter, it will read the 'height (AGL)' not altitude. It will read 'zero' at airfield elevation and after take-off will read height of the aircraft above that specific airfield. If the aircraft is flown to another airfield of different elevation and/or different QFE pressure, the pilot will have to ensure to reset that particular airfield's QFE if he wants the altimeter to read zero on touchdown.

QNH (Q-CODE NAUTICAL HEIGHT)/MSL

QNH is the pressure measured at a station then reduced down to Mean Sea Level (MSL) pressure. When set on the altimeter it will read altitude. If the aircraft is on the tarmac at any given airfield, the altimeter will display the airfield's elevation above mean sea level. This is the most commonly used pressure setting in the commercial world. It is probably the most useful setting to have, as nearly all aviation references to elevation are in relation to mean sea level. The mountain peaks on a map, airfield elevation, target elevation, minimum safe altitudes enroute etc. Incidentally, QNH is given as a regional pressure setting and should be updated with new ones if the aircraft leaves the area of reference into a new QNH pressure region. The QNH is the lowest forecast pressure at mean sea level for a given day (updated hourly) to ensure that safe terrain separation is maintained regardless of the day's variation in pressure. Values of QFE and QNH are obtained from the Airfield Air Traffic Control when needed.

In short, when an aircraft is on the ground, a pressure setting scale of QFE will result in an instrument reading of ground level, i.e. 'zero' within instrument tolerance. A pressure setting of QNH will result in the instrument showing the height of the airfield above mean sea level, within tolerance. Below the transition height the altimeter is commonly set to local QNH (***Figure 8-41***).

In the US and Canada, the transition altitude is fixed at 18 000 feet and the airspace above is known as the Standard Pressure Region. Under conditions of QNH at or above 1 013 mb, FL180 becomes the lowest usable flight level. If the altimeter setting is lower, the lowest usable FL becomes FL190 or even FL200. This restriction ensures that a minimum of 1 000' vertical separation is maintained between the aircraft at 17 000' on QNH and the one at the lowest usable level on Standard Pressure Setting (QNE). In practice, pilots typically change from QNH to QNE (depending upon the altimeter subscale) as they climb through 18 000 feet. If the cleared level is FL180, an early change is prudent. Descending, even when cleared to an altitude at the time cruising level is vacated, the altimeters will remain on standard pressure until just prior to the transition level. It is pertinent to mention that, in reality, the transition level "floats" based on the QNH - the lower the QNH, the higher the transition level - If the QNH is low and the pilots wait until approaching FL180 to change the altimeter subscale, a "Level Bust" is possible. Both primary altimeters are always changed at the same time. In most modern aircraft, not doing so will result in some form of an altimeter mismatch error.

In Europe, and much of the rest of the world, the transition altitude varies from airport to airport. Again, it is a fixed value and is published on the airport documentation, approach plates, AIP etc. The transition level will "float" with the QNH and, again, the TL is

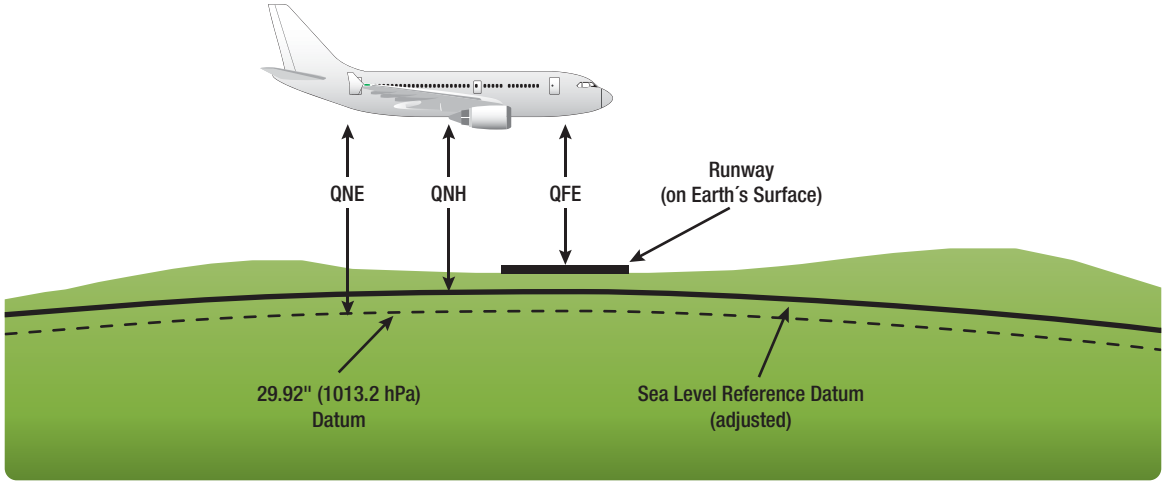


Figure 8-41. A graphical overview of altimetry terms QNE, QFE and QNH.

the lowest assignable level (pressure altitude) that will guarantee minimum vertical separation from an aircraft at the highest assignable altitude using local QNH. The "normal" barometric pressure setting procedure is a little different to that in North America. The procedure is: climbing and cleared to a FL-set Standard Pressure Setting (QNE/1013 mb); descending and cleared to an altitude - set QNH. This is done irrespective of how far above or below the TL/TA, an aircraft is at the time. The only exception is that when the aircraft cleared vertical profile (SID or STAR) has a specific altitude crossing restriction on it i.e. cross DVR at FL160 or cross BRI at 4 000', the altimeter must be left on the appropriate reference, until the restriction has been satisfied. Again, both primary altimeters are changed at the same time.

TEMPERATURE VARIATION

Temperature also affects the accuracy of an altimeter. The aneroid diaphragms used in altimeters are usually made of metal. Their elasticity changes as their temperature changes. This can lead to a false indication, especially at high altitudes when the ambient air is very cold. A bimetallic compensating device is built into many sensitive altimeters to correct for varying temperature. Temperature also affects air density, which has great impact on the performance of an aircraft. Although this does not cause the altimeter to produce an errant reading, flight crews must be aware that performance changes with temperature variations in the atmosphere.

The term density altitude describes altitude corrected for nonstandard temperature. That is, the density altitude is the standard day altitude (pressure altitude) at which an aircraft would experience similar performance as it would on the non-standard day currently being experienced. For example, on a very cold day, the air is denser than on a standard day, so an aircraft performs as though it is at a lower altitude. The density altitude is lower that day. On a very hot day, the reverse is true, and an aircraft performs as though it were at a higher elevation where the air is less dense. The density altitude is higher that day.

Conversion factors and charts have been produced so pilots can calculate the density altitude on any particular day. Inclusion of nonstandard air pressure due to weather systems and humidity can also be factored. So, while the effects of temperature on aircraft performance do not cause an altimeter to indicate falsely, an altimeter indication can be misleading in terms of aircraft performance if these effects are not considered. (***Figure 8-42***)

SCALE ERROR

Scale error is a mechanical error whereby the scale of the instrument is not aligned so the altimeter pointers indicate incorrectly. Periodic testing and adjustment by trained technicians using calibrated equipment ensures scale error is kept to a minimum. The pressure altimeter is connected to the pitot-static system and must receive an accurate sample of ambient air pressure to indicate the correct altitude.

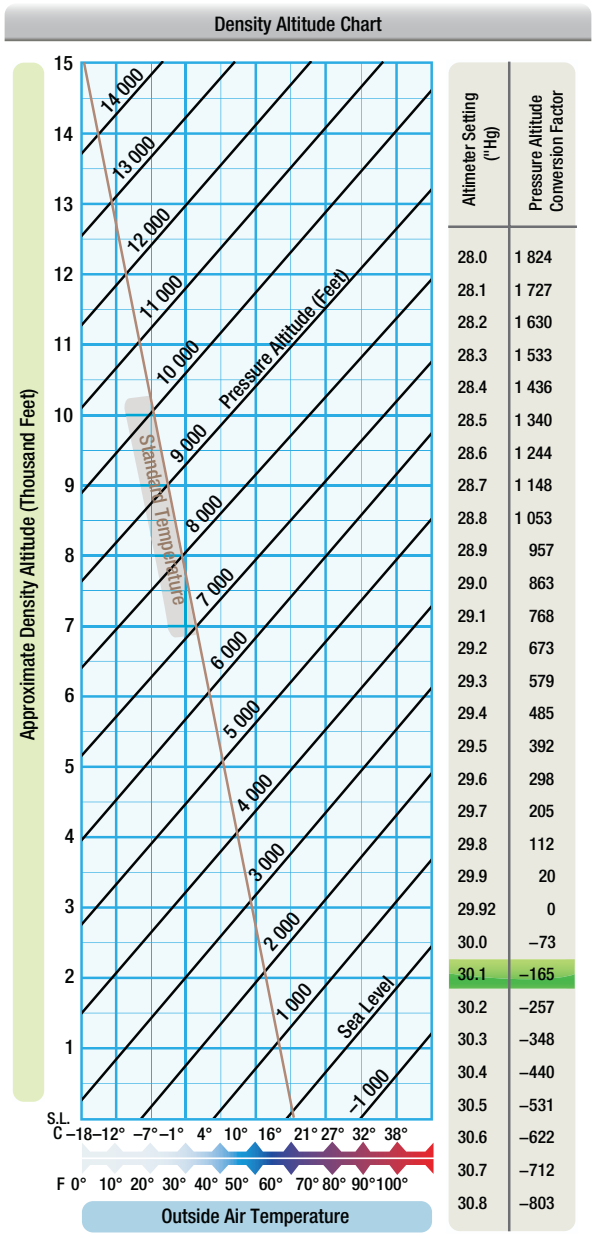


Figure 8-42. The effect of air temperature on aircraft performance is expressed as density altitude.

POSITION ERROR

Position error, or installation error, is that inaccuracy caused by the location of the static vent that supplies the altimeter. While every effort is made to place static vents in undisturbed air, airflow over the airframe changes with the speed and attitude of the aircraft. The amount of this air pressure collection error is measured in test flights, and a correction table showing the variances can be included with the altimeter for the pilot's use. Normally, location of the static vents is adjusted during these test flights so that the position error is minimal. Position error can be removed by the ADC in modern aircraft, so the pilot need not be concerned about this inaccuracy.

SYSTEM LEAKS

Static system leaks can affect the static air input to the altimeter or ADC resulting in inaccurate altimeter indications. It is for this reason that static system maintenance includes leak checks every 24 months, regardless of whether any discrepancy has been noticed. It should also be understood that analog mechanical altimeters are mechanical devices that often reside in a hostile environment. The significant vibration and temperature range swings encountered by the instruments and the pitot-static system (i.e., the tubing connections and fittings) can sometime create damage or a leak, leading to instrument malfunction. Proper care upon installation is the best preventive action. Periodic inspection and testing can also insure integrity.

It should be noted that many modern altimeters are constructed to integrate into flight control systems, autopilots, and altitude monitoring systems, such as those used by ATC. The basic pressure-sensing operation of these altimeters is the same, but a means for transmitting the information is added.

ALTIMETER SERVICEABILITY CHECKS

Altimeters are sensitive devices, more the reason, their serviceability is extremely important. Several basic checks are performed periodically to ensure altimeters work properly. These include visual checks, 'Zeroing' checks and case leak checks using appropriate test equipment.

IMPORTANT ALTITUDE TERMINOLOGIES

MEAN SEA LEVEL

The sea covers vast majority of the earth's total surface area and should provide a suitable datum, but, because of tides and the shape of the earth the level is not constant. An average or mean value has been determined, based on observations made at latitude 45 degrees North and South. Heights and altitudes are given as being Above Mean Sea Level (AMSL).

AERODROME ELEVATION

Aerodrome elevation is the highest point of the landing area (ICAO), measured to an accuracy of 1 ft (or 1/2 m). The aerodrome elevation is prominently displayed at the air traffic control tower. Also called above airfield elevation and airport elevation.

Figure 8-43 shows excerpt of the airport diagram for Tampa International Airport depicted above. The box at the end of runway with the notation "Field Elevation 26" This depicts the highest point on the runway which also coincides with the Touch Down Zone Elevation (TDZE) runway 28. If you look at runway 10 there is a depiction ELEV 14 which is the Runway End or Touchdown Zone Elevation of runway 10. The Touchdown Zone Elevation is the highest elevation in the first 3 000 feet of the landing surface.

The field elevation is not always located at the end of the runway. Look at the airport diagram for Trenton airport depicted below (***Figure 8-44***). The field elevation is depicted at the intersection of taxiway A and runway 16. Since this is within 3 000 feet of the end of the runway, the touchdown zone elevation is the same as the field elevation which is 213.

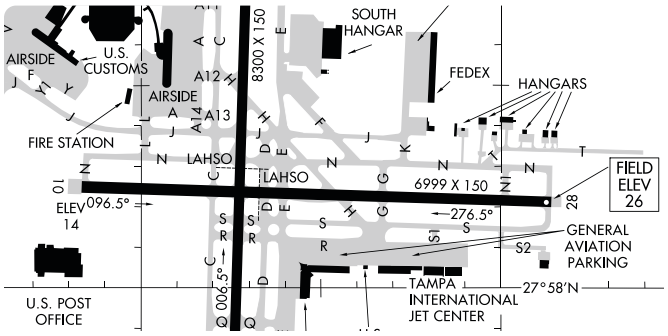


Figure 8-43. Runway elevation depiction in a typical airport chart.

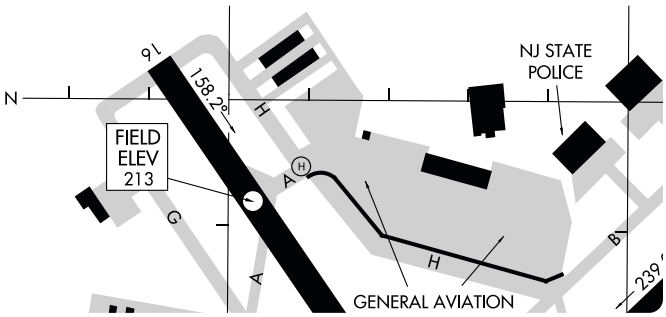


Figure 8-44. Field Elevation (Trenton Airport).

INDICATED ALTITUDE

Indicated altitude is the reading on the altimeter when the altimeter is set to the local barometric pressure at mean sea level. In UK aviation radiotelephony usage, the vertical distance of a level, a point or an object considered as a point, measured from mean sea level; this is referred to over the radio as altitude.

ABSOLUTE ALTITUDE

Absolute altitude is the height of the aircraft above the terrain over which it is flying. It can be measured using a radar altimeter (or "absolute altimeter"). Also referred to as "radar height" or feet/meters above ground level (AGL).

TRUE ALTITUDE

True altitude is the actual elevation above mean sea level. It is indicated altitude corrected for non-standard temperature and pressure. An approximate true altitude can be obtained locally by setting the altimeter to the local QNH.

HEIGHT

Height is the elevation above a ground reference point, commonly the terrain elevation. In UK aviation radiotelephony usage, the vertical distance of a level, a point or an object considered as a point, measured from a specified datum; is referred to over the radio as height, where the specified datum is the airfield elevation.

PRESSURE ALTITUDE

Pressure altitude is the elevation above a standard datum air-pressure plane (typically, 1 013.25 millibars or 29.92" Hg). Pressure altitude is used to indicate "flight level" which is the standard for altitude reporting in the U.S. in Class A airspace (above roughly 18 000 feet). Pressure altitude and indicated altitude are the same when the altimeter setting is 29.92" Hg or 1 013.25 millibars.

DENSITY ALTITUDE

Density altitude is the altitude corrected for non-ISA International Standard Atmosphere atmospheric conditions. Aircraft performance depends on density altitude, which is affected by barometric pressure, humidity and temperature. On a very hot day, density altitude at an airport (especially one at a high elevation) may be so high as to preclude takeoff, particularly for helicopters or a heavily loaded aircraft.

CALIBRATED ALTITUDE

Indicated altitude is corrected for instrument error and pressure error the result is called calibrated altitude.

TRANSITION ALTITUDE

The altitude at or below which the vertical position of an aircraft is controlled by reference to altitudes.

TRANSITION LEVEL

The lowest flight level available for use above the transition altitude.

TRANSITION LAYER

Transition Layer. The airspace between the transition altitude and the transition level.

VERTICAL SPEED INDICATOR

An analog vertical speed indicator (VSI) may also be referred to as a vertical velocity indicator (VVI), or rate of-climb indicator. It is a direct reading, differential pressure gauge that compares static pressure from the aircraft's static system directed into a diaphragm with static pressure surrounding the diaphragm in the instrument case. Air is free to flow unrestricted in and out of the diaphragm but is made to flow in and out of the case through a calibrated orifice. A pointer attached to the diaphragm indicates zero vertical speed when the pressure inside and outside the diaphragm are the same. The dial is usually graduated in 100s of feet per minute. A zeroing adjustment screw, or knob, on the face of the instrument is used to center the pointer exactly on zero, while the aircraft is on the ground. (***Figure 8-45***)

As the aircraft climbs, the unrestricted air pressure in the diaphragm lowers as the air becomes less dense. The case air pressure surrounding the diaphragm lowers more slowly, having to pass through the restriction created by the orifice. This causes unequal pressure inside and outside the diaphragm, which in turn causes

the diaphragm to contract a bit and the pointer indicates a climb. The process works in reverse for an aircraft in a descent. If a steady climb or descent is maintained, a steady pressure differential is established between the diaphragm and case pressure surrounding it, resulting in an accurate indication of the rate of climb via graduations on the instrument face (***Figure 8-46***).

INSTRUMENT LAG

A shortcoming of the rate-of-climb mechanism as described is that there is a lag of six to nine seconds before a stable differential pressure can be established that indicates the actual climb or descent rate of the aircraft. An instantaneous vertical speed indicator (IVSI) has a built-in mechanism to reduce this lag. A small, lightly sprung dashpot, or piston, reacts to the direction change of an abrupt climb or descent. As this small accelerometer does so, it pumps air into or out of the diaphragm, hastening the establishment of the pressure differential that causes the appropriate indication. (***Figure 8-47***)

VARIOMETER

Gliders and lighter-than-air aircraft often make use of a variometer. This is a differential VSI that compares static pressure with a known pressure. It is very sensitive and gives an instantaneous indication. It uses a rotating vane with a pointer attached to it. The vane separates



Figure 8-45. A typical vertical speed indicator.

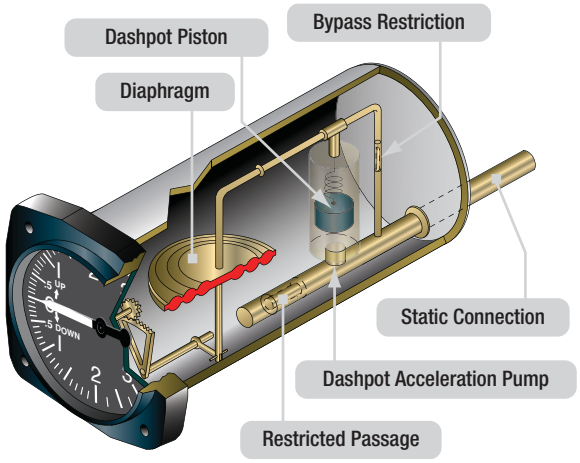


Figure 8-47. The small dashpot in this IVSI reacts abruptly to a climb or descent pumping air into or out of the diaphragm causing an instantaneously vertical speed indication.

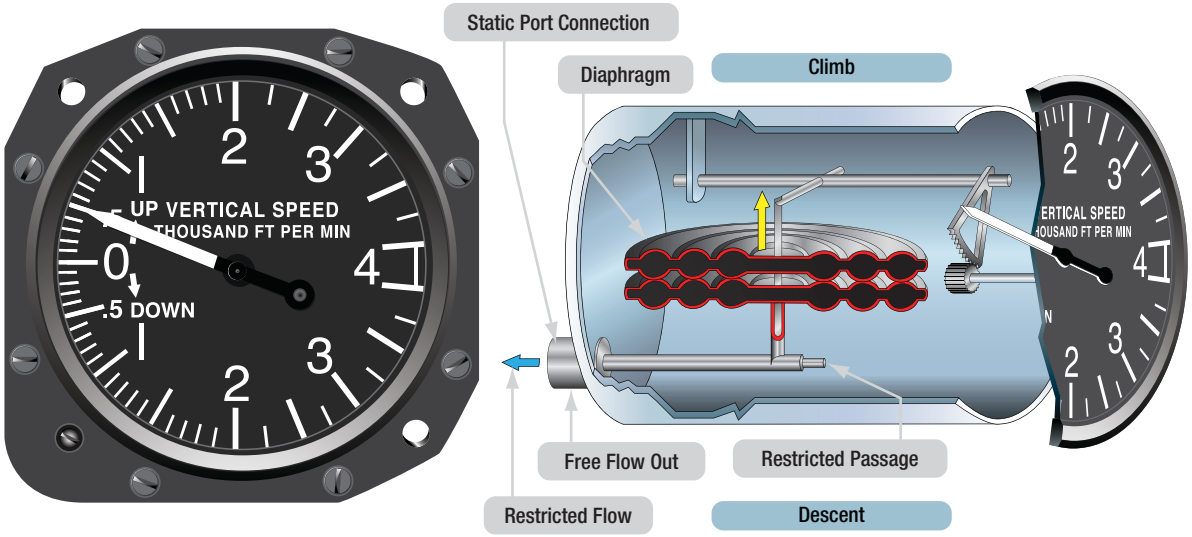


Figure 8-46. The VSI is a differential pressure gauge that compares free-flowing static air pressure in the diaphragm with restricted static air pressure around the diaphragm in the instrument case.

two chambers. One is connected to the aircraft's static vent or is open to the atmosphere. The other is connected to a small reservoir inside the instrument that is filled to a known pressure. As static air pressure increases, the pressure in the static air chamber increases and pushes against the vane. This rotates the vane and pointer, indicating a descent since the static pressure is now greater than the set amount in the chamber with reservoir pressure. During a climb, the reservoir pressure is greater than the static pressure; the vane is pushed in the opposite direction, causing the pointer to rotate and indicate a climb. (***Figure 8-48***)

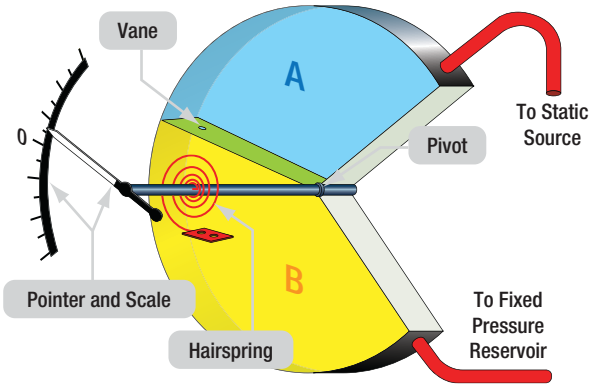


Figure 8-48. A variometer uses differential pressure to indicate vertical speed. A rotating vane separating two chambers (one with static pressure, the other with a fixed pressure reservoir), moves the pointer as static pressure changes.

DIGITAL RATE-OF-CLIMB INDICATORS

The rate-of-climb indication in a digitally displayed instrument system is computed from static air input to the ADC. An aneroid, or solid-state pressure sensor, continuously reacts to changes in static pressure. The digital clock within the computer replaces the calibrated orifice found on an analog instrument. As the static pressure changes, the computer's clock can be used to develop a rate for the change. Using the known lapse rate conversion for air pressure as altitude increases or decreases, a figure for climb or descent in fpm can be calculated and sent to the cockpit. The vertical speed is often displayed near the altimeter information on the primary flight display.

AIRSPEED INDICATOR

The airspeed indicator is another primary flight instrument that is also a differential pressure gauge. Ram air pressure from the aircraft's pitot tube is directed into a diaphragm in an analog airspeed instrument case. Static air pressure from the aircraft static vent(s) is directed into the case surrounding the diaphragm. As the speed of the aircraft varies, the ram air pressure varies, expanding or contracting the diaphragm. Linkage attached to the diaphragm causes a pointer to move over the instrument face, which is calibrated in knots or miles per hour (mph) (***Figure 8-49***). The relationship between the ram air pressure and static air pressure produces the indication known as indicated airspeed. As with the altimeter, there are other factors that must be considered in measuring airspeed throughout all phases of flight. These can cause inaccurate readings or indications that are not useful to the pilot in a particular situation. In analog airspeed indicators, the factors are often compensated for with ingenious mechanisms inside the case and on the instrument dial face.

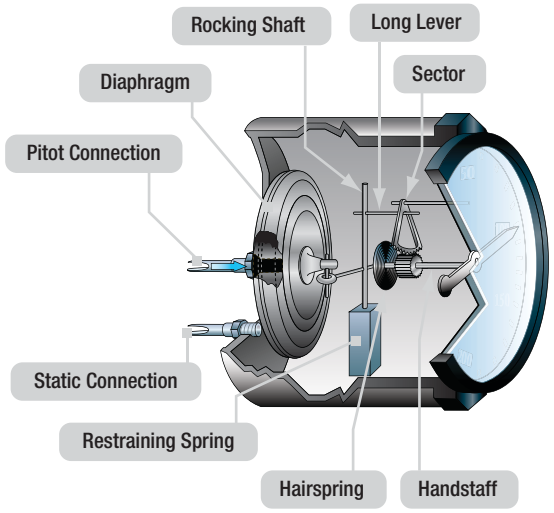


Figure 8-49. An airspeed indicator is a differential pressure gauge that compares ram air pressure with static pressure.

Digital flight instruments can have calculations performed in the ADC so the desired accurate indication is displayed. While the relationship between ram air pressure and static air pressure is the basis for most airspeed indications, it can be more accurate. Calibrated airspeed takes into account errors due to position error of the pitot static pickups. It also corrects for the nonlinear nature of the pitot static pressure differential when it is displayed on a linear scale. Analog airspeed indicators come with a correction chart that allows cross-referencing of indicated airspeed to calibrated airspeed for various flight conditions. These differences are typically very small and often are ignored. Digital instruments have these corrections performed in the ADC.

More importantly, indicated airspeed does not take into account temperature and air pressure differences needed to indicate true airspeed. These factors greatly affect airspeed indication. True airspeed, therefore, is the same as indicated airspeed when standard day conditions exist. But when atmospheric temperature or pressure varies, the relationship between the ram air pressure and static pressure alters. Analog airspeed instruments often include bimetallic temperature compensating devices that can alter the linkage movement between the diaphragm and the pointer movement. There can also be an aneroid inside the airspeed indicator case that can compensate for non-standard pressures. Alternatively, true airspeed indicators exist that allow the pilot to set temperature and pressure variables manually with external knobs on the instrument dial. The knobs rotate the dial face and internal linkages to present an indication that compensates for nonstandard temperature and pressure, resulting in a true airspeed indication. (***Figure 8-50***)



Figure 8-50. An analog true airspeed indicator. The pilot manually aligns the outside air temperature with the pressure altitude scale, resulting in an indication of true airspeed.

MACHMETER

It is important to keep high-speed aircraft from traveling faster than the speed of sound if they are not designed to do so. Even as an aircraft approaches the speed of sound, certain parts on the airframe may experience airflows that exceed it. The problem with this is that near the speed of sound, shock waves can develop that can affect flight controls and, in some cases, can literally tear the aircraft apart if not designed for supersonic airflow. A further complication is that the speed of sound changes with altitude and temperature. So a safe true airspeed at sea level could put the aircraft in danger at altitude due to the lower speed of sound (***Figure 8-51***).

Standard Altitude Temperature and the Speed of Sound		
Altitude (feet)	Temperature (°F)	Speed of Sound (knots)
Sea Level	59	661
2 000	52	657
4 000	48	652
6 000	38	648
8 000	30	643
10 000	23	638
12 000	16	633
14 000	9	629
16 000	2	624
18 000	–5	619
20 000	–12	614
22 000	–19	609
24 000	–27	604
26 000	–34	599
28 000	–41	594
30 000	–48	589
32 000	–55	584
34 000	–62	579
36 000	–69	574
38 000	–70	574
40 000	–70	574
42 000	–70	574
44 000	–70	574
46 000	–70	574
48 000	–70	574
50 000	–70	574

Figure 8-51. As temperatures fall at higher altitudes, the speed of sound is reduced.

In order to safeguard against these dangers, pilots monitor airspeed closely. A maximum allowable speed is established for the aircraft during certification flight testing. This speed is known the Critical Mach number or Mcrit. Mach is a term for the speed of sound. The critical Mach number is expressed as a decimal of Mach such as 0.8 Mach. This means 8/10 of the speed of sound, regardless of what the actual speed of sound is at any particular altitude. Many high performance aircraft are equipped with a Machmeter for monitoring Mcrit. The Machmeter is essentially an airspeed instrument that is calibrated in relation to Mach on the dial. Various scales exist for subsonic and supersonic aircraft (***Figure 8-52***).

In addition to the ram air/static air diaphragm arrangement, Machmeters also contain an altitude sensing diaphragm. It adjusts the input to the pointer so changes in the speed of sound due to altitude are incorporated into the indication.

Some aircraft use a Mach/airspeed indicator as shown in ***Figure 8-53***. This two-in-one instrument contains separate mechanisms to display the airspeed and Mach number. A standard white pointer is used to indicate airspeed in knots against one scale. A red and white striped pointer is driven independently and is read against the Mach number scale to monitor maximum allowable speed.



Figure 8-52. A Machmeter indicates aircraft speed relative to the speed of sound.



Figure 8-53. A combination Mach/airspeed indicator shows airspeed with a white pointer and Mach number with a red and white striped pointer. Each pointer is driven by separate internal mechanisms.

AIR DATA COMPUTERS (ADC) AND DIGITAL AIR DATA COMPUTERS (DADC)

High performance and jet transport category aircraft pitot-static systems may be more complicated. These aircraft frequently operate at high altitude where the ambient temperature falls below -50°C. The compressibility of air is also altered at high speeds and at high altitudes. Airflow around the fuselage changes, making it difficult to pick up consistent static pressure inputs. The pilot must compensate for all factors of air temperature and density to obtain accurate indications from instruments. While many analog instruments have compensating devices built into them, the use of an air data computer (ADC) is common for these purposes on high performance aircraft. Moreover, modern aircraft utilize digital air data computers (DADC). The conversion of sensed air pressures into digital values makes them more easily manipulated by the computer to output accurate information that has compensated for the many variables encountered (***Figure 8-54***).

Essentially, all pressures and temperatures captured by sensors are fed into the ADC. Analog units utilize transducers to convert these to electrical values and manipulate them in various modules containing circuits designed to make the proper compensations for use by different instruments and systems. A DADC usually



Figure 8-54. Teledyne's 90 004 TAS/Plus air data computer (ADC) computes air data information from the pitot-static pneumatic system, aircraft temperature probe, and barometric correction device to help create a clear indication of flight conditions.

receives its data in digital format. Systems that do not have digital sensor outputs will first convert inputs into digital signals via an analog-to-digital converter.

Conversion can take place inside the computer or in a separate unit designed for this function. Then, all calculation and compensations are performed digitally by the computer. Outputs from the ADC are electric to drive servo motors or for use as inputs in pressurization systems, flight control units, and other systems. DADC outputs are distributed to these same systems and the cockpit display using a digital data bus. There are numerous benefits of using ADCs. Simplification of pitot-static plumbing lines creates a lighter, simpler, system with fewer connections, so it is less prone to leaks and easier to maintain.

One time compensation calculations can be done inside the computer, eliminating the need to build compensating devices into numerous individual instruments or units of the systems using the air data. DADCs can run a number of checks to verify the plausibility of data received from any source on the aircraft. Thus, the crew can be alerted automatically of a parameter that is out of the ordinary. Change to an alternate data source can also be automatic so accurate flight deck and systems operations are continuously maintained.

In general, solid state technology is more reliable and modern units are small and lightweight. ***Figure 8-55*** shows a schematic of how a DADC is connected into the aircraft's pitot-static and other systems.

TEMPERATURE INDICATING SYSTEM

The temperature of numerous items must be known for an aircraft to be operated properly. Engine oil, carburetor mixture, inlet air, free air, engine cylinder heads, heater ducts, and exhaust gas temperature of turbine engines are all items requiring temperature monitoring. Many other temperatures must also be known. Different types of thermometers are used to collect and present temperature information.

NON-ELECTRIC TEMPERATURE INDICATORS

The physical characteristics of most materials change when exposed to changes in temperature. The changes are consistent, such as the expansion or contraction of solids, liquids, and gases. The coefficient of expansion of different materials varies and it is unique to each material. Most everyone is familiar with the liquid mercury thermometer. As the temperature of the mercury increases, it expands up a narrow passage that has a graduated scale upon it to read the temperature associated with that expansion. The mercury thermometer has no application in aviation. A bimetallic thermometer is very useful in aviation. The temperature sensing element of a bimetallic thermometer is made of two dissimilar metals strips bonded together. Each metal expands and contracts at a different rate when temperature changes.

One end of the bimetallic strip is fixed, the other end is coiled. A pointer is attached to the coiled end which is set in the instrument housing. When the bimetallic strip is heated, the two metals expand. Since their expansion rates differ and they are attached to each other, the effect is that the coiled end tries to uncoil as the one metal expands faster than the other. This moves the pointer across the dial face of the instrument. When the temperature drops, the metals contract at different rates, which tends to tighten the coil and move the pointer in the opposite direction.

Direct reading bimetallic temperature gauges are often used in light aircraft to measure free air temperature or outside air temperature (OAT). In this application, a collecting probe protrudes through the windshield of the aircraft to be exposed to the atmospheric air. The coiled end of the bimetallic strip in the instrument head is just inside the windshield where it can be read by the pilot. (***Figure 8-56 and Figure 8-57***)

ELECTRICAL TEMPERATURE MEASURING INDICATION

The use of electricity in measuring temperature is very common in aviation. The following measuring and indication systems can be found on many types of aircraft. Certain temperature ranges are more suitably measured by one or another type of system.

ELECTRICAL RESISTANCE THERMOMETER

The principle parts of the electrical resistance thermometer are the indicating instrument, the temperature-sensitive element (or bulb), and the connecting wires and plug

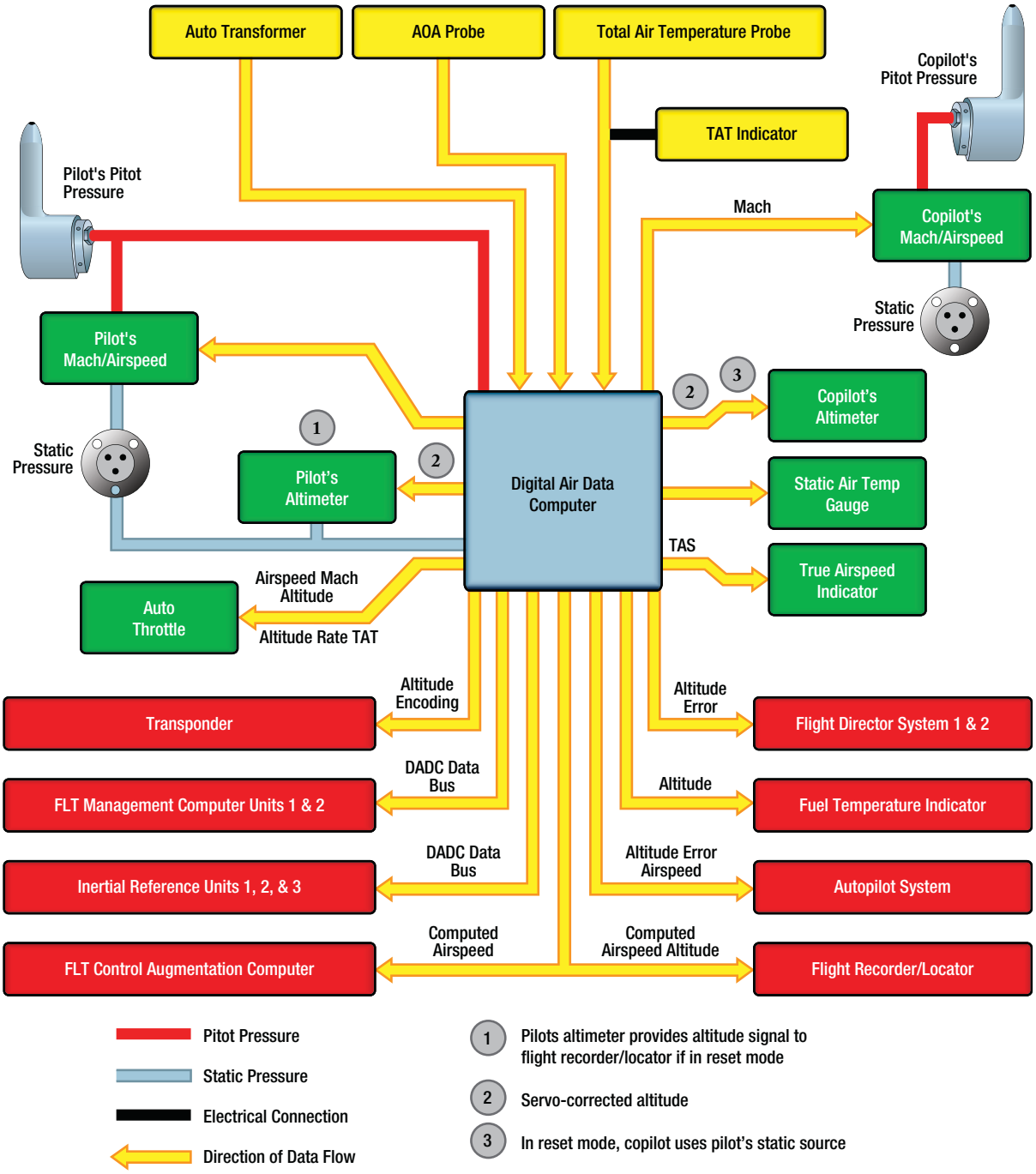


Figure 8-55. ADCs receive input from the pitot-static sensing devices and process them for use by numerous aircraft systems.

connectors. Electrical resistance thermometers are used widely in many types of aircraft to measure carburetor air, oil, free air temperatures, and more. They are used to measure low and medium temperatures in the -70°C to 150°C range.

For most metals, electrical resistance changes as the temperature of the metal changes. This is the principle upon which a resistance thermometer operates. Typically, the electrical resistance of a metal increases as the temperature rises. Various alloys have a high temperature-resistance coefficient, meaning their resistance varies significantly with temperature. This can make them suitable for use in temperature sensing devices. The metal resistor is subjected to the fluid or area in which temperature needs to be measured. It is connected by wires to a resistance measuring device inside the cockpit indicator. The instrument dial is calibrated in degrees Fahrenheit or Celsius as desired rather than in ohms. As the temperature to be measured changes, the resistance of the metal changes and the resistance measuring indicator shows to what extent.

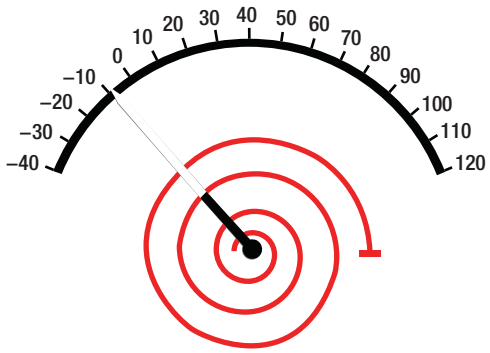


Figure 8-56. A bimetallic temperature gauge works because of the dissimilar coefficients of expansion of two metals bonded together. When bent into a coil, cooling or heating causes the dissimilar metal coil to tighten, or unwind, moving the pointer across the temperature scale on the instrument dial face.



Figure 8-57. A bimetallic outside air temperature gauge and its installation on a light aircraft.

A typical electrical resistance thermometer looks like any other temperature gauge. Indicators are available in dual form for use in multi-engine aircraft. Most indicators are self-compensating for changes in cockpit temperature. The heat-sensitive resistor is manufactured so that it has a definite resistance for each temperature value within its working range. The temperature-sensitive resistor element is a length or winding made of a nickel/ manganese wire or other suitable alloy in an insulating material. The resistor is protected by a closed-end metal tube attached to a threaded plug with a hexagonal head. (***Figure 8-58***)

The two ends of the winding are brazed, or welded, to an electrical receptacle designed to receive the prongs of the connector plug. The indicator contains a resistance-measuring instrument. Sometimes it uses a modified form of the Wheatstone bridge circuit. The Wheatstone-bridge meter operates on the principle of balancing one unknown resistor against other known resistances. A simplified form of a Wheatstone bridge circuit is shown in ***Figure 8-59***. Three equal values of resistance are connected into a diamond shaped bridge circuit. A resistor with an unknown value is also part of the circuit. The unknown resistance represents the resistance of the temperature bulb of the electrical resistance thermometer system. A galvanometer is attached across the circuit at points X and Y.

When the temperature causes the resistance of the bulb to equal that of the other resistances, no potential difference exists between points X and Y in the circuit. Therefore, no current flows in the galvanometer leg



Figure 8-58. An electric resistance thermometer sensing bulb.

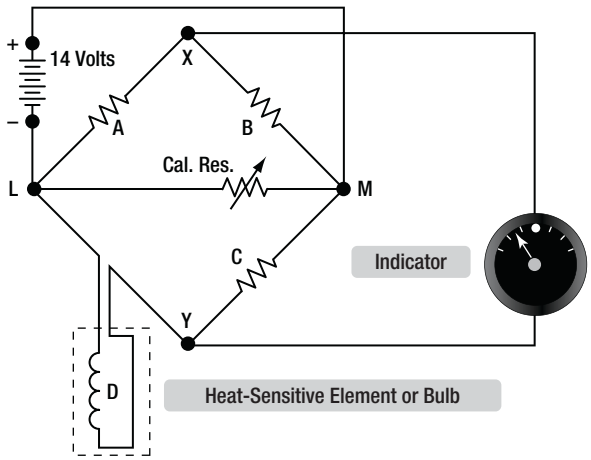


Figure 8-59. The internal structure of an electric resistance thermometer indicator features a bridge circuit, galvanometer. A variable resistor or probe outside the indicator is in the form of the temperature sensor or probe.

of the circuit. If the temperature of the bulb changes, its resistance also changes, and the bridge becomes unbalanced, causing current to flow through the galvanometer in one direction or the other. The galvanometer pointer is actually the temperature gauge pointer. As it moves against the dial face calibrated in degrees, it indicates temperature. Many indicators are provided with a zero adjustment screw on the face of the instrument. This adjusts the zeroing spring tension of the pointer when the bridge is at the balance point (the position at which the bridge circuit is balanced and no current flows through the meter).

RATIOMETER ELECTRICAL RESISTANCE THERMOMETERS

Another way of indicating temperature when employing an electric resistance thermometer is by using a ratiometer. The Wheatstone-bridge indicator is subject to errors from line voltage fluctuation. The ratiometer is more stable and can deliver higher accuracy. As its name suggests, the ratiometer electrical resistance thermometer measures a ratio of current flows. The resistance bulb sensing portion of the ratiometer electric resistance thermometer is essentially the same as described above.

The circuit contains a variable resistance and a fixed resistance to provide the indication. It contains two branches for current flow. Each has a coil mounted on either side of the pointer assembly that is mounted within the magnetic field of a large permanent magnet. Varying current flow through the coils causes different magnetic fields to form, which react with the larger magnetic field of the permanent magnet. This interaction rotates the pointer against the dial face that is calibrated in degrees. Fahrenheit or Celsius, giving a temperature indication (***Figure 8-60***).

The magnetic pole ends of the permanent magnet are closer at the top than they are at the bottom. This causes the magnetic field lines of flux between the poles to be more concentrated at the top. As the two coils produce their magnetic fields, the stronger field interacts and pivots downward into the weaker, less concentrated part of the permanent magnet field, while the weaker coil magnetic field shifts upward toward the more concentrated flux field of the large magnet. This provides a balancing effect that changes but stays in balance as the coil field strengths vary with temperature and the resultant current flowing through the coils. For example, if the resistance of the temperature bulb is equal to the value of the fixed resistance (R), equal values of current flow through the coils. The torques, caused by the magnetic field each coil creates, are the same and cancel any movement in the larger magnetic field. The indicator pointer will be in the vertical position. If the bulb temperature increases, its resistance also increases. This causes the current flow through coil a circuit branch to increase. This creates a stronger magnetic field at coil A than at coil B. Consequently, the torque on coil A increases, and it is pulled downward into the weaker part of the large magnetic field. At the same time, less current flows through the sensor bulb resistor

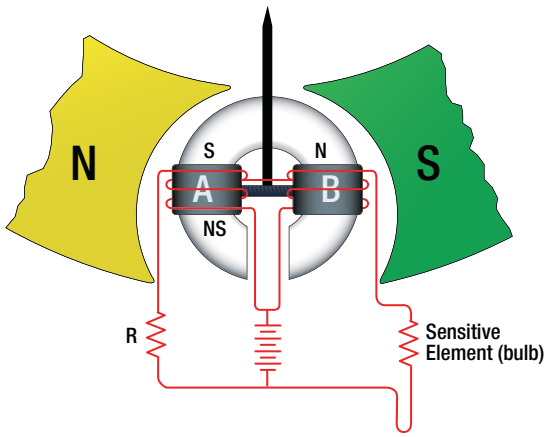


Figure 8-60. A ratiometer temperature measuring indicator has two coils. As the sensor bulb resistance varies with temperature, different amounts of current flow through the coils. This produces varying magnetic fields. These fields interact with the magnetic field of a large permanent magnet, resulting in an indication of temperature.

and coil B, causing coil B to form a weaker magnetic field that is pulled upward into the stronger flux area of the permanent magnet's magnetic field. The pointer stops rotating when the fields reach a new balance point that is directly related to the resistance in the sensing bulb. The opposite of this action would take place if the temperature of the heat-sensitive bulb should decrease.

Ratiometer temperature measuring systems are used to measure engine oil, outside air, carburetor air, and other temperatures in many types of aircraft. They are especially in demand to measure temperature conditions where accuracy is important, or large variations of supply voltages are encountered.

THERMOCOUPLE TEMPERATURE INDICATORS

A thermocouple is a circuit or connection of two unlike metals. The metals are touching at two separate junctions. If one of the junctions is heated to a higher temperature than the other, an electromotive force is produced in the circuit. This voltage is directly proportional to the temperature. So, by measuring the amount of electromotive force, temperature can be determined. A voltmeter is placed across the colder of the two junctions of the thermocouple. It is calibrated in degrees Fahrenheit or Celsius, as needed. The hotter the high temperature junction (hot junction) becomes, the greater the electromotive force produced, and the higher the temperature indication on the meter. (***Figure 8-61***)

Thermocouples are used to measure high temperatures. Two common applications are the measurement of cylinder head temperature (CHT) in reciprocating engines and exhaust gas temperature (EGT) in turbine engines. Thermocouple leads are made from a variety of metals, depending on the maximum temperature to which they are exposed. Iron and constantan, or copper and constantan, are common for CHT measurement. Chromel and alumel are used for turbine EGT thermocouples.

The amount of voltage produced by the dissimilar metals when heated is measured in millivolts. Therefore, thermocouple leads are designed to provide a specific amount of resistance in the thermocouple circuit (usually very little). Their material, length, or cross-sectional size cannot be altered without compensation for the change in total resistance that would result. Each lead that makes a connection back to the voltmeter must be made of the same metal as the part of the thermocouple to which it is connected. For example, a copper wire is connected to the copper portion of the hot junction and a constantan wire is connected to the constantan part. The hot junction of a thermocouple varies in shape depending on its application. Two common types are the gasket and the bayonet.

In the gasket type, two rings of the dissimilar metals are pressed together to form a gasket that can be installed under a spark plug or cylinder hold down nut.

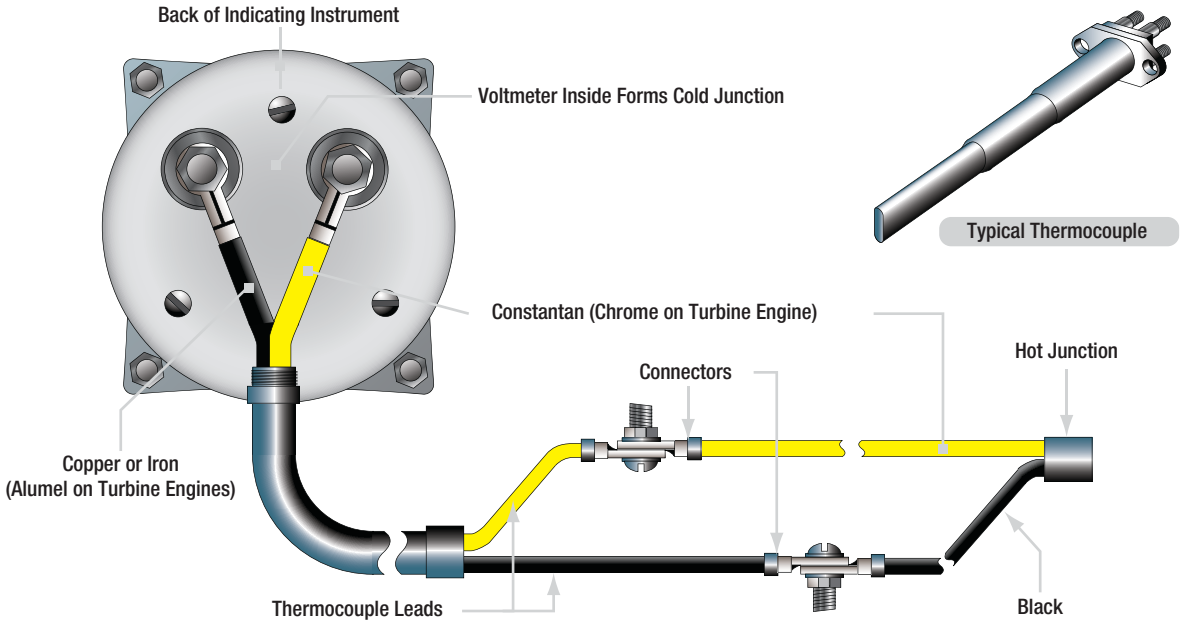


Figure 8-61. Thermocouples combine two unlike metals that cause current flow when heated.

In the bayonet type, the metals come together inside a perforated protective sheath. Bayonet thermocouples fit into a hole or well in a cylinder head. On turbine engines, they are found mounted on the turbine inlet or outlet case and extend through the case into the gas stream. Note that for CHT indication, the cylinder chosen for the thermocouple installation is the one that runs the hottest under most operating conditions. The location of this cylinder varies with different engines (***Figure 8-62***).

The cold junction of the thermocouple circuit is inside the instrument case. Since the electromotive force set up in the circuit varies with the difference in temperature between the hot and cold junctions, it is necessary to compensate the indicator mechanism for changes in cockpit temperature which affect the cold junction. This is accomplished by using a bimetallic spring connected to the indicator mechanism. This actually works the same as the bimetallic thermometer described previously. When the leads are disconnected from the indicator, the temperature of the cockpit area around the instrument panel can be read on the indicator dial. (***Figure 8-63***) Numeric LED indictors for CHT are also common in modern aircraft.

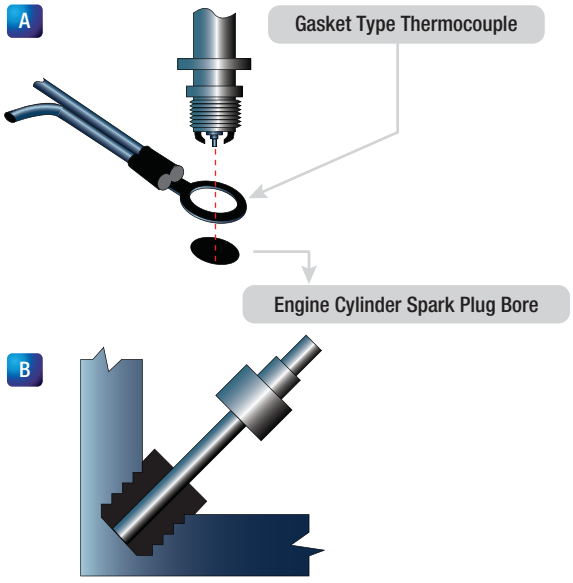


Figure 8-62. A cylinder head temperature thermocouple with a gasket type hot junction is made to be installed under the spark plug or a cylinder hold down nut of the hottest cylinder (A). A bayonet type thermocouple is installed in a bore in the cylinder wall (B).

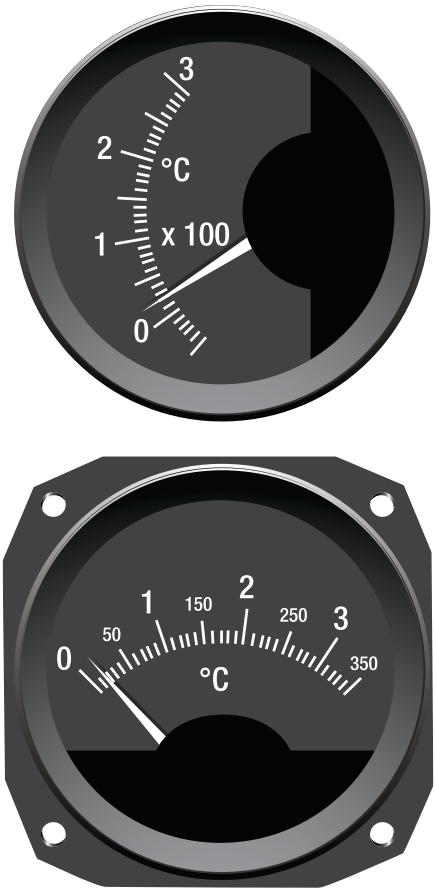


Figure 8-63. Typical thermocouple temperature indicators.

TURBINE GAS TEMPERATURE INDICATING SYSTEMS

EGT is a critical variable of turbine engine operation. The EGT indicating system provides a visual temperature indication in the cockpit of the turbine exhaust gases as they leave the turbine unit. In certain turbine engines, the temperature of the exhaust gases is measured at the entrance to the turbine unit. This is referred to as a turbine inlet temperature (TIT) indicating system. Several thermocouples are used to measure EGT or TIT. They are spaced at intervals around the perimeter of the engine turbine casing or exhaust duct. The tiny thermocouple voltages are typically amplified and used to energize a servomotor that drives the indicator pointer. Gearing a digital drum indication off of the pointer motion is common. (***Figure 8-64***).

The EGT indicator shown is a hermetically sealed unit. The instrument's scale ranges from 0°C to 1 200°C, with a vernier dial in the upper right-hand corner and a power off warning flag located in the lower portion of the dial. A TIT indicating system provides a visual indication at the instrument panel of the temperature

of gases entering the turbine. Numerous thermocouples can be used with the average voltage representing the TIT. Dual thermocouples exist containing two electrically independent junctions within a single probe. One set of these thermocouples is paralleled to transmit signals to the cockpit indicator. The other set of parallel thermocouples provides temperature signals to engine monitoring and control systems. Each circuit is electrically independent, providing dual system reliability.

A schematic for the turbine inlet temperature system for one engine of a four-engine turbine aircraft is shown in ***Figure 8-65***. Circuits for the other three engines are

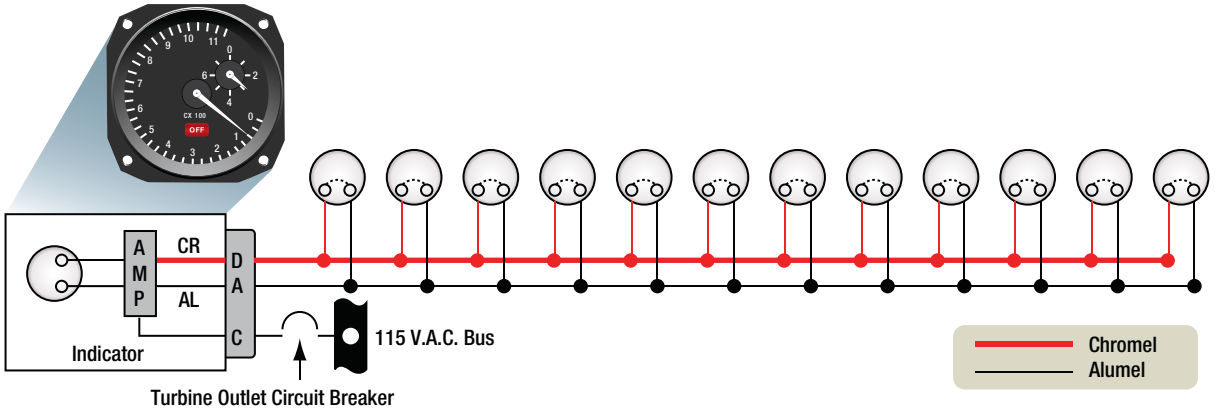


Figure 8-64. A typical exhaust gas temperature thermocouple system.

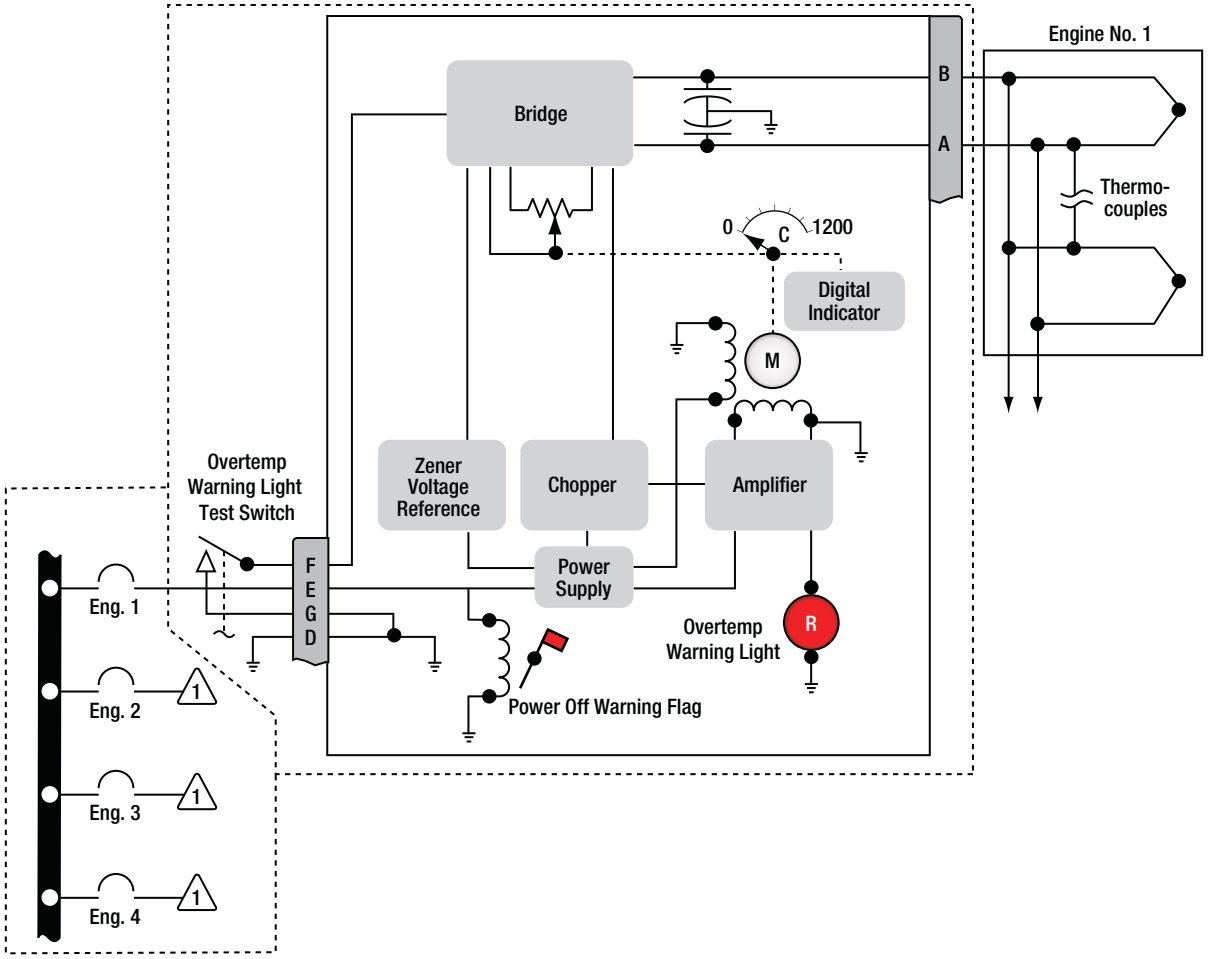


Figure 8-65. A typical analog turbine inlet temperature indicating system.

identical to this system. The indicator contains a bridge circuit, a chopper circuit, a two-phase motor to drive the pointer, and a feedback potentiometer. Also included are a voltage reference circuit, an amplifier, a power-off flag, a power supply, and an over temperature warning light. Output of the amplifier energizes the variable field of the two-phase motor that positions the indicator main pointer and a digital indicator.

The motor also drives the feedback potentiometer to provide a humming signal to stop the drive motor when the correct pointer position, relative to the temperature signal, has been reached. The voltage reference circuit provides a closely regulated reference voltage in the bridge circuit to preclude error from input voltage variation to the indicator power supply.

The over-temperature warning light in the indicator illuminates when the TIT reaches a predetermined limit. An external test switch is usually installed so that over temperature warning lights for all the engines can be tested at the same time. When the test switch is operated, an over-temperature signal is simulated in each indicator temperature control bridge circuit.

Digital cockpit instrumentation systems need not employ resistance-type indicators and adjusted servo-driven thermocouple gauges to provide the pilot with temperature information. Sensor resistance and voltage values are input to the appropriate computer, where they are adjusted, processed, monitored, and output for display on cockpit display panels. They are also sent for use by other computers requiring temperature information for the control and monitoring of various integrated systems.

TOTAL AIR TEMPERATURE MEASUREMENT

Air temperature is a valuable parameter that many performance monitoring and control variables depend on. During flight, static air temperature changes continuously and accurate measurement presents challenges. Below 0.2 Mach, a simple resistance-type or bimetallic temperature gauge can provide relatively accurate air temperature information. At faster speeds, friction, the air's compressibility, and boundary layer behavior make accurate temperature capture more complex. Total air temperature (TAT) is the static air temperature plus any rise in temperature caused by the high-speed movement of the aircraft through the air. The increase in temperature is known as ram rise. TAT-sensing probes are constructed specifically to accurately capture this value and transmit signals for cockpit indication, as well as for use in various engine and aircraft systems.

Simple TAT systems include a sensor and an indicator with a built-in resistance balance circuit. Air flow through the sensor is designed so that air with the precise temperature impacts a platinum alloy resistance element. The sensor is engineered to capture temperature variations in terms of varying the resistance of the element. When placed in the bridge circuit, the indicator pointer moves in response to the imbalance caused by the variable resistor.

More complex systems use signal correction technology and amplified signals sent to a servo motor to adjust the indicator in the cockpit. These systems include closely regulated power supply and failure monitoring. They often use numeric drum type readouts, but can also be sent to an LCD driver to illuminate LCD displays. Many LCD displays are multifunctional, capable of displaying static air temperature and true airspeed.

In fully digital systems, the correction signals are input into the ADC. There, they can be manipulated appropriately for cockpit display or for whichever system requires temperature information. (***Figure 8-66***)

TAT sensor/probe design is complicated by the potential of ice forming during icing conditions. Left unheated, a probe may cease to function properly. The inclusion of a heating element threatens accurate data collection. Heating the probe must not affect the resistance of the sensor element. (***Figure 8-67***). Close attention is paid to airflow and materials conductivity during the design phase. Some TAT sensors channel bleed air through the units to affect the flow of outside air, so that it flows directly onto the platinum sensor without gaining added energy from the probe heater.

Temperature is affected by radiation from the sun and heat from the Earth, which stores the sun's heat in different amounts depending on the surface type. Temperature changes a lot from area to area, depending on the time of day, time of year, atmospheric conditions, global location, and surface type. Knowing the external air temperature is very important for a pilot. The best way



Figure 8-66. Different cockpit TAT displays.

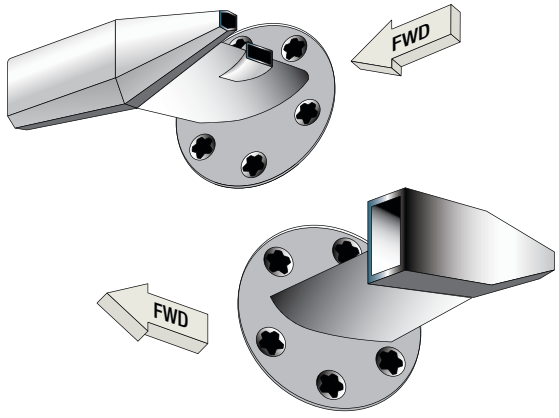


Figure 8-67. Total air temperature (TAT) probes.

to measure it is using a Static Air Temperature (SAT) reading, but unfortunately this can only be measured when the aircraft is still. The friction of air against the aircraft causes a Temperature Rise (TR), which makes the temperature measurement wrong. While moving, the SAT, or Outside Air Temperature (OAT) can be found by subtracting the TR from the Total Air Temperature (TAT). TAT is measured using a Total Air Temperature Probe, also known as a 'Rosemount Probe'. This involves slowing the airflow down to an almost complete stop and measuring the Ram Rise. In a perfect world the airflow would be slowed to a complete stop (stagnation), however this is not possible as there must be airflow through the probe to allow for continual readings. This (along with other factors) causes a very slight error which must be corrected for and is known as the Recovery Factor.

Less efficient instruments will record a Ram Air Temperature (RAT) instead of TAT, which can still be used to calculate OAT but is less accurate. Modern aircraft, such as the Airbus A350A are installed with Multifunctional Probes (MFP) which are capable of performing functions of pitot tube, an angle-of-attack (AoA) sensor and a Total Air Temperature (TAT) probe (***Figure 8-68***). For redundancy, two or more probes are installed to account for reading errors and failure of any one probe/sensor.



Figure 8-68. Multifunction Probes (MFP) installed on the Airbus A350 XWB aircraft.

FUEL QUANTITY INDICATING SYSTEMS

All aircraft fuel systems must have some form of fuel quantity indicator. These devices vary widely depending on the complexity of the fuel system and the aircraft on which they are installed.

RATIOMETER-TYPE

Electric fuel quantity indicators are common in aircraft. Most of these units operate with direct current (DC) and use variable resistance in a circuit to drive a ratiometer type indicator. The movement of a float in the tank moves a connecting arm to the wiper on a variable resistor in the tank unit. This resistor is wired in series with one of the coils of the ratiometer-type fuel gauge in the instrument panel. Changes to the current flowing through the tank unit resistor change the current flowing through one of the coils in the indicator. This alters the magnetic field in which the indicating pointer pivots. The calibrated dial indicates the corresponding fuel quantity (***Figure 8-69***).

Digital indicators are available that work with the same variable resistance signal from the tank unit. They convert the variable resistance into a digital display in the cockpit instrument head (***Figure 8-70***). Fully digital instrumentation systems, such as those found in a glass cockpit aircraft, convert the variable resistance into a digital signal to be processed in a computer and displayed on a flat screen panel.

CAPACITANCE-TYPE

Large and high-performance aircraft typically utilize electronic fuel quantity systems. These more costly systems have the advantage of having no moving parts in the tank sending units. Variable capacitance transmitters are installed in the fuel tanks extending from the top to the bottom of each tank in the usable fuel. Several of these tank units, or fuel probes as they are sometimes called, may be installed in a large tank (***Figure 8-71***).

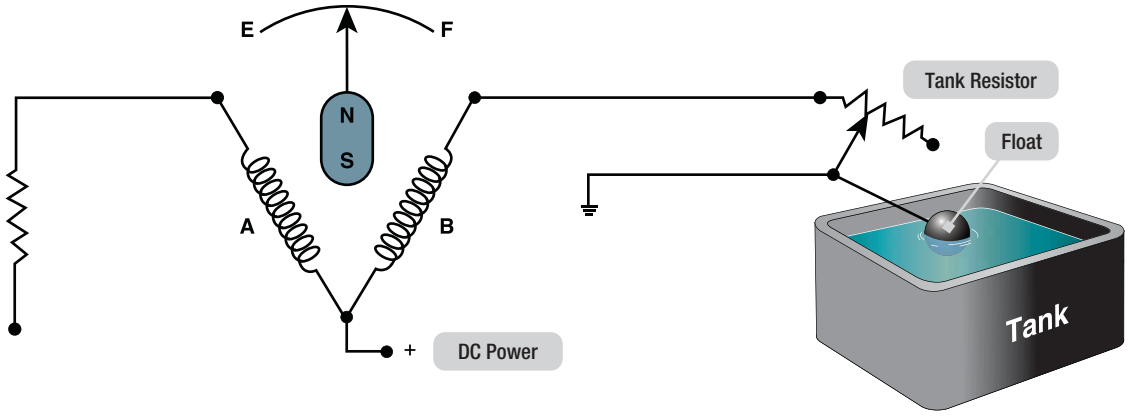


Figure 8-69. A DC electric fuel quantity indicator uses a variable resistor in the tank unit, which is moved by a float arm.

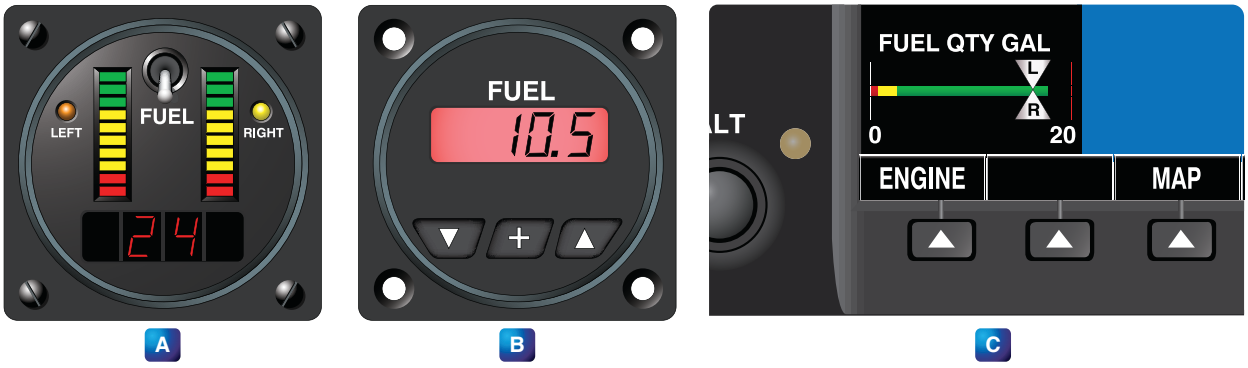


Figure 8-70. Digital fuel quantity gauges that work off of variable resistance from the tank unit are shown in A and B. The fuel quantity indication of a Garmin G-1000 flat screen display is shown in C.

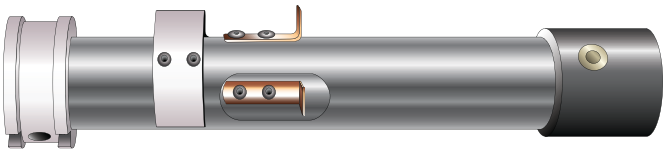


Figure 8-71. A fuel tank transmitter for a capacitance-type fuel quantity indicating system.

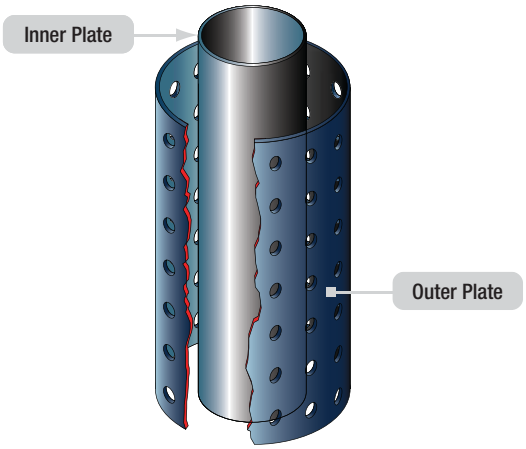


Figure 8-72. The capacitance of tank probes varies in a capacitance- type fuel tank indicator system as the space between the inner and outer plates is filled with varying quantities of fuel and air depending on the amount of fuel in the tank.

They are wired in parallel. As the level of the fuel changes, the capacitance of each unit changes. The capacitance transmitted by all of the probes in a tank is totaled and compared in a bridge circuit by a microchip computer in the tank's digital fuel quantity indicator in the cockpit.

As the aircraft maneuvers, some probes are in more fuel than others due to the attitude of the aircraft. The indication remains steady, because the total capacitance transmitted by all of the probes remains the same. A trimmer is used to match the capacitance output with the pre-calibrated quantity indicator.

A capacitor is a device that stores electricity. The amount it can store depends on three factors: the area of its plates, the distance between the plates, and the dielectric constant of the material separating the plates. A fuel tank unit contains two concentric plates that are a fixed distance apart. Therefore, the capacitance of a unit can change if the dielectric constant of the material separating the plates varies. The units are open at the top and bottom so they can assume the same level of fuel as is in the tanks. Therefore, the material between the plates is either fuel (if the tank is full), air (if the tank is empty), or some ratio of fuel and air depending on how much fuel remains in the tank. (***Figure 8-72***) shows a simplified illustration of this construction.

The bridge circuit that measures the capacitance of the tank units uses a reference capacitor for comparison. When voltage is induced into the bridge, the capacitive reactance of the tank probes and the reference capacitor can be equal or different. The magnitude of the difference is translated into an indication of the fuel quantity in the tank calibrated in pounds. ***Figure 8-73*** represents the nature of this comparison bridge circuit.

The use of tank unit capacitors, a reference capacitor, and a microchip bridge circuit in the fuel quantity indicators is complicated by the fact that temperature affects the dielectric constant of the fuel. A compensator unit (mounted low in the tank so it is always covered with fuel) is wired into the bridge circuit. It modifies current flow to reflect temperature variations of the fuel, which affect fuel density and thus capacitance of the tank units. (***Figure 8-74***)

An amplifier is also needed in older systems. The amplitude of the electric signals must be increased to move the servo motor in the analog indicator. Additionally, the dielectric constant of different turbine-engine fuels approved for a particular aircraft may also vary. Calibration is required to overcome this. A fuel summation unit is part of the capacitance-type fuel quantity indication system. It is used to add the tank quantities from all indicators. This total aircraft fuel quantity can be used by the crew and by flight management computers for calculating optimum

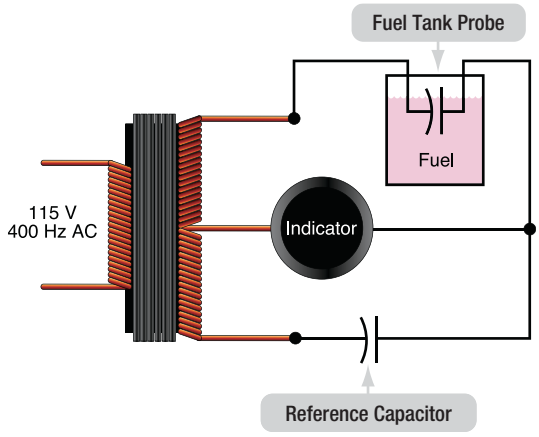


Figure 8-73. A simplified capacitance bridge for a fuel quantity system.

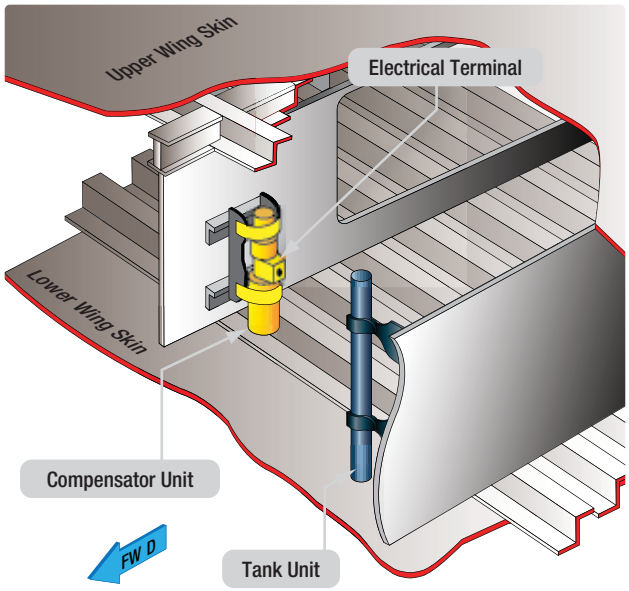


Figure 8-74. A fuel quantity tank unit and compensator unit installed inside a wing tank.

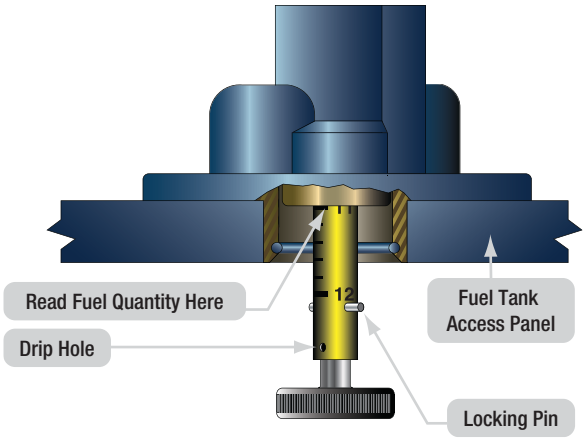


Figure 8-75. A fuel drip stick is lowered from the fuel tank bottom until fuel drips out the hole at the bottom. By reading the calibrated scale and adding readings from all tank drip sticks, a chart can be consulted to arrive at the total fuel quantity on the aircraft by weight or by volume.

airspeed and engine performance limits for climb, cruise, descent, etc. Capacitance-type fuel quantity system test units are available for troubleshooting and ensuring proper functioning and calibration of the indicating system components.

MECHANICAL-TYPE

Many aircraft with capacitance-type fuel indicating systems also use a mechanical indication system to cross-check fuel quantity indications and to ascertain the amount of fuel onboard the aircraft when electrical power is not available. A handful of fuel measuring sticks, or drip sticks, are mounted throughout each tank. When pushed and rotated, the drip stick can be lowered until fuel begins to exit the hole on the bottom of each stick. This is the point at which the top of the stick is equal to the height of the fuel. The sticks have a calibrated scale on them. By adding the indications of all of the drip sticks and converting to pounds or gallons via a chart supplied by the manufacturer, the quantity of the fuel in the tank can be ascertained. (***Figure 8-75***).

ULTRASONIC-TYPE

Ultrasonic level sensors gauge fuel level by measuring the distance from the transmitter (usually located at the tip of a fuel tank) to the surface of the fuel inside the tank. The time for a sound wave to travel back and forth to the fuel surface is used to calculate the distance, and is interpreted by the transmitter electronics as the fuel level. The transmitter electronics module contains all the power, computation, and signal processing circuits and an ultrasonic transducer. The transducer consists of one or more piezoelectric crystals for the transmission and reception of the sound waves. When electrical energy is applied to the piezoelectric crystals, they move to produce a sound signal. When the sound signal is reflected back from the surface of the fuel, the movement of the reflected sound wave generates an electrical signal; this is detected as the return pulse. The transit time, which is measured as the time between the transmitted and return signals, is then used to infer the level of the fuel inside the tank. Ultrasonic level instruments are designed to output a signal corresponding either to the "fillage" or "fullness" of the vessel or the "Ullage" or "emptiness" inside the fuel tank.

$$Fillage = Total\ height - Ullage$$

The sound waves time of flight is the direct function of how much empty space exists between the liquid surface and top of the vessel i.e. "ullage". Fillage becomes the natural measurement and ullage a derived parameter.

$$Ullage = Total\ height - Fillage$$

Most Ultrasonic level instruments are designed to measure Ullage. In this measurement mode, the time of travel of the sound wave is then proportional to how much empty space exists between the liquid surface and the top of the vessel.

GYROSCOPIC INSTRUMENTS

SOURCES OF POWER FOR GYROSCOPIC INSTRUMENTS

Gyroscopic instruments are essential instruments used on all aircraft. They provide the pilot with critical attitude and directional information and are particularly important while flying under IFR. The sources of power for these instruments can vary. The main requirement is to spin the gyroscopes at a high rate of speed. Originally, gyroscopic instruments were strictly vacuum driven.

A vacuum source pulled air across the gyro inside the instruments to make the gyros spin. Later, electricity was added as a source of power. The turning armature of an electric motor doubles as the gyro rotor. In some aircraft pressure, rather than vacuum, is used to induce the gyro to spin. Various systems and powering configuration have been developed to provide reliable operation of the gyroscopic instruments.

PRESSURE-DRIVEN GYROSCOPIC INSTRUMENT SYSTEMS

Gyroscopic instruments are finely balanced devices with jeweled bearings that must be kept clean to perform properly. When early vacuum systems were developed, only oil lubricated pumps were available. Even with the use of air-oil separators, the pressure outputs of these pumps contain traces of oil and dirt. As a result, it was preferred to draw clean air through the gyro instruments with a vacuum system, rather than using pump output pressure that presented the risk of contamination. The development of self-lubricated dry pumps greatly reduced pressure output contaminates. This made pressure gyro systems possible.

At high altitudes, the use of pressure-driven gyros is more efficient. Pressure systems are similar to vacuum systems and make use of the same components, but they are designed for pressure instead of vacuum. Thus, a pressure regulator is used instead of a suction relief valve. Filters are still extremely important to prevent damage to the gyros. Normally, air is filtered at the inlet and outlet of the pump in a pressure gyro system.

ELECTRICALLY-DRIVEN GYROSCOPIC INSTRUMENT SYSTEMS

A spinning motor armature can act as a gyroscope. This is the basis for electrically driven gyroscopic instruments in which the gyro rotor spin is powered by an electric motor. Electric gyros have the advantage of being powered by battery for a limited time if a generator fails or an engine is lost. Since air is not sent through the gyro to spin the rotor, contamination worries are also reduced. Also, elimination of vacuum pumps, plumbing, and vacuum system components saves weight.

On many small, single engine aircraft, electric turn-and-bank or turn coordinators are combined with vacuum powered attitude and directional gyro instruments as a means for redundancy. The reverse is also possible. By combining both types of instruments in the instrument panel, the pilot has more options. On more complex multi-engine aircraft, reliable, redundant electrical systems make use of all electric powered gyro instruments possible.

It should be noted that electric gyro instruments have some sort of indicator on the face of the dial to show when the instrument is not receiving power. Usually, this is in the form of a red flag or mark of some sort often with the word OFF written on it (or a similar word).

PRINCIPLES OF GYROSCOPIC INSTRUMENTS

MECHANICAL GYROS

Three of the most common flight instruments, the attitude indicator, heading indicator, and turn needle of the turn and-bank indicator, are controlled by gyroscopes. To understand how these instruments operate, knowledge of gyroscopic principles and instrument power systems is required. A mechanical gyroscope, or gyro, is comprised of a wheel or rotor with its mass concentrated around its perimeter. The rotor has bearings to enable it to spin at high speeds. (***Figure 8-76A***) Different mounting configurations are available for the rotor and axle, which allow the rotor assembly to rotate about one or two axes perpendicular to its axis of spin. To suspend the rotor for rotation, the axle is first mounted in a supporting ring. (***Figure 8-76B***)

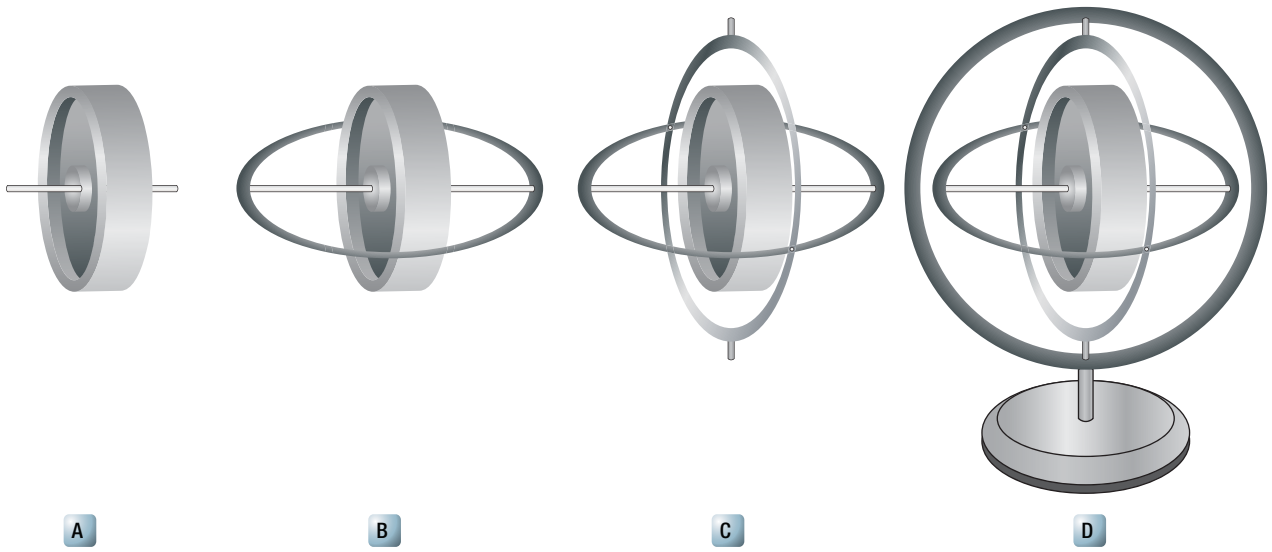


Figure 8-76. Gyroscopes.

If brackets are attached 90 degrees around the supporting ring from where the spin axle attached, the supporting ring and rotor can both move freely 360 degrees. When in this configuration, the gyro is said to be a captive gyro. It can rotate about only one axis that is perpendicular to the axis of spin. (***Figure 8-76C***)

The supporting ring can also be mounted inside an outer ring. The bearing points are the same as the bracket just described, 90 degrees around the supporting ring from where the spin axle attached. Attachment of a bracket to this outer ring allows the rotor to rotate in two planes while spinning. Both of these are perpendicular to the spin axis of the rotor. The plane that the rotor spins in due to its rotation about its axle is not counted as a plane of rotation.

A gyroscope with this configuration, two rings plus the mounting bracket, is said to be a free gyro because it is free to rotate about two axes that are both perpendicular to the rotors spin axis. (***Figure 8-76D***) As a result, the supporting ring with a spinning gyro mounted inside is free to turn 360 degrees inside the outer ring.

Unless the rotor of a gyro is spinning, it has no unusual properties; it is simply a wheel universally mounted. When the rotor is rotated at a high speed, the gyro exhibits a couple of unique characteristics. The first is called gyroscopic rigidity, or rigidity in space. This means that the rotor of a free gyro always points in the same direction no matter which way the base of the gyro is positioned. (***Figure 8-77***)

Gyroscopic rigidity depends upon several design factors

1. Weight for a given size - a heavy mass is more resistant to disturbing forces than a light mass.
2. Angular velocity - the higher the rotational speed, the greater the rigidity or resistance is to deflection
3. Radius at which the weight is concentrated - maximum effect is obtained from a mass when its principal weight is concentrated near the rim, rotating at high speed.
4. Bearing friction - any friction applies a deflecting force to a gyro. Minimum bearing friction keeps deflecting forces at a minimum.

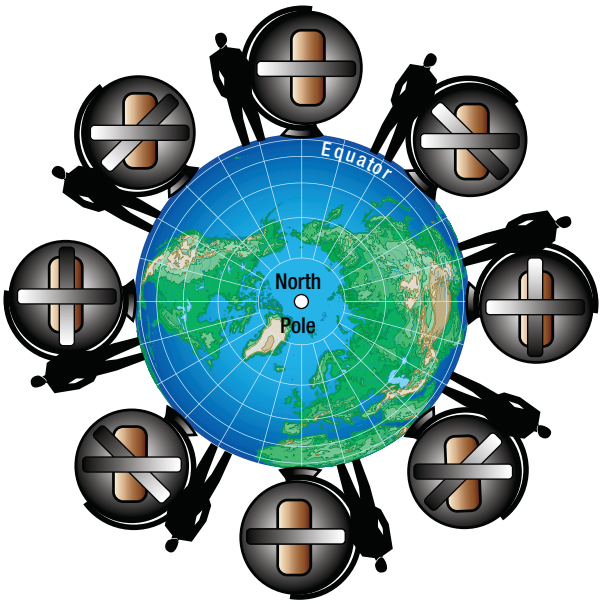


Figure 8-77. Once spinning, a free gyro rotor stays oriented in the same position in space despite the position or location of its base.

This characteristic of gyros to remain rigid in space is exploited in the attitude indicating instruments and the directional indicators that use gyros.

Precession is a second important characteristic of gyroscopes. By applying a force to the horizontal axis of the gyro, a unique phenomenon occurs. The applied force is resisted. Instead of responding to the force by moving about the horizontal axis, the gyro moves in response about its vertical axis. Stated another way, an applied force to the axis of the spinning gyro does not cause the axis to tilt. Rather, the gyro responds as though the force was applied 90 degrees around in the direction of rotation of the gyro rotor. The gyro rotates rather than tilts. (***Figure 8-78***) This predictable controlled precession of a gyroscope is utilized in a turn and bank instrument.

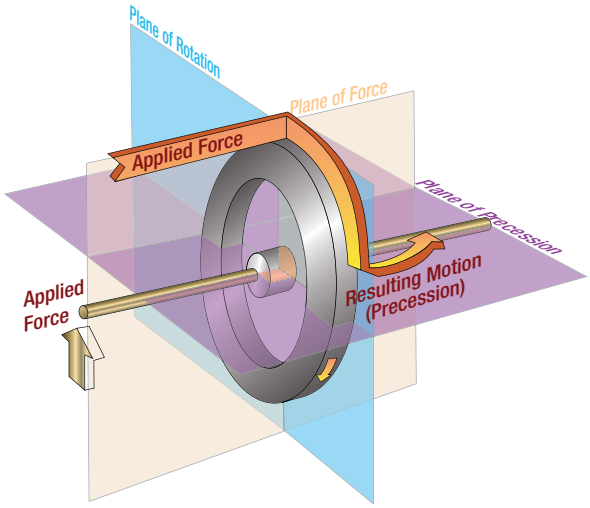


Figure 8-78. When a force is applied to a spinning gyroscope, it reacts as though the force came from 90˚ further around the rotor in the direction it is spinning. The plane of the applied force, the plane of the rotation, and the plane in which the gyro responds (known as the plane of precession), are all perpendicular to each other.

SOLID STATE GYROS AND RELATED SYSTEMS

Improved attitude and direction information is always a goal in aviation. Modern aircraft make use of highly accurate solid-state attitude and directional devices with no moving parts. This results in very high reliability and low maintenance.

RING LASER GYROS (RLG)

The ring laser gyro (RLG) is widely used in commercial aviation. The basis for RLG operation is that it takes time for light to travel around a stationary, non-rotating circular path. Light takes longer to complete the journey if the path is rotating in the same direction as the light is traveling. And, it takes less time for the light to complete the loop if the path is rotating in the direction opposite to that of the light. Essentially, the path is made longer or shorter by the rotation of the path. (***Figure 8-79***) This is known as the Sagnac effect. A laser is light amplification by stimulated emission of radiation. A laser operates by exciting atoms in plasma to release electromagnetic energy, or photons.

A ring laser gyro produces laser beams that travel in opposite directions around a closed triangular cavity. The wavelength of the light traveling around the loop is fixed. As the loop rotates, the path the lasers must travel lengthens or shortens. The light wavelengths compress or expand to complete travel around the loop as the loop changes its effective length. As the wavelengths change, the frequencies also change.

By examining the difference in the frequencies of the two counter rotating beams of light, the rate at which the path is rotating can be measured. A piezoelectric dithering motor in the center of the unit vibrates to prevent lock-in of the output signal at low rotational speeds. It causes units installed on aircraft to hum when operating. (***Figure 8-80***)

An RLG is remotely mounted so the cavity path rotates around one of the axes of flight. The rate of frequency phase shift detected between the counter rotating lasers is proportional to the rate that the aircraft is moving about that axis. On aircraft, an RLG is installed for each axis of flight. Output can be used in analog instrumentation and autopilot systems. It is also easily made compatible for use by digital display computers and for digital autopilot computers.

RLGs are rugged and have a long service life with little maintenance due to their lack of moving parts. They measure movement about an axis extremely quickly and provide continuous output. They are extremely accurate and generally are considered superior to mechanical gyroscopes.

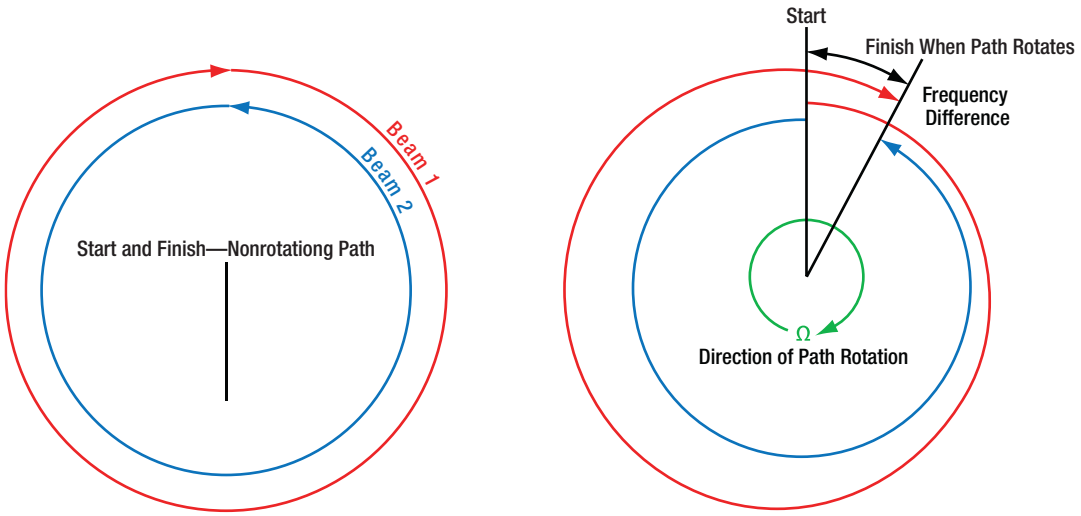


Figure 8-79. Light traveling in opposite directions around a non-rotating path arrives at the end of the loop at the same time (top). When the path rotates, light traveling with the rotation must travel farther to complete one loop. Light traveling against the rotation completes the loop sooner (bottom).

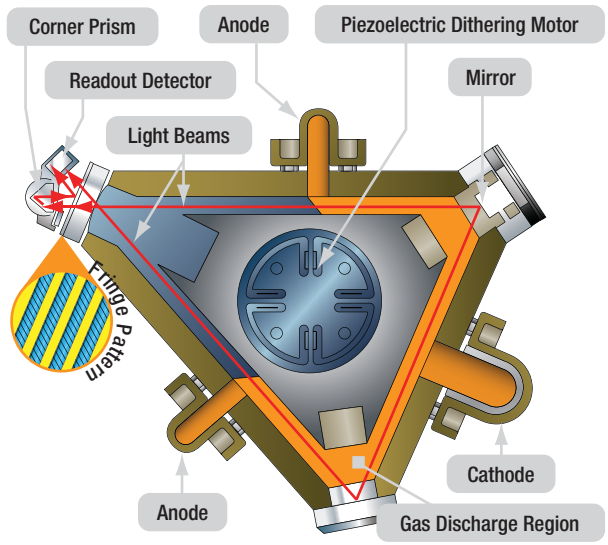


Figure 8-80. The ring laser gyro is rugged, accurate, and free of friction.

MICROELECTROMECHANICAL BASED ATTITUDE AND DIRECTIONAL SYSTEMS (MEMS)

On aircraft, microelectromechanical systems (MEMS) devices save space and weight. Through the use of solid-state MEMS devices, reliability is increased primarily due to the lack of moving parts. The development of MEMS technology for use in aviation instrumentation integrates with the use of air data computers ADCs. This newest improvement in technology is low cost and promises to proliferate through all forms of aviation. MEMS for gyroscopic applications are used in small, general aviation aircraft, as well as larger commercial aircraft. Tiny vibration-based units with resistance and capacitance measuring pick-offs are accurate and reliable and only a few millimeters in length and width. They are normally integrated into a complete micro-electronic solid-state chip designed to yield an output after various conditioning processes are performed. The chips, which are analogous to tiny circuit boards, can be packaged for installation inside a dedicated computer or module that is installed on the aircraft.

While a large mechanical gyroscope spins in a plane, its rigidity in space is used to observe and measure the movement of the aircraft. The basis of operation of many MEMS gyroscopes is the same despite their tiny size. The difference is that a vibrating or oscillating piezoelectric device replaces the spinning, weighted ring of the mechanical gyro. Still, once set in motion, any out of-plane motion is detectable by varying microvoltages or capacitances detected through geometrically arranged pickups.

Since piezoelectric substances have a relationship between movement and electricity, microelectrical stimulation sets a piezoelectric gyro in motion and the tiny voltages produced via the movement in the piezo are extracted. They are then input as the required variables needed to compute attitude or direction information. (***Figure 8-81***)

ATTITUDE HEADING AND REFERENCE SYSTEMS

In many modern aircraft, (AHRS) have taken the place of the gyroscope and other individual instruments. While MEMS devices provide part of the attitude information for the system, GPS, solid state magnetometers, solid state accelerometers, and digital air data signals are all combined in an AHRS to compute and output highly reliable information for display on a cockpit panel. (***Figure 8-82***)



Figure 8-82. Instrumentation displayed within a glass cockpit using an attitude heading and reference system (AHRS) computer.

COMMON GYROSCOPIC INSTRUMENTS

VACUUM-DRIVEN ATTITUDE GYROS

The attitude indicator, or artificial horizon, is one of the most essential flight instruments. It gives the pilot pitch and roll information that is especially important when flying without outside visual references. The attitude indicator operates with a gyroscope rotating in the horizontal plane. Thus, it mimics the actual horizon through its rigidity in space. As the aircraft pitches and rolls in relation to the actual horizon, the gyro gimbals allow the aircraft and instrument housing to pitch and roll around the gyro rotor that remains parallel to the ground.

A horizontal representation of the aeroplane in miniature is fixed to the instrument housing. A painted semisphere simulating the horizon, the sky, and the ground is attached to the gyro gimbals. The sky and ground meet at what is called the horizon bar.

The relationship between the horizon bar and the miniature aeroplane are the same as those of the aircraft and the actual horizon. Graduated scales reference the degrees of pitch and roll. Often, an adjustment knob allows pilots of varying heights to place the horizon bar at an appropriate level. (***Figure 8-83***)

In a typical vacuum driven attitude Gyro system, air is sucked through a filter and then through the attitude indicator in a manner that spins the gyro rotor inside. An erecting mechanism is built into the instrument to assist in keeping the gyro rotor rotating in the intended plane. Precession caused by bearing friction makes this necessary. After air engages the scalloped drive on the rotor, it flows from the instrument to the vacuum pump through four ports. These ports all exhaust the same amount of air when the gyro is rotating in plane.

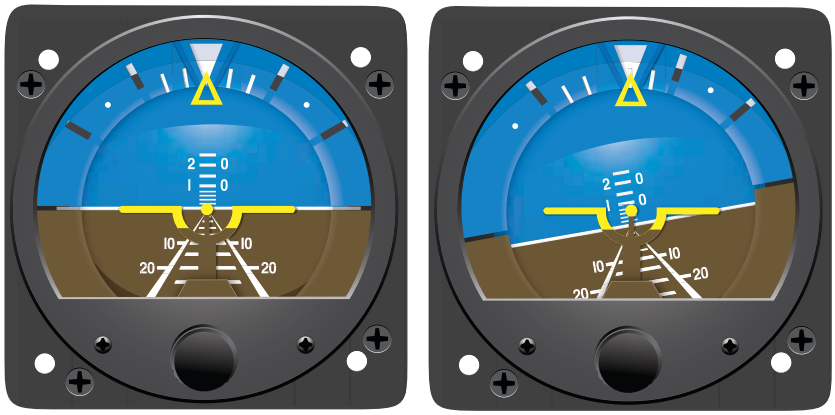


Figure 8-83. A typical vacuum-driven attitude indicator shown with the aircraft in level flight (left) and in a climbing right turn (right).

When the gyro rotates out of plane, air tends to port out of one side more than another. Vanes close to prevent this, causing more air to flow out of the opposite side. The force from this unequal venting of the air re-erects the gyro rotor. (***Figure 8-84***)

Early vacuum-driven attitude indicators were limited in how far the aircraft could pitch or roll before the gyro gimbals contacted stops, causing abrupt precession and tumbling of the gyro. Many of these gyros include a caging device. It is used to erect the rotor to its normal operating position prior to flight or after tumbling. A flag indicates that the gyro must be uncaged before use. More modern gyroscopic instruments are built so they do not tumble, regardless of the angular movement of the aircraft about its axes. In addition to the contamination potential introduced by the air-drive system, other shortcomings exist in the performance of vacuum-driven attitude indicators.

Some are induced by the erection mechanism. The pendulous vanes that move to direct airflow out of the gyro respond not only to forces caused by a deviation from the intended plane of rotation, but centrifugal force experienced during turns also causes the vanes to allow asymmetric porting of the gyro vacuum air. The result is inaccurate display of the aircraft's attitude, especially in skids and steep banked turns. Also, abrupt acceleration and deceleration imposes forces on the gyro rotor. Suspended in its gimbals, it acts similar to an accelerometer, resulting in a false nose-up or nose-down indication. Pilots must learn to recognize these errors and adjust accordingly.

ELECTRIC ATTITUDE INDICATORS

Electric attitude indicators are very similar to vacuum driven gyro indicators. The main difference is in the drive mechanism. Inside the gimbals of an electric gyro, a small squirrel cage electric motor is the rotor. It is typically driven by 115-volt, 400-cycle AC. It turns at approximately 21 000 rpm.

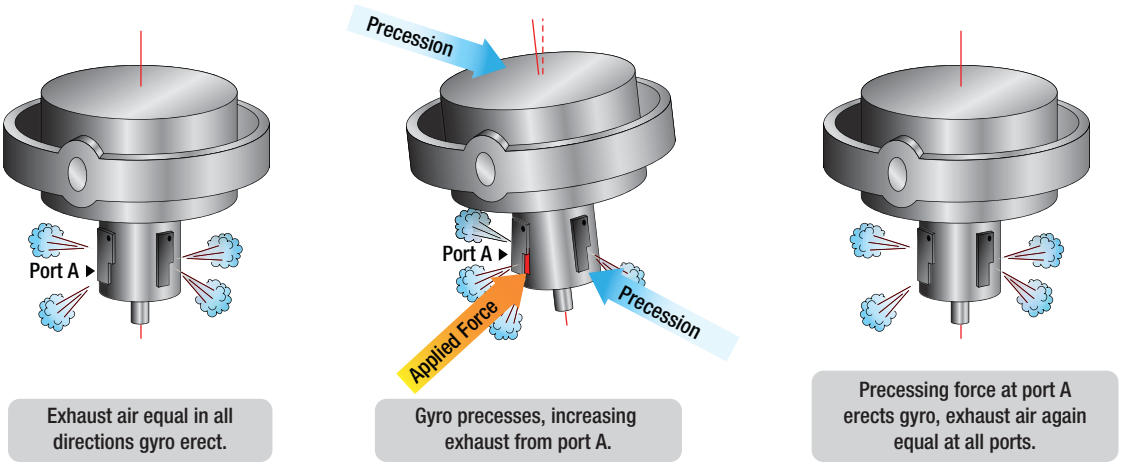


Figure 8-84. The erecting mechanism of a vacuum-driven attitude indicator.

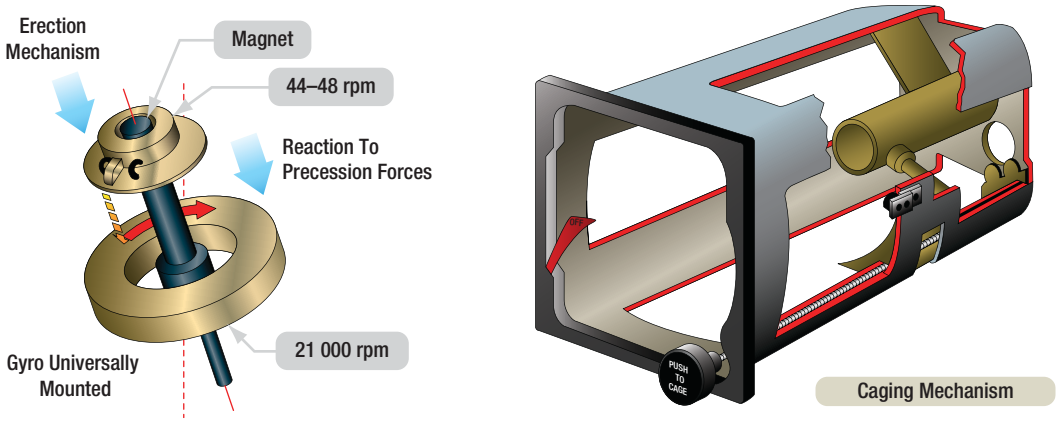


Figure 8-85. Erecting and caging mechanisms of an electric attitude indicator.

Other characteristics of the vacuum-driven gyro are shared by the electric gyro. The rotor is still oriented in the horizontal plane. The free gyro gimbals allow the aircraft and instrument case to rotate around the gyro rotor that remains rigid in space. A miniature aeroplane fixed to the instrument case indicates the aircraft's attitude against the moving horizon bar behind it. Electric attitude indicators address some of the shortcomings of vacuum-driven attitude indicators.

Since there is no air flowing through an electric attitude indicator, air filters, regulators, plumbing lines and vacuum pump(s) are not needed. Contamination from dirt in the air is not an issue, resulting in the potential for longer bearing life and less precession. Erection mechanism ports are not employed, so pendulous vanes responsive to centrifugal forces are eliminated.

It is still possible that the gyro may experience precession and need to be erected. This is done with magnets rather than vent ports. A magnet attached to the top of the gyro shaft spins at approximately 21 000 rpm. Around this magnet, but not attached to it, is a sleeve that is rotated by magnetic attraction at approximately 44 to 48 rpm. Steel balls are free to move around the sleeve.

If the pull of gravity is not aligned with the axis of the gyro, the balls fall to the low side. The resulting precession re-aligns the axis of rotation vertically. Typically, electric attitude indicator gyros can be caged manually by a lever and cam mechanism to provide rapid erection. When the instrument is not getting sufficient power for normal operation, an off flag appears in the upper right hand face of the instrument. (***Figure 8-85***)

GYROSCOPIC DIRECTION INDICATOR OR DIRECTIONAL GYRO (DG)

The gyroscopic direction indicator or directional gyro (DG) is often the primary instrument for direction. Because a magnetic compass fluctuates so much, a gyro aligned with the magnetic compass gives a much more stable heading indication. Gyroscopic direction indicators are located at the center base of the instrument panel basic T. A vacuum-powered DG is common on many light aircraft. Its basis for operation is the gyro's rigidity in space. The gyro rotor spins in the vertical plane and stays aligned with the direction to which it is set. The aircraft and instrument case moves around the rigid gyro. This causes a vertical compass card that is geared to the rotor gimbal to move. It is calibrated in degrees, usually with every 30 degrees labeled. The nose of a small, fixed aeroplane on the instrument glass indicates the aircraft's heading. (***Figure 8-86***)

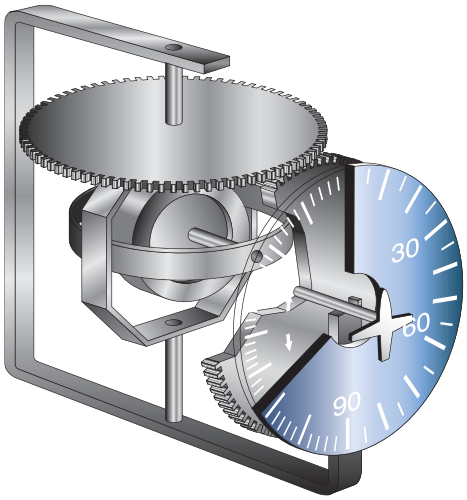


Figure 8-86. A typical vacuum-powered gyroscopic direction indicator, also known as a directional gyro.

Vacuum-driven direction indicators have many of the same basic gyroscopic instrument issues as attitude indicators. Built-in compensation for precession varies and a caging device is usually found. Periodic manual realignment with the magnetic compass by the pilot is required during flight.

TURN COORDINATORS

Many aircraft make use of a turn coordinator. The rotor of the gyro in a turn coordinator is canted upwards 30°. As such, it responds not only to movement about the vertical axis, but also to roll movements about the longitudinal axis. This is useful because it is necessary to roll an aircraft to turn it about the vertical axis. Instrument indication of roll, therefore, is the earliest possible warning of a departure from straight-and-level flight. Typically, the face of the turn coordinator has a small aeroplane symbol. The wing tips of the aeroplane provide the indication of level flight and the rate at which the aircraft is turning (***Figure 8-87***)

TURN-AND-SLIP INDICATOR

The turn-and-slip indicator may also be referred to as the turn-and-bank indicator, or needle-and-ball indicator. Regardless, it shows the correct execution of a turn while banking the aircraft and indicates movement about the vertical axis of the aircraft (yaw). Most turn-and-slip indicators are located below the airspeed indicator of the instrument panel basic T, just to the left of the direction indicator. The turn-and-slip indicator is actually two separate devices built into the same instrument housing: a turn indicator pointer and slip indicator ball. The turn pointer is operated by a gyro that can be driven by a vacuum, air pressure, or by electricity. The ball is a completely independent device. It is a round agate, or steel ball, in a glass tube filled with dampening fluid. It moves in response to gravity and centrifugal force experienced in a turn. Turn indicators vary. They all indicate the rate at which the aircraft is turning.

Three degrees of turn per second cause an aircraft to turn 360 degrees in 2 minutes. This is considered a standard turn. This rate can be indicated with marks right and left of the pointer, which normally rests in the vertical position. Sometimes, no marks are present and the width of the pointer is used as the calibration device. In this case, one pointer width deflection from vertical is equal to the 3° per second standard 2 minute turn rate. Faster aircraft tend to turn more slowly and have graduations or labels that indicate 4 minute turns. In other words, a pointer's width or alignment with a graduation mark on this instrument indicates that the aircraft is turning a 1-1/2° per second and completes a 360 degree turn in 4 minutes. It is customary to placard the instrument face with words indicating whether it is a 2 or 4 minute turn indicator. (***Figure 8-88***)

The turn pointer indicates the rate at which an aircraft is turning about its vertical axis. It does so by using the precession of a gyro to tilt a pointer. The gyro spins in a vertical plane aligned with the longitudinal axis of the aircraft. When the aircraft rotates about its vertical axis during a turn, the force experienced by the spinning gyro is exerted about the vertical axis. Due to precession, the reaction of the gyro rotor is 90 degrees further around the gyro in the direction of spin. This



Figure 8-87. A turn coordinator senses and indicates the rate of both roll and yaw.



Figure 8-88. Turn-and-slip indicator.

means the reaction to the force around the vertical axis is movement around the longitudinal axis of the aircraft. This causes the top of the rotor to tilt to the left or right. The pointer is attached with linkage that makes the pointer deflect in the opposite direction, which matches the direction of turn. So, the aircraft's turn around the vertical axis is indicated around the longitudinal axis on the gauge. This is intuitive to the pilot when regarding the instrument, since the pointer indicates in the same direction as the turn. (***Figure 8-89***)

The slip indicator (ball) part of the instrument is an inclinometer. The ball responds only to gravity during coordinated straight-and-level flight. Thus, it rests in the lowest part of the curved glass between the reference wires. When a turn is initiated and the aircraft is banked, both gravity and the centrifugal force of the turn act upon the ball. If the turn is coordinated, the ball remains in place. Should a skidding turn exist, the centrifugal force exceeds the force of gravity on the ball and it moves in the direction of the outside of the turn.

During a slipping turn, there is more bank than needed, and gravity is greater than the centrifugal force acting on the ball. The ball moves in the curved glass toward the inside of the turn. As mentioned previously, often power for the turn-and-slip indicator gyro is electrical if the attitude and direction indicators are vacuum powered. This allows limited operation off battery power should the vacuum system and the electric generator fail. The directional and attitude information from the turn-and-slip indicator, combined within formation from the pitot-static instruments, allow continued safe emergency operation of the aircraft.

Electrically powered turn-and-slip indicators are usually DC powered. Vacuum-powered turn-and-slip indicators are usually run on less vacuum (approximately 2" Hg) than fully gimbaled attitude and direction indicators. Regardless, proper vacuum must be maintained for accurate turn rate information to be displayed.

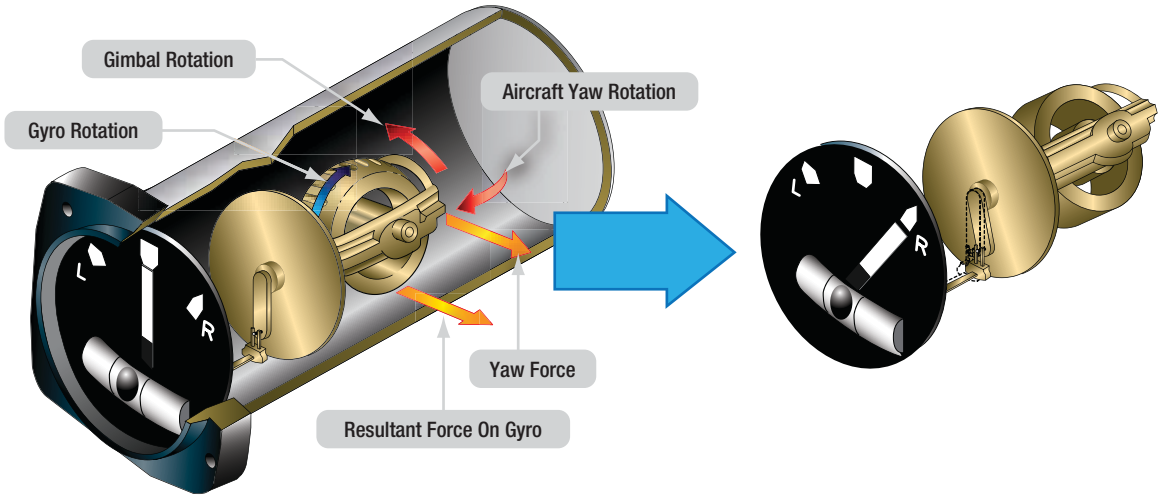


Figure 8-89. The turn-and-slip indicator's gyro reaction to the turning force in a right hand turn. The yaw force results in a force on the gyro 90° around the rotor in the direction it is turning due to precession. This causes the top of the rotor to tilt to the left. Through connecting linkage, the pointer tilts to the right.

TERRAIN AWARENESS AND WARNING SYSTEM (GPWS & EGPWS)

Terrain Awareness and Warning System (TAWS) help alert pilots when aircraft are unintentionally flown near to ground, the water, a mountain, or to an obstacle. The TAWS system works by using digital data of height and other instrumental data, in order to predict if a future position of the aircraft intersects with the ground.

The flight crew is thus informed in advance with aural and visual warnings. The actual systems in current use are known as Ground Proximity Warning System (GPWS) and Enhanced GPWS (EGPWS).

GROUND PROXIMITY WARNING SYSTEM (GPWS)

The purpose of GPWS is to alert the flight crew about the existence of an unsafe condition when the aircraft is near the terrain (terrain proximity).

MODE ANNUNCIATIONS

This system is based on 7 Modes. GPWS modes are annunciated to the flight crew by means of aural messages and visual indications. The enhanced features of GPWS alerts the flight crew to early descent on approach near airports and about terrain ahead of the airplane. It is called EGPWS.

There are 7 Modes of operations.

- Mode 1: Large or Excessive Descent rate.
- Mode 2: Large or Excessive Closure rate when approaching terrain that is rising.
- Mode 3: Descent after take-off or GA.
- Mode 4: Insufficient Terrain Clearance when aircraft in not in landing configuration.
- Mode 5: Descent below glide scope.
- Mode 6: Aural Call outs when descending through selected radar altitude.
- Mode 7: Warning for wind shear.

SYSTEM DESCRIPTION

The main components are:

- GPWC
- GPWC Warning Lights
- Control Panel

GPWC

GPWC compares the aircraft flight profile, flap and gear position and terrain clearance to determine an alert situation. It receives information from different systems e.g. TAS, barometric altitude, barometric altitude rate, radio altitude, glide scope deviation, present position, heading data, etc. This computer compares the input with the programmed limits of each mode and if it finds that aircraft have entered in any dangerous condition then it generates appropriate warnings to the pilot.

WARNING LIGHTS

Two warning lights are provided to give visual indication of ground proximity warnings.

GROUND PROXIMITY PULL UP LIGHT (RED)

This light indicates mode 1 & mode 2 PULL UP warning condition. It is typically located on the captain's instrument panel. In glass cockpit aircraft this warning is indicated on the primary flight display instead of a dedicated light.

FLAP, GEAR LIGHT, SWITCH GROUND PROXIMITY, GLIDE SCOPE, INHIBIT LIGHT SWITCH

This AMBER light switch is used to indicate advisory or alert mode 1 through 5. When the switch is pressed, the mode 5 (below G/S) aural and visual indications are inhibited. This light switch is typically located on the captain's instrument panel or on the landing gear panel (B-777).

GROUND PROXIMITY FLAP OVERRIDE LIGHT, SWITCH AND GEAR OVERRIDE SWITCH

This light switch is used to simulate flap down (25° or more) or landing gear down position respectively. It is alternate action P/B switches. The white light illuminates when override function is activated. Normally these light switches are located on the First's officer's instrument panel.

WARNING LIGHTS GPWS CONTROL PANEL

This control panel provides

- Flap/Gear Inhibit LIGHT/SW
- SELF TEST FACILITY
- SYSTEM INOP LIGHT

B-777 GPWS CONTROL PANEL INOP LIGHT

In some installations, an amber INOP light is illuminated when the computer or input signal failure is detected or a self-test is being performed. Self-test can be initiated on ground as well as in flight.

Mode 1 - Large or Excessive Descent Rate

Mode 1 provides alert and warning for large descent rate into terrain. This mode is independent of landing gear and flap positions. Mode 1 alert and warning occurs between 30 ft and 2 450 ft RA. The rate of descent determines at which alert is initiated. When the barometric altitude rate exceeds a threshold value the alert is generated by GPWC.
(***Figure 8-90 and Figure 8-91***)

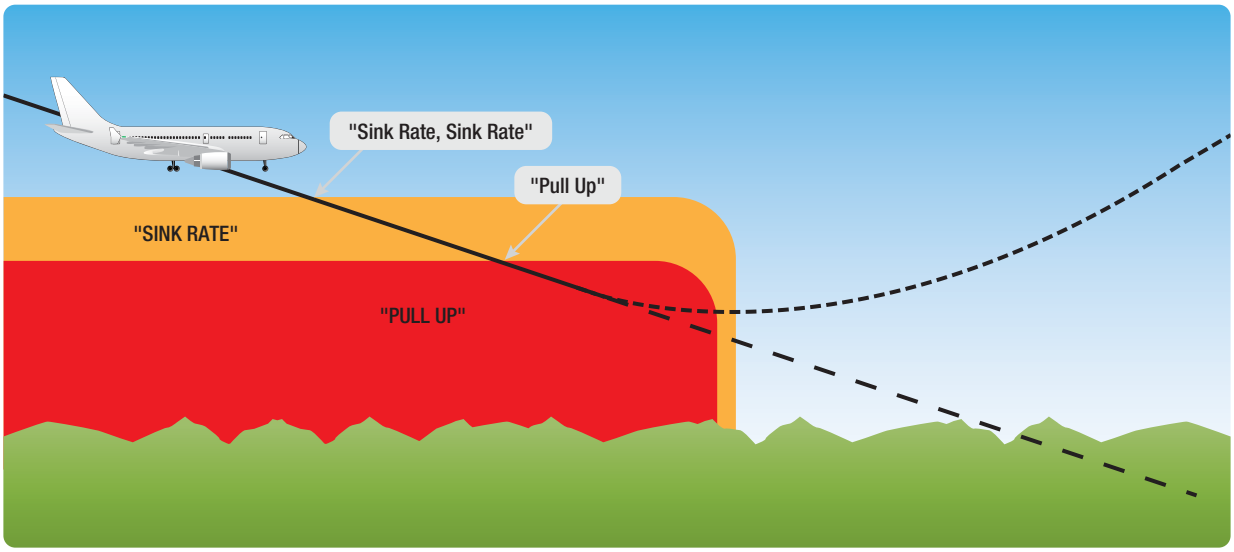


Figure 8-90. Mode 1 - Excessive Decent Rate.

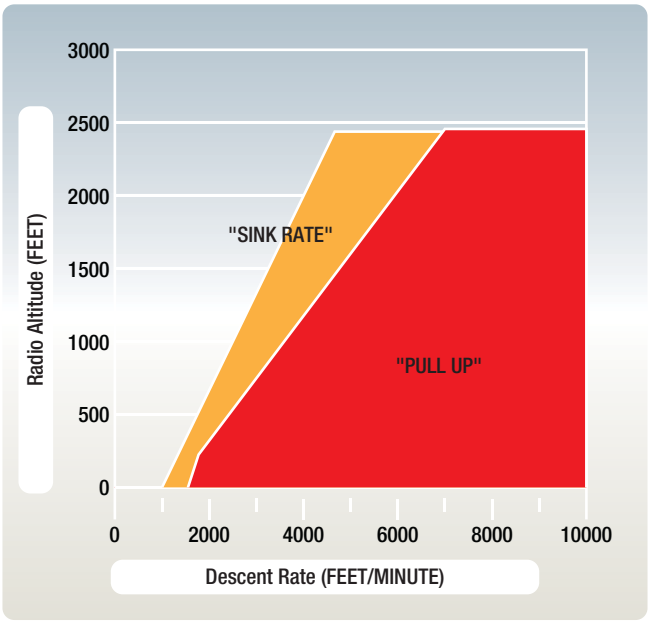


Figure 8-91. Mode 1 - Excessive Descent Rate Chart for Turbofan (Jet) aircraft.

Mode 2 - Excessive Closure Rate

Mode 2 provides a warning based system for too large closure rate with respect to a rising terrain. (***Figure 8-92 and Figure 8-93***)

Mode 2 has two sub modes:

- Mode 2A is active when FLAPS up (less than 15 units). This mode is based on RA.
- Mode 2B is active when FLAPS are down (more than 15 units). This mode is based on RA.

Mode 2B - Excessive Closure Rate

Mode 2B gives annunciations for a large closure rate when the flaps are down (more than15 units).

Mode 3- Descent After Take Off

Mode 3 provides alerts for a large loss of altitude, after take-off or during a missed approach, with the landing gear up and flaps are less than 25 units. (***Figure 8-94***) When the aircraft descends a large predetermined amount of altitude, "GROUND PROXIMITY" light comes on and the aural message DON'T SINK is repeated. The annunciations end when a positive rate of climb occurs. (***Figure 8-95***)

Mode 4

Insufficient Terrain Clearance when aircraft is not in the Landing configuration.

- Mode 4 provides alerts for terrain clearance that is not safe when the aircraft is not in the landing configuration. Mode 4 alerts between 30 ft and 1 000ft. The GPWC uses this data to detect mode 4 alert: 1. RA 2. CAS 3. Flap and gear position Mode 4 is in two parts: 4A and4B. Insufficient Terrain Clearance when aircraft in not in Landing configuration.

Mode 4A is active during cruise and approach with the landing gear up. (***Figure 8-96***)

- When the aircraft has RA below 500 feet at an airspeed less than 190 knots, the mode 4A aural message is the repeated "too low gear "

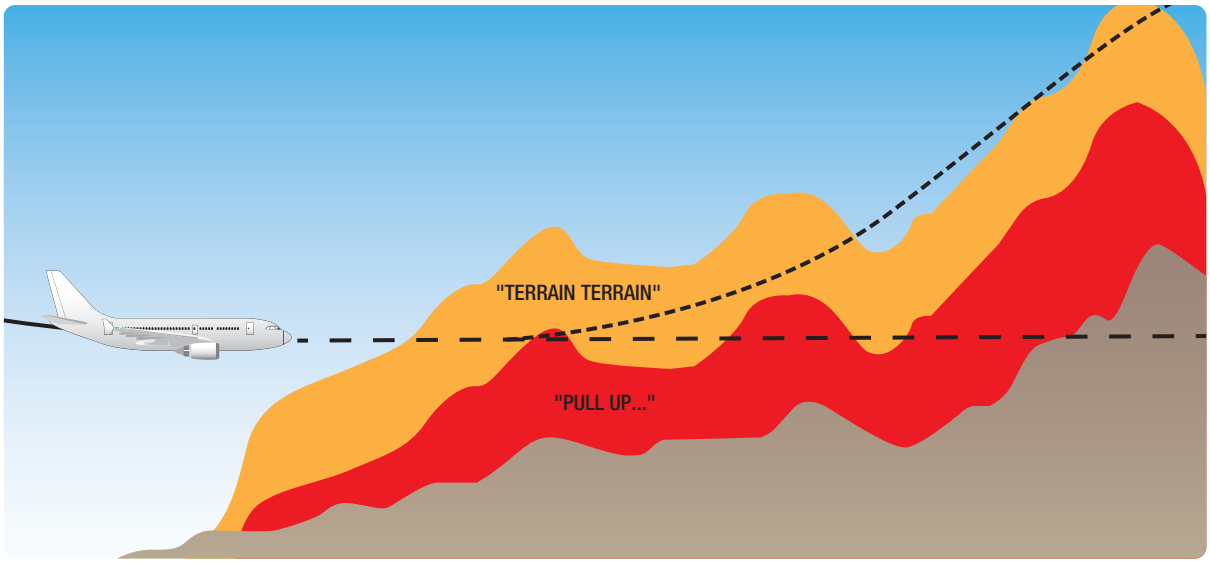


Figure 8-92. Mode 2 - Excessive Closure to Terrain.

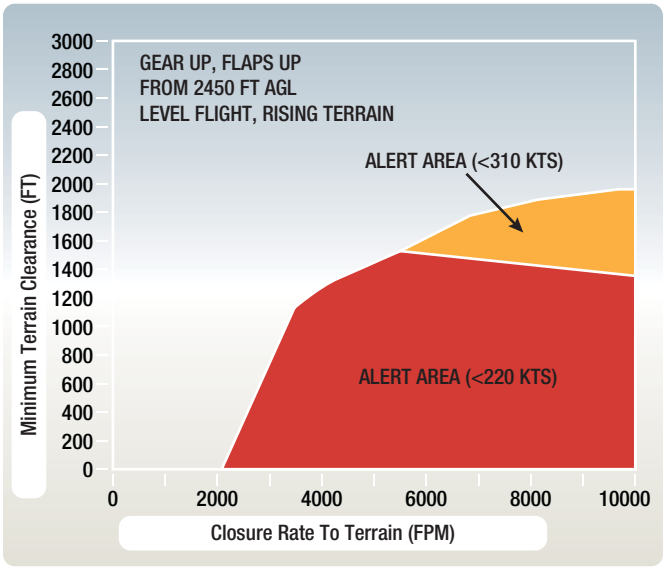


Figure 8-93. Mode 2 - Alert Variation Depending upon Aircraft Speed (Knots).

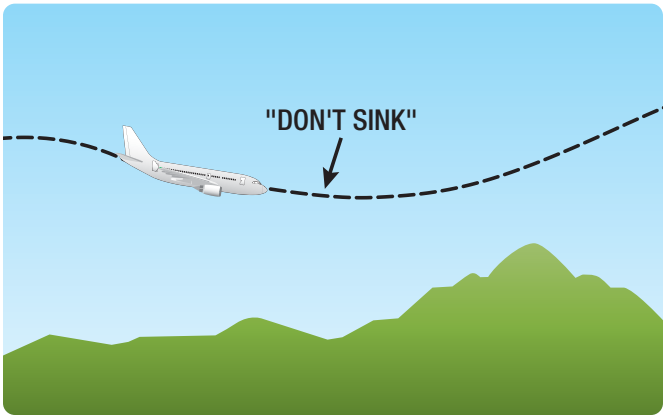


Figure 8-94. Mode 3 - Descent After Takeoff.

- When airspeeds is increases and it becomes more than 190 knots, the aural message is the repeated "too low terrain"
- The GND PROX light comes ON for either of these aural alerts

Mode 4B is active in cruise and approach, but with the landing gear down and flaps are less than 15 units. When the aircraft goes below 245 feet RA altitude at an airspeed less than 159 knots, the Mode 4B aural message is the repeated 'too low flaps. (***Figure 8-97***)

Mode 5

Descent Below Glideslope - Descent Below GS

This Mode 5 provides alerts when the aircraft is too far below the glide slope beam on the ILS course approach with the landing gear down. Mode 5 does not operate during back course approaches. This mode operates between 30 to 1 000ft. Mode 5 has two annunciation levels. Low level and normal alert.

When the aircraft descends below 1 000 feet radio altitude and the glide slope deviation is more than 1.3 dots. The aural message GLIDE SLOPE is repeated at half the sound intensity level of the other GPWS modes. GND PROX light/SW light comes ON.

When the aircraft descends below 300 feet radio altitude and the glide slope deviation is more than 2 dots:

- The aural message GLIDE SLOPE is repeated at the normal sound level.

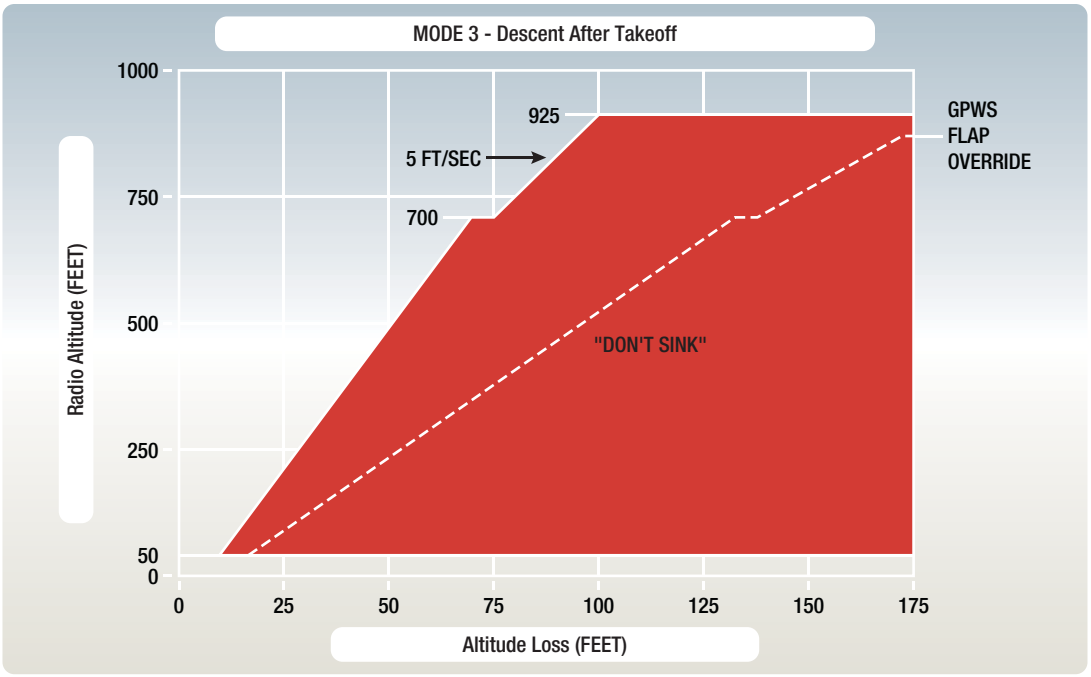


Figure 8-95. Mode 3 - Alert Chart for Loss of Altitude immediately after Take-off.

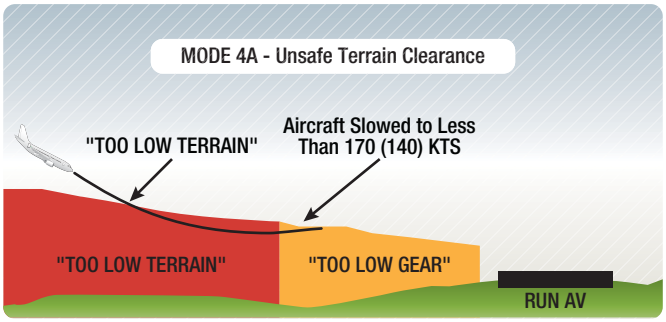


Figure 8-96. Mode 4A - Insufficient Terrain Clearance.

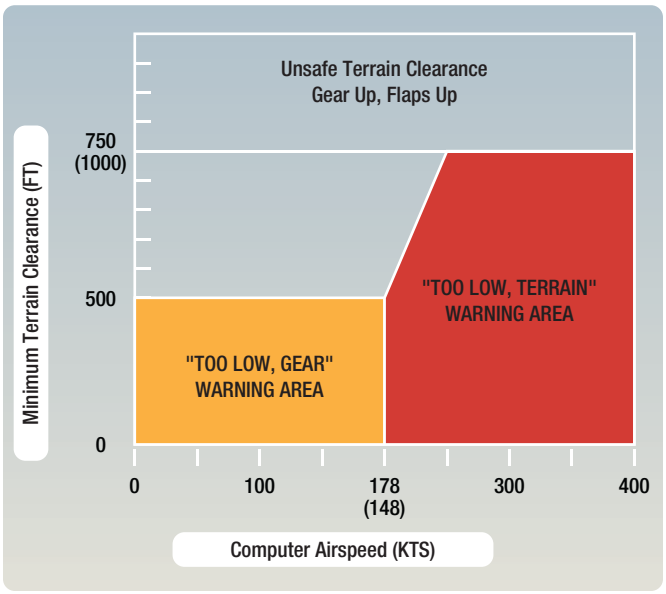


Figure 8-97. Mode 4A - Alert Chart and Aural Messages.

- The glide slope message is repeated more quickly as the terrain clearance decreases and/or the glide slope deviation increases.
- By pushing GND PROX light/SW, both visual and aural alert can be canceled.

Mode 6

Aural Call outs when descending through selected RA.

An optional GPWC feature is Mode 6 radio altitude aural call outs. Mode 6 produces alerts and/or call outs for altitudes below 1 000 feet radio altitude with the landing gear down. Mode 6 only produces call out this mode does not give visual displays. A program pin selection determines the call out configuration. Descent below selected DH, GPWC uses this data to determine Mode 6 alerts. (***Figure 8-98***)

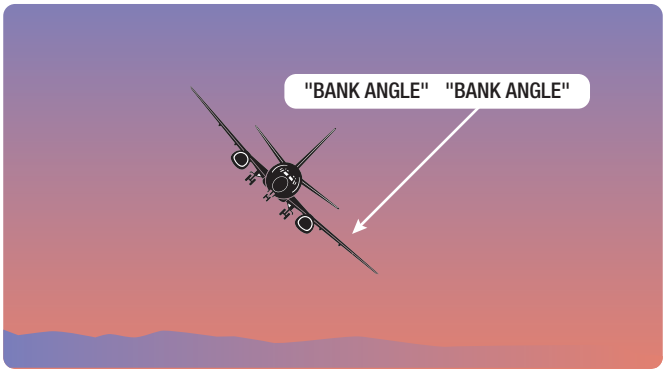


Figure 8-98. Mode 6 - Bank Angle Call-out.

- Radar Altitudes
- Gear Position
- Program Pin Selection

If airplane rolls greater than some specific bank angle (35 degrees in B-777) then GPWC generate call-out of "Bank Angle".

Mode 7

Wind Shear

An optional GPWS Mode 7 is wind shear. Wind shear can happen anywhere in the atmosphere. It is a bad weather condition which consists of large volumes of air which move quickly in different (usually opposite) directions. One type of wind shear which is most dangerous to aircraft is the microburst, which has a column of downward moving air.

Micro bursts are most dangerous below 500 feet, where pilots have little time and airspace to recover. Mode 7 produces warnings for flight into a wind shear condition during takeoff or final approach (below 1 500 feet radio altitude). The initial effect of microburst when an aircraft enters is in increase in airspeed and altitude. As the aircraft moves continuously in a microburst then the effect quickly changes to loss in airspeed and altitude. The GPWC examines this sudden change in airplane condition (e.g. airspeed, altitude from ADC) and finally generates a WINDSHEAR warning.

A wind shear warning provides these annunciations, a siren sounds and then there is an aural message:

- "WIND SHEAR, WIND SHEAR, WIND SHEAR"
- A RED WIND SHEAR warning message shows on the Primary Flight Display.
- RED light comes ON.

FAULT ANNUNCIATION

Ground Proximity Warning System faults are annunciated on the lower EICAS display unit by the calling status page or by the maintenance message. The same message is available for display during the self-test.

- FAULT ANNUNCIATION
- MAINTENANCE MESSAGE
- The maintenance message GND PROX BITE on the maintenance page can be called up by pressing the ECS/MSG (Environmental Control System – Maintenance Message) push button switch on the EICAS maintenance panel.

- EGPWS is modified version of GPWS. It contains all the modes with some additional features.
- In this system, GPWC has world-wide terrain database.
- EGPWS has two additional features.
  1. Terrain Awareness (TA)
  2. Terrain Clearance Floor (TCF)

EGPWS

With the use of accurate GPS and FMS information, the EGPWS locates the position of the aircraft and then present a graphical colored view of the aircraft position to advise the flight crew of any potential conflict with the terrain or obstacle as most accidents occur near airports; all the data related with the airports are present in the database of EGPWS.

This additional EGPWS contains worldwide terrain data base. This additional feature compares airplane position, track, speed, heading with the database to find if there is an alert condition. This function is called TA (Terrain Awareness). The GPWC has a world-wide terrain data base to which the aircraft position is compared. If GPWC finds any terrain threat then warnings are generated The GPWC makes a colored map of the terrain forward of the aircraft on ND or WXR display or dedicated display. The map has different colors to show terrain altitude relative to the airplane altitude. (***Figure 8-99***)

TERRAIN CLEARANCE FLOOR (TCF)

TCF function alerts the pilot when airplane descends too low on approach. TCF uses airplane position and runway database to determine if an alert condition

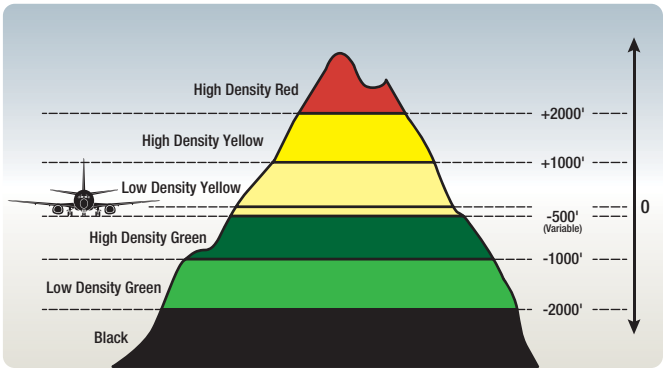


Figure 8-99. Terrain Alerting & Display. Each specific color and intensity represents terrain (and obstacles) below, at, or above the aircraft's altitude based on the aircraft's position with respect to the terrain in the database.

exists. GPWC has a data base which contains location of the entire hard surface runway in the world that is 3 500 ft to more in length. TCF makes a terrain clearance envelope around each airport. The altitude of envelop increases as the distance from the airport increases.

GPWC compares the airplane position and radio altitude with the TCF envelope data. If aircraft descend below this envelope then GPWC generates an alert for the pilot. "Too low Terrain" aural alert is generated once and it is repeated again for each 20% loss in altitude. Ground proximity warning light comes on. This warning remains until the aircraft climbs above the TCF envelop. (***Figure 8-100***)

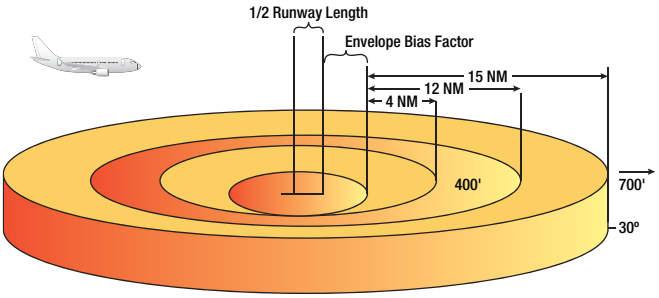


Figure 8-100. Terrain Clearance Floor (TCF) Alert Envelope.

DIRECTION INDICATING INSTRUMENTS

A myriad of techniques and instruments exist to aid the pilot in navigation of the aircraft. An indication of direction is part of this navigation. While the next chapter deals with communication and navigation, this section discusses some of the magnetic direction indicating instruments. Additionally, a common, reliable gyroscopic direction indicator is discussed in the gyroscopic instrument section of this chapter.

DIRECT INDICATING MAGNETIC COMPASS

Having an instrument on board an aircraft that indicates direction can be invaluable to the pilot. In fact, it is a requirement that all certified aircraft have some sort of magnetic direction indicator. The magnetic compass is a direction finding instrument that has been used for navigation for hundreds of years. It is a simple instrument that takes advantage of the earth's magnetic field.

Figure 8-101 shows the earth and the magnetic field that surrounds it. The magnetic north pole is very close to the geographic North Pole of the globe, but they are not the same. An ordinary permanent magnet that is free to do so, aligns itself with the direction of the earth's magnetic field. Upon this principle, an instrument is constructed that the pilot can reference for directional orientation. Permanent magnets are attached under a float that is mounted on a pivot so it is free to rotate in the horizontal plane. As such, the magnets align with the earth's magnetic field.

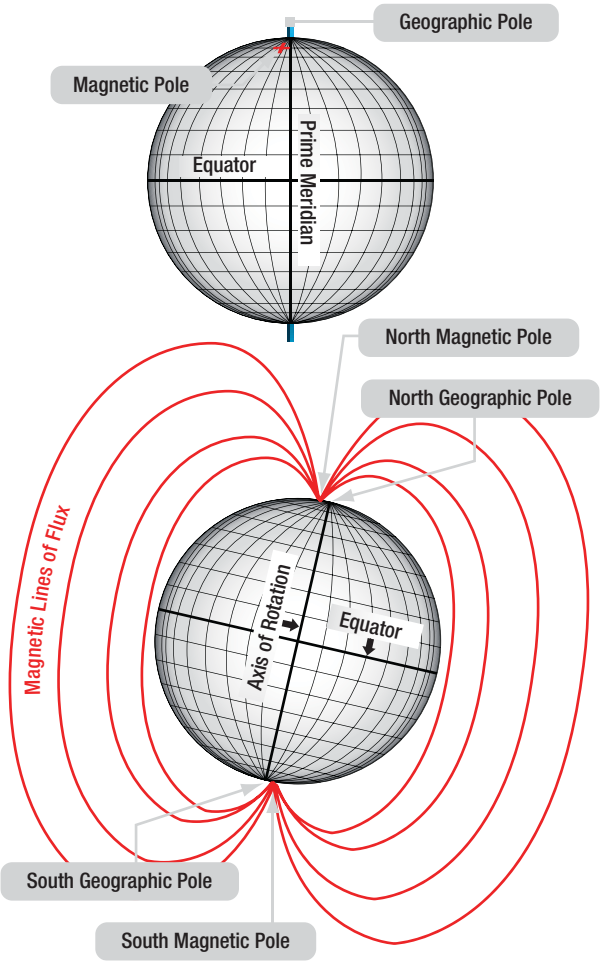


Figure 8-101. The earth and its magnetic field.

A numerical compass card, usually graduated in 5° increments, is constructed around the perimeter of the float. It serves as the instrument dial. The entire assembly is enclosed in a sealed case that is filled with a liquid similar to kerosene. This dampens vibration and oscillation of the moving float assembly and decreases friction. On the front of the case, a glass face allows the numerical compass card to be referenced against a vertical lubber line. The magnetic heading of the aircraft is read by noting the graduation on which the lubber line falls. Thus, direction in any of 360 degrees can be

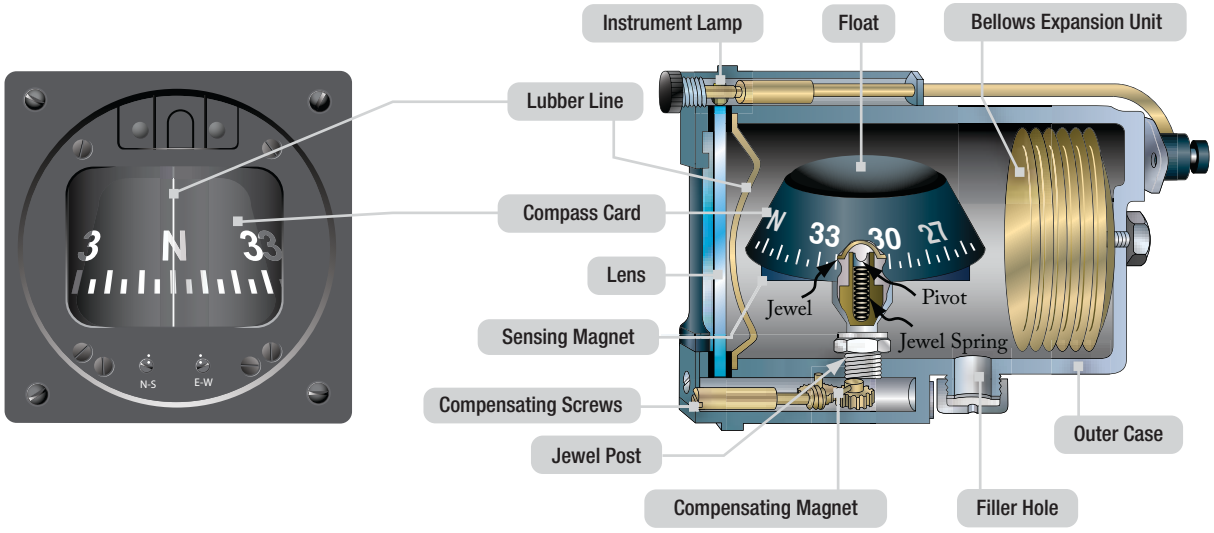


Figure 8-102. The parts of a typical magnetic compass.

read off the dial as the magnetic float compass card assembly holds its alignment with magnetic north, while the aircraft changes direction. The liquid that fills the compass case expands and contracts as altitude changes and temperature fluctuates. A bellows diaphragm expands and contracts to adjust the volume of the space inside the case so it remains full. (***Figure 8-102***)

There are accuracy issues associated with using a magnetic compass. The main magnets of a compass align not only with the earth's magnetic field, they actually align with the composite field made up of all magnetic influences around them, meaning local electromagnetic influence from metallic structures near the compass and operation aircraft's electrical system. This is called magnetic deviation. It causes a magnet's alignment with the earth's magnetic field to be altered. Compensating screws are turned, which move small permanent magnets in the compass case to correct for this magnetic deviation. The two set-screws are on the face of the instrument and are labeled N-S and E-W. They position the small magnets to counterbalance the local magnetic influences acting on the main compass magnets.

The process for knowing how to adjust for deviation is known as swinging the compass. It is described in the instrument maintenance pages near the end of this chapter. Magnetic deviation cannot be overlooked. It should never be more than 10 degrees. Using nonferrous mounting screws and shielding or twisting the wire running to the compass illuminating lamp are additional steps taken to keep deviation to a minimum.

Another compass error is called magnetic variation. It is caused by the difference in location between the earth's magnetic poles and the geographic poles. There are only a few places on the planet where a compass pointing to magnetic north is also pointing to geographic North. A line drawn through these locations is called the Agonic line. At all other points, there is some variation between that which a magnetic compass indicates as north and geographic (true) North. Isogonic lines drawn on aeronautical charts indicate points of equal variation. Depending on the location of the aircraft, airmen must add or subtract degrees from the magnetic indication to obtain true geographic location information. (***Figure 8-103***)

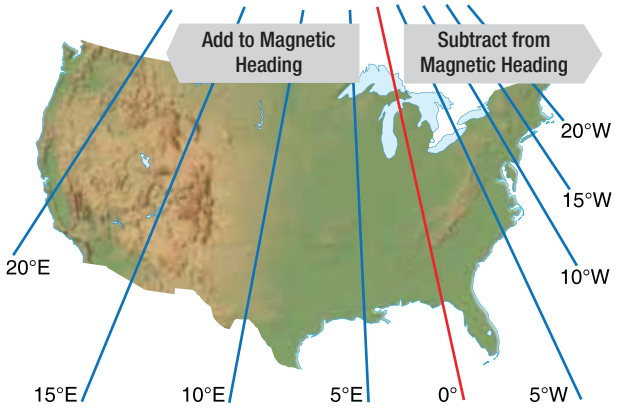


Figure 8-103. Aircraft located along the agonic line have 0˚ of variation between magnetic north and true north. Locations on and between the isogonic lines require addition or subtraction, as shown, to magnetic indications to arrive at a true geographic direction.

The earth's magnetic field exits the poles vertically and arches around to extend past the equator horizontally or parallel to the earth's surface. (***Figure 8-101***) Operating an aircraft near the magnetic poles causes what is known as dip error. The compass magnets pull downward toward the pole, rather than horizontally, as is the case near the equator. This downward motion causes inaccuracy in the indication.

Although the compass float mechanism is weighted to compensate, the closer the aircraft is to the north or south magnetic poles, the more pronounced the errors. Dip errors manifest themselves in two ways. The first is called acceleration error. If an aircraft is flying on an east-west path and simply accelerates, the inertia of the float mechanism causes the compass to swing to the north. Rapid deceleration causes it to swing southward. Second, if flying toward the North Pole and a banked turn is made, the downward pull of the magnetic field initially pulls the card away from the direction of the turn. The opposite is true if flying south from the North Pole and a banked turn is initiated. In this case, there is initially a pull of the compass indicator toward the direction of the turn. These kinds of movements are called turning errors.

Another peculiarity exists with the magnetic compass that is not dip error. Look again at the magnetic compass in ***Figure 8-102***. If flying north or toward any indicated heading, turning the aircraft to the left causes a steady decrease in the heading numbers. But, before the turn is made, the numbers to the left on the compass card are actually increasing. The numbers to the right of the lubber line rotate behind it on a left turn. So, the compass card rotates opposite to the direction of the intended turn. This is because, from the pilot's seat, you are actually looking at the back of the compass card. While not a major problem, it is more intuitive to see the 360 degrees of direction oriented as they are on an aeronautical chart or a hand held compass.

VERTICAL MAGNETIC COMPASS

Solutions to the shortcomings of the simple magnetic compass described above have been engineered. The vertical magnetic compass is a variation of the magnetic compass that eliminates the reverse rotation of the compass card just described. By mounting the main indicating magnets of the compass on a shaft rather than a float, through a series of gears, a compass card can be made to turn about a horizontal axis. This allows the numbers for a heading, towards which the pilot wants to turn, to be oriented correctly on the indicating card. In other words, when turning right, increasing numbers are to the right; when turning left, decreasing numbers rotate in from the left. (***Figure 8-104***)

Many vertical magnetic compasses have also replaced the liquid-filled instrument housing with a dampening cup that uses eddy currents to dampen oscillations. Note that a vertical magnetic compass and a directional gyro look very similar and are often in the lower center position of the instrument panel basic T. Both use the nose of an aircraft as the lubber line against which a rotating compass card is read. Vertical magnetic compasses are characterized by the absence of the hand adjustment knob found on directional gyros, which is used to align the gyro with a magnetic indication.

REMOTE INDICATING COMPASS

Magnetic deviation is compensated for by swinging the compass and adjusting compensating magnets in the instrument housing. A better solution to deviation is to remotely locate the magnetic compass in a wing tip or vertical stabilizer where there is very little interference with the earth's magnetic field. By using a synchro remote indicating system, the magnetic compass float assembly can act as the rotor of the synchro system. As the float mechanism rotates to align with magnetic north in the remotely located compass, a varied electric current can be produced in the transmitter. This alters the magnetic field produced by the coils of the indicator



Figure 8-104. A vertical magnetic direction indicator provides a realistic reference of headings.

in the cockpit, and a magnetic indication relatively free from deviation is displayed. Many of these systems are of the magnesyn type.

REMOTE INDICATING SLAVED GYRO COMPASS

FLUX GATE COMPASS

An elaborate and very accurate method of direction indication has been developed that combines the use of a gyro, a magnetic compass, and a remote indicating system. It is called the slaved gyro compass or flux gate compass system. A study of the gyroscopic instruments section of this chapter assists in understanding this device.

A gyroscopic direction indicator is augmented by magnetic direction information from a remotely located compass. The type of compass used is called a flux valve or flux gate compass. It consists of a very magnetically permeable circular segmented core frame or spider. The earth's magnetic field flows through this iron core and varies its distribution through segments of the core as the flux valve is rotated via the movement of the aircraft. Pickup coil windings are located on each of the core's spider legs that are positioned 120 degrees apart. (***Figure 8-105***)

The distribution of earth's magnetic field flowing through the legs is unique for every directional orientation of the aircraft. A coil is placed in the center of the core and is energized by AC current. As the AC flow passes through zero while changing direction, the earth's magnetic field is allowed to flow through the core. Then, it is blocked or gated as the magnetic field of the core current flow builds to its peak again. The cycle is repeated at the frequency of the AC supplied to the excitation coil. The result is repeated flow and non-flow of the earth's flux across the pickup coils.

During each cycle, a unique voltage is induced in each of the pickup coils reflecting the orientation of the aircraft in the earth's magnetic field. The electricity that flows from each of the pickup coils is transmitted out of the flux valve via wires into a second unit. It contains an autosyn transmitter, directional gyro, an amplifier, and a triple wound stator that is similar to that found in the indicator of a synchro system. Unique voltage is induced in the center rotor of this stator which reflects the voltage received from the flux valve pickup coils sent through the stator coils. It is amplified and used to augment the position of the DG. The gyro is wired to be the rotor of an autosyn synchro system, which transmits the position of the gyro into an indicator unit located in the cockpit. In the indicator, a vertical compass card is rotated against a small aeroplane type lubber line like that in a vertical magnetic compass. (***Figure 8-106 and Figure 8-107***)

Further enhancements to direction finding systems of this type involving the integration of radio navigation aids are common. The radio magnetic indicator (RMI) is one such variation. In addition to the rotating direction indicator of the slaved gyro compass, it contains two pointers. One indicates the bearing to a very high frequency (VHF) omnidirectional range (VOR) station and the other indicates the bearing to a non-directional automatic direction finder (ADF) beacon. These and

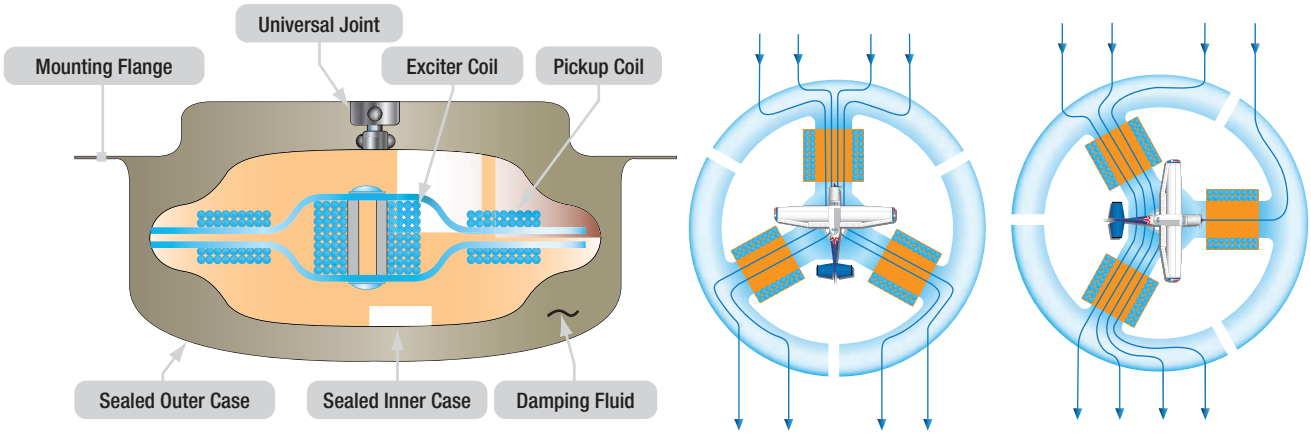


Figure 8-105. As the aircraft turns in the earth's magnetic field, the lines of flux flow lines vary through the permeable core of flux gate, creating variable voltages at the three pickoffs.

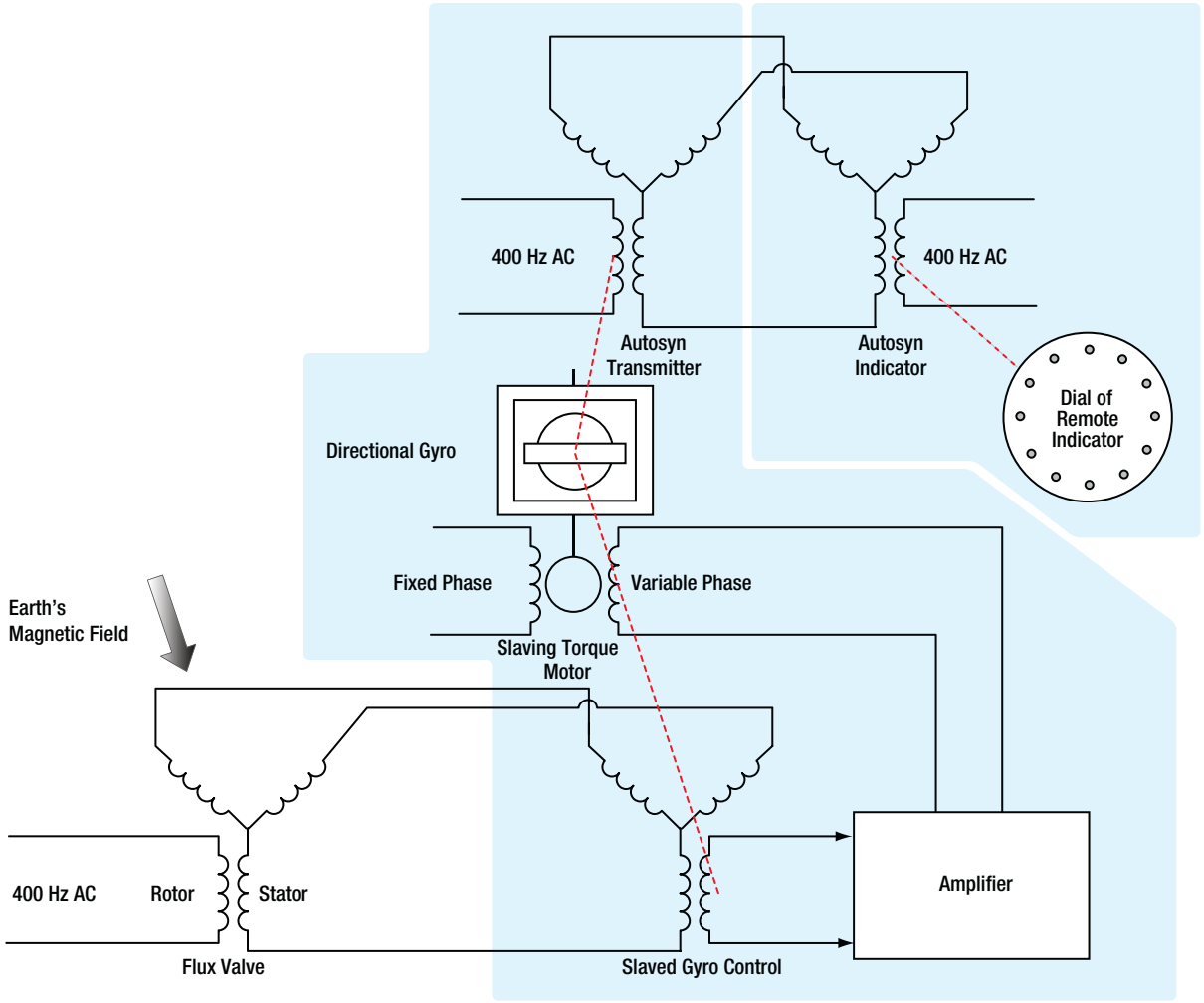


Figure 8-106. A simplified schematic of a flux gate, or slaved gyro, compass system.

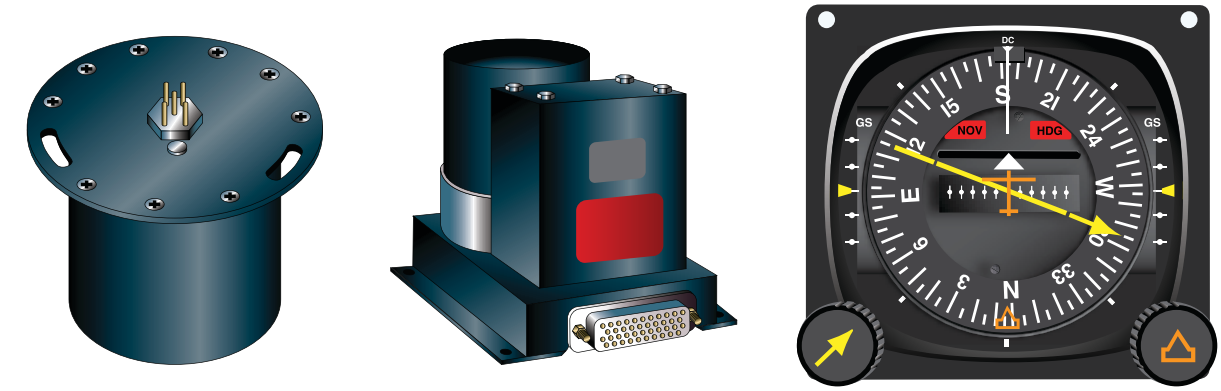


Figure 8-107. Solid state magnetometer units.

other radio navigation aids are discussed further in the communications and navigation chapter of this handbook. It should also be noted that integration of slaved gyro direction indicating system information into auto-pilot systems is also possible.

SOLID STATE MAGNETOMETERS

Solid state magnetometers are used on many modern aircraft. They have no moving parts and are extremely accurate. Tiny layered structures react to magnetism on a molecular level resulting in variations in electron activity. These low power consuming devices can sense

not only the direction to the earth's magnetic poles, but also the angle of the flux field. (***Figure 8-108***) They are free from oscillation that plagues a standard magnetic compass. They feature integrated processing algorithms and easy integration with digital systems.

FLIGHT DATA RECORDER (FDR)

DESCRIPTION

Flight Data Recorder (FDR) - device used to record specific aircraft performance parameters. The purpose of an FDR is to collect and record data from a variety of aircraft sensors onto a medium designed to survive an accident. (***Figure 8-109***)

An FDR has historically been one of two types of "flight recorder" carried on aircraft, the other being a cockpit voice recorder (CVR). Where both types of recorder are fitted, they are now sometimes combined into a single unit (ICAO Definition: Combination recorders). Combination recorders need to meet the flight recorder equipage requirements as specifically detailed in ICAO Annex 6 - Operation of Aircraft.



Figure 8-108. A radio magnetic indicator (RMI) combines a slaved gyro heading indication (red triangle at top of gauge) with magnetic bearing information to a VOR station (solid pointer) and an ADF station (hollow pointer).



Figure 8-109. Flight Data Recorder.

ICAO REQUIREMENTS

According to the provisions in ICAO Annex 6 - Operation of Aircraft, a Type I FDR shall record the parameters required to determine accurately the aeroplane flight path, speed, attitude, engine power, configuration and operation. Types II and IIA FDRs shall record the parameters required to determine accurately the aeroplane flight path, speed, attitude, engine power and configuration of lift and drag devices.

The detailed list of parameters to be recorded by FDRs is provided in section 6.3 "Flight Recorders" and at Attachment D to Annex 6. Furthermore, provisions in section 6.3 specify the aircraft equipage requirements depending on the maximum certificated take-off mass and the date of first issue of the individual certificate of airworthiness. For example, provision 6.3.6 of Annex 6, states that, all aeroplanes of a maximum certificated take-off mass of over 5 700 kg for which the individual certificate of airworthiness is first issued after 1 January 2005 shall be equipped with a Type IA FDR.

According to ICAO SARPS, combination recorders (FDR/CVR) can only be used to meet the flight recorder equipage requirements as specifically indicated in ICAO Annex 6.

OBJECTIVE

The recorder is installed in the most crash survivable part of the aircraft, usually the tail section. The data collected in the FDR system can help investigators determine whether an accident was caused by pilot error, by an external event (such as windshear), or by an airplane system problem. This data have contributed to airplane system design improvements and the ability to predict potential difficulties as airplanes age. An example of is using FDR data to monitor the condition of a high-hours engine. Evaluating the data could be useful in making a decision to replace the engine before a failure occur.

HISTORY

Flight data recorders were first introduced in the 1950s. Many first generation FDRs used metal foil as the recording medium. This metal foil was housed in a crash survivable box installed in the aft end of an

airplane. Beginning in 1965, FDRs (commonly known as "black boxes") were required to be painted bright orange or bright yellow, making them easier to locate at a crash site.

Second generation FDRs were introduced in the 1970s as the requirement to record more data increased, but they were unable to process the larger amounts of incoming sensor data. The solution was development of the flight data acquisition unit (FDAU). A flight-data acquisition unit is a unit that receives various discrete, analog and digital parameters from a number of sensors and avionic systems and then routes them to a flight data recorder (FDR) and, if installed, to a Quick Access Recorder. Information from the FDAU to the FDR is sent via specific data frames, which depend on the aircraft manufacturer. Integration of FDAU functions into software required by other aircraft system components is now being seen, as in the case of the Enhanced Airborne Flight Recorder (EAFR) installed on the Boeing 787.

The second-generation digital FDR (DFDR) uses tape similar to audio recording tape. The tape is 300 to 500 ft long and can record up to 25 hours of data. It is stored in a cassette device mounted in a crash-protected enclosure.

FAA rule changes in the late 1980s required the first-generation FDRs to be replaced with digital recorders. Many of the older FDRs were replaced with second-generation magnetic tape recorders that can process incoming data without a Flight Data Acquisition Unit. Most of these DFDRs can process up to 18 input parameters (signals). This requirement was based upon an airplane with four engines and a requirement to record 11 operational parameters for up to 25 hours.

Most recent recorders utilise solid state technology. Solid state uses stacked arrays of memory chips, so they don't have moving parts. With no moving parts, there are fewer maintenance issues and a decreased chance of something breaking during a crash. Data from both the cockpit voice recorder (CVR) and FDR is stored on stacked memory boards inside the Crash-survivable Memory Unit.

The most modern FDR systems incorporate an Emergency Locator Transmitter (ELT) and some up to date recorders are also equipped with an Underwater Locator Beacon to assist in locating in the event of an over water accident. A device called a "pinger" is automatically activate when the recorder is immersed in water. It transmits an acoustic signal on a frequency of 37.5 KHz that can be detected with a suitable receiver. In the case of the latest recorders, these transmissions are detectable at all but the most extreme oceanic depths but since they are battery-powered, their transmissions only continue for a limited period.

PRINCIPLES OF OPERATION

The FDR onboard the aircraft records many different operating conditions of the flight. By regulation, newly manufactured aircraft must monitor at least eighty-eight important parameters such as time, altitude, airspeed, heading, and aircraft attitude. In addition, some FDRs can record the status of more than 1 000 other in-flight characteristics that can aid in the investigation. The items monitored can be anything from flap position to auto-pilot mode or even smoke alarms. It is required by regulations that, on an annual basis, an FDR verification check (readout) is performed in order to verify that all mandatory parameters are recorded.

- Magnetic Tape - The introduction of the CVR in the late 1960s and DFDRs in the early 1970s made magnetic tape the recording medium of choice until the introduction of solid-state flight recorders in the late 1980s. There were a variety of tapes and tape transports used by the various recorder manufacturers. The most widely used tapes were mylar, kapton, and metallic. The tape transports were even more varied, using designs such as coplaner reel to reel, coaxial reel-to-reel, endless loop reel packs and endless loop random storage. Tape CVRs record four channels of audio for 30 minutes, and the DFDR records 25 hours of data. CVRs and FDRs record over the oldest data with the newest data in an endless loop recording pattern.
- Digital Recording - Most DFDRs require a flight data acquisition unit (FDAU) to provide an interface between the various sensors and the DFDR. The FDAU converts analog signals from the sensors to digital signals that are then multiplexed into a serial data stream suitable for recording by the DFDR. Industry standards dictated the format of the data stream, which for the vast majority of tape-based DFDRs is 64 12-bit data words per second. The recording capacity of the tape DFDR is limited by the length of tape that

can be crash-protected and the data frame format. The capacity of the tape DFDRs was adequate for the first generation of wide-body transports, but was quickly exceeded when aircraft like the Boeing 767 and Airbus A320 with digital avionics were introduced.

- Solid State Technology - The introduction of solid-state flight recorders in the late 1980s marked the most significant advance in evolution of flight recorder technology. The use of solid-state memory devices in flight recorders has expanded recording capacity, enhanced crash/fire survivability, and improved recorder reliability. It is now possible to have 2-hour audio CVRs and DFDRs that can record up to 256 12-bit data words per second, or 4 times the capacity of magnetic tape DFDRs.

CURRENT SURVIVABILITY STANDARDS

TSO C123a (CVR) and C124a (DFDR)

- Fire (High Intensity) - 1 100°C flame covering 100% of recorder for 30 minutes. (60 minutes if ED56 test protocol is used).
- Fire (Low Intensity) - 260°C Oven test for 10 hours
- Impact Shock - 3 400 GS for 6.5 ms
- Static Crush - 5 000 pounds for 5 minutes on each axis
- Fluid Immersion - Immersion in aircraft fluids (fuel, oil etc.) for 24 hours
- Water Immersion - Immersion in sea water for 30 days
- Penetration Resistance - 500 lb. Dropped from 10 ft. with a 1/4-inch diameter contact point
- Hydrostatic Pressure - Pressure equivalent to depth of 20 000 ft.

ELECTRONIC DISPLAYS



Figure 8-110. Equivalent Electromechanical Flight and Navigation Instruments on the left.

The early EIS displays mimicked the analog display formats for ease in pilot training as the crew transitioned from older analog displays to digital displays that were driven by aircraft data computers, known as display processors or symbol generators. ***Figure 8-110*** depicts an early model Boeing 737 instrument panel with an analog Attitude Direction Indicator (ADI) and analog Horizontal Situation Display (HSI) in the right picture, and a later model B737 instrument panel with electronic ADI (EADI) and electronic HSI (EHSI) displays shown in the left picture.

The ADI or EADI is an artificial horizon with lateral bars superimposed to display computer-generated pitch and bank steering commands from the Flight Director computer. The HSI or EHSI is similar to a heading indicator, except that it combines navigation commands from the VHF Omni-Range (VOR) or Global Positioning System (GPS) receivers which are used for en-route guidance, or from the Instrument Landing System, which is used for terminal guidance. Besides heading, the HSI/EHSI also provides actual track, desired track, track angle error, drift angle, cross track deviation, and distance to destination information from the Distance Measuring Equipment or Inertial Navigation System (INS). (***Figure 8-111***)

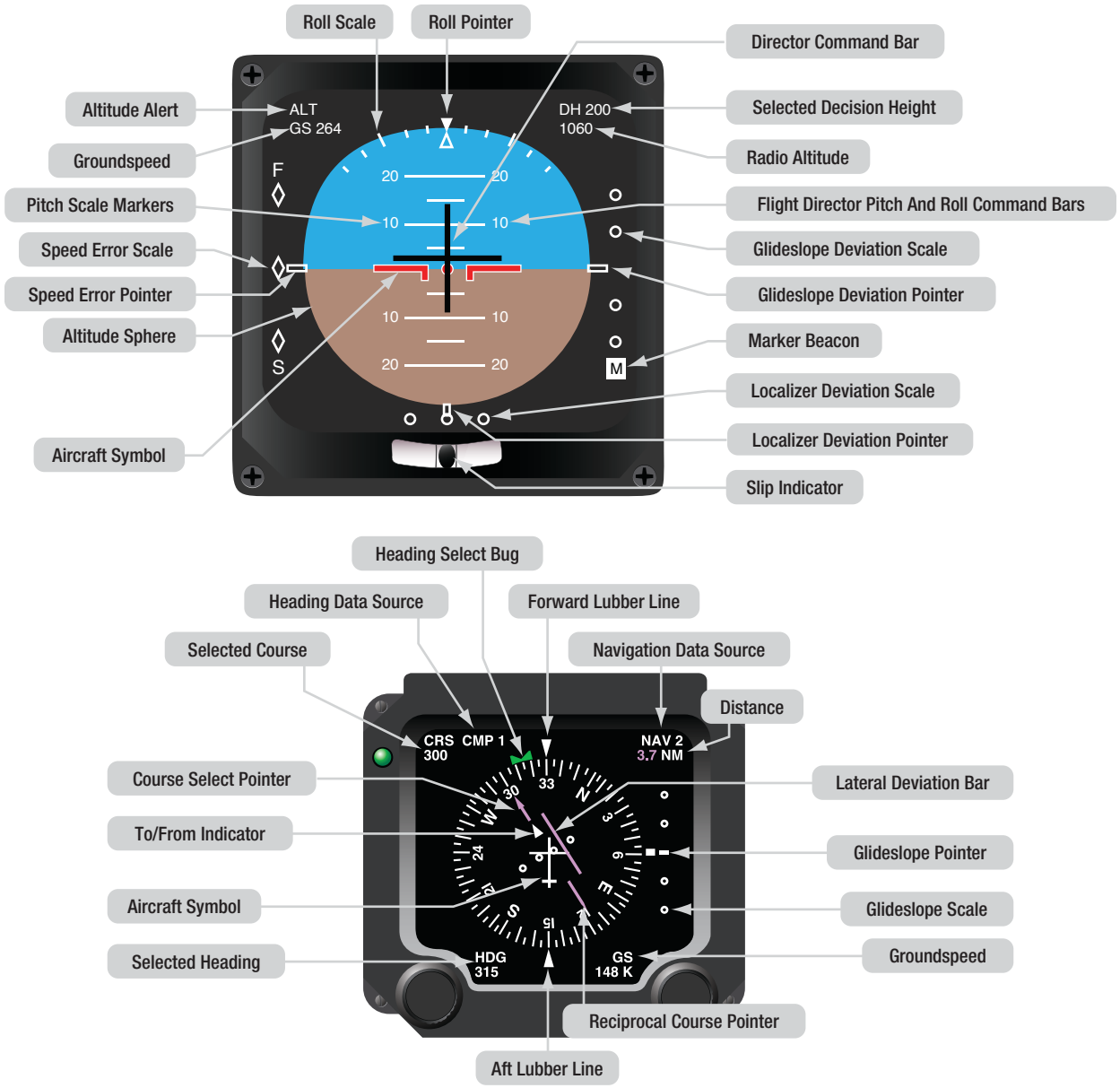


Figure 8-111. Typical EADI (top) and EHSI (bottom) display symbology.

The pilot and the co-pilot not only have independent EADI and EHSI displays, but they also have independent Display Processor Units, also known as Symbol Generators, to drive their displays. (***Figure 8-112***)

Display formats are produced by the Symbol Generators that receive inputs from the crew and various on-board systems. The Flight Director Systems, Navigation Systems, Air Data Systems, and Weather Radar provide inputs to the Symbol Generators, along with commands from the each crewmember's display control panel. The Symbol Generators produce the graphics for the EADI, EHSI, and an optional Multi-Function Display (MFD) that is mounted in the center instrument panel. The MFD, which is physically identical to the EADI and EHSI, is typically used to display weather radar information; however, it can also be used to display either light information or navigational information in the event of an EADI or EHSI failure. The following section will discuss the Boeing 777 EIS, which is a more advanced example of the one just covered.

ELECTRONIC FLIGHT INSTRUMENT SYSTEM

The Boeing 777, which first entered service in 1995, has six 8" inch square multi-color LCD displays as shown in ***Figure 8-113***. The B777 EIS consists of a dual redundant Electronic Flight Instrument Systems (EFIS) and Engine Indication and Crew Alerting System

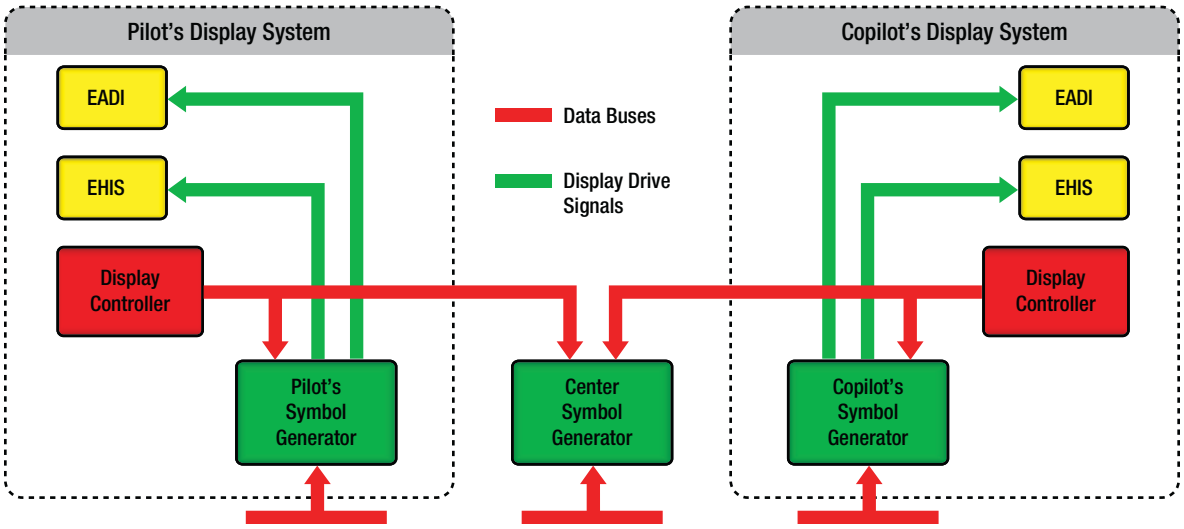


Figure 8-112. Electronic Displays are driven by Symbol Generators.

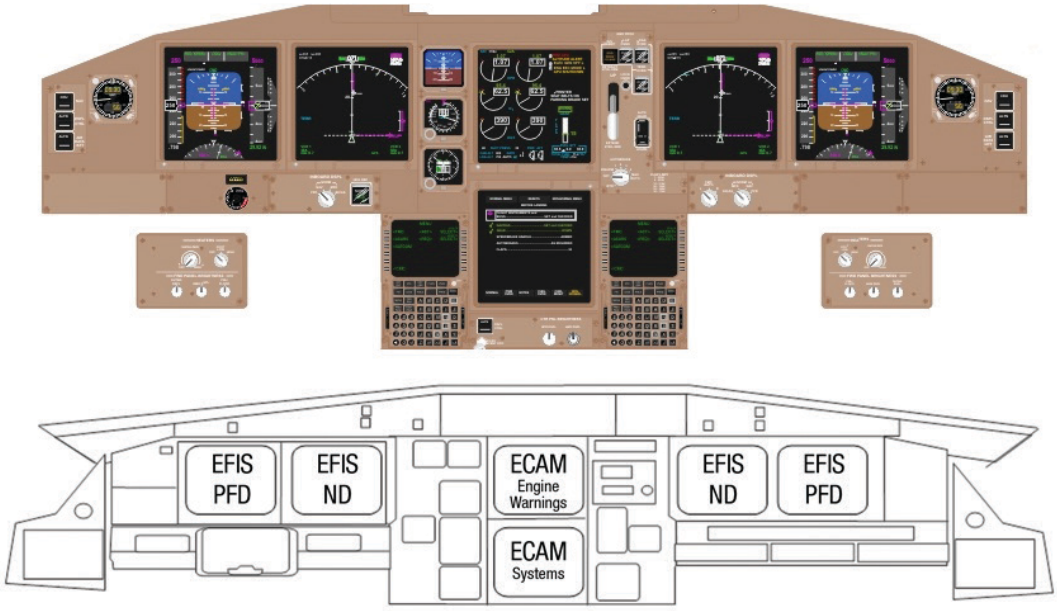


Figure 8-113. Boeing 777 Electronic Instrument System has 6 LCD Displays.

(EICAS). On the left side of the instrument panel is the Captain's EFIS, consisting of a Primary Flight Display (PFD) located outboard and a Navigation Display (ND) located inboard. The co-pilot's EFIS located on the right instrument panel has an identical PFD and ND, located outboard and inboard respectively. All the displays are interchangeable to reduce the number of required spares. The information shown on each display, whether for flight or navigation, is determined by what each crew member selects on their individual display control panels.

The PFD takes the place of the EADI and displays all the information critical to flight, including attitude, airspeed, barometric altitude, vertical speed, heading, light modes, radio altitude, ILS data, and Traffic Alert and Collision Avoidance System (TCAS) resolution advisory. The PFDs are designed to increase the crew's situational awareness by integrating all of this information into a single composite display instead of the crew having to monitor several independent analog instruments. Also, the colors on the display change to alert the crew to potentially hazardous flight conditions, such as low airspeed, high rate of descent, etc. ***Figure 8-114*** is a typical Primary Flight Display

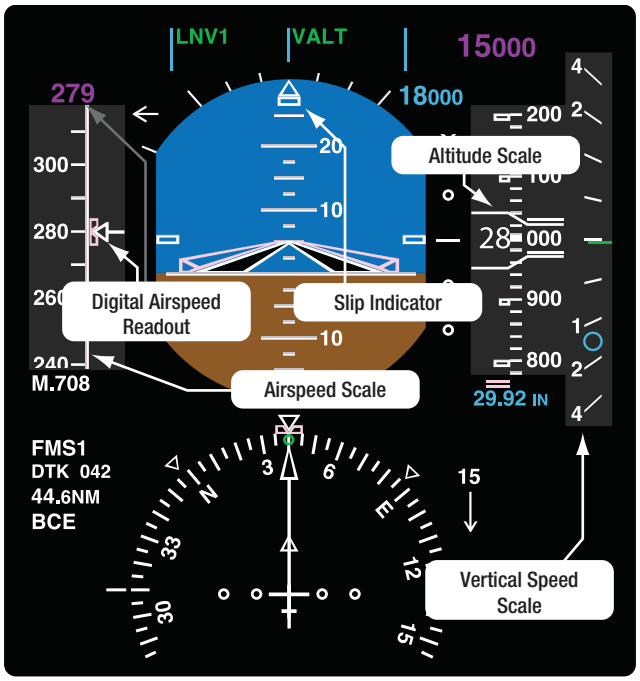


Figure 8-114. Primary flight display format.

format showing the artificial horizon in the center of the display, airspeed on the left side, altitude on the right side, heading on the bottom, and flight modes on the top of the display. Notice how the moving ladder format used for altitude and airspeed provide both absolute and relative information so the crew knows not only the exact numeric value, but also the rate that the altitude and airspeed is changing.

The Navigation Display, shown in ***Figure 1-115***, takes the place of the EHSI display to show the requisite information to navigate the aircraft, including heading, VOR, GPS, and ILS guidance. The ND has the ability to overlay additional information on the navigation page to eliminate the need for separate dedicated displays. Some examples of information that is typically overlaid on the ND include weather information from either the onboard weather radar or ground based sensors, and digital maps showing pre-programmed routes and waypoints from the Flight Management System.

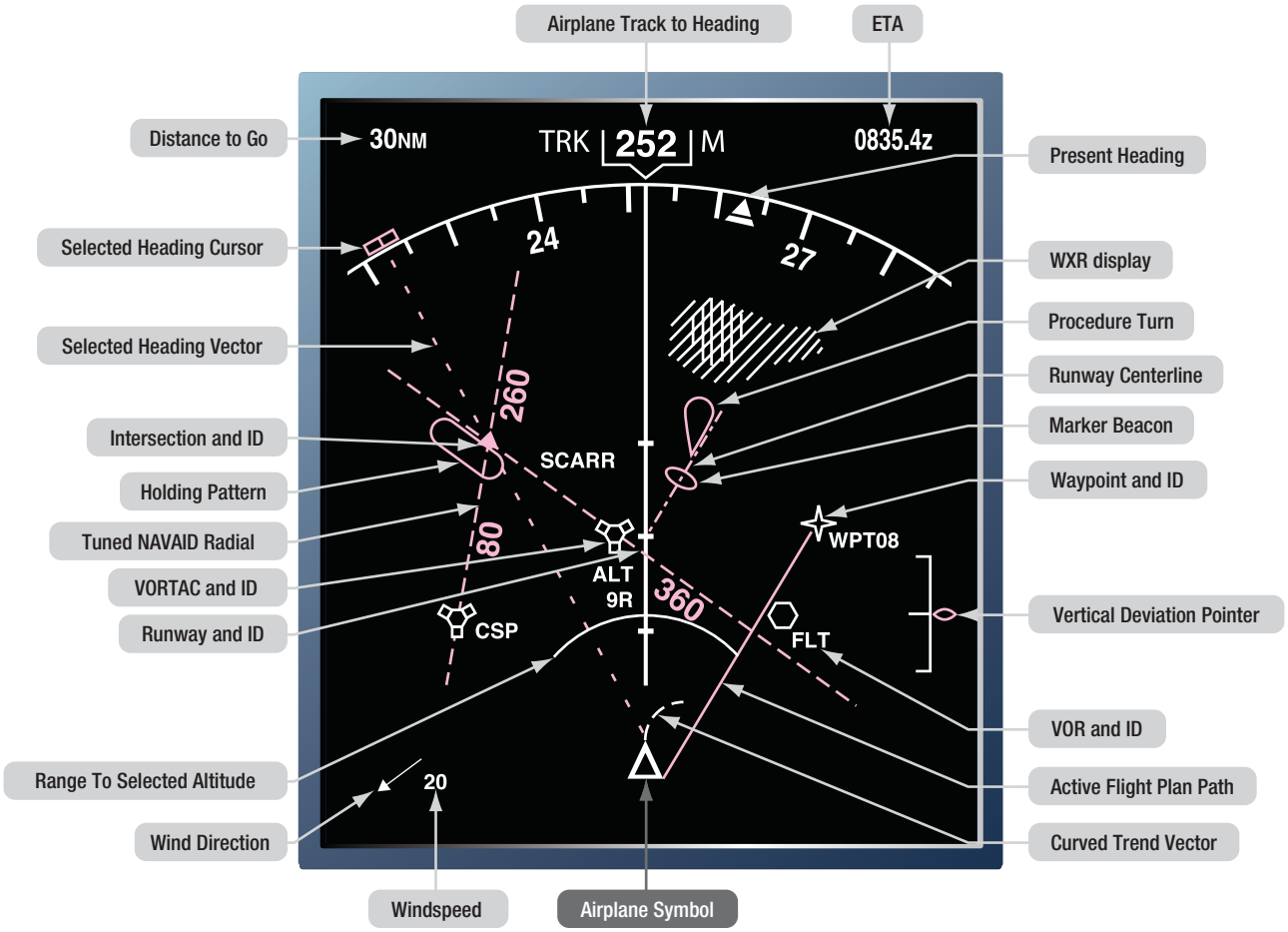


Figure 8-115. Navigation map display format.

ENGINE INDICATION AND CREW ALERTING SYSTEM (EICAS)

The Engine Indication and Crew Alerting System (EICAS), also called an Electronic Centralized Aircraft Monitor (ECAM) on Airbus aircraft, performs the monitoring of aircraft systems that was previously per formed by the Flight Engineer in three crew member cockpits.

As previously shown in ***Figure 1-113***, the two EICAS displays on the B777 are located in the center instrument panel. The upper EICAS display shows engine performance data, such as pressure ratio, N1 rotor speed, exhaust gas temperature, total air temperature, thrust mode, etc., in addition to cabin pressure, flap/slat position, landing gear position, and crew status alerts. (***Figure 8-116***)



Figure 8-116. EICAS engine display format.

The EICAS engine display format mimics the round analog instruments, while also providing digital readouts of the parameters. EICAS improves situational awareness by allowing the crew to see systems operation in graphical format and alerting them to any failures or impending failures. For example, if low oil pressure is detected, the EICAS will provide an aural alert and show to the oil pressure page on a lower display with a red box outlining which engine has low oil pressure. The Airbus ECAM system provides the crew with the following levels of warning along with detailed messages as to the nature of the problem and suggested courses of action.

- Level 3: An over speed, fire, or stall condition will cause a repetitive chime aural warning with a bright red flashing light.
- Level 2: A system failure, but not a safety of flight issue, will result in a single chime aural warning and a steady amber light.
- Level 1: Failure leading to system degradation results only in an amber light.
- Mode or System Status: If everything is normal, a green light will illuminate.

The lower EICAS display is called a Multi-Function Display because it provides auxiliary information to the flight crew and maintenance crew. The MFD can be used as a secondary engine display, status display, communications display, maintenance page, or electronic checklist. The MFD formats also include synoptic displays that provide system status diagrams for the fuel, electrical, hydraulic, flight control, and environmental control systems, in addition to showing door and landing gear positions. On some aircraft, the MFD is also used to display images from the ground maneuvering camera system. ***Figure 8-117*** is a schematic diagram of an Engine Indication and Crew Alerting System with all its associated components. The display select panel allows the crew to choose which computer is actively supplying information. It also controls the display of secondary engine information and system status displays on the lower monitor.

EICAS has a unique feature that automatically records the parameters of a failure event to be regarded afterwards by maintenance personnel. Pilots that suspect a problem may be occurring during flight can press the event record button on the display select panel. This also records the parameters for that flight period to be studied later by maintenance. Hydraulic, electrical, environmental, performance, and Auxiliary Power Unit (APU) data are examples of what may be recorded. EICAS uses Built-In-Test Equipment (BITE) for systems and components. A maintenance control panel is included for technicians. When the aircraft is on the ground, push-button switches display information pertinent to various systems for analysis. (***Figure 8-118***)

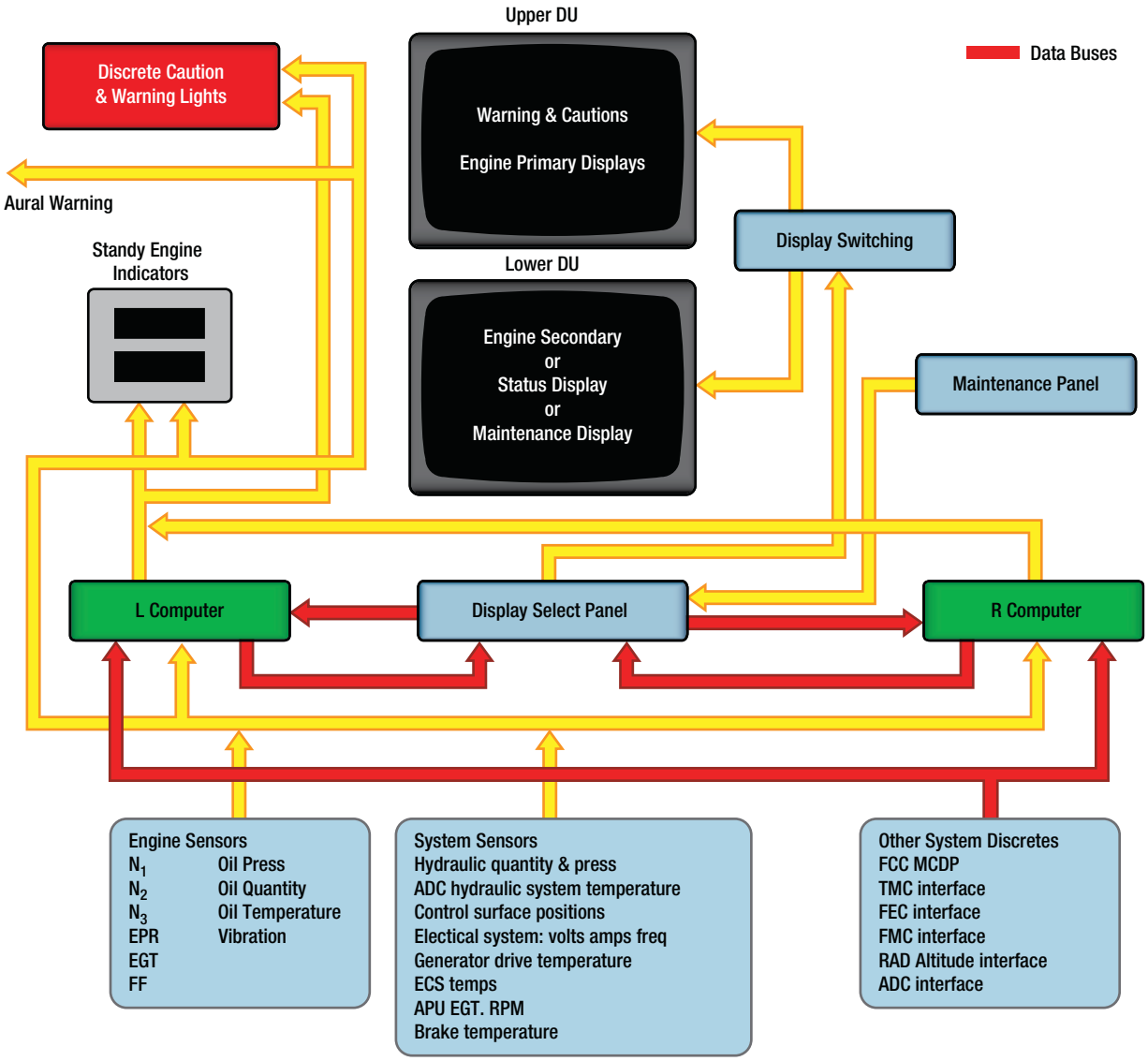


Figure 8-117. EICAS schematic diagram.

This section contained an overview of an aircraft cockpit with its Electronic Instrument System. The following section will discuss how digital data streams are formed and processed by aircraft computers and then sent over digital data buses to cockpit displays to provide essential information for the flight crew and maintenance crew.

ELECTRONIC INSTRUMENT COMPUTING SYSTEMS

The Boeing 777 was the first commercial aircraft to make use of the Integrated Modular Avionics (IMA) concept in what Boeing calls its Airplane Information Management System (AIMS), the main computer for controlling the electronic instrument system. AIMS integrates multiple functions that require large quantities of data collection and processing. As shown in ***Figure 8-119***, AIMS provides the display processing and symbol generation for the two PFDs, two NDs, EICAS and MFD display units (DU). AIMS interfaces with approximately 130 LRUs, sensors, switches and indicators through multiple data buses, in addition to analog and discrete (ON/OFF) signals, to permit the integration of information from a majority of aircraft systems in one place. The onboard maintenance system uses AIMS for the BITE computing function. Besides driving the six LCD electronic displays, AIMS accepts control inputs from the two EFIS control panels, two cursor control panels, two display switching panels, two instrument source select panels, a center display control panel, and display select panel. (***Figure 8-120***)

The Captain and First Officer display switching panels select the desired display format (PFD, NAV, MFD or EICAS) on the inboard display units. The normal

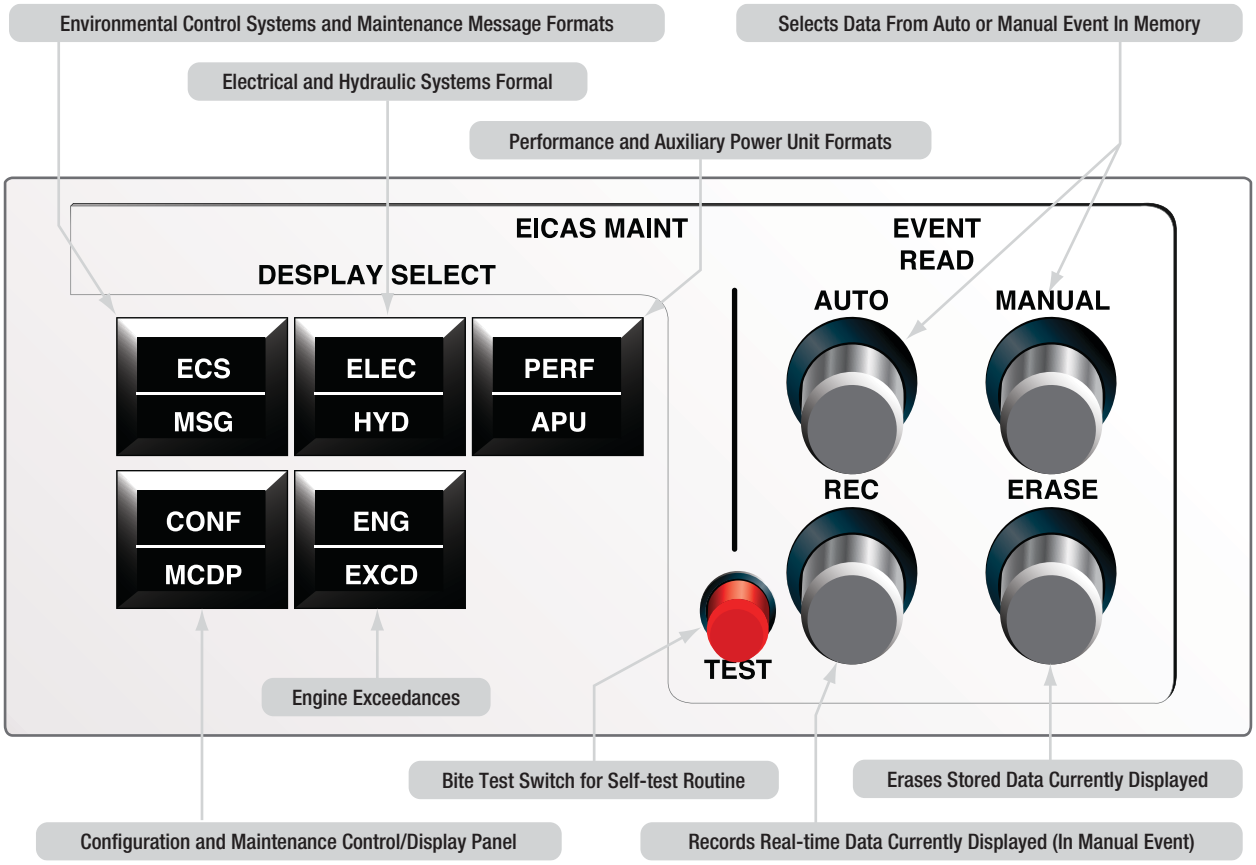


Figure 8-118. EICAS maintenance control panel.

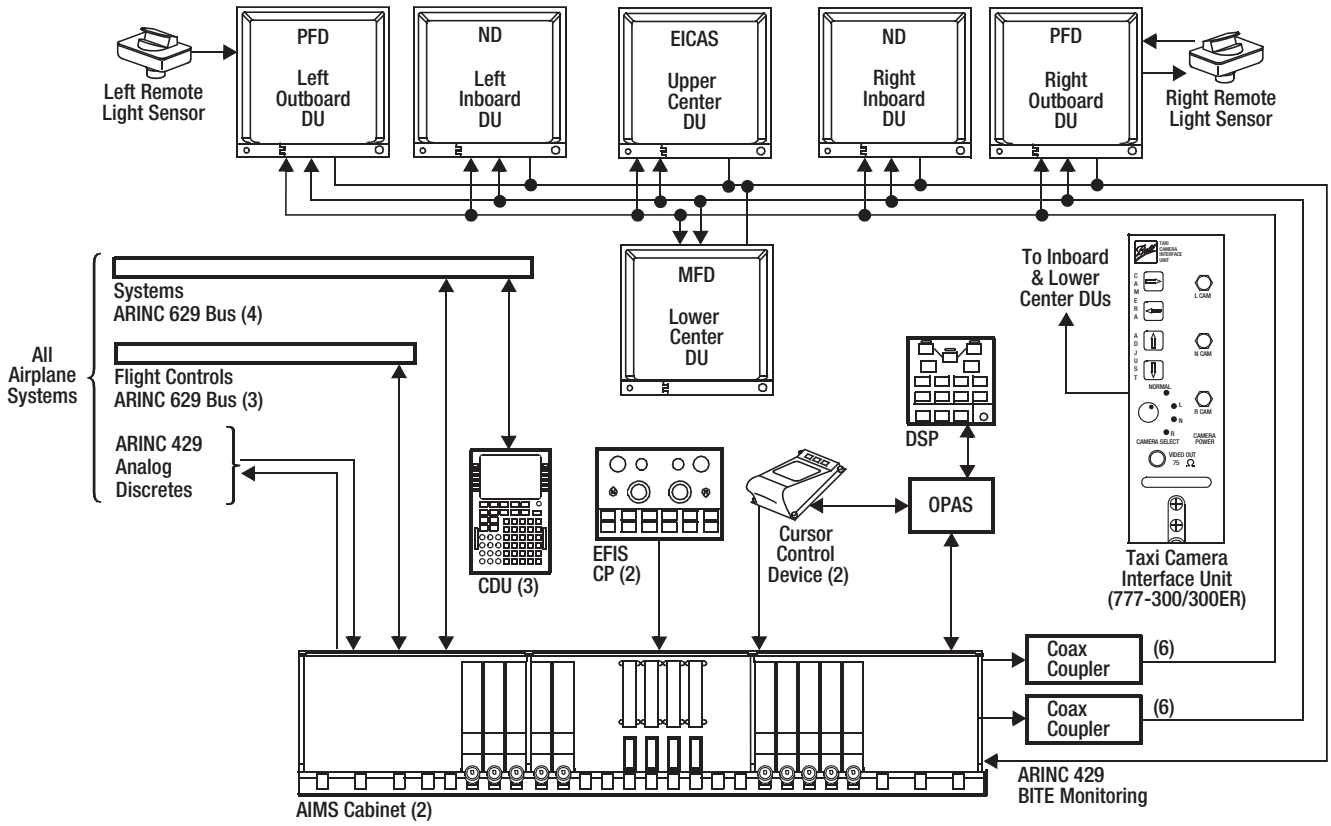


Figure 8-119. IMS is the EIS computing system for the Boeing 777.

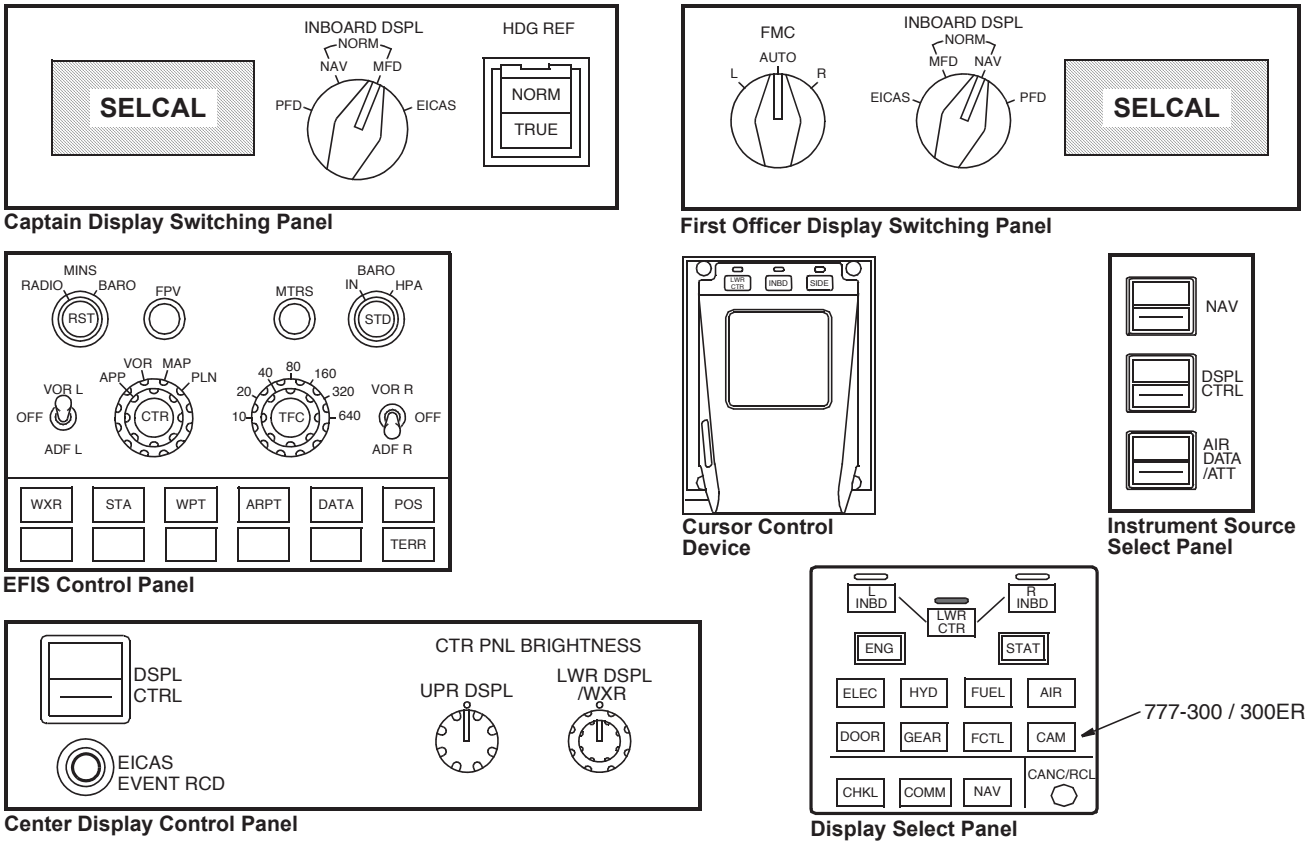


Figure 8-120. AIMS accepts control inputs from multiple display control panels.

modes for the inboard displays are either the Navigation (NAV) or Multi Function Display (MFD) pages. The two cursor control devices are used to select and activate items on the MFDs, much like a mouse on a personal computer.

The EFIS (Electronic Flight Instrumentation System) controls are used to select the desired source of the EFIS data. For the PFD (Primary Flight Display), the EFIS control panel selects the barometric altitude reference, radio altitude decision height, flight path vector and altitude reference. For the ND (Navigation Display), it selects the display mode format (map, plan, approach, or VOR navigation) and range, and turns on/off the VOR (VHF Omni-Range) and ADF (Automatic Direction Finder) pointers, Weather Radar (WXR) display, TCAS (Traffic Alert and Collision Avoidance System) display, and other navigation and surveillance inputs.

Electronic instrument computing systems are triple redundant to allow for several automatic reversionary and manual override modes to switch out faulty display management computers if the event of a failure. ***Figure 8-121*** shows the architecture of the Airbus A330 electronic instrument system (EIS) in normal mode where DMC-1 is driving the Captain's EFIS PFD and ND, DMC-2 is driving the First Officer's EFIS PFD and ND, and DMC-3 is driving the ECAM (Electronic Centralized Aircraft Monitor) engine warning display and system display. Note that the system display on the Airbus aircraft provides the same function as an MFD on Boeing aircraft. In the event that an outboard display unit fails, the altitude, attitude, and airspeed information that was displayed on the PFD will automatically revert to being displayed on the inboard display unit since flying the aircraft always takes precedence over navigation. Likewise, if the top center DU fails, engine and warning information will revert to being displayed on the system display.

Figure 8-122 depicts how the system would reconfigure in the event of a multiple DMC failure. In this example, the EFIS computer fails in DMC-2 and both the EFIS and ECAM computers fail in DMC-3. In this case, the first officer's EFIS would be driven by DMC-1 with the same display formats as the captain, and the ECAM would run off of DMC-1 or DMC-3.

RECONFIGURATION POSSIBILITIES - ARCHITECTURE

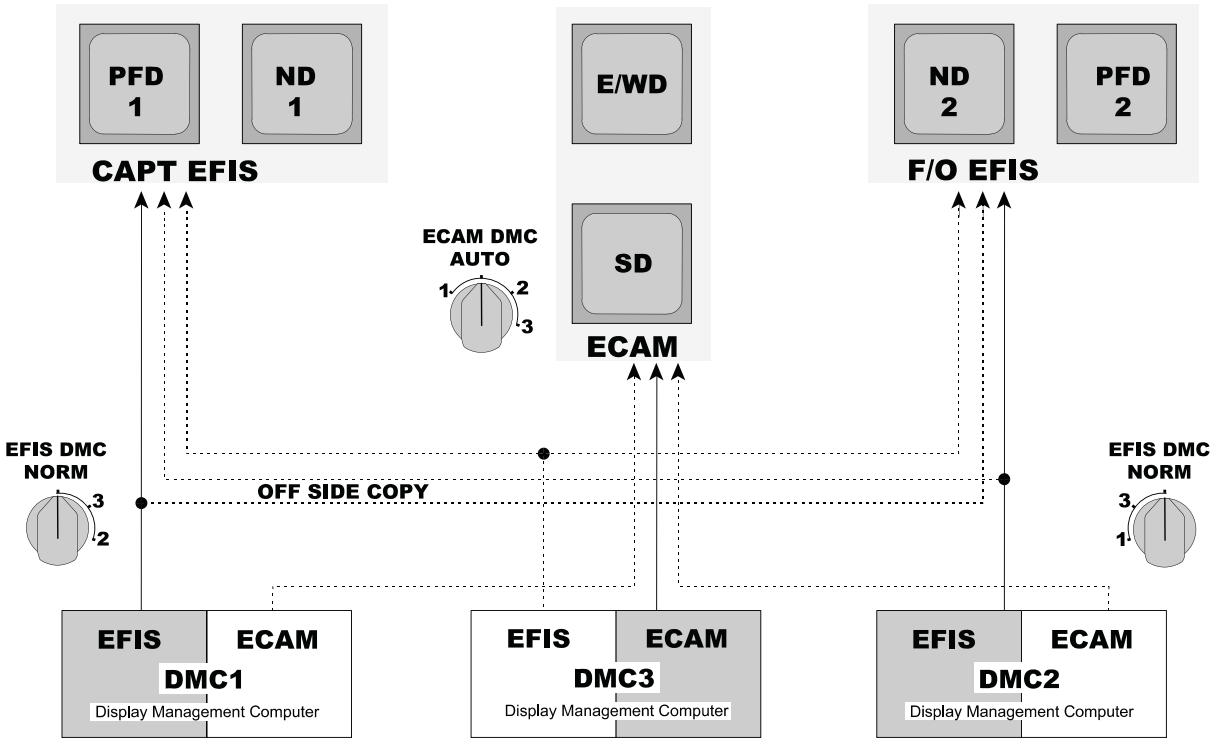


Figure 8-121. Airbus A330 EIS reconfiguration possibilities.

Reconfiguration - ECAM on DMC1 + F/O on EFIS DMC1

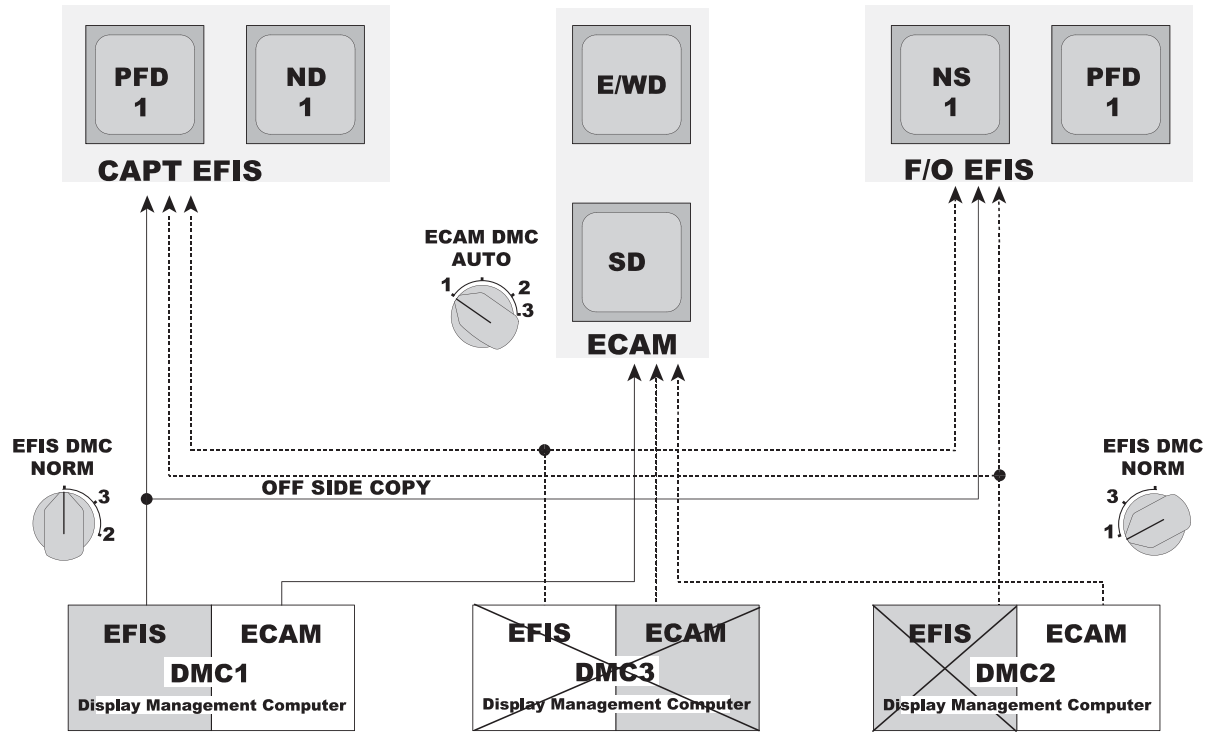


Figure 8-122. One of many reversionary modes in the event of multiple DMC failures.

STALL WARNING AND ANGLE OF ATTACK (AOA) INDICATORS

An aircraft's angle of attack (AOA) is the angle formed between the wing chord centerline and the relative wind. At a certain angle, airflow over the wing surfaces is insufficient to create enough lift to keep the aircraft flying, and a stall occurs. An instrument that monitors the AOA allows the pilot to avoid such a condition. The simplest form of AOA indicator is a stall warning device that does not have a gauge located in the cockpit. It uses an aural tone to warn of an impending stall due to an increase in AOA. This is done by placing a reed in a cavity just aft of the leading edge of the wing. The cavity has an open passage to a precise point on the leading edge.

In flight, air flows over and under a wing. The point on the wing leading edge where the oncoming air diverges is known as the point of stagnation. As the AOA of the wing increases, the point of stagnation moves down below the open passage that leads inside the wing to the reed. Air flowing over the curved leading edge speeds up and causes a low pressure. This causes air to be sucked out of the inside of the wing through the passage. The reed vibrates as the air rushes by making a sound audible in the cockpit. (***Figure 8-123***)

Another common device makes use of an audible tone as the AOA increases to near the point where the aircraft will stall. This stall warning device includes an electric switch that opens and closes a circuit to a warning horn audible in the cockpit. It may also be wired into a warning light circuit.



Figure 8-123. A reed-type stall warning device is located behind this opening in the leading edge of the wing. When the angle of attack increases to near the point of a stall, low-pressure air flowing over the opening causes a suction, which audibly vibrates the reed.

The switch is located near the point of stagnation on the wing leading edge. A small lightly sprung tab activates the switch. At normal AOA, the tab is held down by air that diverges at the point of stagnation and flows under the wing. This holds the switch open so the horn does not sound nor the warning light illuminate. As the AOA increases, the point of stagnation moves down.

The divergent air that flows up and over the wing now pushes the tab upward to close the switch and complete the circuit to the horn or light. (***Figure 8-124***)

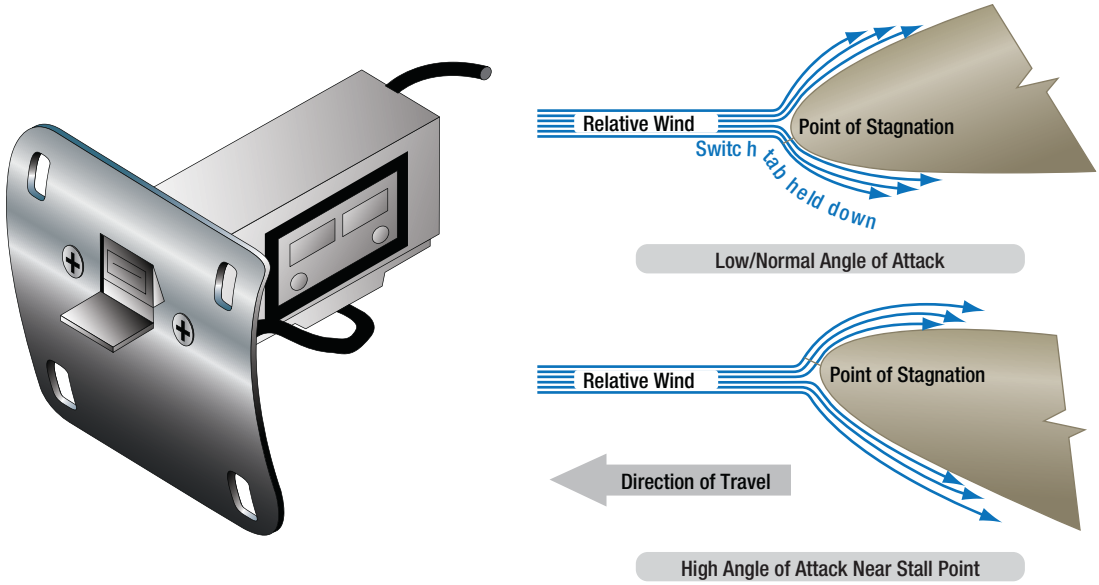


Figure 8-124. A popular stall warning switch located in the wing leading edge.

A true AOA indicating system detects the local AOA of the aircraft and displays the information on a cockpit indicator. It also may be designed to furnish reference information to other systems on high-performance aircraft. The sensing mechanism and transmitter are usually located on the forward side of the fuselage. It typically contains a heating element to ensure ice free operation. Signals are sent from the sensor to the cockpit or computer(s) as required. An AOA indicator may be calibrated in actual angle degrees, arbitrary units, percentage of lift used, symbols, or even fast/slow. (***Figure 8-125***)

There are two main types of AOA sensors in common use. Both detect the angular difference between the relative wind and the fuselage, which is used as a reference plane. One uses a vane, known as an alpha vane, externally mounted to the outside of the fuselage. It is free to rotate in the wind.

As the AOA changes, air flowing over the vane changes its angle. The other uses two slots in a probe that extends out of the side of the fuselage into the airflow. The slots lead to different sides of movable paddles in a chamber of the unit just inside the fuselage skin. As the AOA varies, the air pressure ported by each of the slots changes and the paddles rotate to neutralize the pressures.

The shaft upon which the paddles rotate connects to a potentiometer wiper contact that is part of the unit. The same is true of the shaft of the alpha vane. The changing resistance of the potentiometer is used in a balanced bridge circuit to signal a motor in the indicator to move the pointer proportional to the AOA. (***Figure 8-126 and Figure 8-127***)



Figure 8-125. Angle of attack indicator.

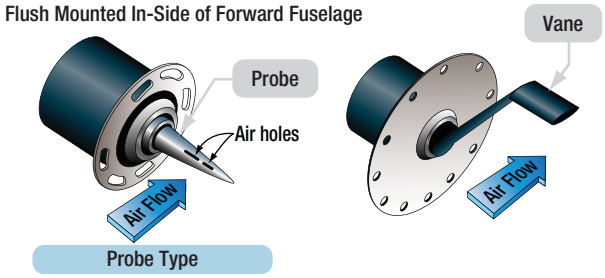


Figure 8-126. A slotted AOA probe and an alpha vane.

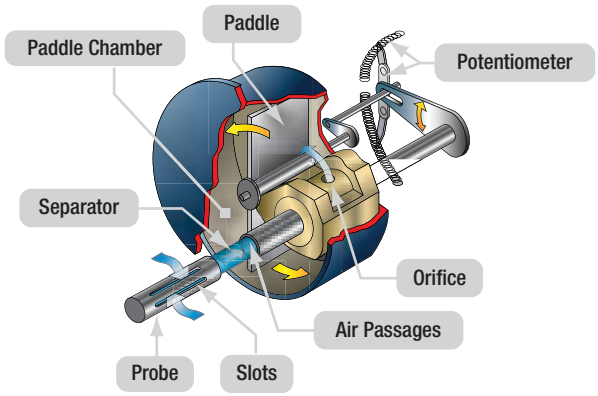


Figure 8-127. The internal structure of a slotted probe airstream direction detector.

Modern aircraft AOA sensor units send output signals to the ADC. There, the AOA data is used to create an AOA indication, usually on the primary flight display. AOA information can also be integrated with flap and slat position information to better determine the point of stall. Additionally, AOA sensors of the type described are subject to position error since airflow around the alpha vane and slotted probe changes somewhat with airspeed and aircraft attitude. The errors are small, but can be corrected in the ADC.

To incorporate a warning of an impending stall, many AOA systems signal a stick shaker motor that literally shakes the control column to warn the pilot as the aircraft approaches a stall condition. Electrical switches are actuated in the AOA indicator at various preset AOA to activate the motor that drives an unbalanced weighted ring, causing the column to shake. Some systems include a stick pusher actuator that pushes the control yoke forward, lowering the nose of the aircraft when the critical AOA is approached. Regardless of the many existing variations for warning of an impending stall, the AOA system triggers all stall warnings in high performance aircraft.

VIBRATION MEASUREMENT AND INDICATION

A turbine engine has an extremely low vibration level and a change in vibration, due to an impending or partial failure, may go unnoticed. Many engines are therefore fitted with vibration indicators that continually monitor the vibration level of the engine. (***Figure 8-128***)

Early vibration transducers were of the moving coil type and up to three could be located at strategic locations around the engine (HP Compressor case, LP Turbine case, etc). The units of vibration for these systems were in terms of relative amplitude.

An alternative system consists of a piezo-electric crystal and a mass inside a casing. As the engine vibrates, the mass will exert a force upon the crystal which will emit a small alternating current of a frequency equal to the frequency of vibration. This is then amplified and displayed in the flight deck via an ammeter. (***Figure 8-129***)

More modern systems have a pair of piezoelectric crystals contained within one housing. This provides for dual channel redundancy. Each transducer detects a broadband vibration signal that reflects all the vibrations in the engine. This broadband signal is processed by a microprocessor and the frequency of the rotating spools

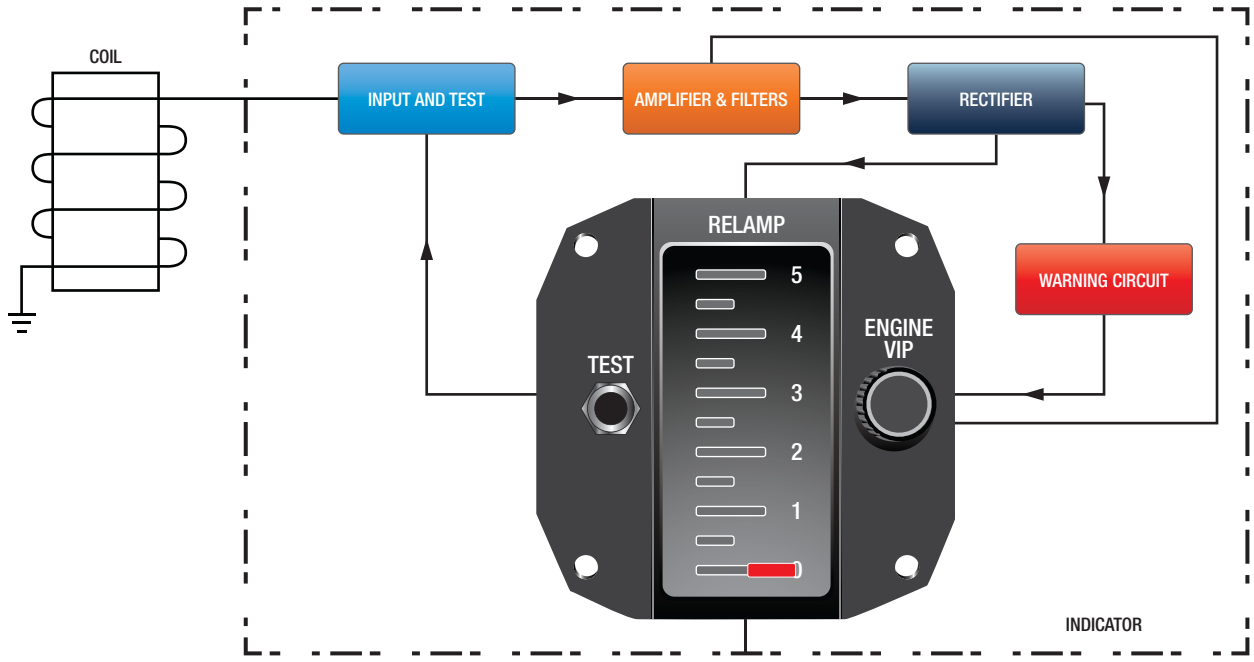


Figure 8-128. Vibration monitoring system.

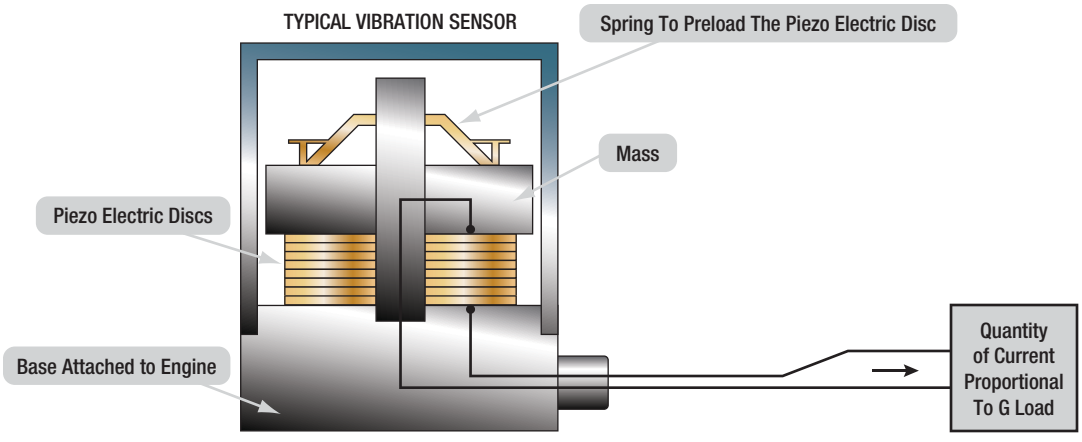


Figure 8-129. Piezo-electric vibration transducer.

(N1, N2 and for RR engines N3) so that the amplitude of vibration of these major assemblies can be displayed, usually on EICAS or ECAM. (***Figure 8-130***)

VIBRATION MEASUREMENT AND INDICATION

Engine vibration is a feature of engine operation which cannot be eliminated entirely even with turbine engines, which, unlike piston engines, have no reciprocating parts. Thus, by accurate balancing of such components as crankshafts, compressor and turbine rotor discs, vibration must be kept down to the lowest levels acceptable under all operating conditions. In respect of turbine engine operation, however, there is always the possibility of these levels being exceeded as a result of certain mechanical failures occurring. For example, a turbine blade may crack or 'creep', or an uneven temperature distribution around turbine blades and rotor discs may be set up; either of these will give rise to unbalanced conditions of the main rotating assemblies and possible disintegration. In order, therefore, to indicate when the maximum amplitude of vibration of an engine exceeds the preset level, monitoring systems.

A block diagram of a typical system is shown in the ***Figure 8-131***. It consists of a vibration pick-off, or sensor, mounted on an engine at right angles to its axis, an amplifier monitoring unit, and a moving coil calibrated to show vibration amplitude in thousandths of an inch (mils).

The sensor is a linear-velocity detector that converts the mechanical energy of vibration into an electrical signal of proportional magnitude. It does this by means of a spring supported permanent magnet suspended in a coil attached to the interior of the case. An engine vibrates, a sensor unit and core move with its magnet, however, tends to remain fixed in space because of inertia.

In other words, its function is similar to that of an accelerometer. The motion of the coil causes the turns to cut the field of the magnet, thus inducing a voltage in the coil and providing a signal to the amplifier unit. The signal, after amplification and integration by an electrical filter network, is fed to the indicator via a rectifying section.

An amber indicator light also forms part of the system, together with a test switch. The light is supplied with DC from the amplifier rectifying section and it comes on when the maximum amplitude of vibration exceeds the preset value. The test switch permits functional checking of the system's electrical circuit.

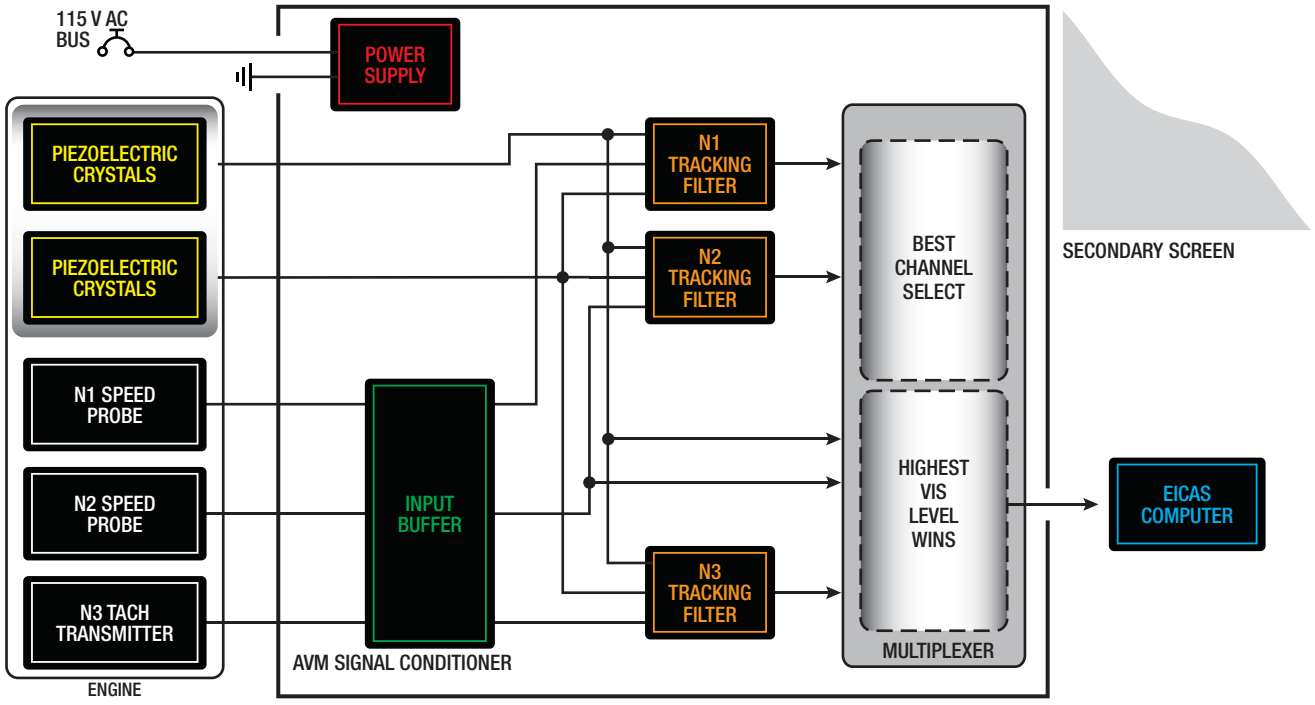


Figure 8-130. AVM system operation.

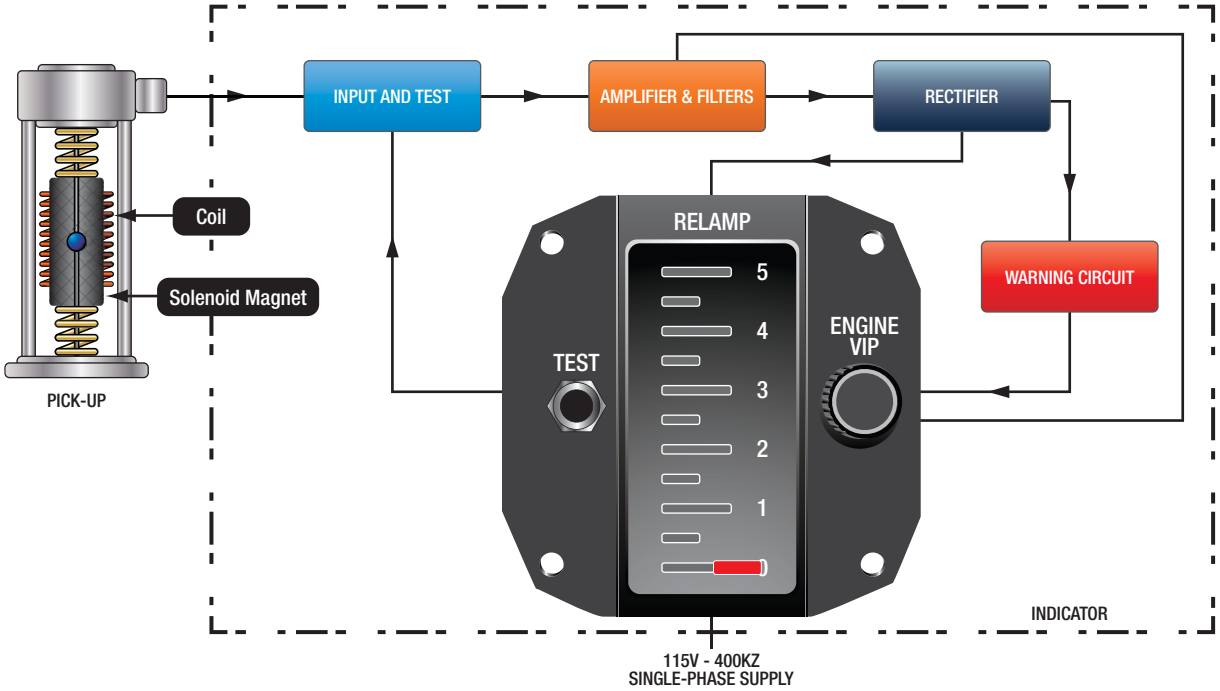


Figure 8-131. Schematic of a turbine engine vibration sensing system.

In some engine installations, two sensors may be fitted to an engine: for example, in a typical turbofan engine, one monitors vibration levels around the fan section, and the other around the engine core section. In systems developed for use in conjunction with LCD and CRT display indicators, the vibration sensors are of the type whereby vibration causes signals to be induced in a piezoelectric stack.

INSTRUMENT WARNING SYSTEM

WARNING AND INDICATING LIGHTS

Warning and indicator lights are used to alert the flight crew to conditions affecting the operation of aircraft systems. The lights may be divided into different categories according to function they perform, and in general, we find that they fall into three main categories:

I. Warning lights
II. Caution lights
III. Indicating or advisory lights.

WARNING LIGHTS

These are designed to alert the flight crew of unsafe conditions and are accordingly colored red.

CAUTION LIGHTS

These are amber in color to indicate abnormal but not necessarily dangerous conditions requiring caution, e.g. hydraulic system pressure running low.

INDICATING OR ADVISORY LIGHTS

These lights, which are either green or blue, are provided to indicate that a system is operable or has assumed a safe condition, e.g. a landing gear down and locked

Warning and indicator light assemblies are, basically, of simple construction, consisting of a bulb contained within a casing which incorporates electrical contacts and terminals for connection into the appropriate circuit. The colored lens is contained within a cap which fits over the casing and bulb. Provision for testing the bulb to ensure that its filament is intact is also incorporated in many types of light assemblies. The lens cap is so mounted on the casing that it can be pressed in to connect the bulb directly to the main power supply. Such an arrangement is referred to as a "press-to-test" facility.

Lights may also include a facility for dimming and usually this may be done in either of two ways. A dimming resistor may be included in light circuit, or the lens cap may incorporate an iris type diaphragm which

can be opened or closed by rotating cap. Lights used for warning purposes do not usually include the dimming facility because of the danger involved in having a dimmed warning light escaping notice.

The power supplies for warning and indicator lights are derived from the DC distribution system and the choice of busbar for their connection must be properly selected. For example, if the failure of a system or a component is caused by the loss of supply to an auxiliary busbar, then it is obvious that if the warning light system is fed from the same busbar warning indications will also be lost. To avoid this risk it is necessary for warning lights to be supplied from busbars different from those feeding the associated service, and preferably on or as close as possible electrically to the busbar. Caution and indicating lights may also, in some cases, be supplied in a similar manner, but usually they are supplied from the same busbar as the associated service.

MAGNETIC INDICATORS

In many types of aircraft system, components require electrical control; for example, in a fuel system, electric actuators position valves which permit-the supply of fuel from the main tanks to the engines and also for cross-feeding the fuel-supply. All such devices are, in the majority of cases, controlled by switches on the appropriate systems panel, and to confirm the completion of movement of the device an indicating system is necessary.

The indicating system can either be in the form of a scale and pointer type of instrument, or an indicator light, but both methods can have certain disadvantages. The use of an instrument is rather space-consuming particularly where a number of actuating devices are involved, and unless it is essential for a pilot or systems engineer to know exactly the position of a device at any one time, instruments are uneconomical. Indicator lights are of course simpler, cheaper and consume less power, but the liability of their filaments to failure without warning contributes a hazard particularly in the case where "light out" is intended to indicate a "safe" condition of a system. Furthermore, in systems requiring a series of constant indications of prevailing conditions, constantly illuminated lamps can lead to confusion and misinterpretation on the part of the pilot or systems engineer. Therefore to enhance the reliability of indication, indicators containing small electromagnets operating a shutter or similar moving element are installed on the systems panels of many present day aircraft.

In its simplest form, some aircraft differential pressure indicating devices consist of visual indicators or associated warning lights on aircraft instrument panel entirely built upon magnetic or mechanically latched buttons or pins that extend when the differential pressure exceeds that allowed for a serviceable element. (***Figure 8-132***) When this increased pressure reaches a specific value, inlet pressure forces the spring-loaded magnetic piston downward, breaking the magnetic attachment between the indicator button and the magnetic piston. This allows the red indicator to pop out, signifying that the element must be cleaned. The button or pin, once extended, remains in that position until manually reset and provides a permanent (until reset) warning of a loaded element. This feature is particularly useful where it is impossible for an operator to continuously monitor the visual indicator, such as in a remote location on the aircraft.

Some button indicators have a thermal lockout device incorporated in their design that prevents operation of the indicator below a certain temperature. The lockout prevents the higher differential pressure generated at cold temperatures by high fluid viscosity from causing a false indication of a loaded filter element.

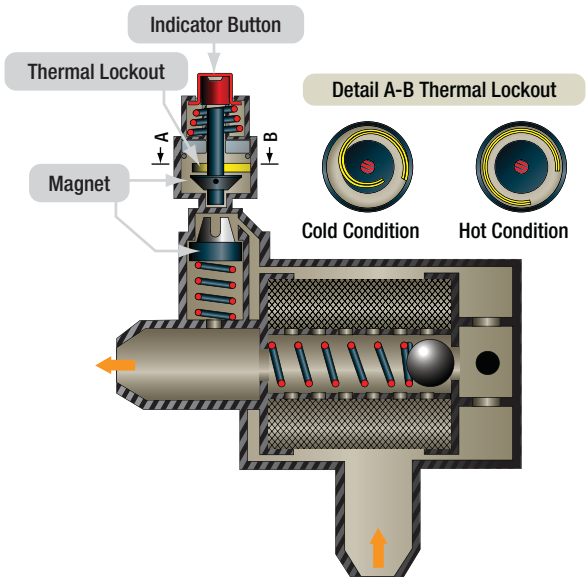


Figure 8-132. Aircraft filter bypass valve with incorporated magnetic indicator.

The pictorial presentations offered by these indicators is further improved by the painting of "flow lines" on the appropriate panels so that they interconnect the indicators with the system control switches, essential indicators and warning lights. A typical application of "flow lines" is shown in ***Figure 8-133***.

CENTRAL WARNING SYSTEMS

In the development of large types of aircraft and their associated systems, it became apparent that the use of warning and indicator lights in increasing numbers, and widely dispersed throughout flight compartments, would present a problem and that a new approach would be necessary. As a result, systems referred to as "central warning systems" were developed.

In its basic form, a system comprises a centralized group of warning and indicator lights connected to signal circuits actuated by the appropriate systems of the aircraft, each light displaying a legend denoting the system, and a malfunction or advisory message. All the lights are contained on an annunciator panel installed within a pilot's visual range.

An example of a system containing master warning and caution annunciator lights is shown in ***Figure 8-134***. The lights are centrally grouped according to systems, on a glare shield panel directly in front of the pilots and over their main instrument panels. The lights are also interconnected with systems indicating lights on an overhead control panel.

When a fault occurs in one of the systems, the overhead panel light for that system will illuminate, but as this may not always be readily observed by the pilots, their attention will be drawn to the fault situation by the simultaneous illumination of the annunciator light for the system, and of the master caution light. The lights are illuminated via a "fault pulser" and SCR circuit

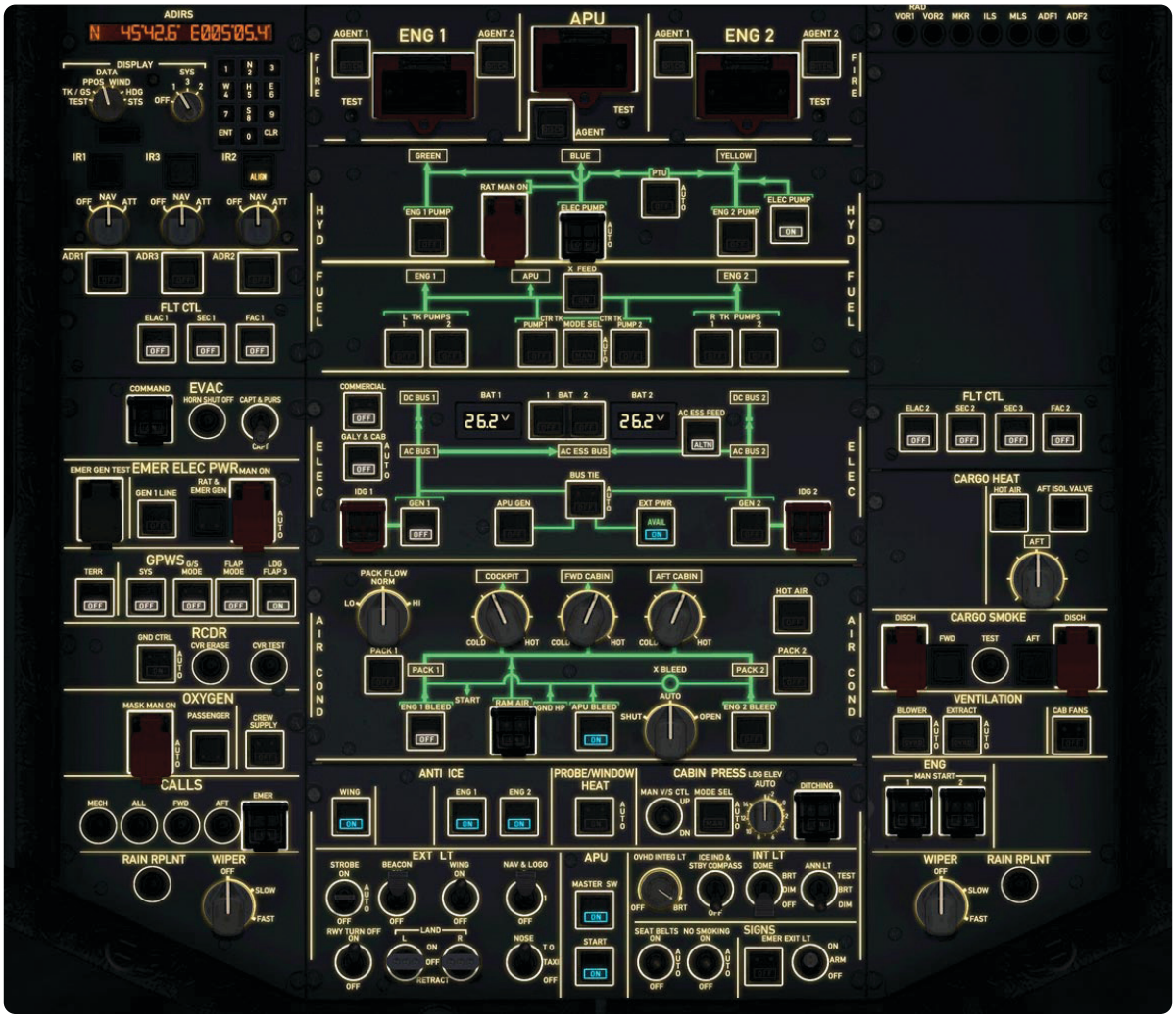


Figure 8-133. Application of "Flow Lines" on the aircraft overhead panel.



Figure 8-134. Master Caution and System Annunciator lights, left and right (Boeing 737).

arrangement Identification of the faulted system is cross-checked by observation of its control section of the overhead panel, and once this has been made, it is unnecessary for the master caution and annunciator lights to remain illuminated.

They can therefore be extinguished by pressing the cap of either master caution light. If there is a need to recall the faulted system on an annunciator panel this can be accomplished by pressing the cap of the corresponding annunciator light. If the fault is not corrected a "recall pulser" circuit will retrigger the SCR and so illuminate the system annunciator light. In aircraft carrying a flight engineer, a panel is also installed at his station and is functionally integrated with the pilot's panel.

A flight engineer's panel is illustrated in ***Figure 8-135*** and may be taken as an example of central warning displays. In this case, the panel is made up of a number of blue lights which are advisory of normal operating conditions, a number of amber lights, a red "master warning" light and an amber "master caution" light.

When a fault occurs in a system, a fault-sensing device transmits a signal which illuminates the appropriate amber light. The signal is also transmitted to an electronic device known as a logic controller, the function of which is to determine whether the fault is of a hazardous nature or is one requiring caution. If the fault is hazardous, then the controller output signal illuminates the red "master warning" light; if caution is required, then the signal will illuminate only amber "master caution" light.

Each master light incorporates a switch unit so that when the caps are pressed in, the active signal circuits are disconnected to extinguish the lights and, at the same time, they are reset to accept signals from faults which might subsequently occur in any other of the systems in the aircraft. The system lights are not of the resetting type and remain illuminated until the system fault is corrected. Dimming of lights and testing of bulb filaments is carried out by means of switches mounted adjacent to the ANNUNCIATOR panel.



Figure 8-135. Red, Amber and Green Color Coded Annunciator Lights (Cessna).

In an effort to increase the safety of operating complicated aircraft, computers and computer systems have been incorporated. Flight instrumentation and engine and airframe monitoring are areas particularly well suited to gain advantages from the use of computers. They contribute by helping to reduce instrument panel clutter and focusing the pilot's attention only on matters of imminent importance.

"Glass Cockpit" is a term that refers to the use of flat-panel display screens in cockpit instrumentation. In reality, it also refers to the use of computer-produced images that have replaced individual mechanical gauges. Moreover, computers and computer systems monitor the processes and components of an operating aircraft beyond human ability while relieving the pilot of the stress from having to do so. Computerized electronic flight instrument and maintenance systems have additional benefits. The solid-state nature of the components increases reliability. Also, microprocessors, data buses, and LCDs all save space and weight. Technicians interface with EICAS (Engine Indicating and Crew Alerting System) and ECAM (Electronic Centralized Aircraft Monitoring) systems through control panels to gather operating and maintenance data. (***Figure 5-136***)

These systems have been developed and utilized on aircraft for a number of years. New systems and computer architecture development is ongoing. Detains on the operation and use of these glass cockpit maintenance aids are located in the manufacturer's maintenance manual.

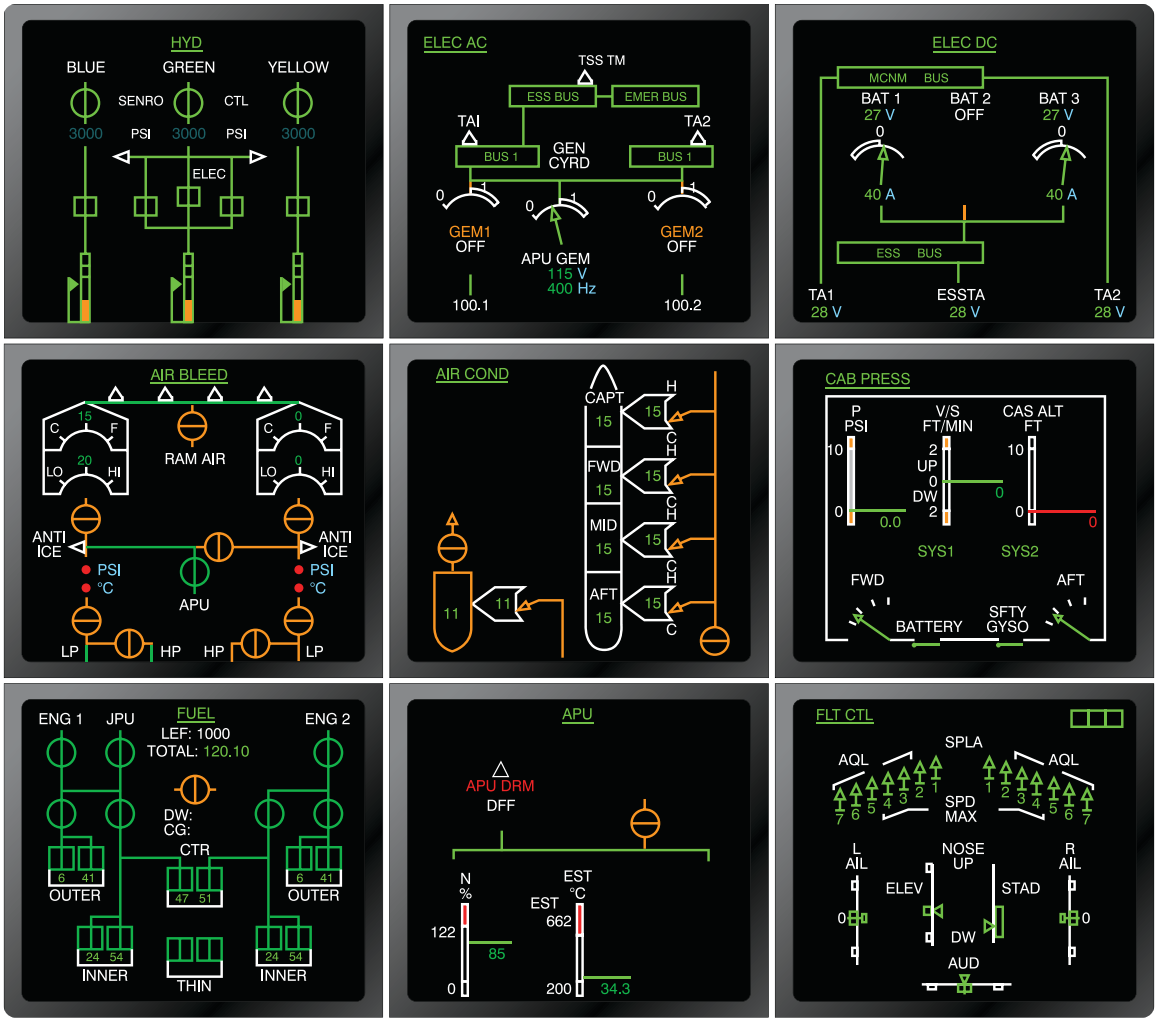


Figure 8-136. Glass cockpit displays allow system operating and maintenance data to be accessed by the technician. Nine of the 12 available system diagrams from an ECAM system are shown. The technician uses a control panel to select the desired system for display.

QUESTIONS

Question: 8-1
The three types of aircraft instruments are classified as? _____, _____, and _____ instruments.

Question: 8-2
Density of air is measured as? _____ per unit _____. It is _____ proportional to pressure, and _____ proportional to temperature.

Question: 8-3
Absolute pressure is pressure relative to a near perfect _____. Differential pressure is the _____ between _____ pressure sources.

Question: 8-4
Piezoelectric devices, when subject to _____ forces, generate a _____?

Question: 8-5
Servo assisted altimeters, are _____ accurate than basic capsule altimeters. They incorporate an _____ and a _____ bar as a transducer to convert the capsule movement into an _____ _____.

Question: 8-6
A Mach-meter measures the aircrafts speed relative to the speed of _____. It has the basic components of an airspeed indicator with the addition of an additional _____ to determine the _____ so that Mach speed can be determined.

Question: 8-7
A thermocouple is a circuit consisting of _____, _____ metals? It has a _____ and a _____ junction. They are used to measure _____ temperatures.

Question: 8-8
Large aircraft use a _____ type fuel measuring system using probes connected in _____. As the fuel level varies the _____ of the probes varies which accurately determines the fuel state with the aid of an _____.

ANSWERS

Answer: 8-1
flight, engine, navigation.

Answer: 8-2
mass; volume; directly; inversely.

Answer: 8-3
vacuum; difference; two.

Answer: 8-4
pressure; voltage.

Answer: 8-5
"E"; "I"; electric current.

Answer: 8-6
sound; diaphragm; altitude.

Answer: 8-7
two; unlike; hot; cold; high.

Answer: 8-8
capacitive; parallel; di-electric; amplifier.

Question: 8-9
Old gyroscopic instruments used a _____ source to provide rotational energy; most modern mechanical types are _____. Gyros have two inherent characteristics called? _____ and _____. Ring laser gyros utilize _____ rays traveling in _____ directions around a _____ to determine the _____ of change of the _____. There are _____ cavities, one for each of the aircraft's _____.

Question: 8-10
In relation to a Turn and Bank indicator, a Turn Coordinator has its rotor gimbal inclined at about _____ degrees to the _____ so that the instrument will respond to movements not only about the _____ axis, but also to movements about the _____ axis. It will detect changes in _____ and _____ attitudes

Question: 8-11
A GPWS has _____ basic modes. EGPWS has an additional feature to provide _____, _____ to the pilot. The aircraft's current position is compared to a terrain _____ that is programmed into the unit. Different _____ on the display indicate different _____ relative to the aircraft's _____.

Question: 8-12
Remote compass systems use a _____, _____ to detect the earth's _____ field.

Question: 8-13
Modern Flight Data Recorders are solid _____. They record both _____ and _____ with much _____ memory space than earlier models – so they have more capacity.

Question: 8-14
EFIS displays provide _____ information than the space used by conventional instruments, and are more _____. Automatic switching between displays of priority data and information is provided in the event of individual display _____.

Question: 8-15
The AOA is the angle formed between the wing _____, _____ and the _____ air flow. A large aircraft stall warning system uses a _____ to warn the pilot of an impending stall.

Question: 8-16
Modern vibration detectors are _____ devices.

ANSWERS

Answer: 8-9
vacuum; electric; rigidity; precession; light; opposite; cavity; rate; frequency; three; axes.

Answer: 8-10
thirty; horizontal; normal; longitudinal; yaw; roll.

Answer: 8-11
seven; terrain; awareness; database; colors; altitudes; altitude.

Answer: 8-12
flux; valve; magnetic.

Answer: 8-13
state; voice; data; larger.

Answer: 8-14
more; reliable; failure.

Answer: 8-15
chord; center-line; relative; stick-shaker.

Answer: 8-16
piezo-electric.



PART-66 SYLLABUS **LEVELS**

CERTIFICATION CATEGORY → **B2**

Sub-Module 09

LIGHTS (ATA 33)

Knowledge Requirements

13.9 - Lights (ATA 33)	
External: navigation, landing, taxiing, ice; Internal: cabin, cockpit, cargo; Emergency.	3

AIRCRAFT LIGHTING SYSTEMS

Aircraft lighting systems provide illumination for both exterior and interior use. Lights on the exterior provide illumination for such operations as landing at night, inspection of icing conditions, and safety from midair collision. Interior lighting provides illumination for instruments, cockpits, cabins, and other sections occupied by crew-members and passengers. Certain special lights, such as indicator and warning lights, indicate the operation status of equipment.

EXTERIOR LIGHTS

Position, anti-collision, landing, and taxi lights are common examples of aircraft exterior lights. Some lights are required for night operations. Other types of exterior lights, such as wing inspection lights, are of great benefit for specialized flying operations.

NAVIGATION / POSITION LIGHTS

Aircraft operating at night must be equipped with position lights that meet minimum requirements. A set of position lights consist of one red, one green, and one white light. (***Figures 9-1 and 9-2***)



Figure 9-1. A left wing tip position light (red) and a white strobe light.



Figure 9-2. A right wing tip position light, also known as a navigation light.

On some types of installations, a switch in the cockpit provides for steady or flashing operation of the position lights. On many aircraft, each light unit contains a single lamp mounted on the surface of the aircraft. Other types of position light units contain two lamps and are often streamlined into the surface of the aircraft structure. The green light unit is always mounted at the extreme tip of the right wing. The red unit is mounted in a similar position on the left wing. The white unit is usually located on the vertical stabilizer in a position where it is clearly visible through a wide angle from the rear of the aircraft. ***Figure 9-3*** illustrates a schematic diagram of a position light circuit. Position lights are also known as navigation lights.

There are, of course, many variations in the position light circuits used on different aircraft. All circuits are protected by fuses or circuit breakers, and many circuits include flashing and dimming equipment. Small aircraft are usually equipped with a simplified control switch and circuitry. In some cases, one control knob or switch is used to turn on several sets of lights; for example, one type utilizes a control knob, the first movement of which turns on the position lights and the instrument panel lights. Further rotation of the control knob increases the intensity of only the panel lights. A flasher unit is seldom included in the position light circuitry of very light aircraft but is used in small twin-engine aircraft. Traditional position lights use incandescent light bulbs. LED lights have been introduced on modern aircraft because of their good visibility, high reliability, and low power consumption.

ANTI-COLLISION LIGHTS

An anti-collision light system may consist of one or more lights. They are rotating beam lights that are usually installed on top of the fuselage or tail in such a location that the light does not affect the vision of the crew member or detract from the visibility of the

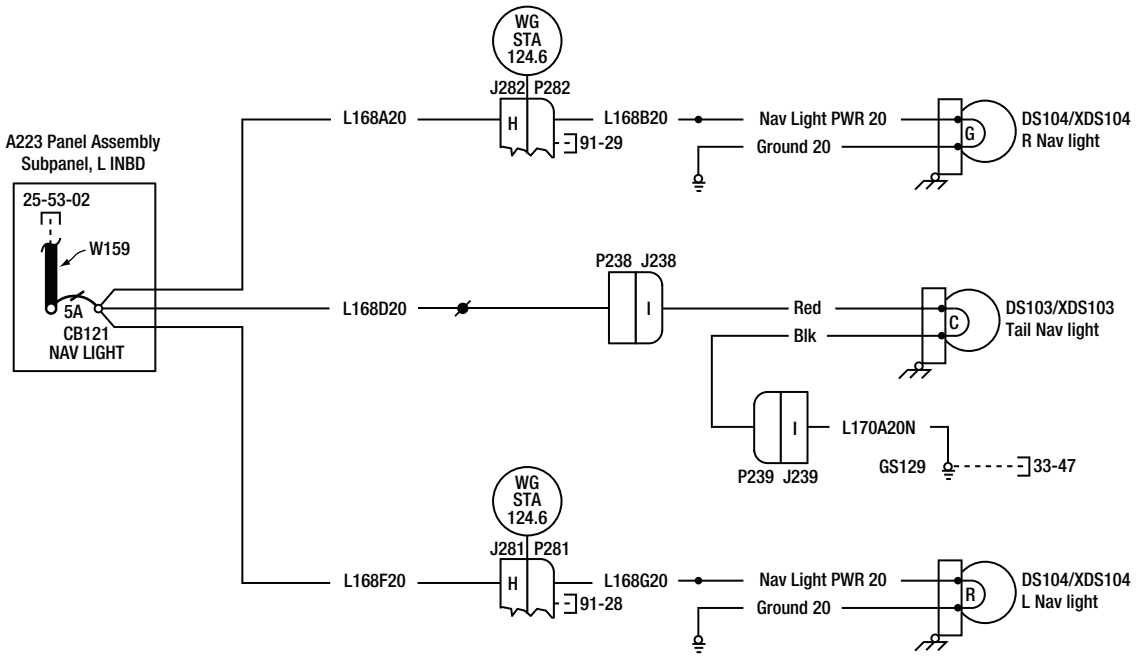


Figure 9-3. Navigation light system schematic.

position lights. Large transport type aircraft use an anti-collision light on top and one on the bottom of the aircraft. ***Figure 9-4*** shows a typical anti-collision light installation in a vertical stabilizer.

An anti-collision light unit usually consists of one or two rotating lights operated by an electric motor. The light may be fixed but mounted under rotating mirrors inside a protruding red glass housing. The mirrors rotate in an arc, and the resulting flash rate is between 40 and 100 cycles per minute. Newer aircraft designs use a LED type of anti-collision light. The anti-collision light is a safety light to warn other aircraft, especially in congested areas.

A white strobe light is a second type of anti-collision light that is also common. Usually mounted at the wing tips and, possibly, at empennage extremities, strobe lights produce an extremely bright intermittent flash of white light that is highly visible. The light is produced by a high voltage discharge of a capacitor. A dedicated power pack houses the capacitor and supplies voltage to a sealed xenon-filled tube. The xenon ionizes with a flash when the voltage is applied. A strobe light is shown in ***Figure 9-1***.

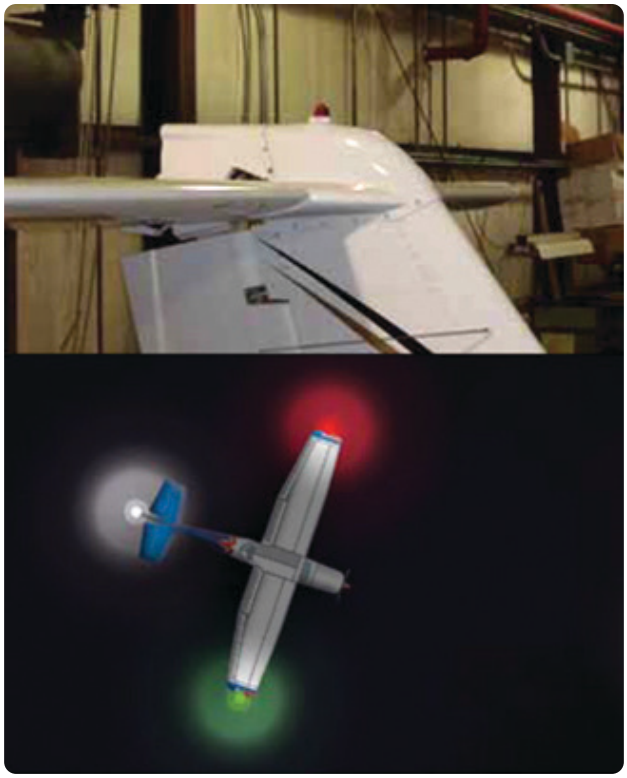


Figure 9-4. Anticollision lights.

LANDING LIGHTS

Landing lights are installed in aircraft to illuminate runways during night landings. These lights are very powerful and are directed by a parabolic reflector at an angle providing a maximum range of illumination. Landing lights of smaller aircraft are usually located midway in the leading edge of each wing or streamlined into the aircraft surface. Landing lights for larger transport category aircraft are usually located in the leading edge of the wing close to the fuselage.

Each light may be controlled by a relay, or it may be connected directly into the electric circuit. On some aircraft, the landing light is mounted in the same area with a taxi light. (***Figure 9-5***) A sealed beam, halogen, or high intensity xenon discharge lamp is used.



Figure 9-5. Landing lights.

TAXI LIGHTS

Taxi lights are designed to provide illumination on the ground while taxiing or towing the aircraft to or from a runway, taxi strip, or in the hangar area. (***Figure 9-6***) Taxi lights are not designed to provide the degree of illumination necessary for landing lights. On aircraft with tricycle landing gear, either single or multiple taxi lights are often mounted on the non-steerable part of the nose landing gear. They are positioned at an oblique angle to the center line of the aircraft to provide illumination directly in front of the aircraft and also some illumination to the right and left of the aircraft's path. On some aircraft, the dual taxi lights are supplemented by wingtip clearance lights controlled by the same circuitry. Taxi lights are also mounted in the recessed areas of the wing leading edge, often in the same area with a fixed landing light.

Many small aircraft are not equipped with any type of taxi light, but rely on the intermittent use of a landing light to illuminate taxiing operations. Still other aircraft utilize a dimming resistor in the landing light circuit to provide reduced illumination for taxiing. A typical circuit for taxi lights is shown in ***Figure 9-7***. Some large aircraft are equipped with alternate taxi lights located on the lower surface of the aircraft, aft of the nose radome. These lights, operated by a separate switch from the main taxi lights, illuminate the area immediately in front of and below the aircraft nose.



Figure 9-6. Taxi lights.

WING ICE INSPECTION LIGHTS

Some aircraft are equipped with wing inspection lights to illuminate the leading edge of the wings to permit observation of icing and general condition of these areas during flight. These lights permit visual detection of ice formation on wing leading edges while flying at night. They are usually controlled through a relay by an on/off toggle switch in the cockpit.

Some wing inspection light systems may include or be supplemented by additional lights, sometimes called nacelle lights, that illuminate adjacent areas, such a cowl flaps or the landing gear. These are normally the same type of lights and can be controlled by the same circuits.

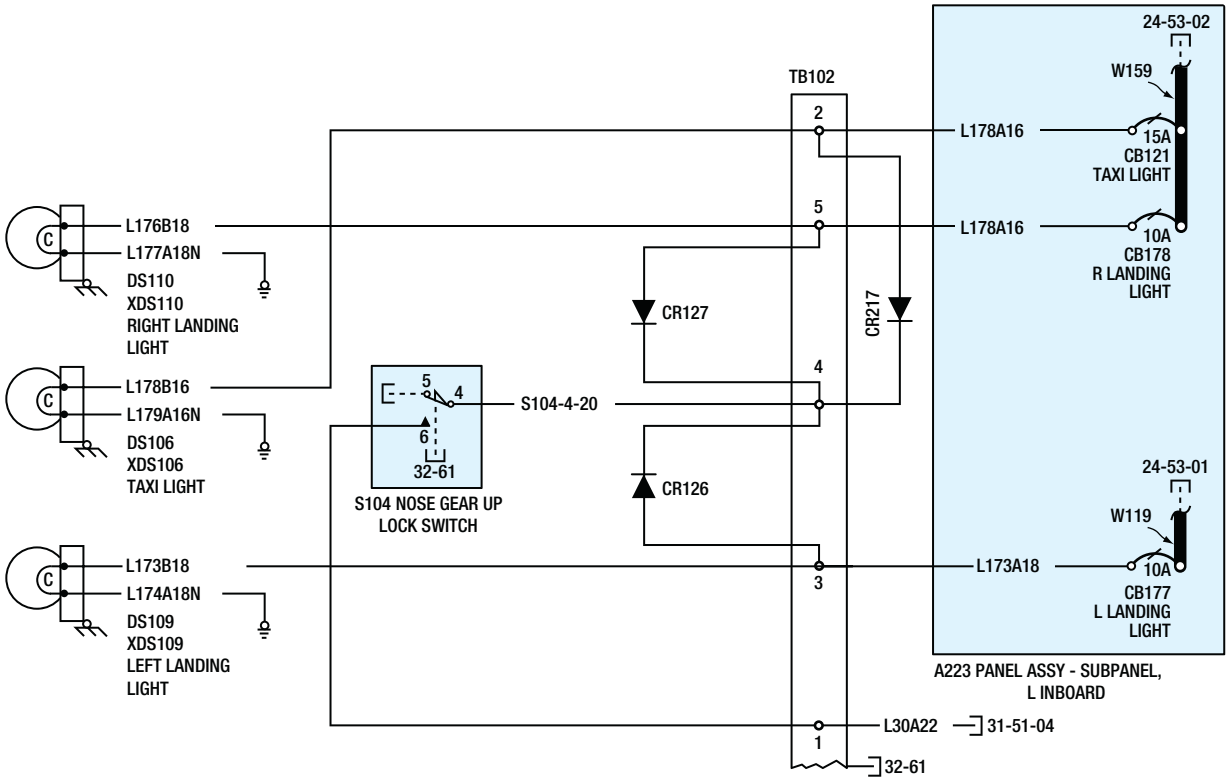


Figure 9-7. Taxi light circuit.

INTERIOR LIGHTS

Aircraft are equipped with interior lights to illuminate the cabin. (***Figure 9-8***) Often white and red light settings are provided on the flight deck. Commercial aircraft have a variety of independent lighting systems that illuminate the flight deck, instrument panels, passenger cabin, cargo compartments and more. Interior lights incorporate the use of both incandescent and fluorescent lights that operate off a variety of AC and DC electrical buses.

FLIGHT DECK/CONTROL CABIN LIGHTING

On an airliner flight deck, it is normal to have lighting for general illumination of the control cabin as well as local lighting for panels, instruments and controls. Fluorescent background lights are also used. A centrally located panel, typically an overhead panel, houses the controls for many interior and exterior lights. Independent light controls may also be located on appropriate panels.

Normal lighting requirements are met using 28 VAC power with key lights positioned for part-power and no power situations typically run off a 28 VDC bus. On Boeing Aircraft, 115 VAC is used for fluorescent lights.

PASSENGER CABIN LIGHTS

Independent lighting systems are used in the passengers cabin. A combination of incandescent and fluorescent lights in overhead and window positions provide general illumination. (***Figure 9-9***) These normally use AC power. Threshold and doorway entry lights are used as well as a variety of illuminated information signs. Galley and lavatories have their own lighting circuits. Overhead passenger service units (PSUs) in each seat row contain independent reading lights and service call lights for each seat. On the most modern aircraft such as the Boeing 777, the myriad of lights in the passenger compartment are controlled by a central control unit such as the Cabin Service System (CSS). Interface panels are located for easy access by the cabin crew. A menu provides numerous options for not only which lights are illuminated but also light intensity.

CARGO COMPARTMENT LIGHTING

Cargo and service compartments also have lighting. Dome lights, flood lights and explosion-proof lights as required are installed with independent circuits protected by circuit breakers. The lights are controlled by switches near the entrance to each area or inside the compartments.



Figure 9-8. Interior cockpit and cabin light system.



Figure 9-9. Incandescent and fluorescent lights are used to illuminate modern aircraft cabins.

Often, a control panel for a cargo area includes light switches in addition to door and cargo system operating controls. Sidewall, overhead and door mounted lights are common. Door and door sill lights are positioned so that they illuminate the cargo compartment doorway as well as the area just outside the compartment to facilitate work while loading cargo.

EMERGENCY LIGHTING

Emergency lights are installed in the cabin to illuminate escape routes for passengers and crew during a failure of AC power systems. Lighting strips in the floor and exit lights automatically illuminate when power is lost. (***Figure 9-10***) Emergency lights are used to illuminate the over-wing area at the emergency exits and on the escape slides. Lavatories and the control cabin also have emergency lighting.

Various configurations exist for automatic switching of certain emergency lights to the hot DC battery bus (or similar) in case of partial electrical failure. Some interior lighting is designed to always be connected to a DC bus so no switching is required. Total electrical failure causes most emergency lights to revert to dedicated batteries that are an integral part of the lighting installation. Emergency EXIT/area lights for example, may contain a battery in the assembly that includes the lamps, cover lens, solid state switching logic and battery-charging control circuits. In some cases, the light/battery assembly can be removed from its mounted location and used as a portable flashlight. NiCad batteries are typical. In other configurations, the dedicated emergency light battery is remotely located in the same area as the light.

Regardless, emergency lighting is ARMED by a switch on the flight deck or at the passenger cabin lighting control panel. The all-in-one emergency light assemblies also have a switch that must be set to ARMED when the unit is installed. Inspection of an aircraft's emergency lighting system normally includes checking the condition and security of all visible wiring, connections, terminals, fuses, and switches and light units. A continuity light or meter can be used in making these checks, since the cause of many troubles can often be located by systematically testing each circuit for continuity.

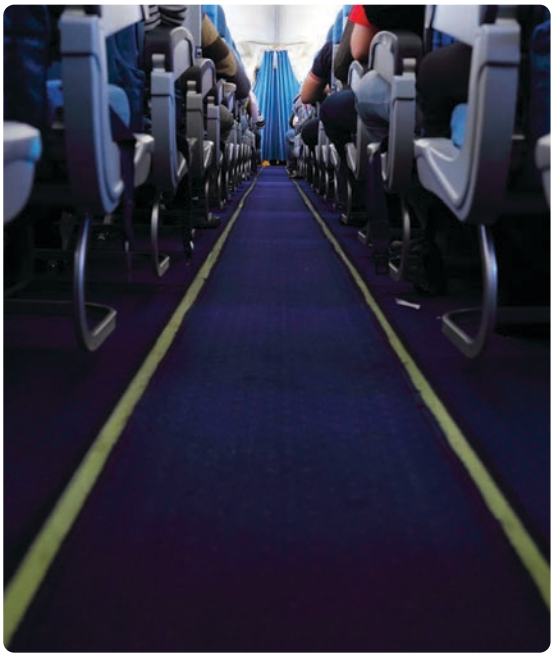


Figure 9-10. Emergency light strips in the aisle floor guide passengers and crew to the exits in case of emergency.

Question: 9-1
Another name for a "navigation light" is a _____ light.

Question: 9-2
A _____ navigation light is always mounted on the extreme tip of the right wing.

Question: 9-3
A rotating light on the exterior of an aircraft is a(n) _____ light.

Question: 9-4
Typically, aircraft have _____ lights because landing lights are too bright and focused at an angle beneficial for seeing the runway from the air.

Question: 9-5
A light that illuminates the leading edge of the wing is used to observe _____ as well as the general condition in this area during flight.

Question: 9-6
Cabin lighting in normal situations is typically powered by _____ current.

Question: 9-7
Cargo compartment lighting is controlled from _____.

Question: 9-8
Cabin exit emergency lights often contain integral _____.

ANSWERS

Answer: 9-1
position.

Answer: 9-2
green.

Answer: 9-3
anti-collision light.

Answer: 9-4
taxi.

Answer: 9-5
ice.

Answer: 9-6
alternating.

Answer: 9-7
a control panel near the entrance to the compartment.

Answer: 9-8
battery.
solid state switching logic circuits.
battery charging circuits.
operational arming switches.



PART-66 SYLLABUS **LEVELS**

CERTIFICATION CATEGORY → **B2**

Sub-Module 10

ON BOARD MAINTENANCE SYSTEMS (ATA 45)

Knowledge Requirements

13.10 - On Board Maintenance Systems (ATA 45) — 3

Central maintenance computers;
Data loading system;
Electronic library system;
Printing;
Structure monitoring (damage tolerance monitoring).

INTRODUCTION

On board maintenance systems (also known as central maintenance systems (CMS)) are electronic/computer based systems used to monitor and record the performance status of systems on-board modern aircraft. The structure and configuration of on board maintenance systems vary from one aircraft type to another. However, the core functions of these systems are broadly similar. They monitor the aircraft for faults, record and store the fault data, and provide information about these faults to flight crews and maintenance personnel.

The data collected by on board maintenance systems can be accessed both in flight and on the ground. In flight, the system advises the flight crew of faults that may affect aircraft operation. On the ground, maintenance crews use the system for testing and troubleshooting purposes. In some applications, the aircraft can relay fault information to the ground whilst in flight.

On board maintenance systems also store procedural data contained in maintenance manuals, troubleshooting manuals, flight manuals, and other publications. The systems allow maintenance personnel to access these publications without having to carry books and papers to the aircraft. On board maintenance systems allow such technical data to be uploaded, downloaded, viewed, and printed by maintenance personnel. The systems are used for both line and base maintenance.

For the most part, the information provided here is general. When aircraft specific information is given, it should be noted that system details and terminology differ between aircraft manufacturers.

CENTRAL MAINTENANCE SYSTEMS (CMS)

The primary function of a CMS is to initiate airplane system tests, record and store test results, monitor airplane systems status, recall test results from previous flight legs, upload/download data and software updates, isolate and identify faults. Furthermore, a CMS may interface with other on-board computers to provide data for display to flight crew, initiate warnings and cautions, and to enable or disable some CMS functions depending on the flight phase or configuration of the airplane.

A typical CMS consists of single or dual central maintenance computers connected to a printer, data loader, a data reporting/transmission system and multifunction control display units (MCDU). ***Figure 10-1*** illustrates the structure of a typical CMS.

CENTRAL MAINTENANCE COMPUTER (CMC)

The central maintenance computer (CMC) is the main processing unit of the CMS. The CMC acquires, processes and outputs data to and from various airplane systems. These inputs and outputs may be digital, analogue, in a particular format (such as ARINC 429) and discrete (of a specific value such as a voltage, current, pressure, etc). Typical inputs to a CMC would originate from airplane system built in test results initiated by the CMS, airplane system operating modes and configuration status, airplane flight modes, and commands from flight and maintenance crews.

An input from the airplane's landing gear 'weight on wheels' switch would indicate to the CMC that the airplane is either airborne or on the ground. This information is stored to record the flight phase of the airplane and is used to restrict or inhibit particular tests that may be requested via the CMC.

Outputs from the CMC are provided for flight deck visual and aural displays (EFIS, EICAS/ECAM), to initiate airplane system built in test equipment (BITE), data transfer to other storage media such as remote memory devices, printers, transmission via airplane condition and reporting systems (ACARS) and data

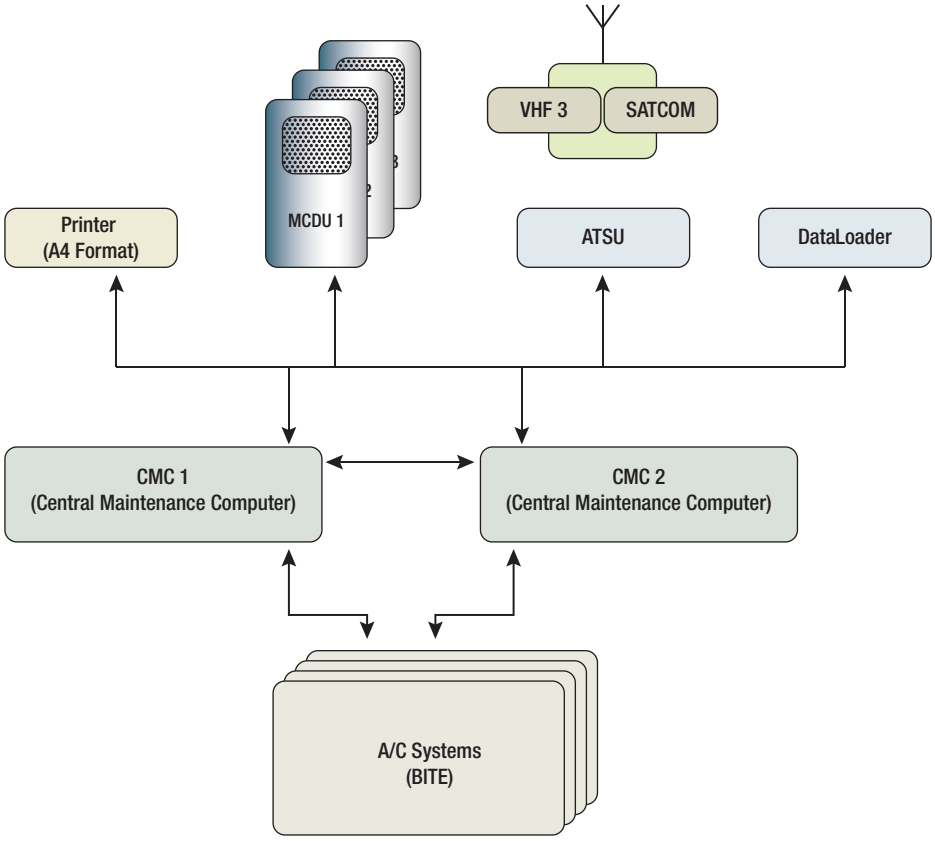


Figure 10-1. Typical central maintenance system.

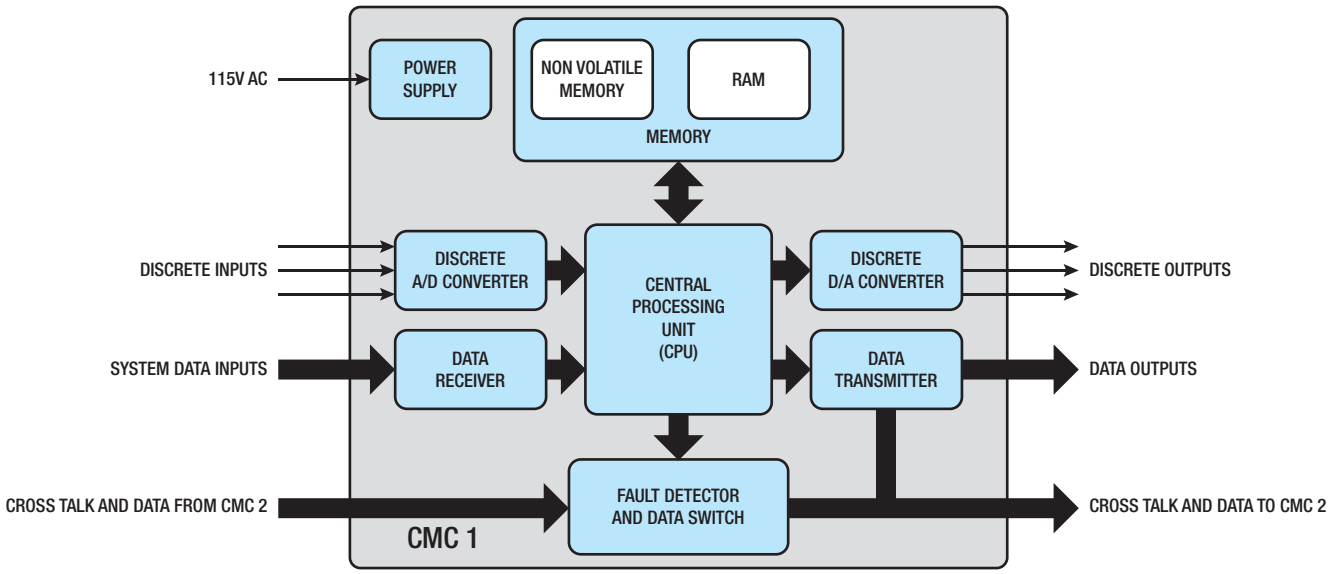


Figure 10-2. Central maintenance computer schematic.

communication systems (VHF, HF or SATCOM). In order to perform these functions a CMC consists of a number of modules or circuits which are typical in any computer system. The schematic diagram in ***Figure 10-2*** shows the modules needed to support the main processor within the CMC.

The power supply module receives the airplane power supply voltage (usually 115V AC) and provides the appropriate regulated DC voltages to the logic circuits in the CMC. Operating software and fault history are stored in the non-volatile memory (NVM). Approximately 500 faults can be stored in the NVM

for later retrieval by maintenance crews. The random access memory is used to store active faults during the current flight leg. Discrete inputs and outputs are used to convey BITE status to the CMC or to initiate a BITE sequence in an airplane system. In each case it may be necessary to convert the discrete signals from analogue to digital (A/D conversion) formats to allow the CPU to process information or from digital to analogue (D/A conversion) to activate a system test sequence or discrete indicator such as a caption warning LED.

Data receivers multiplex data inputs from the airplane systems and convert data streams from serial to parallel. The transmitters convert processed data from the CPU into a serial output for transmission to airplane systems and other CMS components. In modern airplane systems ARINC 429 is the most common data bus used to interconnect airplane systems to the CMC.

It is common, particularly on large commercial airplanes, to have a dual CMC installation in the CMS. With dual CMCs, all data inputs are available to both units. One CMC will be "active" at any given time with the other CMC on "standby." The active CMC is the one providing outputs. If the active CMC should fail, the standby can be substituted for it either automatically or manually by the crew.

A "crosstalk" bus enables the two CMC's to share and monitor faults within each CMC. A fault monitoring circuit within each CMC controls a "data switch" which, upon detecting a fault in its own CMC, isolates the defective CMC and allows the other CMC's data to be passed directly through the faulty unit. A discrete input to the CMC from a manually operated switch in the flight deck enables the crew to manually select the other CMC.

The central processing unit (CPU) performs all of the arithmetic and control functions within the CMC. By comparing outputs from airplane systems with expected values based on known inputs and the airplanes configuration, attitude and flight phase, the CMC will generate a fault message/code which is then notified to the crew, when necessary, and recorded. A CMC can generate up to 10 000 different fault codes.

The CMC can be a self-contained line replaceable unit (LRU) installed in its own mounting rack in an avionics compartment as found in Airbus or Boeing airplanes or as a removable printed circuit board (PCB)/card unit installed in a card rack such as the integrated avionics processing system (IAPS) in the Rockwell Collins Proline system.

FAULT CLASSIFICATION

Faults occurring in aircraft systems either on the ground or in flight may, or may not, affect the performance and/or capability of the aircraft. Some faults would require immediate action by the crew whereas others may require crew awareness or no crew involvement at all. Clearly, some faults need to be brought to the attention of the crew more rapidly than others and therefore must be classified in order of priority, displayed and recorded as such. A system of warnings, cautions and advisories is adopted on most monitoring and display systems.

Warnings are the highest priority fault classification and are usually displayed as a RED visual indication accompanied by a continuous aural warning to the crew. These indicate that the fault requires an immediate action from the crew and will continue to be displayed whilst the fault remains.

Cautions are usually displayed as an AMBER visual display, accompanied by a single aural tone indicate that the fault requires immediate crew awareness but no immediate action. Advisories, displayed as a WHITE visual indication, have no accompanying aural tone and require general crew awareness.

A CMS records these faults in accordance with the classification of a Level A (Warnings), Level B (Cautions) or Level C (Advisories). Airbus Industries adopt a similar classification system for the Airbus range of aircraft where warnings and cautions are a Class 1 fault, Class 2 is an advisory or system status notification and a Class 3 fault is not displayed to the crew but is recorded by the CMS for retrieval by maintenance crews on the ground.

BUILT-IN TEST EQUIPMENT (BITE)

Units that are monitored by the CMC may contain circuits known as built in test equipment (BITE). BITE is installed in many systems throughout the aircraft including navigation systems, flight control systems, environmental control systems, and others. Within each system, the BITE circuitry tests numerous individual parameters to determine whether the system is functioning properly. The individual system BITE circuits are connected to the CMC by a digital data bus. ARINC 429 buses are used for this purpose in many aircraft. Other data buses, such as ARINC 629, may also be used.

Typical BITE functions initiated by the CMC serve to detect the fault, isolate the faulty unit and record the fault parameters. The level of test performed will depend on the flight phase of the aircraft, system configuration and complexity. Generally, tests fall under a number of categories the most common of which are;

- Initial/Power up Test – Performed immediately at system switch on or following a power interrupt. The BITE will check the functionality of system circuits (power supplies, processors, input/output ports, etc). System software may also be reset or loaded during this test. If a power interrupt were to occur in flight this test may only be restricted to essential checks only.
- Cyclic/Continuous Test – These tests may occur frequently (once or twice a second) whilst the system is in operation. These tests do not affect the performance of the system and are primarily a monitoring function.
- Interruptive/Maintenance Test – These are extensive tests performed on the ground to aid troubleshooting. These usually involve the simulation of an input signal to a system and or component and monitoring the output for its accuracy or appropriate operation. This may require moving surfaces to be actuated to various positions during the course of the test.

If any fault is detected by the BITE during these tests, an output is generated and sent to the CMS. If anything that is being monitored fails, BITE will alert the CMC automatically.

Crews can initiate the BITE power-up check for a given system from the CMS at any time. This capability is provided as a CMS menu item on the MCDU. This function can be useful when troubleshooting the system. Some LRUs containing BITE have indicator lights that indicate the status of the LRU. Green lights indicate a normal condition, red lights indicate that the BITE detected a fault in the LRU. Amber lights indicate that an input, from an external source, to the LRU has been lost or is corrupted. ***Figure 10-3*** shows an LRU with BITE indicators. BITE systems also have the capability of storing fault history. The history is kept in non-volatile memory. Non-volatile memory retains stored data even after the system has been powered off.

In order that the BITE does not adversely affect the operation of the aircraft at any given flight phase the CMS receives inputs from flight guidance computers (FGC) and other system modes/configurations such as, landing gear up/down and aircraft weight on wheels switching. These inputs to the CMC determine the level of test initiated by the CMC and performed by BITE. Some tests are completely disabled during flight and others are limited to non-interruptive tests.



Figure 10-3. Built In Test Equipment (BITE) Indicators.

MULTIFUNCTION CONTROL DISPLAY UNIT (MCDU)

CMCs are accessed via control units in the flight deck, multifunction control display units (MCDU) are found on most medium to large airplanes in service today. ***Figures 10-4 and 10-5*** show a typical MCDU and locations on the center pedestal of a typical large airplane. The number of MCDUs fitted may vary due to the size and complexity of the airplane. Smaller airplanes may have a single MCDU for pilot use, whereas larger airplanes would have two (pilot and co-pilot) or three with one acting as a standby or maintenance use.

The MCDUs enable the user to navigate the on board maintenance system by selecting from various on-screen menus. Navigating the menus allow the user to view and select current/present faults, previous faults, upload and download databases, and initiate system BITEs. The user can also view component and database information such as, part numbers, serial numbers and database versions.

Current faults are, of course, important for determining the aircraft's current status prior to dispatch or when troubleshooting. Fault history is used to monitor fault trends, such as recurring failures of a particular component. In addition, a CMC menu permits the user to check the current status of individual systems, even if no fault condition is present.

Figure 10-6 shows an example of navigating the central maintenance system menus on an MCDU.

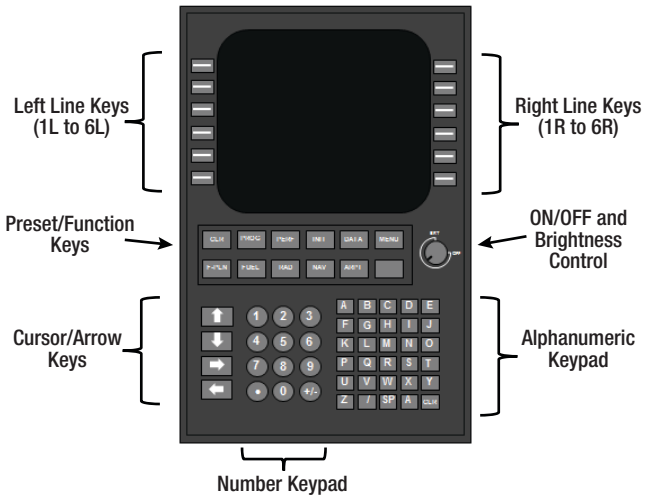


Figure 10-4. Multifunction Control Display Unit.

Menus are displayed on the screen and the user uses the line keys on the left and/or right to select the menu item. In the example shown, there are three faults: A bleed air system fault, and anti-ice system fault, and an electrical system fault. The menus allow the user to access specific information about each fault that was sensed, such as the date and time the fault occurred.

In addition to the cockpit-mounted control units, some on board maintenance systems allow for the connection of a remote device such as a laptop computer. When the laptop computer is connected, it can be used to access the data stored in the CMC. Present fault reports and previous fault reports can be downloaded to the remote device.

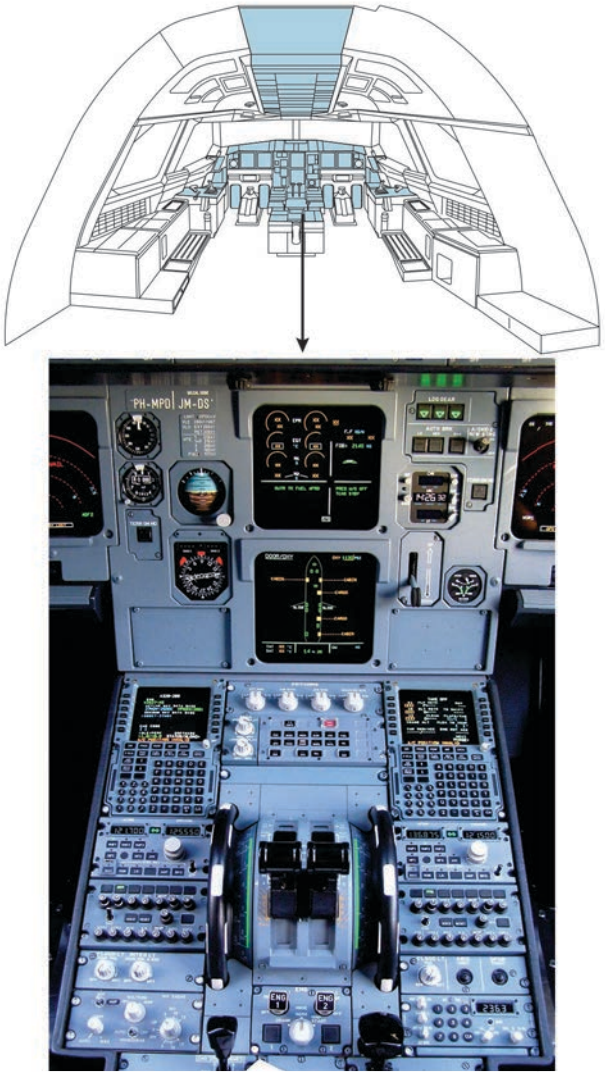


Figure 10.5. Typical flight deck center pedestal.

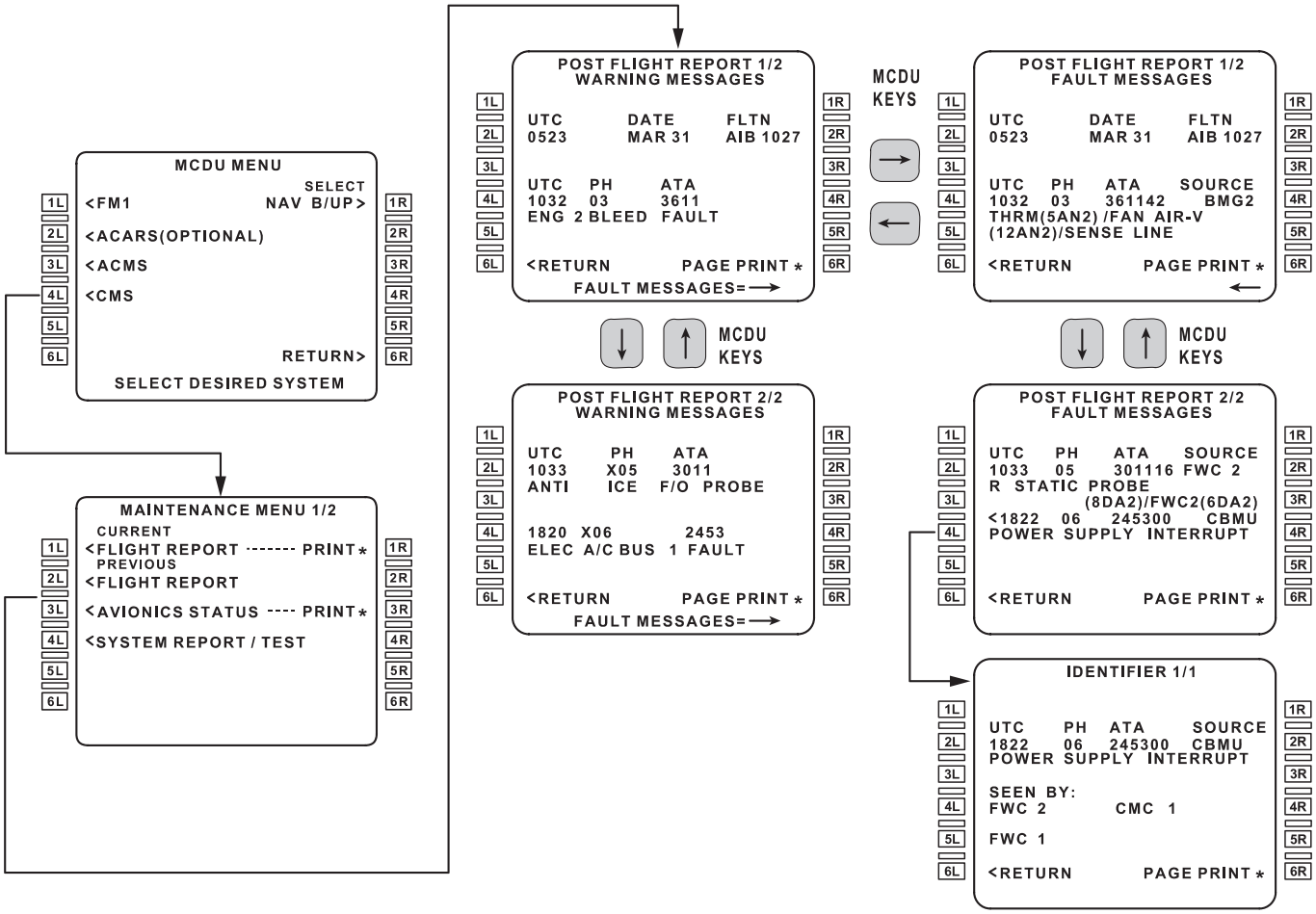


Figure 10-6. Maintenance Menus on the A330.

AIRCRAFT COMMUNICATIONS ADDRESSING AND REPORTING SYSTEM (ACARS)

Data from the CMS can be relayed to ground stations via an on-board data management system such as an aircraft communications and reporting system (ACARS) or an air traffic information management system (ATIMS). These systems utilise a central management unit (CMU) to gather and concentrate data concerning aircraft position, attitude, speed, fuel status, etc, as well as data from the CMS. This data is then 'packaged' and sent to the airplane operators ground operation stations where the information may be analysed and the performance of the airplane be monitored.

Airplane fault data relayed to the ground station can be used to prepare the maintenance crews with sufficient information to allow them to make a diagnosis before the airplane arrives at its destination where tooling and spare components will have been made ready to fix the airplane.

The ACARS data is transmitted to the ground using VHF communications radio, HF radio or satellite communications (SATCOM). Worldwide, a network of ground stations is able to communicate digitally with aircraft using the system. ***Figure 10-7*** shows the ACARS system network philosophy. ACARS is a stand-alone system and is used to transmit data from any number of systems on board the airplane. It can be used by the flight crew to send messages manually, and automatically send reports. This mode of operation is referred to as 'Demand' mode. Data can also be requested from the ground stations where an ACARS data transmission may be 'triggered' by a signal from the ground. This mode of operation is referred to as 'Polled' mode. When in demand mode ACARS will automatically send messages containing data relating to any condition that affects the flight safety of the airplane.



Figure 10-7. Aircraft communications addressing and reporting system network.

DATA LOADING SYSTEMS

An aircraft's data loading system provides a means to upload data to, and download data from various airplane systems, including the CMS. The data loading system can be used with any digital system that requires data uploads and downloads while installed in the aircraft. Early data loading systems used floppy disks as the data storage medium. An example of this is the multipurpose disk drive unit (MDDU) used on many Airbus Industrie airplanes. The MDDU uses 3.5 inch floppy disks for uploading, downloading, and data storage. In the Airbus system, a Data Loader Selector (DLS) is used to select the system where data should be loaded to or from. This is enabled by a data router installed in the system, the data router ensures that data is transferred to the correct computer from the MDDU as selected by the DLS. A typical data loading panel which incorporates a MDDU and DLS is shown in ***Figure 10-8***.

A DLS may be found on the flight deck overhead panel or on the center pedestal. An MDDU may be installed on the center pedestal or on the co-pilots side panel.

On the Boeing 777, data loading is accomplished through a maintenance access terminal (MAT) on the flight deck. ***Figure 10-9*** shows the MAT.

Data loading systems also allow for the use of other forms of storage media. Newer systems can be connected to a laptop computer through a USB (universal serial bus) cable. A CD-ROM disk, or a USB memory stick or "flash drive" may also be used. In some aircraft, there are multiple locations to connect external devices to the data loading system. For example, the 777 has two laptop maintenance access terminal interfaces. One is located on the flight deck, and one is located in the main equipment center below the flight deck.

The primary uses for the data loading system are the uploading of program updates, the uploading of database updates, and the downloading of reports. An example of a unit requiring program updates is the central maintenance computer, which contains an operating program that is upgraded from time to time. The program upgrades to the CMC are input through the data loading

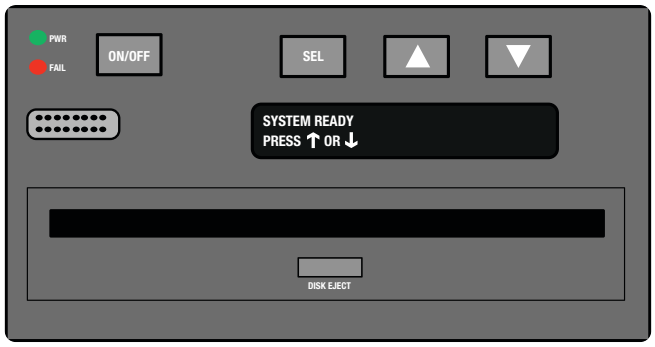


Figure 10-8. Data loading panel.

system. The same is true for other aircraft systems with internal programming. The number of systems that require program updating varies from aircraft to aircraft.

An example of a database that requires updating is the navigation database which forms a part of the flight management system (FMS). The navigation database contains a great deal of information used by the flight crew. This includes the locations of airports, airways, waypoints, and intersections, the locations and frequencies of radio navigation aids, and other information needed to create and follow a flight plan. Because changes to this information occur from time to time, the navigation database requires periodic updates. These updates are uploaded through the data loading system. The standard frequency for navigation database updates is every 28 days. ***Figure 10-10*** shows examples of navigation database update software.

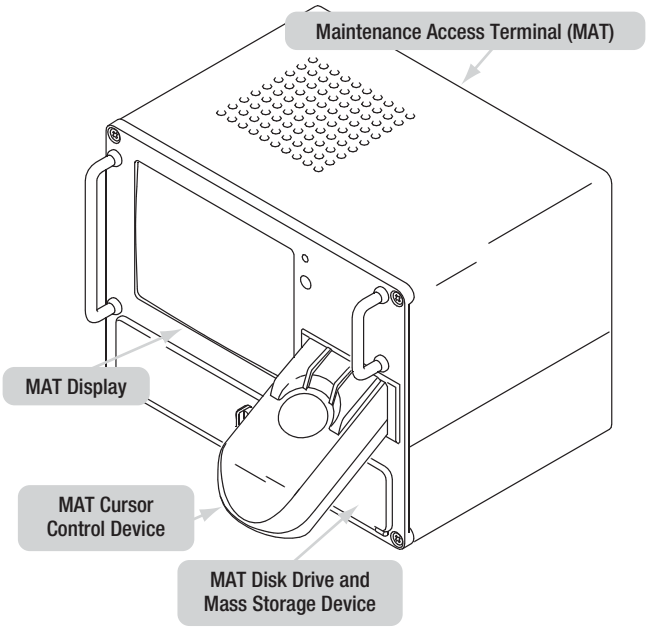


Figure 10-9. Boeing 777 Maintenance Access Terminal (MAT).

The data loading system can also be used to download reports from the aircraft. An example of this is the report of faults stored within the central maintenance computer. Reports on both current faults and fault history can be downloaded.



Figure 10-10. Navigation system update software.

ELECTRONIC LIBRARY SYSTEM

An electronic library system (ELS) consists of databases containing information used by flight crews and maintenance personnel. These databases can include maintenance manuals, illustrated parts catalogs, wiring diagram manuals, troubleshooting manuals, flight manuals, service bulletins, and many other kinds of documentation from the manufacturer or the aircraft operator. The ELS takes the place of paper manuals. This results in a weight savings, and can make accessing the information in the manuals quicker and easier.

The databases in an electronic library system can be accessed through an on-board display terminal and keyboard. They can also be accessed by an external personal computer, or through another digital device such as an iPad or tablet. The laptop or other external device is typically connected to the system using a serial bus cable. The databases in an ELS must be updated periodically as revisions are made to the technical data contained in the manuals. These revisions can be input through the data loading system.

PRINTING

Many aircraft have capability to print out paper copies of reports from the on board maintenance system, as well as other documents. Aircraft printers typically conform to ARINC Standard 744A, which gives technical requirements for such printers. These printers are able to print high-resolution alphanumeric text, as well as graphical images. The printers can print on paper up to 8.5 inches wide. ***Figure 10-11*** shows an example of an aircraft printer.

The speeds of aircraft printers vary, depending on the specific model of printer, and on what is being printed. Text generally prints faster, and images take longer. Some printers can print a page of text in as little as 5 seconds, while others are slower. Print resolution also varies. A standard resolution is 300 dots per inch (dpi), but some printers are capable of greater resolution. The paper supply for aircraft printers comes in the form of rolls. The paper rolls are typically 150 feet long, and may be perforated or non-perforated.

Inside the printer, an electric motor is used to advance or 'Slew' the paper. The printer uses a thermal print head, and the paper is heat sensitive. For this reason, care must be taken to keep the paper away from heat sources and out of direct sunlight while it is being stored. Exposure to heat can darken the paper, making it unusable for printing.

Aircraft printers receive input from CMCs, the ACARS system, and other sources by means of data lines, which may be ARINC 429 buses or Ethernet cables. Some printers are capable of receiving input wirelessly, and operated as part of a wireless LAN (local area network).

A typical aircraft printer is equipped with an indicator light to show whether the power is on or off. It will also give an alert when the paper supply is running low. Some printers perform a self-test on power-up, and will provide an indication if a fault is found during the test.



Figure 10-11. An aircraft printer.

STRUCTURE MONITORING

Structure monitoring, also known as damage tolerance monitoring, has been recognized as an important function in aircraft maintenance. As aircraft age, their structures becomes more susceptible to damage caused by fatigue. Repeated cabin pressurization cycles cause fatigue. Repairs and alterations can change the structural characteristics of an aircraft, introducing different stresses than were present with the original design. Corrosion can seriously weaken an aircraft's structure. Also, events such as hard landings can lead to structural damage which may be difficult to detect.

Certification regulations require aircraft manufacturers to identify critical areas of the aircraft's structure. These areas are known as fatigue critical structures (FCS). These critical structures are identified by performing fatigue testing on test articles, which are subjected to repeated load cycles until they fail. The results of this testing are analyzed to determine the FCS for the aircraft.

Aircraft operators are required to monitor all FCS on their aircraft. This monitoring is intended to detect cracks and other structural deformations before they reach critical proportions, resulting in catastrophic failure. The FCS monitoring process is accomplished by performing damage tolerance inspections (DTIs). DTIs are inspections focused specifically on fatigue critical structures. The aircraft's DTI program will state when and where to inspect, how to inspect, and how often to repeat the required inspections.

DTI inspections may be accomplished using visual inspection, eddy current, penetrant, X-ray, or other methods. However, drawbacks to these methods are that the inspections are a 'snapshot' of the airplanes structural condition at the time that the inspection was performed and, although trends can be determined, it is not a real-time monitor of the airplane.

These inspections usually form part of a planned maintenance schedule. A condition based maintenance (CBM) strategy ensures that maintenance is carried out on an airplane only when the maintenance is needed. A fully integrated vehicle health monitoring (IVHM) system supports a CBM programme by providing a constant source of airplane system performance and integrity data to allow maintenance analysts to schedule maintenance as and when it is needed.

An onboard structure health monitoring system utilizes sensors deployed at key points on the airplane structure. Strain sensors are bonded to a critical point on the structure. If the structure at that point becomes deformed, the strain sensor also becomes deformed. This deformation changes the electrical characteristics (typically the resistance) of the sensor indicating that damage, stress or strain is evident in the structure.

The sensors are controlled and monitored by central sensor controllers which are linked to the airplane on-board maintenance computers. This data is then recorded for download and analysis by ground maintenance staff. Data can also be relayed to ground stations whilst the airplane is in flight.

A number of technologies have been or are being developed for structure and systems monitoring. Ultrasonic, comparative vacuum monitoring and smart aircraft structure systems have come some way to realise the effectiveness and viability of integrated SHM systems.

The ultrasonic method uses a series of ultrasonic radiating elements and receiver sensors attached to the airplane structure. In ***Figure 10-12*** the waves radiating from the 'transmitting' elements are altered by damage to the structure. When the received waves are compared by analysts with waves from an undamaged structure, the level and extent of damage can be determined.

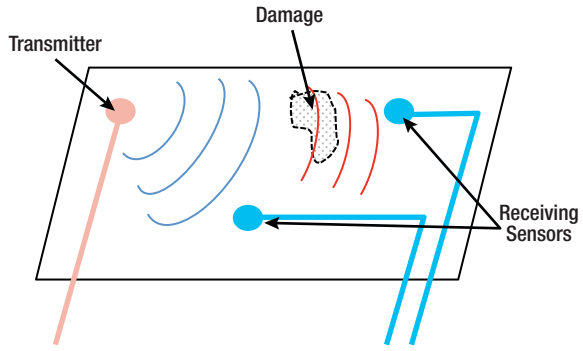


Figure 10-12. Ultrasonic sensor system.

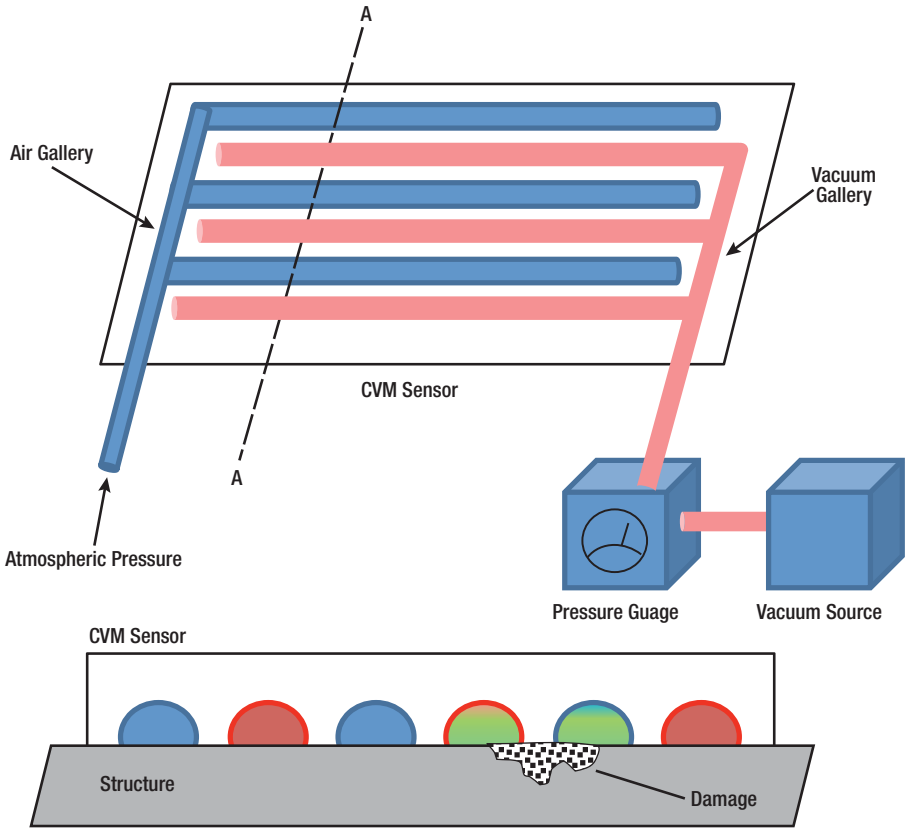


Figure 10-13. Comparative vacuum monitoring sensor.

A comparative vacuum monitoring method uses sensors containing a vacuum manifold and an air manifold. These manifolds are referred to as 'galleries' and are inter-meshed in the sensor material (***Figure 10-13***). The sensor is bonded to a point on the aircraft structure. A vacuum source is used to create a low vacuum in the vacuum 'gallery', the air gallery is pressurized to atmosphere.

Under normal conditions a differential pressure exists between the two galleries which, is measured by a pressure monitor. If a crack forms in the structure across the sensor galleries air will be allowed to leak from the atmospheric gallery to the vacuum gallery reducing the differential pressure between them. This pressure reduction is detected by the pressure monitor. The sensor controller processes the data from each sensor to determine the location and size of the crack. Very small cracks can be detected using this system as the air and vacuum manifolds in each sensor are placed between 100 to 300 micro-meters apart.

Increased use of composite materials in airplane manufacture has introduced new methods and strategies for maintenance and repair of airplane structures. Composite materials are more brittle and susceptible to damage than aluminum alloys traditionally used in airplane structures.

Where lower electrical conduction qualities in composite materials pose a challenge with regard to airplane electrical bonding, lightning strike and grounding, developments in newer composite materials known as 'nano-composites' possess semi-conductor properties which can be used in the manufacture of 'smart aircraft structure systems'.

Components manufactured from 'nano-composites' could be monitored by on-board systems where any change in the components semi-conductive properties is measured to determine if any damage has occurred.

Effective structure monitoring is crucial for preventing accidents caused by structural failure. For this reason, all data gathered during damage tolerance inspections must be recorded and carefully evaluated to ensure that the aircraft remains structurally sound.

Question: 10-1
What is a primary function of a central maintenance system?
A. Isolate and identify faults
B. Repair software faults
C. Restrict the airplane operation when faults are detected

Question: 10-2
What is the function of an A/D Converter in a central maintenance computer?
A. Convert digital outputs into a discrete signal
B. Change a discrete input to a digital signal
C. Process analogue and digital data

Question: 10-3
Why does the central maintenance computer require an input from the airplane landing gear weight on wheels switch?
A. To trigger tests on the ground
B. To restrict or inhibit some BITE tests in flight
C. This enables the ACARS transmitter

Question: 10-4
What happens following a failure of the active CMC in a dual CMC system?
A. The data switch in the failed CMC operates to allow data from the standby CMC to be routed to the data outputs
B. The data switch in the standby CMC operates to disconnect the failed CMC
C. The standby CMC switches the data switch in the failed CMC to isolate it

Question: 10-5
What is a function of the data receiver in the CMC?
A. Parallel to serial data conversion
B. Analogue to digital conversion
C. Multiplexing

Question: 10-6
How many faults can be stored in the CMC non-volatile memory?
A. 10 000
B. None, they are stored in volatile memory
C. Approximately 500

Question: 10-7
How are cautions displayed to the flight crew?
A. Red indication accompanied by a continuous sound
B. White indication accompanied by a single chime
C. Amber indication accompanied by a single chime

Question: 10-8
Which BITE can only be activated on the ground?
A. Interruptive
B. Initial
C. Cyclic

ANSWERS

Answer: 10-1
A. Isolate and identify faults

Answer: 10-2
B. Change a discrete input to a digital signal

Answer: 10-3
B. To restrict or inhibit some BITE tests in flight

Answer: 10-4
A. The data switch in the failed CMC operates to allow data from the standby CMC to be routed to the data outputs

Answer: 10-5
C. Multiplexing

Answer: 10-6
C. Approximately 500

Answer: 10-7
C. Amber indication accompanied by a single chime

Answer: 10-8
A. Interruptive



PART-66 SYLLABUS **LEVELS**

CERTIFICATION CATEGORY → **B2**

Sub-Module 11

AIR CONDITIONING AND CABIN PRESSURIZATION (ATA 21)

Knowledge Requirements

13.11 - Air Conditioning and Cabin Pressurization (ATA 21)

11.1 - Air Supply

Sources of air supply including engine bleed, APU and ground cart;	2

11.2 - Air Conditioning

Air conditioning systems;	2
Air cycle and vapor cycle machines;	3
Distribution systems;	1
Flow, temperature and humidity control system;	3

11.3 - Pressurization

Pressurization systems;	3
Control and indication including control and safety valves;	
Cabin pressure controllers;	

11.4 - Safety and Warning Devices

Protection and warning devices.	3

SOURCES OF SUPPLY AIR

The source of air to pressurize and air condition an aircraft varies mainly with engine type. Reciprocating aircraft have pressurization sources different from those of turbine-powered aircraft. Note that the compression of air raises its temperature. A means for keeping cabin air cool is built into most pressurization systems. It may be in the form of a heat exchanger, using cold ambient air to modify the temperature of the air from the pressurization source. However, a full air cycle air conditioning system with expansion turbine is typically used on commercial transport aircraft. It provides the advantage of cabin temperature control on the ground and at low altitudes.

BLEED AIR

The main principle of operation of a turbine engine involves the compression of large amounts of air to be mixed with fuel and burned. Air bled from the compressor section of the engine is relatively free of contaminants. As such, compressor air is a great source of air for cabin pressurization and air conditioning. However, the volume of air for engine power production is reduced by bleeding air off of the compressor. And, even though the amount of bleed air to the overall amount of air compressed for combustion is relatively small, it should still be minimized. Modern, large-cabin turbofan engine aircraft contain recirculation fans to reuse up to 50 percent of the air in the cabin. This reduces bleed air volume and helps maintain high engine output.

There are different ways hot, high-pressure bleed air can be exploited for cabin pressurization and air conditioning. Smaller turbine aircraft, or sections of a large aircraft, may make use of a jet pump flow multiplier. With this device, bleed air is tapped off of the turbine engine's compressor section. It is ejected into a venturi jet pump mounted in air ducting that has one end open to the ambient air and the other end directed into the compartment to be pressurized. Due to the low pressure established in the venturi by the bleed air flow, air is drawn in from outside the aircraft. It mixes with the bleed air and is delivered to the pressure vessel to pressurize it. An advantage of this type of pressurization is the lack of moving parts. (***Figure 11-1***)

Disadvantages are that only a relatively small volume of space can be pressurized in this manner and there is no means for cooling the ambient air.

Another method of pressurizing an aircraft using turbine engine compressor bleed air is to have the bleed air drive a separate compressor that has an ambient air intake. A turbine turned by bleed air rotates a compressor impeller mounted on the same shaft. Outside air is drawn in and compressed. It is mixed with the bleed air outflow from the turbine and is sent to the pressure vessel. Turboprop aircraft often use this device, known as a turbo-compressor. (***Figure 11-2***)

The most common method of pressurizing and air conditioning turbine-powered aircraft is with an air cycle air conditioning and pressurization system. Bleed air is used, and through an elaborate system

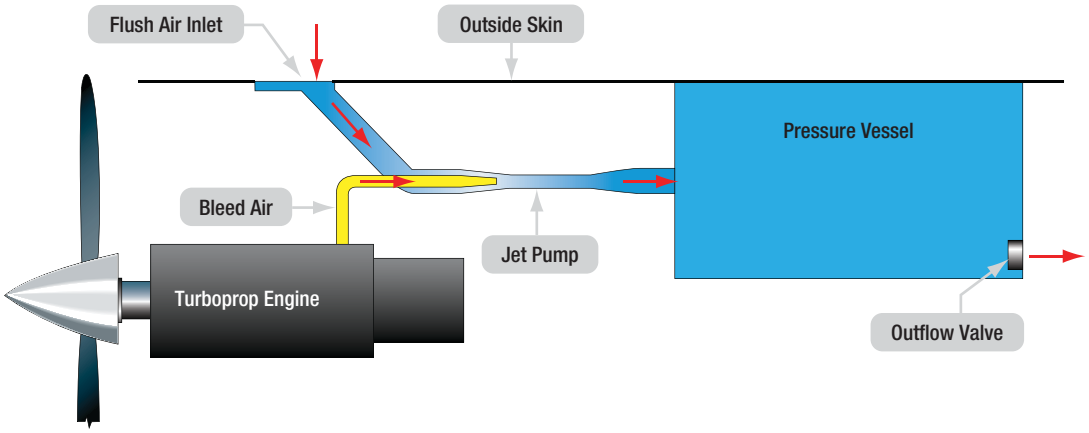


Figure 11-1. A jet pump flow multiplier ejects bleed air into a venturi which draws air for pressurization from outside the aircraft.

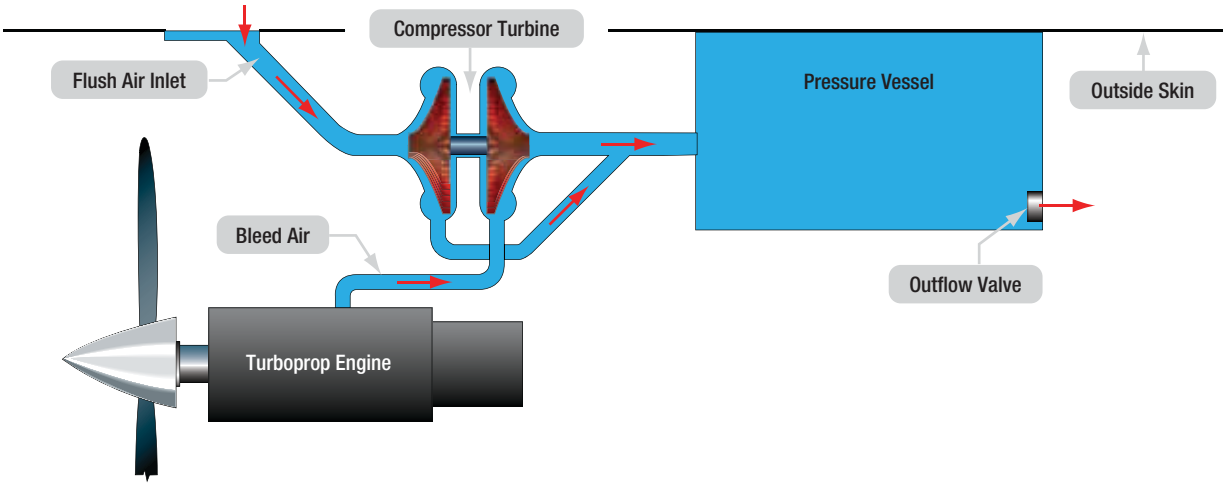


Figure 11-2. A turbo compressor used to pressurize cabins mostly in turboprop aircraft.

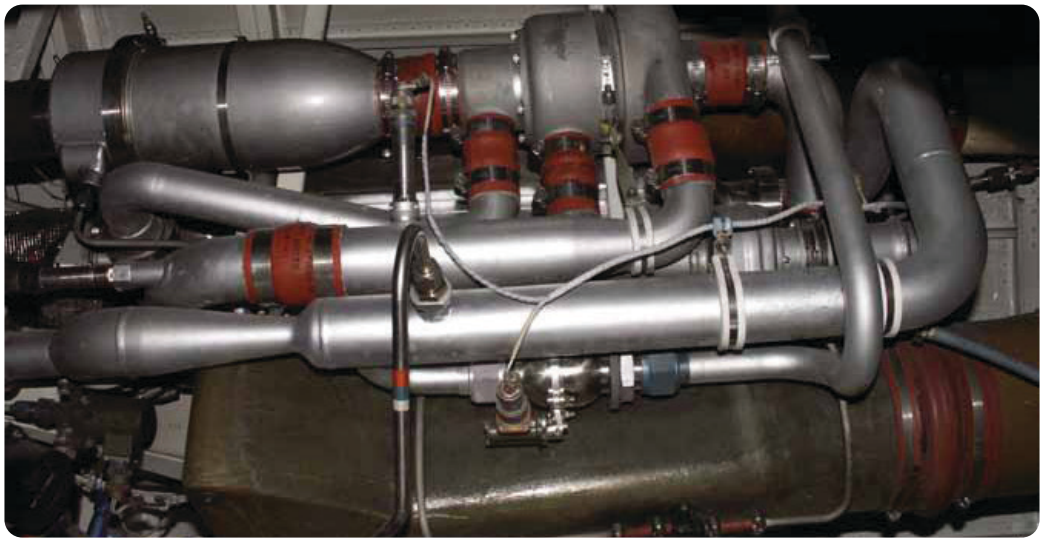


Figure 11-3. An air cycle air conditioning system used to pressurize and regulate the temperature of the cabin of a business jet aircraft.

that includes heat exchangers, a compressor, and an expansion turbine, cabin air temperature and pressure are controlled. Air cycle air conditioning is discussed in greater detail below. (***Figure 11-3***)

AUXILLARY POWER UNIT - APU

The source of air for air conditioning does not always have to be bleed air from the compressor section of the main engines. The aircraft's Auxiliary Power Unit (APU) is designed to deliver pressurized bleed air for engine starting and for operation of the aircraft's air conditioning packages. Sufficient pneumatic air from the APU is routed through ducting to the packs so that the cabin can be cooled while the aircraft is on the ground with the engine not running. This is one common method for keeping the cabin at a comfortable temperature which passengers are boarding.

GROUND CART

When an aircraft is on the ground, operating the engines or the APU to provide air for air conditioning is expensive. It increases the time in service of these expensive components and expedites costly mandatory overhauls that are performed at specified time intervals. A ground cart is available to provide the pressurized source of air for the air conditioning packs. This is typically a portable powerplant that drives a high volume air compressor.

The cart is towed to the aircraft's location on the ramp and is connected into the aircraft's pneumatic system ducting with a 4 inch diameter hose. The connection point is upstream of the air conditioning packs. Cart air is regulated to the normal pneumatic system pressure and can also be used for pneumatic system trouble shooting without the expense of running the APU or main engines.

An even better and more economical solution for cooling the aircraft while it is stationary on the ground exists. Most high-performance, medium-size and larger turbine-powered aircraft are fitted with a receptacle in the air distribution system. The air distribution system basically consists of a series of ducts that carry conditioned air from the packs to wherever it is needed on the aircraft. This is discussed further below. A ground source of conditioned air can be connected directly into the distribution system receptacle. Cool air from a ground-based air conditioner is blown directly into the cabin through the aircraft's distribution system ducting (***Figure 11-4***).

This makes operating the aircraft's air conditioning packs unnecessary. The ground-based air conditioning unit can be a large air conditioner mounted on a truck or it can be a fixed type such as those used in homes and businesses.



Figure 11-4. A duct hose installed on this airliner distributes hot or cold air from a ground-based source throughout the cabin using the aircraft's own air distribution system ducting.

AIR CONDITIONING SYSTEMS

There are two types of air conditioning systems commonly used on aircraft. Air cycle air conditioning is used on most turbine-powered aircraft. It makes use of engine bleed air or APU pneumatic air during the conditioning process. Vapor cycle air conditioning systems are often used on reciprocating aircraft. This type system is similar to that found in homes and automobiles. Note that some turbine-powered aircraft also use vapor cycle air conditioning.

AIR CYCLE AIR CONDITIONING

Air cycle air conditioning prepares engine bleed air to pressurize the aircraft cabin. The temperature and quantity of the air must be controlled to maintain a comfortable cabin environment at all altitudes and on the ground. The air cycle system is often called the air conditioning package or pack. It is usually located in the lower half of the fuselage or in the tail section of turbine-powered aircraft. (***Figure 11-5***)

SYSTEM OPERATION

Even with the frigid temperatures experienced at high altitudes, bleed air is too hot to be used in the cabin without being cooled. It enters the air cycle system and is routed through a heat exchanger where ram air cools the bleed air. This cooled bleed air is directed into an air cycle machine. There, it is compressed before flowing through a secondary heat exchanger that cools the air again with

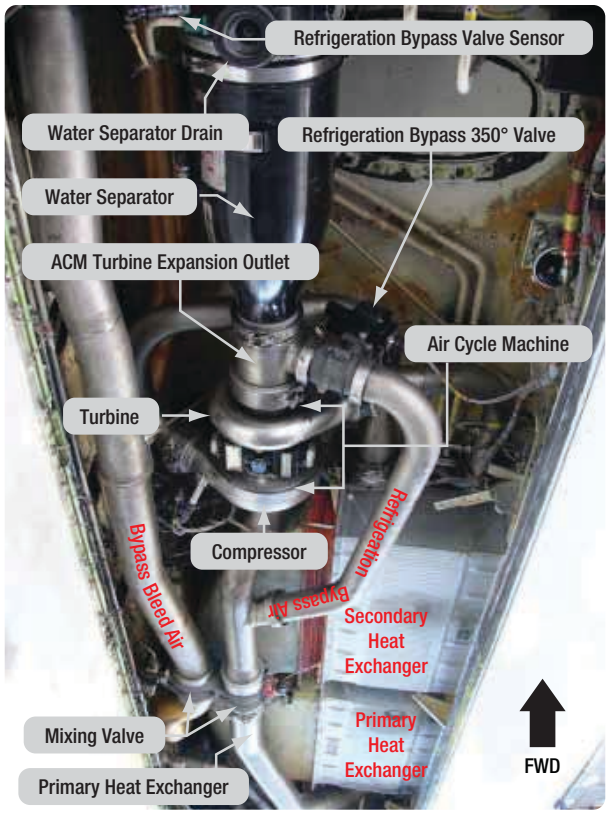


Figure 11-5. Boeing 737 air cycle system. The photo is taken looking up into the air conditioning bay located in the lower fuselage on each side of the aircraft.

ram air. The bleed air then flows back into the air cycle machine where it drives an expansion turbine and cools even further. Water is then removed and the air is mixed with bypassed bleed air for final temperature adjustment. It is sent to the cabin through the air distribution system. By examining the operation of each component in the air cycle process, a better understanding can be developed of how bleed air is conditioned for cabin use. Refer to ***Figure 11-6***, which diagrams the air cycle air conditioning system of the Boeing 737.

PNEUMATIC SYSTEM SUPPLY

During normal flight, the pneumatic system is supplied by bleed air tap-offs located on each engine compressor section. It consists of a pneumatic manifold, valves, regulators and ducting. Typically, a pneumatic system manifold contains hot air between 30 and 75 psi. The air conditioning packs are supplied by this manifold as are other critical airframe systems, such as the anti-ice and hydraulic pressurization system.

COMPONENT OPERATION

Pack Valve

The pack valve is the valve that regulates bleed air from the pneumatic manifold into the air cycle air conditioning system. It is controlled with a switch from the air conditioning panel in the cockpit. Many pack valves are electrically controlled and pneumatically operated. Also known as the supply shutoff valve, the pack valve opens, closes, and modulates to allow the air cycle air conditioning system to be supplied with a designed volume of hot, pressurized air. (***Figure 11-7***) When an overheat or other abnormal condition requires that the air conditioning package be shut down, a signal is sent to the pack valve to close.

Bleed Air Bypass

A means for bypassing some of the pneumatic air supplied to the air cycle air conditioning system around the system is present on all aircraft. This warm bypassed air must be mixed with the cold air produced by the air cycle system so the air delivered to the cabin is a comfortable temperature. In the system shown in ***Figure 11-6***, this is accomplished by the mixing valve. It simultaneously controls the flow of bypassed air and air to be cooled to meet the requirements of the auto temperature controller. It can also be controlled manually with the cabin temperature selector in manual mode. Other air cycle systems may refer to the valve that controls the air bypassed around the air cycle cooling system as a temperature control valve, trim air pressure regulating valve, or something similar.

Primary Heat Exchanger

Generally, the warm air dedicated to pass through the air cycle system first passes through a primary heat exchanger. It acts similarly to the radiator in an automobile. A controlled flow of ram air is ducted over and through the exchanger, which reduces the temperature of the air inside the system. (***Figure 11-8***) A fan draws air through the ram air duct when the aircraft is on the ground so that the heat exchange is possible when the aircraft is stationary. In flight, ram air doors are modulated to increase or decrease ram air flow to the exchanger according to the position of the wing flaps. During slow flight, when the flaps are extended, the doors are open. At higher speeds, with the flaps retracted, the doors move toward the closed position reducing the amount of ram air to the exchanger.

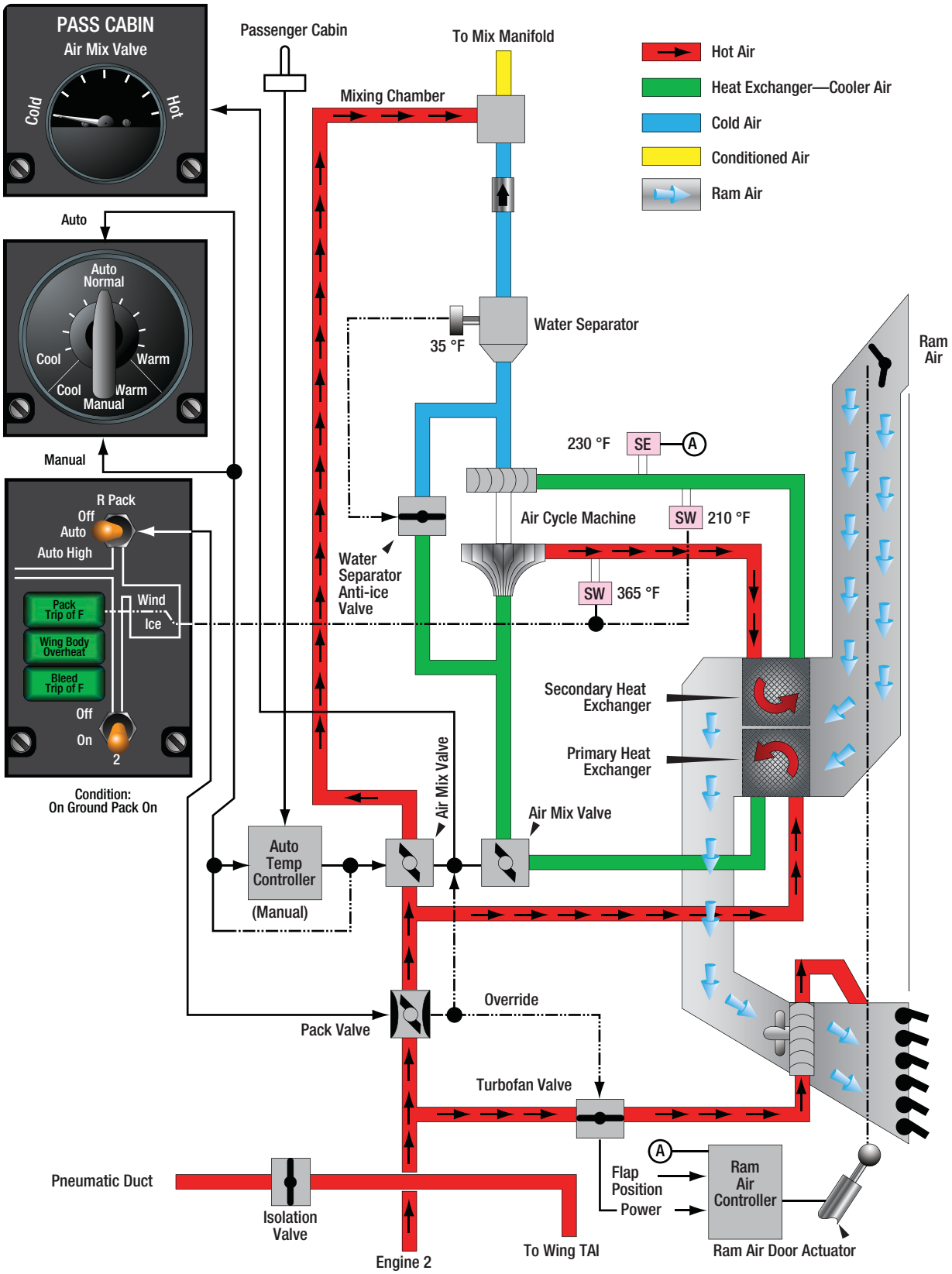


Figure 11-6. The air cycle air conditioning system on a Boeing 737.

Similar operation is accomplished with a valve on smaller aircraft. (***Figure 11-9***)

Refrigeration Turbine Unit or Air Cycle Machine and Secondary Heat Exchanger

The heart of the air cycle air conditioning system is the refrigeration turbine unit, also known as the air cycle machine (ACM). It is comprised of a compressor that is driven by a turbine on a common shaft. System air flows from the primary heat exchanger into the compressor side of the ACM. As the air is compressed, its temperature rises. It is then sent to a secondary heat exchanger, similar to the primary heat exchanger located in the ram air duct. The elevated temperature of the ACM compressed air facilitates an easy exchange of heat energy to the ram air. The cooled system air, still under pressure from the continuous system air flow and the ACM compressor, exits the secondary heat exchanger. It is directed into the turbine side of the ACM.

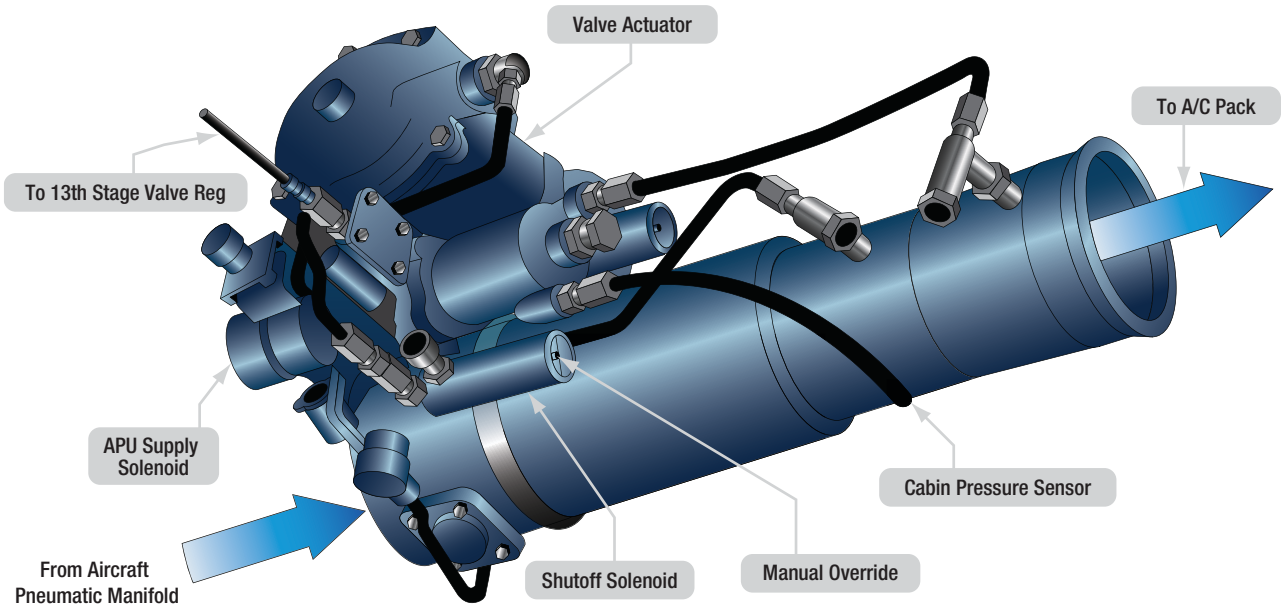


Figure 11-7. This pack valve drawing illustrates the complexity of the valve, which opens, closes, and modulates. It is manually actuated from the cockpit and automatically responds to supply and air cycle system parameter inputs.

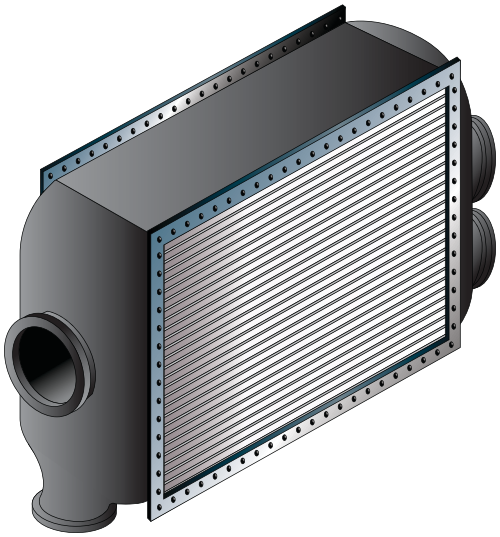


Figure 11-8. The primary and secondary heat exchangers in an air cycle air conditioning system are of similar construction. They both cool bleed air when ram air passes over the exchanger coils and fins.



Figure 11-9. A ram air door controls the flow of air through the primary and secondary heat exchangers.

The steep blade pitch angle of the ACM turbine extracts more energy from the air as it passes through and drives the turbine. Once through, the air is allowed to expand at the ACM outlet, cooling even further. The combined energy loss from the air first driving the turbine and then expanding at the turbine outlet lowers the system air temperature to near freezing. (***Figure 11-10***)

Water Separator

The cool air from the air cycle machine can no longer hold the quantity of water it could when it was warm. A water separator is used to remove the water from the saturated air before it is sent to the aircraft cabin. The separator operates with no moving parts. Foggy air from the ACM enters and is forced through a fiberglass sock that condenses and coalesces the mist into larger water drops. The convoluted interior structure of the separator swirls the air and water. The water collects on the sides of the separator and drains down and out of the unit, while the dry air passes through. A bypass valve is incorporated in case of a blockage. (***Figure 11-11***)

Refrigeration Bypass Valve

As mentioned, air exiting the ACM turbine expands and cools. It becomes so cold, it could freeze the water in the water separator, thus inhibiting or blocking airflow. A temperature sensor in the separator controls a refrigeration bypass valve designed to keep the air flowing through the water separator above freezing temperature. The valve is also identified by other names such as a temperature control valve, 35° valve, anti-ice valve, and similar. It bypasses warm air around the ACM when opened. The air is introduced into the expansion ducting, just upstream of the water separator, where it heats the air just enough to keep it from freezing. Thus, the refrigeration bypass valve regulates the temperature of the ACM discharge air so it does not freeze when passing through the water separator. This valve is visible in ***Figure 11-5*** and is diagrammed in the system in ***Figure 11-6***.

All air cycle air conditioning systems use at least one ram air heat exchanger and an air cycle machine with expansion turbine to remove heat energy from the bleed

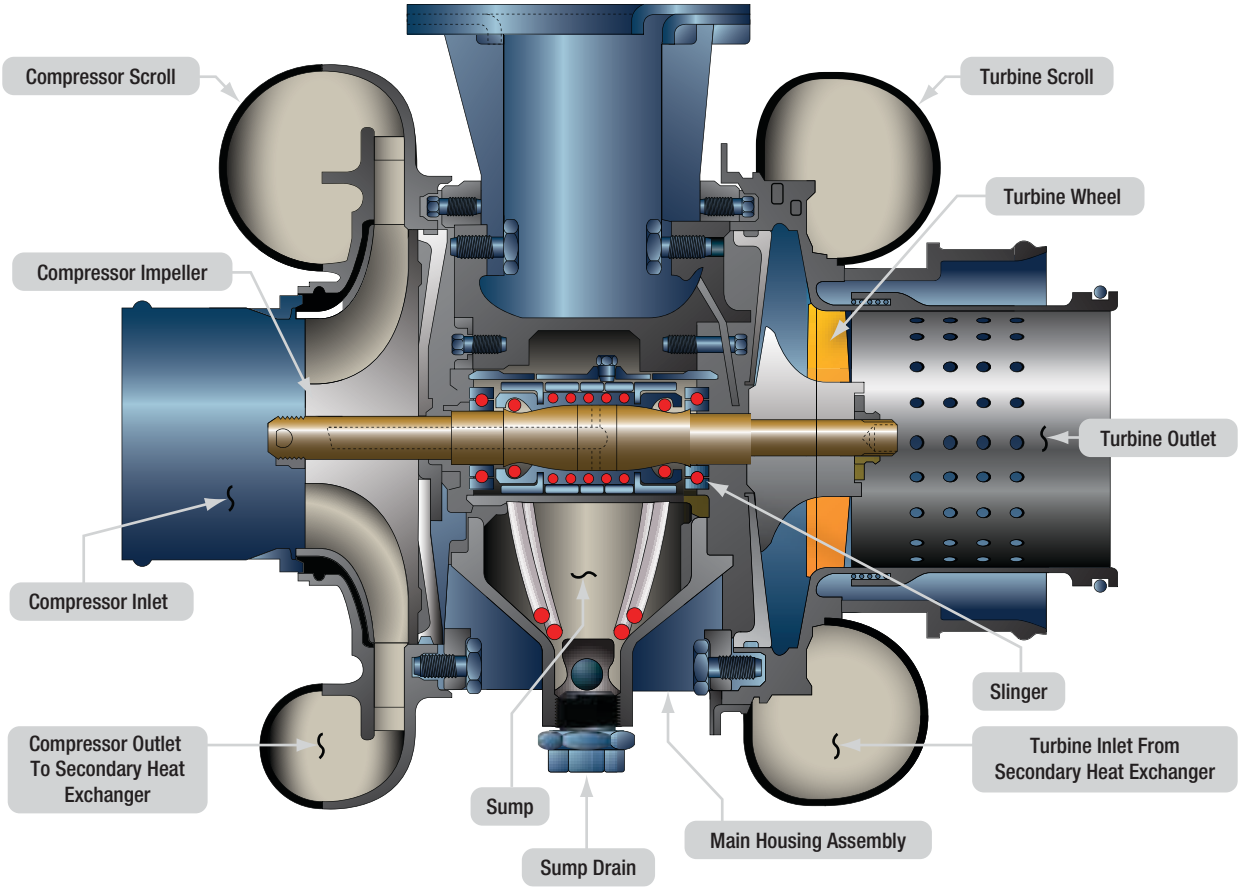


Figure 11-10. A cutaway diagram of an air cycle machine. The main housing supports the single shaft to which the compressor and turbine are attached. Oil lubricates and cools the shaft bearings.

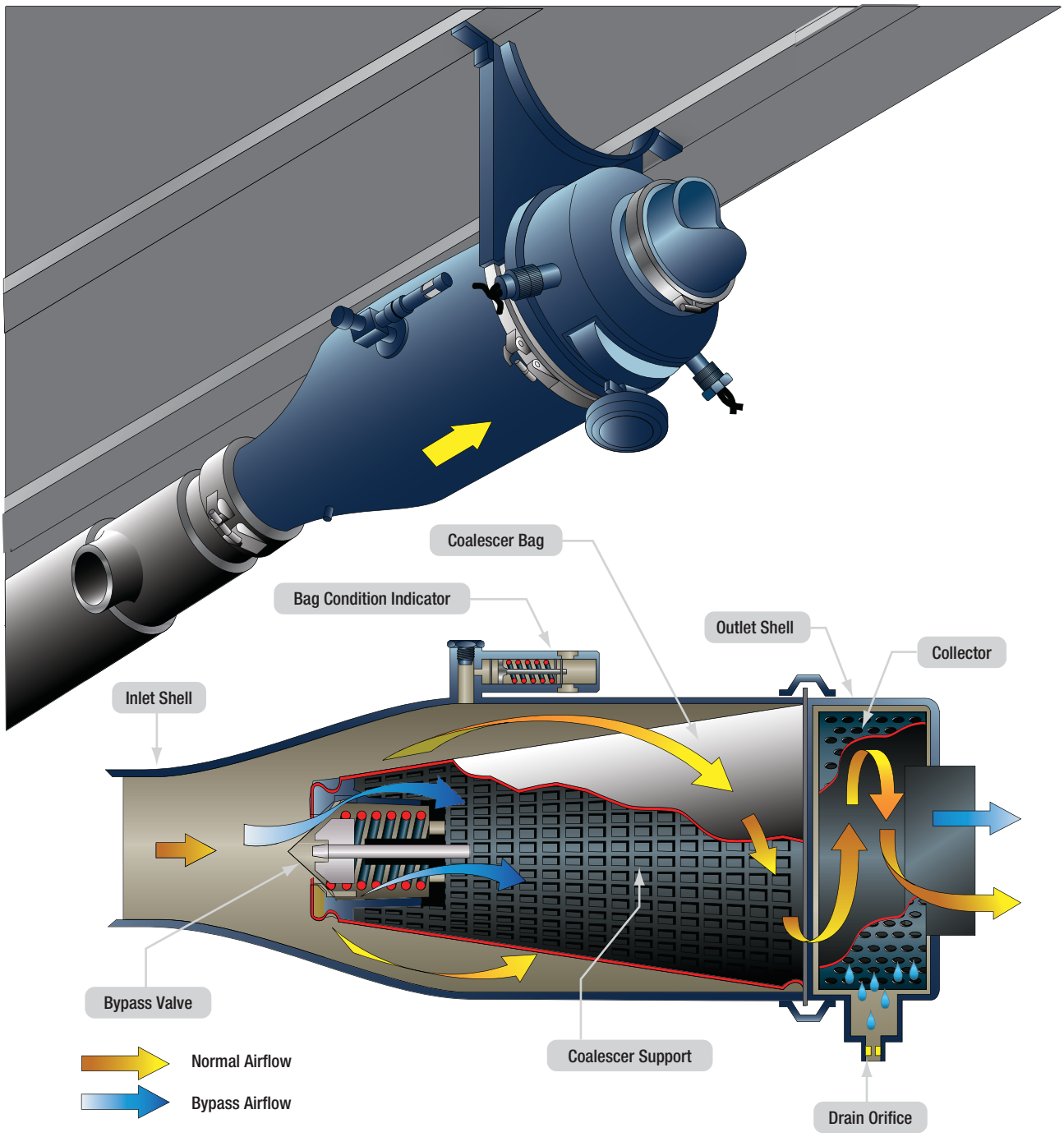


Figure 11-11. A water separator coalesces and removes water by swirling the air/water mixture from ACM expansion turbine. Centrifugal force sends the water to the walls of the collector where it drains from the unit.

air, but variations exist. An example of a system different from that described above is found on the McDonnell Douglas DC-10. Bleed air from the pneumatic manifold is compressed by the air cycle machine compressor before it flows to a single heat exchanger. Condensed water from the water separator is sprayed into the ram air at its entrance to the exchanger to draw additional heat from the compressed bleed air as the water evaporates.

A trim air valve for each cabin zone mixes bypassed bleed air with conditioned air in response to individual temperature selectors for each zone. When cooling air demands are low, a turbine bypass valve routes some heat exchanger air directly to the conditioned air manifold. (***Figure 11-12***)

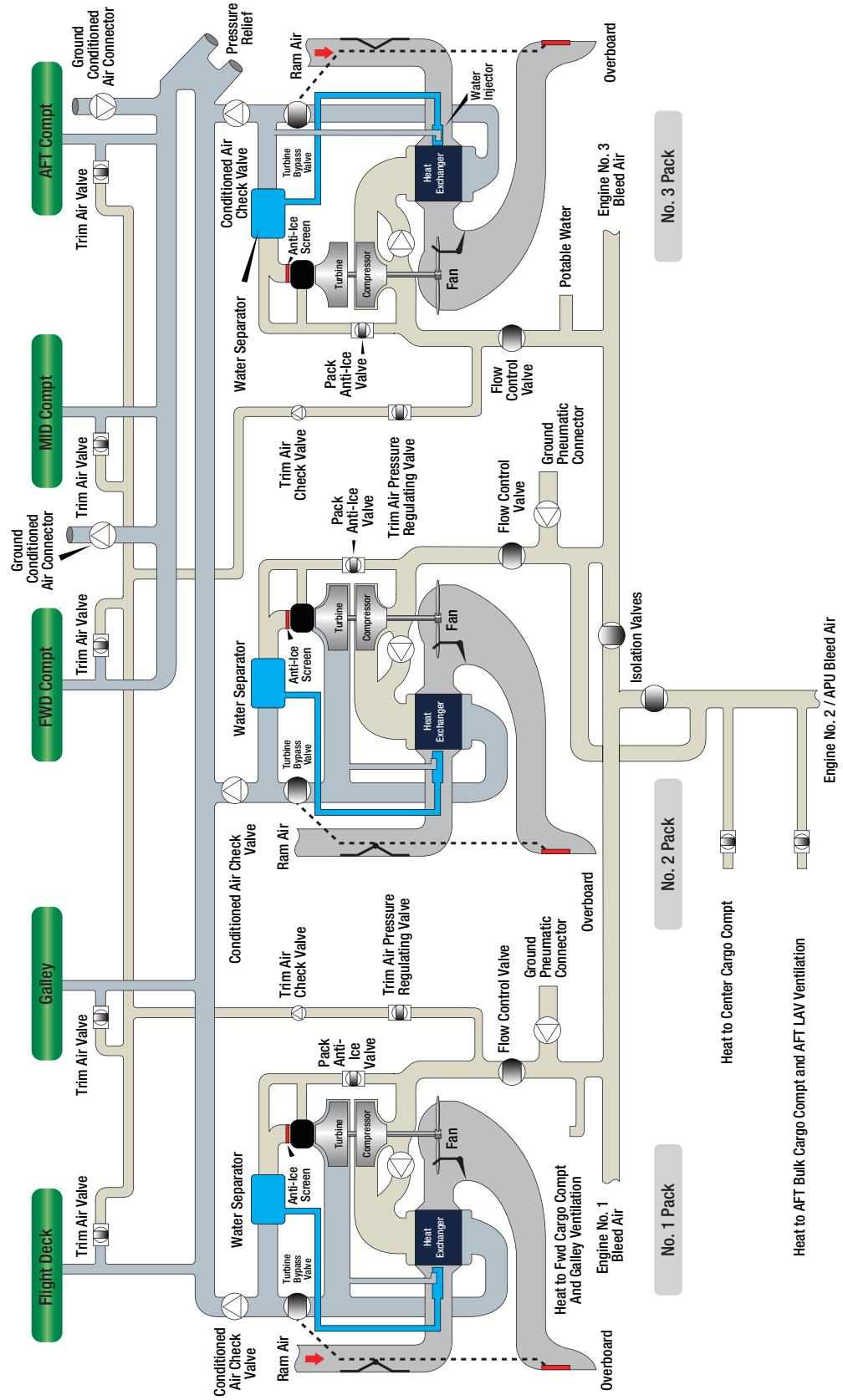


Figure 11-12. A water separator coalesces and removes water by swirling the air/water mixture from ACM expansion turbine. Centrifugal force sends the water to the walls of the collector where it drains from the unit.

VAPOR CYCLE AIR CONDITIONING

Some turbine powered business class aircraft and older transport category aircraft use vapor cycle air conditioning. The absence of a bleed air source on reciprocating engine aircraft makes the use of an air cycle system impractical for conditioning cabin air. Vapor cycle air conditioning is used on most non-turbine aircraft that are equipped with air conditioning. However, it is not a source of pressurizing air as the air cycle system conditioned air is on turbine powered aircraft. The vapor cycle system only cools the cabin.

If an aircraft equipped with a vapor cycle air conditioning system is pressurized, it uses a different source of air for pressurization.

Vapor cycle air conditioning is a closed system used solely for the transfer of heat from inside the cabin to outside of the cabin. It can operate on the ground and in flight.

THEORY OF REFRIGERATION

Energy can be neither created nor destroyed; however, it can be transformed and moved. This is what occurs during vapor cycle air conditioning. Heat energy is moved from the cabin air into a liquid refrigerant. Due to the additional energy, the liquid changes into a vapor. The vapor is compressed and becomes very hot. It is removed from the cabin where the very hot vapor refrigerant transfers its heat energy to the outside air. In doing so, the refrigerant cools and condenses back into a liquid. The refrigerant returns to the cabin to repeat the cycle of energy transfer. (***Figure 11-13***)

Heat is an expression of energy, typically measured by temperature. The higher the temperature of a substance, the more energy it contains. Heat always flows from hot to cold. These terms express the relative amount of energy present in two substances. They do not measure the absolute amount of heat present. Without a difference in energy levels, there is no transfer of energy (heat).

Adding heat to a substance does not always raise its temperature. When a substance changes state, such as when a liquid changes into a vapor, heat energy is absorbed. This is called latent heat. When a vapor condenses into a liquid, this heat energy is given off. The temperature of a substance remains constant during its change of state. All energy absorbed or given off, the latent heat, is used for the change process. Once the change of state is complete, heat added to a substance raises the temperature of the substance. After a substance changes state into a vapor, the rise in temperature of the vapor caused by the addition of still more heat is called superheat.

The temperature at which a substance changes from a liquid into a vapor when heat is added is known as its boiling point. This is the same temperature at which a vapor condenses into a liquid when heat is removed. The boiling point of any substance varies directly with pressure. When pressure on a liquid is increased, its boiling point increases, and when pressure on a liquid is decreased, its boiling point also decreases. For example, water boils at 212 °F at normal atmospheric temperature (14.7 psi). When pressure on liquid water is increased to 20 psi, it does not boil at 212 °F. More energy is required to overcome the increase in pressure. It boils at approximately 226.4 °F. The converse is also true. Water can also boil at a much lower temperature simply by reducing the pressure upon it. With only 10 psi of pressure upon liquid water, it boils at 194 °F.
(***Figure 11-14***)

Vapor pressure is the pressure of the vapor that exists above a liquid that is in an enclosed container at any given temperature. The vapor pressure developed by various substances is unique to each substance. A substance that is said to be volatile, develops high vapor pressure at standard day temperature (59 °F). This is because the boiling point of the substance is much lower. The boiling point of tetrafluoroethane (R134a), the refrigerant used in most aircraft vapor cycle air conditioning systems, is approximately -15 °F. Its vapor pressure at 59 °F is about 71 psi. The vapor pressure of any substance varies directly with temperature.

BASIC VAPOR CYCLE

Vapor cycle air conditioning is a closed system in which a refrigerant is circulated through tubing and a variety of components. The purpose is to remove heat from the aircraft cabin. While circulating, the refrigerant changes state. By manipulating the latent heat required to do so, hot air is replaced with cool air in the aircraft cabin.

To begin, R134a is filtered and stored under pressure in a reservoir known as a receiver dryer. The refrigerant is in liquid form. It flows from the receiver dryer

through tubing to an expansion valve. Inside the valve, a restriction in the form of a small orifice blocks most of the refrigerant. Since it is under pressure, some of the refrigerant is forced through the orifice. It emerges as a spray of tiny droplets in the tubing downstream of the valve. The tubing is coiled into a radiator- type assembly known as an evaporator. A fan is positioned to blow cabin air over the surface of the evaporator. As it does, the heat in the cabin air is absorbed by the refrigerant, which uses it to change state from a liquid to a vapor. So much heat is absorbed that the cabin air blown by the fan across the evaporator cools significantly. This is the vapor cycle conditioned air that lowers the temperature in the cabin.

The gaseous refrigerant exiting the evaporator is drawn into a compressor. There, the pressure and the temperature of the refrigerant are increased. The high-pressure high-temperature gaseous refrigerant flows through tubing to a condenser. The condenser is like a radiator comprised of a great length of tubing with fins attached to promote heat transfer. Outside air is directed over the condenser. The temperature of the refrigerant inside is higher than the ambient air temperature, so heat is transferred from the refrigerant to the outside air.

The amount of heat given off is enough to cool the refrigerant and to condense it back to a high-pressure liquid. It flows through tubing and back into the receiver dryer, completing the vapor cycle.

There are two sides to the vapor cycle air conditioning system. One accepts heat and is known as the low side. The other gives up heat and is known as the high side. The low and high refer to the temperature and pressure of the refrigerant. As such, the compressor and the expansion valve are the two components that separate the low side from the high side of the cycle. (***Figure 11-15***) Refrigerant on the low side is characterized as having low pressure and temperature. Refrigerant on the high side has high pressure and temperature.

VAPOR CYCLE AIR CONDITIONING SYSTEM COMPONENTS

By examining each component in the vapor cycle air conditioning system, greater insight into its function can be gained.

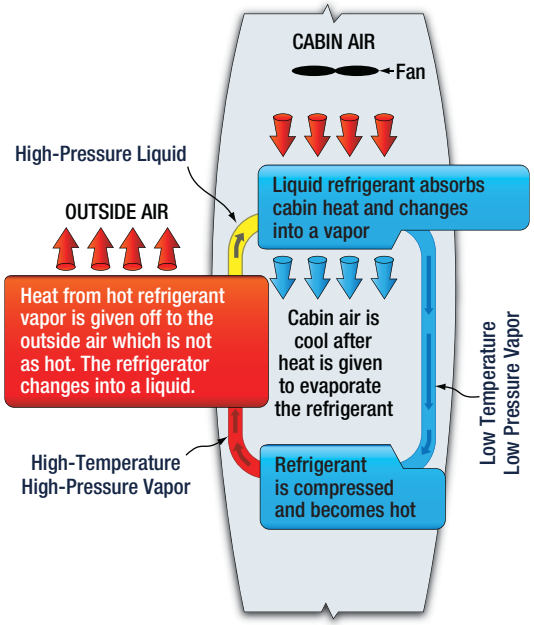


Figure 11-13. In vapor cycle air conditioning, heat is carried from the cabin to the outside air by a refrigerant which changes from a liquid to a vapor and back again.

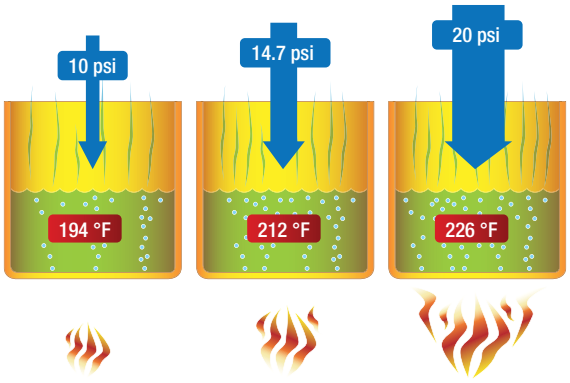


Figure 11-14. Boiling point of water changes as pressure changes.

Refrigerant

For many years, dichlorodifluoromethane (R12) was the standard refrigerant used in aircraft vapor cycle air conditioning systems. Some of these systems remain in use today. R12 was found to have a negative effect on the environment; in particular, it degraded the earth's protective ozone layer. In most cases, it has been replaced by tetrafluoroethane (R134a), which is safer for the environment. R12 and R134a should not be mixed, nor should one be used in a system designed for the other. Possible damage to soft components, such as hoses and seals, could result causing leaks and or malfunction. Use only the specified refrigerant when servicing vapor cycle air conditioning systems. (***Figure 11-16***)

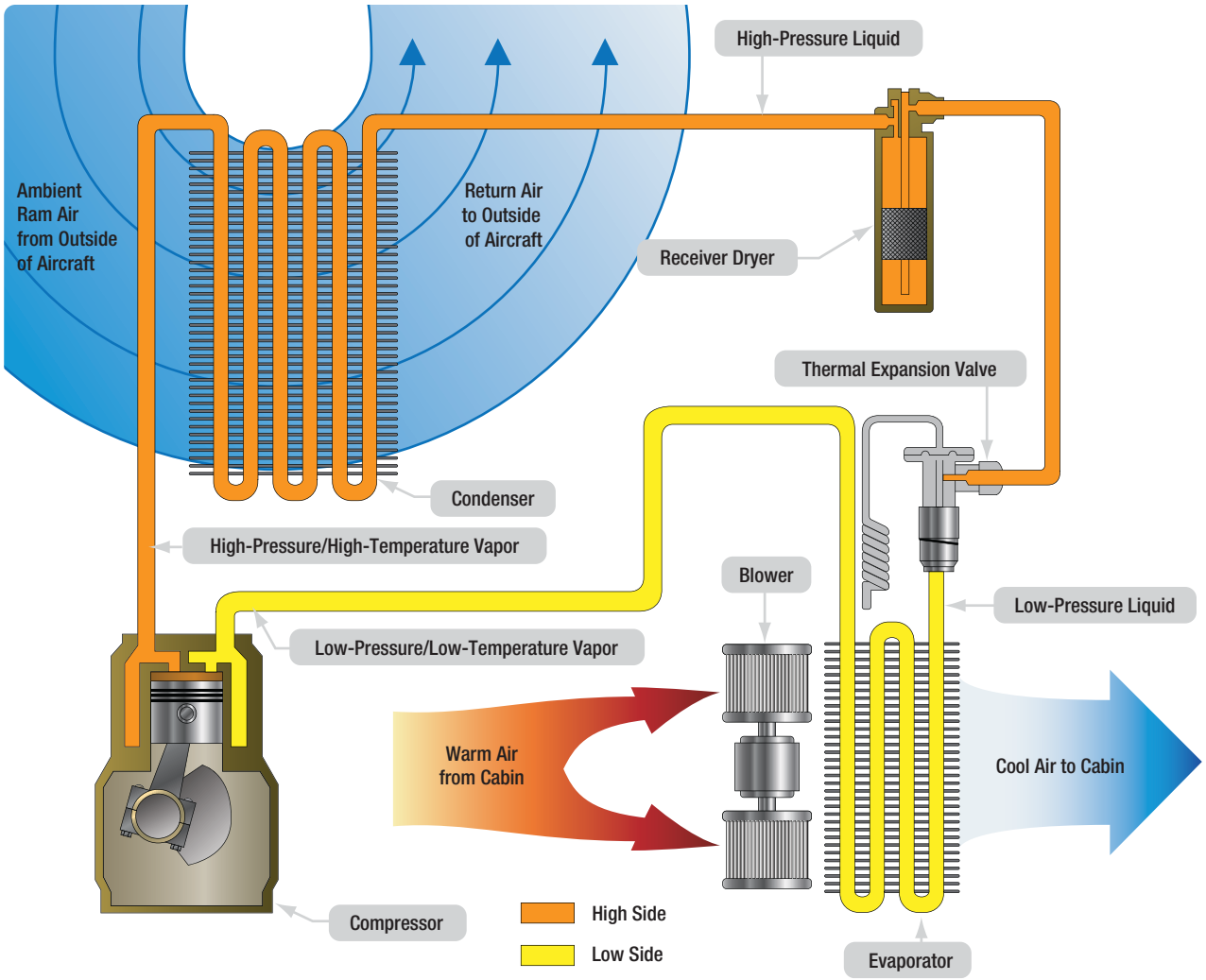


Figure 11-15. A basic vapor cycle air conditioning system. The compressor and the expansion valve are the two components that separate the low side from the high side of the cycle. This figure illustrates this division. Refrigerant on the low side is characterized as having low pressure and temperature. Refrigerant on the high side has high pressure and temperature.

R12 and R134a behave so similarly that the descriptions of the R134a vapor cycle air conditioning system and components in the following paragraphs also apply to an R12 system and its components.

R134a is a halogen compound (CF3CFH2). As mentioned, it has a boiling point of approximately -15 °F. It is not poisonous to inhale in small quantities, but it does displace oxygen. Suffocation is possible if breathed in mass quantity.

Regardless of manufacturer, refrigerants are sometimes called Freon, which is a trade name owned by the Dupont Company. Caution should be used when handling any refrigerant. Because of the low boiling points, liquid refrigerants boil violently at typical atmospheric temperatures and pressure. They rapidly absorb heat energy from all surrounding matter. If a drop lands on skin, it freezes, resulting in a burn. Similar tissue damage can result if a drop gets in one's eye. Gloves and other skin protection, as well as safety goggles, are required when working with refrigerant.

Receiver Dryer

The receiver dryer acts as the reservoir of the vapor cycle system. It is located downstream of the condenser and upstream of the expansion valve.

When it is very hot, more refrigerant is used by the system than when temperatures are moderate. Extra refrigerant is stored in the receiver dryer for this purpose. Liquid refrigerant from the condenser flows into the receiver dryer. Inside, it passes through filters and a desiccant material. The filters remove any



Figure 11-16. A small can of R134a refrigerant used in vapor cycle air conditioning systems.

foreign particles that might be in the system. The desiccant captures any water in the refrigerant. Water in the refrigerant causes two major problems. First, the refrigerant and water combine to form an acid. If left in contact with the inside of the components and tubing, the acid deteriorates the materials from which these are made. The second problem with water is that it could form ice and block the flow of refrigerant around the system, rendering it inoperative.

Ice is particularly a problem if it forms at the orifice in the expansion valve, which is the coldest point in the cycle.

Occasionally, vapor may find its way into the receiver dryer, such as when the gaseous refrigerant does not completely change state to a liquid in the condenser. A stand tube is used to remove refrigerant from the receiver dryer. It runs to the bottom of the unit to ensure liquid is withdrawn and forwarded to the expansion valve. At the top of the stand tube, a sight glass allows the technician to see the refrigerant. When enough refrigerant is present in the system, liquid flows in the sight glass. If low on refrigerant, any vapor present in the receiver dryer may be sucked up the stand tube causing bubbles to be visible in the sight glass. Therefore, bubbles in the sight glass indicate that the system needs to have more refrigerant added. (***Figure 11-17***)

Expansion Valve

Refrigerant exits the receiver dryer and flows to the expansion valve. The thermostatic expansion valve has

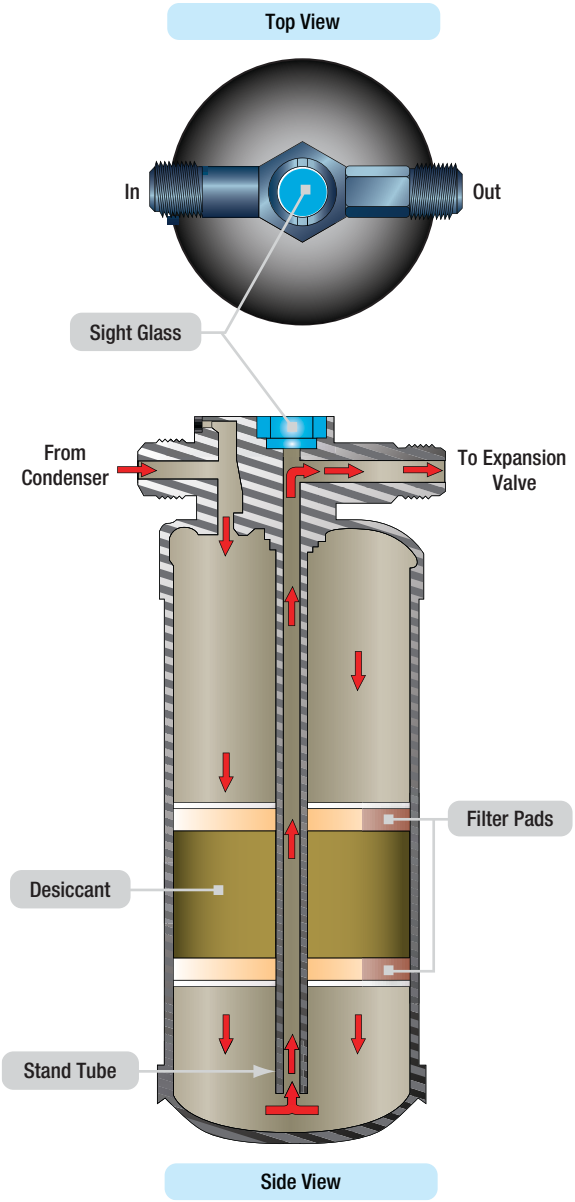


Figure 11-17. A receiver dryer acts as reservoir and filter in a vapor cycle system. Bubbles viewed in the sight glass indicate the system is low on refrigerant and needs to be serviced.

an adjustable orifice through which the correct amount of refrigerant is metered to obtain optimal cooling. This is accomplished by monitoring the temperature of the gaseous refrigerant at the outlet of the next component in the cycle, the evaporator. Ideally, the expansion valve should only let the amount of refrigerant spray into the evaporator that can be completely converted to a vapor.

The temperature of the cabin air to be cooled determines the amount of refrigerant the expansion valve should spray into the evaporator. Only so much is needed to completely change the state of the refrigerant from a liquid to a vapor. Too little causes the gaseous refrigerant

to be superheated by the time it exits the evaporator. This is inefficient. Changing the state of the refrigerant from liquid to vapor absorbs much more heat than adding heat to already converted vapor (superheat). The cabin air blowing over the evaporator will not be cooled sufficiently if superheated vapor is flowing through the evaporator. If too much refrigerant is released by the expansion valve into the evaporator, some of it remains liquid when it exits the evaporator. Since it next flows to the compressor, this could be dangerous.

The compressor is designed to compress only vapor. If liquid is drawn in and attempts are made to compress it, the compressor could break, since liquids are essentially incompressible. The temperature of superheated vapor is higher than liquid refrigerant that has not totally vaporized. A coiled capillary tube with a volatile substance inside is located at the evaporator outlet to sense this difference. Its internal pressure increases and decreases as temperature changes. The coiled end of the tube is closed and attached to the evaporator outlet. The other end terminates in the area above a pressure diaphragm in the expansion valve. When superheated refrigerant vapor reaches the coiled end of the tube, its elevated temperature increases the pressure inside the tube and in the space above the diaphragm. This increase in pressure causes the diaphragm to overcome spring tension in the valve. It positions a needle valve that increases the amount of refrigerant released by the valve. The quantity of refrigerant is increased so that the refrigerant only just evaporates, and the refrigerant vapor does not superheat.

When too much liquid refrigerant is released by the expansion valve, low-temperature liquid refrigerant arrives at the outlet of the evaporator. The result is low pressure inside the temperature bulb and above the expansion valve diaphragm. The superheat spring in the valve moves the needle valve toward the closed position, reducing the flow of refrigerant into the evaporator as the spring overcomes the lower pressure above the diaphragm. (***Figure 11-18***)

Vapor cycle air conditioning systems that have large evaporators experience significant pressure drops while refrigerant is flowing through them. Externally equalized expansion valves use a pressure tap from the outlet of the evaporator to help the superheat spring balance the diaphragm. This type of expansion valve is easily recognizable by the additional small-diameter line that comes from the evaporator into the valve (2 total). Better control of the proper amount of refrigerant allowed through the valve is attained by considering both the temperature and pressure of the evaporator refrigerant. (***Figure 11-19***)

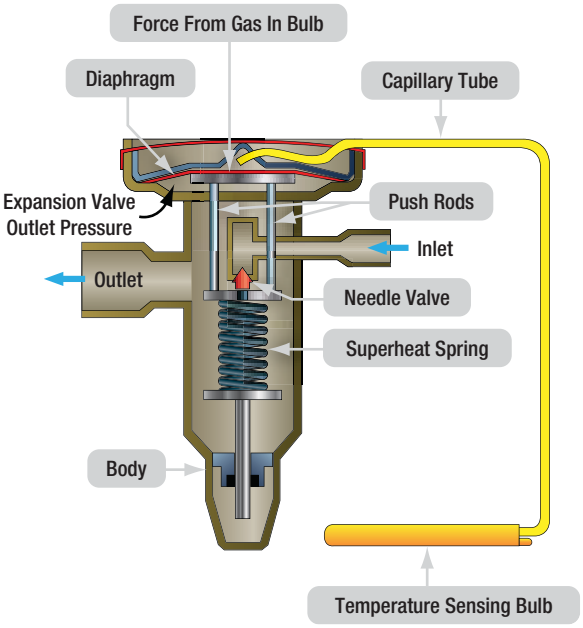


Figure 11-18. An internally equalized expansion valve.

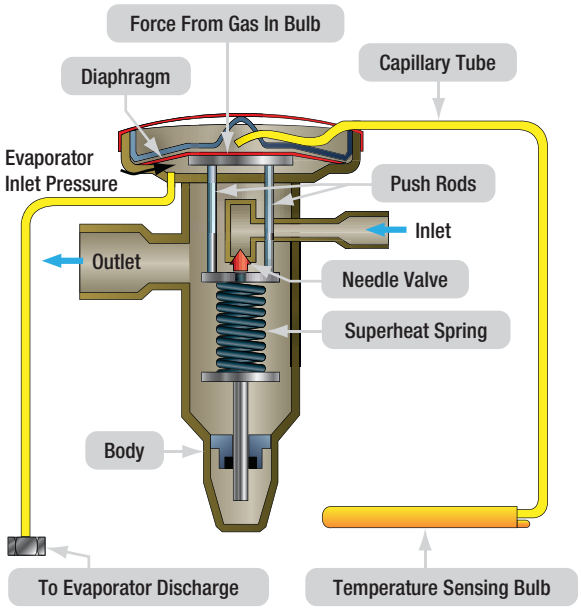


Figure 11-19. An externally equalized expansion valve uses evaporator discharge temperature and pressure to regulate the amount of refrigerant passing through the valve and into the evaporator.

Evaporator

Most evaporators are constructed of copper or aluminum tubing coiled into a compact unit. Fins are attached to increase surface area, facilitating rapid heat transfer between the cabin air blown over the outside of the evaporator with a fan and the refrigerant inside.

The expansion valve located at the evaporator inlet releases high-pressure, high-temperature liquid refrigerant into the evaporator. As the refrigerant absorbs heat from the cabin air, it changes into a low-pressure vapor. This is discharged from the evaporator outlet to the next component in the vapor cycle system, the compressor. The temperature and pressure pickups that regulate the expansion valve are located at the evaporator outlet. The evaporator is situated in such a way that cabin air is pulled to it by a fan. The fan blows the air over the evaporator and discharges the cooled air back into the cabin. (***Figure 11-20***)

This discharge can be direct when the evaporator is located in a cabin wall. A remotely located evaporator may require ducting from the cabin to the evaporator and from the evaporator back into the cabin. Sometimes the cool air produced may be introduced into an air distribution system where it can blow directly on the occupants through individual delivery vents. In this manner, the entire vapor cycle air conditioning system may be located fore or aft of the cabin. A multi-position fan switch controlled by the pilot is usually available.

Figure 11-21 diagrams the vapor cycle air conditioning system in a Cessna Mustang very light jet. It has two evaporators that share in the cooling, with outlets integrated into a distribution system and cockpit mounted switches for the fans, as well as engaging and disengaging the system. When cabin air is cooled by flowing over the evaporator, it can no longer retain the water that it could at higher temperature. As a result, it condenses on the outside of the evaporator and needs to be collected and drained overboard. Pressurized aircraft may contain a valve in the evaporator drain line that opens only periodically to discharge the water, to maintain pressurization. Fins on the evaporator must be kept from being damaged, which could inhibit airflow. The continuous movement of warm cabin air around the fins keeps condensed water from freezing. Ice on the evaporator reduces the efficiency of the heat exchange to the refrigerant.

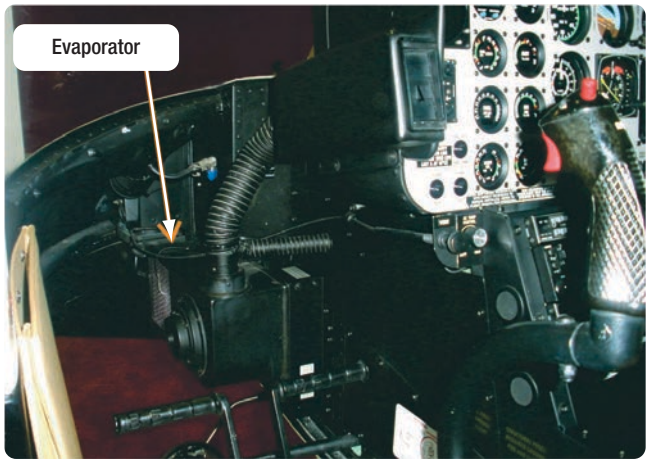


Figure 11-20. The evaporator of this aircraft's vapor cycle air conditioning system is visible in the forward cabin sidewall behind the right rudder pedal.

Compressor

The compressor is the heart of the vapor cycle air conditioning system. It circulates the refrigerant around the vapor cycle system. It receives low-pressure, low-temperature refrigerant vapor from the outlet of the evaporator and compresses it. As the pressure is increased, the temperature also increases. The refrigerant temperature is raised above that of the outside air temperature. The refrigerant then flows out of the compressor to the condenser where it gives off the heat to the outside air.

The compressor is the dividing point between the low side and the high side of the vapor cycle system. Often it is incorporated with fittings or has fittings in the connecting lines to it that are designed to service the system with refrigerant. Access to the low and high sides of the system are required for servicing, which can be accomplished with fitting upstream and downstream of the compressor.

Modern compressors are either engine-driven or driven by an electric motor. Occasionally, a hydraulically driven compressor is used. A typical engine-driven compressor, similar to that found in an automobile, is located in the engine nacelle and operated by a drive belt off of the engine crankshaft. An electromagnetic clutch engages when cooling is required, which causes the compressor to operate. When cooling is sufficient, power to the clutch is cut, and the drive pulley rotates but the compressor does not. (***Figure 11-22***)

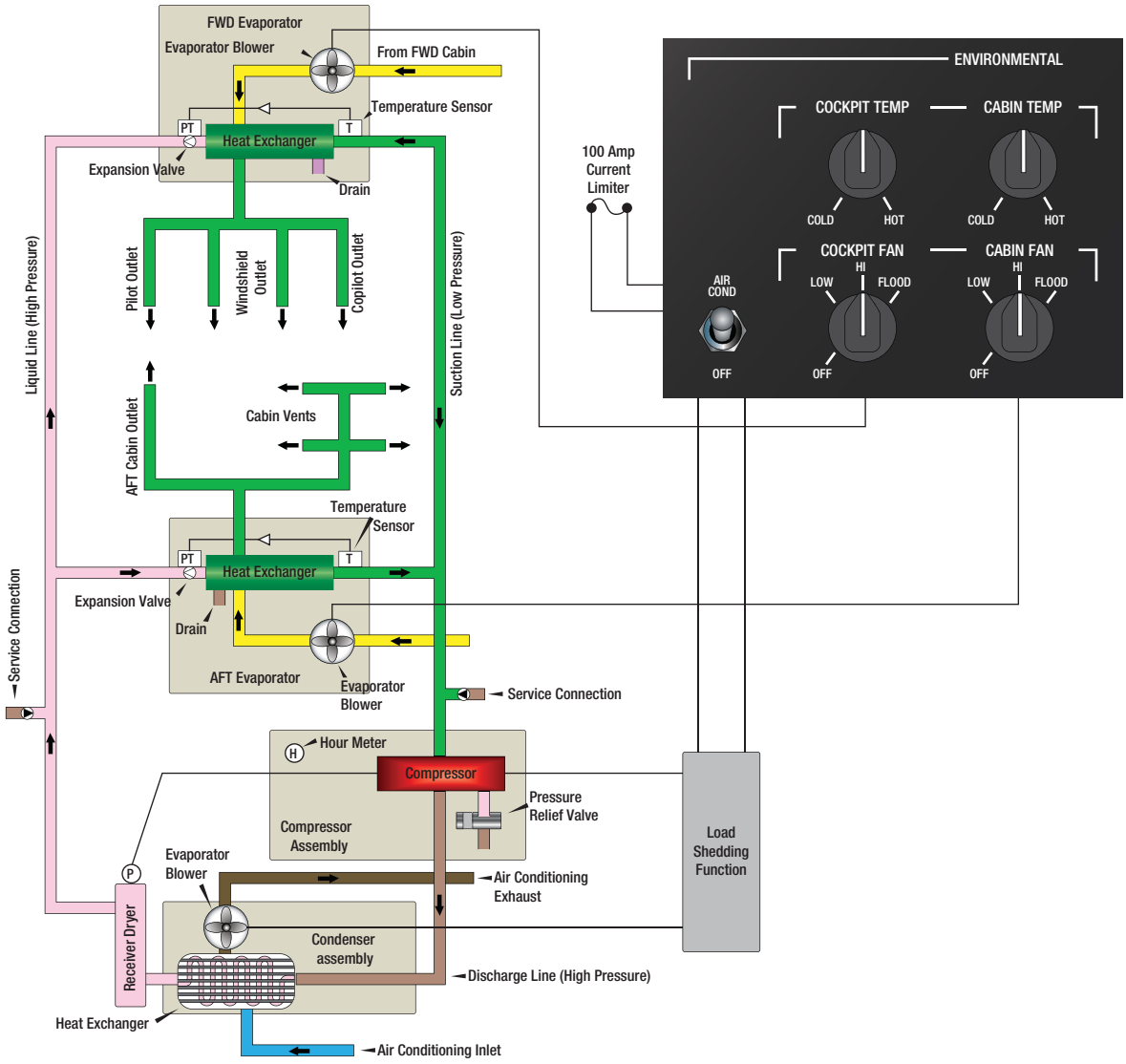


Figure 11-21. The vapor cycle air conditioning system on a Cessna Mustang has two evaporators, one for the cockpit and one for the cabin. Each evaporator assembly contains the evaporator, a blower, a thermal expansion valve and the temperature feedback line from the outlet of the evaporator to the expansion valve.

Dedicated electric motor-driven compressors are also used on aircraft. Use of an electric motor allows the compressor to be located nearly anywhere on the aircraft, since wires can be run from the appropriate bus to the control panel and to the compressor. (***Figure 11-23***) Hydraulically driven compressors are also able to be remotely located. Hydraulic lines from the hydraulic manifold are run through a switch activated solenoid to the compressor. The solenoid allows fluid to the compressor or bypasses it. This controls the operation of the hydraulically driven compressor.

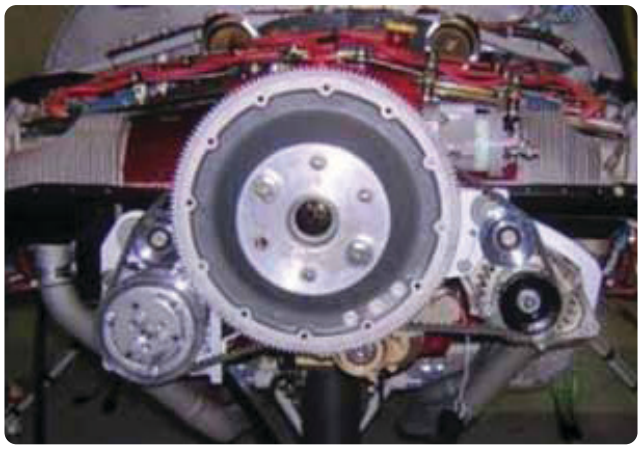


Figure 11-22. A typical belt drive engine driven compressor. The electromagnetic clutch pulley assembly in the front starts and stops the compressor depending on cooling demand.

Regardless of how the vapor cycle air conditioning compressor is driven, it is usually a piston type pump. It requires use of a lightweight oil to lubricate and seal

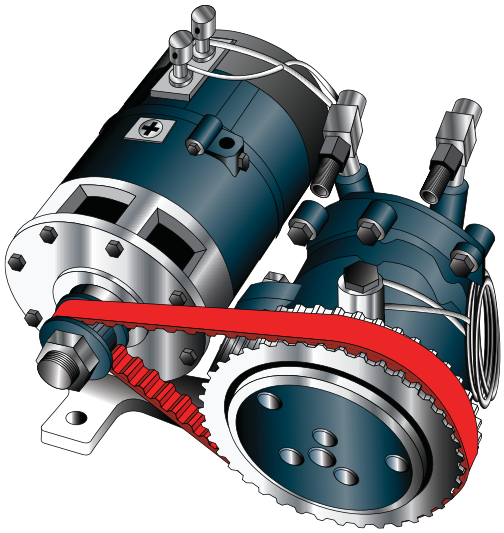


Figure 11-23. Examples of electric motor driven vapor cycle air conditioning compressors.

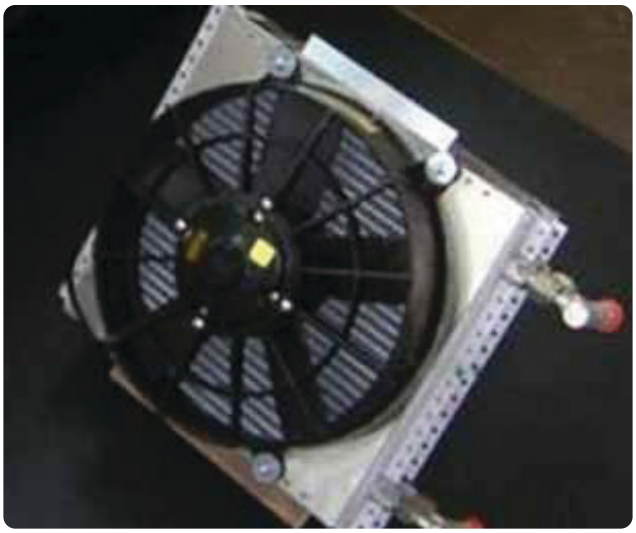


Figure 11-24. A vapor cycle air conditioning condenser assembly with an integral fan pulling outside air through the unit during ground operations.

the unit. The oil is entrained by the refrigerant and circulates with it around the system. The crankcase of the compressor retains a supply of the oil, the level of which can be checked and adjusted by the technician. Valves exist on some compressor installations that can be closed to isolate the compressor from the remainder of the vapor cycle system while oil servicing takes place.

Condenser

The condenser is the final component in the vapor cycle. It is a radiator-like heat exchanger situated so that outside air flows over it and absorbs heat from the high-pressure, high temperature refrigerant received from the compressor. A fan is usually included to draw the air through the compressor during ground operation. On some aircraft, outside air is ducted to the compressor. On others, the condenser is lowered into the airstream from the fuselage via a hinged panel.

Often, the panel is controlled by a switch on the throttle levers. It is set to retract the compressor and streamline the fuselage when full power is required. (***Figure 11-24***)

The outside air absorbs heat from the refrigerant flowing through the condenser. The heat loss causes the refrigerant to change state back into a liquid. The high-pressure liquid refrigerant then leaves the condenser and flows to the receiver dryer. A properly engineered system that is functioning normally fully condenses all the refrigerant flowing through the condenser.

Service Valves

All vapor cycle air conditioning systems are closed systems; however, access is required for servicing. This is accomplished through the use of two service valves. One valve is located in the high side of the system and the other in the low side. A common type of valve used on vapor cycle systems that operate with R12 refrigerant is the Schrader valve. It is similar to the valve used to inflate tires. (***Figure 11-25***) A central valve core seats and unseats by depressing a stem attached to it. A pin in the servicing hose fitting is designed to do this when screwed onto the valve's exterior threads. All aircraft service valves should be capped when not in use.

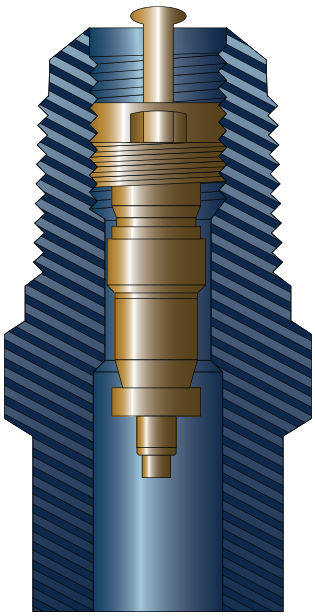


Figure 11-25. Cross-section of an R12 refrigerant service valve.

R134a systems use valves that are very similar to the Schrader valve in function, operation, and location. As a safety device to prevent inadvertent mixing of refrigerants, R134a valve fittings are different from Schrader valve fittings and do not attach to Schrader valve threads. The R134a valve fittings are a quick-disconnect type.

Another type of valve called a compressor isolation valve is used on some aircraft. It serves two purposes. Like the Schrader valve, it permits servicing the system with refrigerant. It also can isolate the compressor so the oil level can be checked and replenished without opening the entire system and losing the refrigerant charge. These valves are usually hard mounted to the inlet and outlet of the compressor.

A compressor isolation valve has three positions. When fully open, it back seats and allows the normal flow of refrigerant in the vapor cycle. When fully closed or front seated, the valve isolates the compressor from the rest of the system and servicing with oil, or even replacement of the compressor, is possible without losing the refrigerant charge. When in an intermediate position, the valve allows access to the system for servicing. The system can be operated with the valve in this position, but should be back seated for normal operation. The valve handle and service port should be capped when servicing is complete. (***Figure 11-26***)

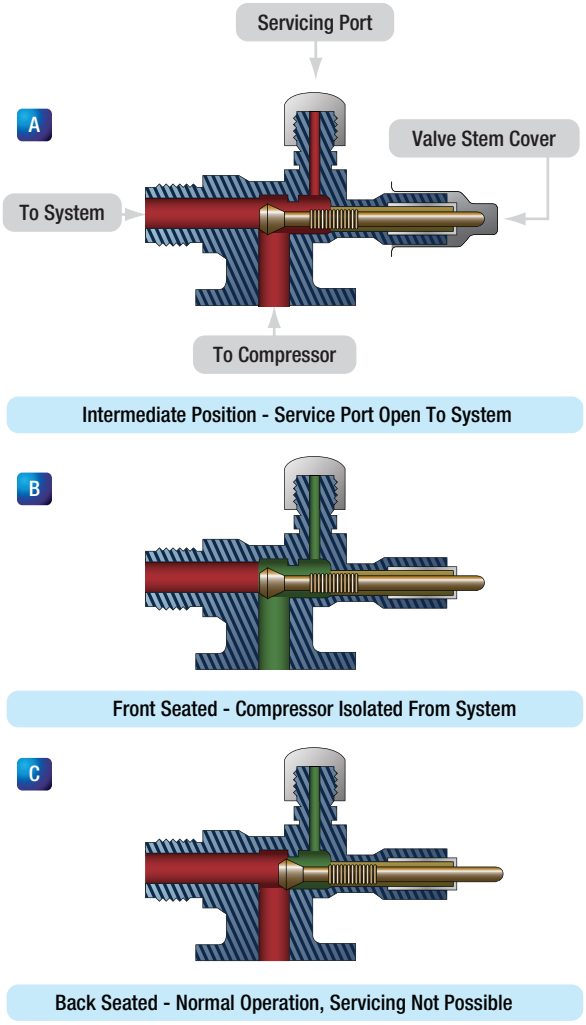


Figure 11-26. Compressor isolation valves isolate the compressor for maintenance or replacement. They also allow normal operation and servicing of the vapor cycle air conditioning system with refrigerant.

AIR CONDITIONING SERVICING

Special servicing equipment is used to service vapor cycle air conditioning systems. The U.S. Environmental Protection Agency (EPA) has declared it illegal to release R12 refrigerant into the atmosphere. Equipment has been designed to capture the refrigerant during the servicing process. Although R134a does not have this restriction, it is illegal in some locations to release it to the atmosphere, and it may become universally so in the near future. It is good practice to capture all refrigerants for future use, rather than to waste them or to harm the environment by releasing them into the atmosphere.

Capturing the refrigerant is a simple process designed into the proper servicing equipment. The technician should always be vigilant to use the approved refrigerant for the system being serviced and should follow all manufacturer's instruction.

MANIFOLD SET, GAUGES, HOSES, AND FITTINGS

In the past, the main servicing device for vapor cycle air conditioning systems was the manifold set. It contains three hose fittings, two O-ring sealed valves, and two gauges. It is essentially a manifold into which the gauges, fittings, and valves are attached. The valves are positioned to connect or isolate the center hose with either fitting.

Hoses attach to the right and left manifold set fittings and the other ends of those hoses attach to the service valves in the vapor cycle system. The center fitting also has a hose attached to it. The other end of this hose connects to either a refrigerant supply or a vacuum pump, depending on the servicing function to be performed. All servicing operations are performed by manipulating the valves. (***Figure 11-27***)



Figure 11-27. A basic manifold set for servicing a vapor cycle air conditioning system.

The gauges on the manifold set are dedicated - one for the low side of the system and the other for the high side. The low-pressure gauge is a compound gauge that indicates pressures above or below atmospheric pressure (0 gauge pressure). Below atmospheric pressure, the gauge is scaled in inches of mercury down to 30 inches. This is to indicate vacuum. 29.92 inches equals an absolute vacuum (absolute zero air pressure). Above atmospheric pressure, gauge pressure is read in psi.

The scale typically ranges from 0 to 60 psi, although some gauges extend up to 150 psi. The high-pressure gauge usually has a range from zero up to about 500 psi gauge pressure. It does not indicate vacuum (pressure lower than atmospheric). These gauges and their scales can be seen in ***Figure 11-28***.

The low-pressure gauge is connected on the manifold directly to the low side fitting. The high-pressure gauge connects directly to the high side fitting. The center fitting of the manifold can be isolated from either of the gauges or the high and low service fittings by the hand valves. When these valves are turned fully clockwise, the center fitting is isolated. If the low pressure valve is opened (turned counterclockwise), the center fitting is opened to the low pressure gauge and the low side service line. The same is true for the high side when the high-pressure valve is opened. (***Figure 11-28***)

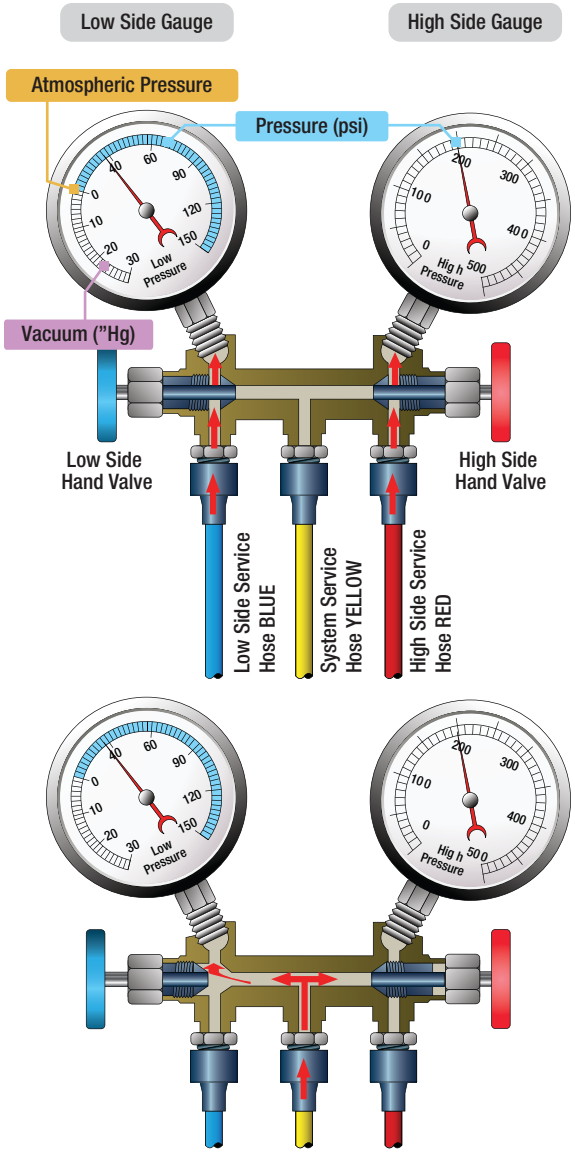


Figure 11-28. The internal workings of a manifold set with the center fitting isolated (top). Opening a valve connects the center hose to that side of the system and the gauge (bottom).

Special hoses are attached to the fittings of the manifold valve for servicing the system. The high-pressure charging hose is usually red and attaches to the service valve located in the high side of the system. The low-pressure hose, usually blue, attaches to the service valve that is located in the low side of the system. The center hose attaches to the vacuum pump for evacuating the system, or to the refrigerant supply for charging the system. Proper charging hoses for the refrigerant specific service valves must be used. When not using the manifold set, be sure the hoses are capped to prevent moisture from contaminating the valves.

AIR CONDITIONING SERVICE

Regulations that require capture of all vapor cycle refrigerant have limited the use of the manifold set. It can still be used to charge a system. The refrigerant container is attached to the center hose and the manifold set valves are manipulated to allow flow into the low or high side of the system as required. But, emptying a system of refrigerant requires a service unit made to collect it.

Allowing the refrigerant to flow into a collection container attached to the center hose will not capture the entire refrigerant charge, as the system and container pressures equalize above atmospheric pressure. An independent compressor and collection system is required.

Modern refrigeration recharging and recovery units are available to perform all of the servicing functions required for vapor cycle air conditioning systems. These all-in-one service carts have the manifold set built into the unit. As such, the logic for using a manifold set still applies. Integral solenoid valves, reservoirs, filters, and smart controls allow the entire servicing procedure to be controlled from the unit panel once the high side and low side services hoses are connected. A built-in compressor enables complete system refrigerant purging. A built-in vacuum pump performs system evacuation. A container and recycling filters for the refrigerant and the lubricating oil allow total recovery and recycling of these fluids. The pressure gauges used on the service unit panel are the same as those on a manifold set. Top-of-the-line units have an automatic function that performs all of the servicing functions sequentially and automatically once the hoses are hooked up to the vapor cycleair conditioning system and the system quantity of refrigerant has been entered. (***Figure 11-29***)

REFRIGERANT SOURCE

R134a comes in containers measured by the weight of the refrigerant they hold. Small 12-ounce to 2½-pound cans are common for adding refrigerant. Larger 30 and 50 pound cylinders equipped with shutoff valves are often used to charge an evacuated system, and they are used in shops that service vapor cycle systems frequently. (***Figure 11-30***) These larger cylinders are also used in the full servicing carts described above. The amount of refrigerant required for any system is measured in pounds. Check the manufacturer's service data and charge the system to the level specified using only the approved refrigerant from a known source.



Figure 11-29. A modern refrigerant recovery/recycle/charging service unit. Electronic control of solenoid activated valves combine with a built-in system for recovering, recycling, and recharging. A built-in vacuum pump and heated refrigerant reservoir are also included.



Figure 11-30. A 30 pound R134a refrigerant container with dual fittings. The fitting controlled by the blue valve wheel opens to the vapor space above the liquid refrigerant for connection to the low side of the vaporcycle system. The fitting controlled by the red valve wheel draws liquid refrigerant from the bottom of the cylinder through a stand tube. This fitting is connected to the system high side. On containers without dual fittings, the container must be inverted to deliver liquid refrigerant through a connected hose.

VACUUM PUMPS

Vacuum pumps used with a manifold set, or as part of a service cart, are connected to the vapor cycle system so that the system pressure can be reduced to a near total vacuum. The reason for doing this is to remove all of the water in the system. As mentioned, water can freeze, causing system malfunction and can also combine with the refrigerant to create corrosive compounds.

Once the system has been purged of its refrigerant and it is at atmospheric pressure, the vacuum pump is operated. It gradually reduces the pressure in the system. As it does, the boiling point of any water in the system is also reduced. Water boils off or is vaporized under the reduced pressure and is pulled from the system by the pump, leaving the system moisture free to be recharged with refrigerant. (***Figure 11-31***)

The strength and efficiency of vacuum pumps varies as does the amount of time to hold the system at reduced pressure specified by manufacturers. Generally, the best established vacuum is held for 15-30 minutes to ensure all water is removed from the system. Follow the manufacturer's instructions when evacuating a vapor cycle air conditioning system. (***Figure 11-32***)

LEAK DETECTORS

Even the smallest leak in a vapor cycle air conditioning system can cause a loss of refrigerant. When operating normally, little or no refrigerant escapes. A system that requires the addition of refrigerant should be suspected of having a leak.

Electronic leak detectors are safe, effective devices used to find leaks. There are many types available that are able to detect extremely small amounts of escaped refrigerant. The detector is held close to component and hose connections where most leaks occur. Audible and visual alarms signal the presence of refrigerant. A detector specified for the type of refrigerant in the system should be chosen. A good leak detector is sensitive enough to detect leaks that would result in less than ½-ounce of refrigerant to be lost per year. (***Figure 11-33***)

Other leak detection methods exist. A soapy solution can also be applied to fittings and inspected for the formation of bubbles indicating a leak. Special leak detection dyes compatible for use with refrigerant can be injected into the vapor cycle system and can be seen when they are forced out at a leak. Many of these are made to be visible under UV light. Occasionally, a leak can be detected upon close visual inspection. Oil in the system can be forced out of a leak, leaving a visible residue that is usually on the bottom side of a leaky fitting.

Old hoses may become slightly porous and leak a significant amount of refrigerant over time. Because of the length and area through which the

Inches of Vacuum on Low Side Gauge (inches Hg)	Temperature at Which Water Boils (°F)	Absolute Pressure (psi)
0	212	14.696
4.92	204.98	12.279
9.23	194	10.152
15.9 4	176	6.866
20.7 2	158	4.519
24.0 4	140	2.888
26.2 8	122	1.788
27.7 5	104	1.066
28.6 7	86	0.614
28.9 2	80.06	0.491
29.0 2	75.92	0.442
29.1 2	71.96	0.393
29.2 2	69.08	0.344
29.3 2	64.04	0.295
29.4 2	59	0.246
29.5 2	53.06	0.196
29.6 2	44.96	0.147
29.7 4	32	0.088
29.8 2	21.02	0.0049
29.8 7	6.08	0.00245
29.9 1	–23.98	0.00049

Figure 11-31. When the temperature is low, a greater amount of vacuum is needed to boil off and remove any water in the vapor cycle system.



Figure 11-32. A vacuum pump is used to lower the pressure in the vapor cycle air conditioning system. This reduces the boiling point of water in the system, which vaporizes and is drawn out by the pump.



Figure 11-33. This electronic infrared leak detector can detect leaks that would lose less than ¼ ounce of refrigerant per year.

refrigerant is lost, this type of leak may be difficult to detect, even with leak detecting methods. Visibly deteriorated hoses should be replaced.

SYSTEM SERVICING

Vapor cycle air conditioning systems can give many hours of reliable, maintenance-free service. Periodic visual inspections,tests, and refrigerant level and oil level checks may be all that is required for some time. Follow the manufacturer's instructions for inspection criteria and intervals.

Visual Inspection

All components of any vapor cycle system should be checked to ensure they are secure. Be vigilant for any damage, misalignment, or visual signs of leakage. The evaporator and condenser fins should be checked to ensure they are clean, unobstructed, and not folded over from an impact. Dirt and inhibited airflow through the fins can prevent effective heat exchange to and from the refrigerant. Occasionally, these units can be washed. Since the condenser often has ram air ducted to it or extends into the airstream, check for the presence of debris that may restrict airflow. Hinged units should be checked for security and wear. The mechanism to extend and retract the unit should function as specified, including the throttle position switch present on many systems. It is designed to cut power to the compressor clutch and retract the condenser at full power settings. Condensers may also have a fan to pull air over them during ground operation. It should be checked to ensure it functions correctly. (***Figure 11-34***)

Be sure the capillary temperature feedback sensor to the expansion valve is securely attached to the evaporator outlet. Also, check the security of the pressure sensor and thermostat sensor if the system has them. The evaporator should not have ice on the outside.

This prevents proper heat exchange to the refrigerant from the warm cabin air blown over the unit. The fan blower should be checked to ensure it rotates freely. Depending on the system, it should run whenever the cooling switch is selected and should change speeds as the selector is rotated to more or less cooling. Sometimes systems low on refrigerant can cause ice on the evaporator, as can a faulty expansion valve or feedback control line. Ice formation anywhere on the outside of a vapor cycle air conditioning system should be investigated for cause and corrected. (***Figure 11-35***)

Security and alignment of the compressor is critical and should be checked during inspection. Belt-driven compressors need to have proper belt tension to function properly. Check the manufacturer's data for information on how to determine the condition and tension of the belt, as well as how to make adjustments. Oil level should be sufficient. Typically, ¼-ounce of oil is added for each pound of refrigerant added to the system. When changing a component, additional oil may need to be added to replace that which is trapped in the replaced unit. Always use the oil specified in the manufacturer's maintenance manual.

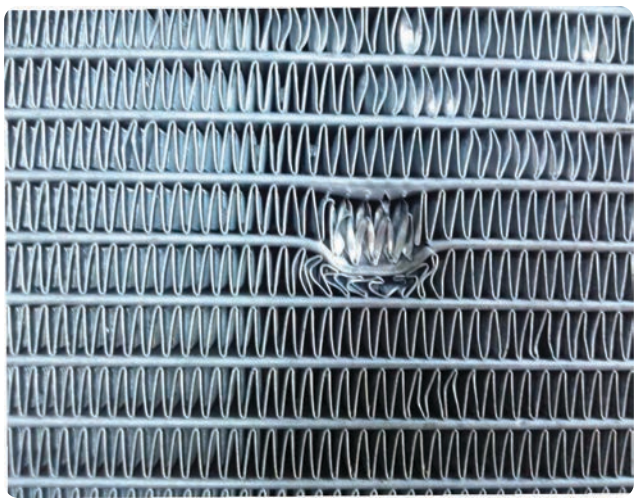


Figure 11-34. Damaged fins on a condenser.

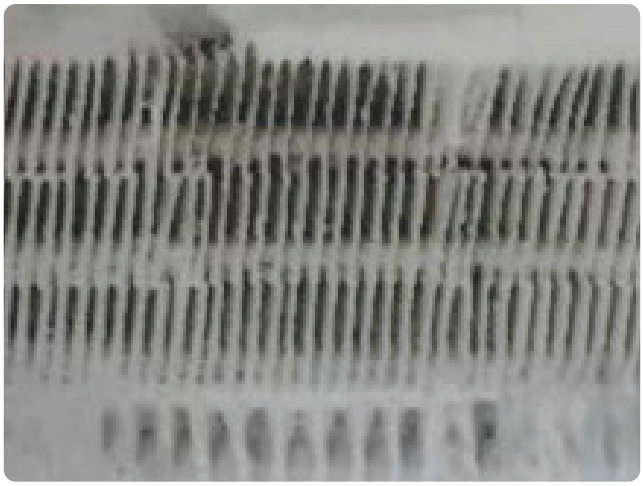


Figure 11-35. Ice on the evaporator coils is cause for investigation. It prevents proper heat exchange to the refrigerant.

Leak Test

As mentioned under the leak detector section above, leaks in a vapor cycle air conditioning system must be discovered and repaired. The most obvious sign of a possible leak is a low refrigerant level. Bubbles present in the sight glass of the receiver dryer while the system is operating indicate more refrigerant is needed. A system check for a leak may be in order. Note that vapor cycle systems normally lose a small amount of refrigerant each year. No action is needed if this amount is within limits.

Occasionally, all of the refrigerant escapes from the system. No bubbles are visible in the sight glass, but the complete lack of cooling indicates the refrigerant has leaked out. To locate the leak point, the system needs to be partially charged with refrigerant so leak detection methods can be employed. About 50 psi of refrigerant in the high and low sides should be sufficient for a leak check. By introducing the refrigerant into the high side, pressure indicated on the low side gauge verifies the orifice in the expansion valve is not clogged. When all refrigerant is lost due to a leak, the entire system should be checked. Each fitting and connection should be inspected visually and with a leak detector.

When a vapor cycle air conditioning system loses all of its refrigerant charge, air may enter the system. Water may also enter since it is in the air. This means that a full system evacuation must be performed after the leak is found and repaired. By establishing only a 50 psi charge in a depleted system, the leak(s) becomes detectable, but time and refrigerant are not wasted prior to evacuation. System evacuation is discussed below.

Performance Test

Verification of proper operation of a vapor cycle air conditioning system is often part of a performance test. This involves operating the system and checking parameters to ensure they are in the normal range. A key indication of performance is the temperature of the air that is cooled by the evaporator. This can be measured at the air outflow from the evaporator or at a nearby delivery duct outlet. An ordinary thermometer should read 40 - 50 °F, with the controls set to full cold after the system has been allowed to operate for a few minutes. Manufacturer's instructions include information on where to place the thermometer and the temperature range that indicates acceptable performance.

Pressures can also be observed to indicate system performance. Typically, low side pressure in a vapor cycle system operating normally is 10-50 psi, depending on ambient temperature. High side pressure is between 125 and 250 psi, again, depending on ambient temperature and the design of the system. All system performance tests are performed at a specified engine rpm (stable compressor speed) and involve a period of time to stabilize the operation of the vapor cycle. Consult the manufacturer's instructions for guidance.

Feel Test

A quick reference field test can be performed on a vapor cycle air conditioning system to gauge its health. In particular, components and lines in the high side (from the compressor to the expansion valve) should be warm or hot to the touch. The lines on both sides of the receiver dryer should be the same temperature. Low side lines and the evaporator should be cool. Ice should not be visible on the outside of the system. If any discrepancies exist, further investigation is needed. On hot, humid days, the cooling output of the vapor cycle system may be slightly compromised due to the volume of water condensing on the evaporator.

Purging the System

Purging the system means emptying it of its refrigerant charge. Since the refrigerant must be captured, a service cart with this capability should be used. By connecting the hoses to the high side and low side service valves and selecting recover, cart solenoid valves position so that a system purging compressor pumps the refrigerant out of the vapor cycle system and into a recovery tank.

Vapor cycle systems must be properly purged before opening for maintenance or component replacement. Once opened, precautions should be taken to prevent contaminants from entering the system. When suspicion exists that the system has been contaminated, such as when a component has catastrophically failed, it can be flushed clean. Special fluid flush formulated for vapor cycle air conditioning systems should be used. The receiver dryer is removed from the system for flushing and a new unit is installed, as it contains fresh filters. Follow the aircraft manufacturer's instructions.

Checking Compressor Oil

The compressor is a sealed unit in the vapor cycle system that is lubricated with oil. Any time the system

is purged, it is an opportunity to check the oil quantity in the compressor crankcase. This is often done by removing a filler plug and using a dip stick. Oil quantity should be maintained within the proper range using oil recommended by the manufacturer. Be certain to replace the filler plug after checking or adding oil. (***Figure 11-36***)

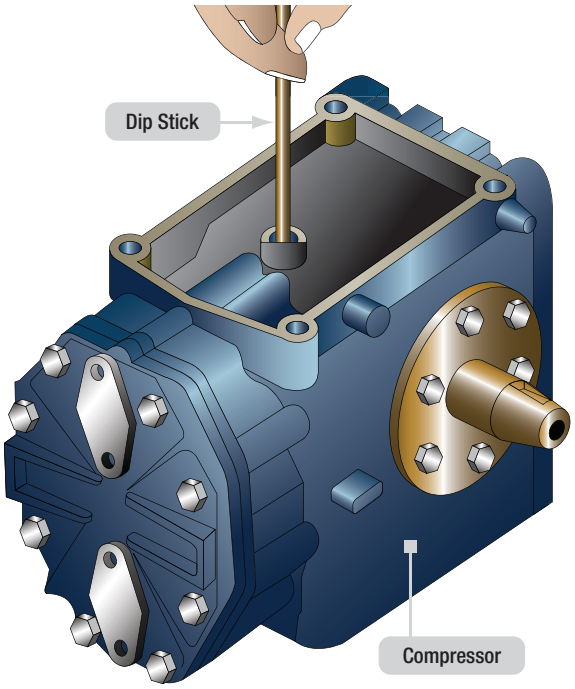


Figure 11-36. Checking the compressor oil when the system is open.

EVACUATING THE SYSTEM

Only a few drops of moisture can contaminate a vapor cycle air conditioning system. If this moisture freezes in the expansion valve, it could completely block the refrigerant flow. Water is removed from the system by evacuation. Anytime the system refrigerant charge falls below atmospheric pressure, the refrigerant is lost, or the system is opened, it must be evacuated before recharging.

Evacuating a vapor cycle air conditioning system is also known as pumping down the system. A vacuum pump is connected and pressure inside the system is reduced to vaporize any water that may exist. Continued operation of the vacuum pump draws the water vapor from the system. A typical pump used for evacuating an air conditioning system can reduce system pressure to about 29.62 "Hg (gauge pressure). At this pressure, water boils at 45 ˚F. Operate the vacuum pump to achieve the recommended gauge pressure. Hold this vacuum for as long as the manufacturer specifies.

As long as a vapor cycle air conditioning system retains a charge higher than atmospheric pressure, any leak forces refrigerant out of the system. The system pressure prevents air (and water vapor) from entering. Therefore, it is permissible to recharge or add refrigerant to a system that has not dropped below atmospheric pressure without evacuating the system.

CHARGING THE SYSTEM

Charging capacity of a vapor cycle air conditioning system is measured by weight. The aircraft manufacturer's maintenance manual specifies this amount and the amount and type of oil to be put into the system when filling. Preweighing the refrigerant or setting the refrigerant weight into the servicing cart input ensures the system is filled to capacity.

Charging a vapor cycle air conditioning system should be undertaken immediately after evacuation of the system is completed. With the hoses still connected to the high and low side service valves, selecting charge on the service cart panel positions solenoid operated valves so that the refrigerant supply is available. First, refrigerant is released into the high side of the system. Observe the low side gauge. When the low side gauge begins to indicate pressure, it is known that refrigerant is passing through the tiny orifice in the expansion valve. As pressure builds in the high side, the flow of refrigerant into the system stops.

To complete the charge of the system, refrigerant needs to be drawn in by the compressor. A major concern is to avoid damage to the compressor by having liquid refrigerant enter the compressor inlet. After the initial release of refrigerant into the high side, the high side service valve is closed and the remaining charge is made through the low side service valve. The engine is started and run at a specified rpm, usually a high idle speed. Full cool is selected on the air conditioning control panel in the cockpit. As the compressor operates, it draws vapor into the low side until the correct weighed amount of refrigerant is in the system. Charging is completed with a full performance test.

Charging with a manifold set is accomplished in the same way. The manifold center hose is connected to the refrigerant source that charges the system. After opening the valve on the container (or puncturing the seal on a small can), the center hose connection on the manifold

set should be loosened to allow air in the hose to escape. Once the air is bled out of the hose, the refrigerant can enter the system through whichever service valve is opened. The sequence is the same as above and all manufacturer instructions should be followed.

Oil quantity added to the system is specified by the manufacturer. Refrigerant premixed with oil is available and may be permissible for use. This eliminates the need to add oil separately. Alternately, the amount of oil to be put into the system can be selected on the servicing cart. Approximately ¼-ounce of oil for each pound of refrigerant is a standard amount; however, follow the manufacturer's specifications.

DISTRIBUTION SYSTEMS

Distribution of cabin air on pressurized aircraft is managed with a system of air ducts leading from the pressurization source into and throughout the cabin. Typically, air is ducted to and released from ceiling vents, where it circulates and flows out floor-level vents. The air then flows aft through the baggage compartments and under the floor area. It exits the pressure vessel through the outflow valve(s) mounted low, on, or near the aft pressure bulkhead. The flow of air is nearly imperceptible. Ducting is hidden below the cabin floor and behind walls and ceiling panels depending on the aircraft and system design. Valves to select pressurization air source, ventilating air, temperature trim air, as well as in line fans and jet pumps to increase flow in certain areas of the cabin, are all components of the air distribution system. Temperature sensors, overheat switches, and check valves are also common.

On turbine powered aircraft, temperature controlled air from the air conditioning system is the air that is used to pressurize the cabin. The final regulation of the temperature of that air is sometimes considered part of the distribution system. Mixing air conditioned air with bleed air in a duct or a mixing chamber allows the crew to select the exact temperature desired for the cabin. The valve for mixing is controlled in the cockpit or cabin by a temperature selector. Centralized manifolds from which air can be distributed are common. (***Figure 11-37***)

Large aircraft may be divided into zones for air distribution. Each zone has its own temperature selector and associated valve to mix conditioned and bleed air so that each zone can be maintain at a temperature independent of the others. The air distribution system on most aircraft makes provisions for ducting and circulating cooling air to electronics equipment bays. It also contains a gasper air system. This is air ducted from the cold air manifold or duct to an overhead adjustable delivery nozzle at each passenger station. An inline fan controlled from the cockpit supplies a steady stream of gasper air that can be regulated or shut off with the delivery nozzle(s). (***Figure 11-38***)

FLOW CONTROL

Most commercial airliners have some sort of mass air flow control system. The parameters of flight constantly change. Adjustments to engine speeds affects bleed air flow. Output from the air conditioning pack varies. Altitude and air density vary. To ensure a stable mass of air flow through the cabin a mass airflow controller adjusts the flow to a predetermined level. Flow from the cabin air source is regulated to meet these demands. Flow of air leaving the cabin is also controlled. Mechanisms to control the flow of bleed air vary on different aircraft and different engines. During takeoff and climb, engine rpm is high. Air flow, temperature and pressure through the engine compressor are also high.

Locating a bleed port at an early stage of compression provides sufficient bleed air for cabin environmental control systems and other needs. But as engine throttle settings are reduced during cruise and decent, a bleed port located on a higher stage of compression is required for the same volume and pressure of bleed air to be

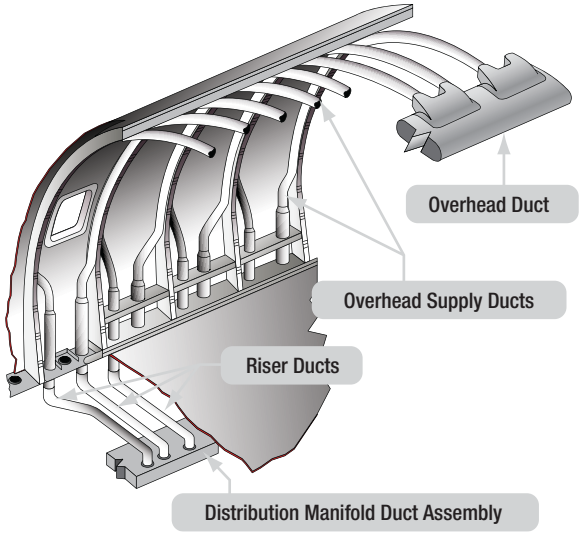


Figure 11-37. Centralized manifolds from which air can be distributed are common.

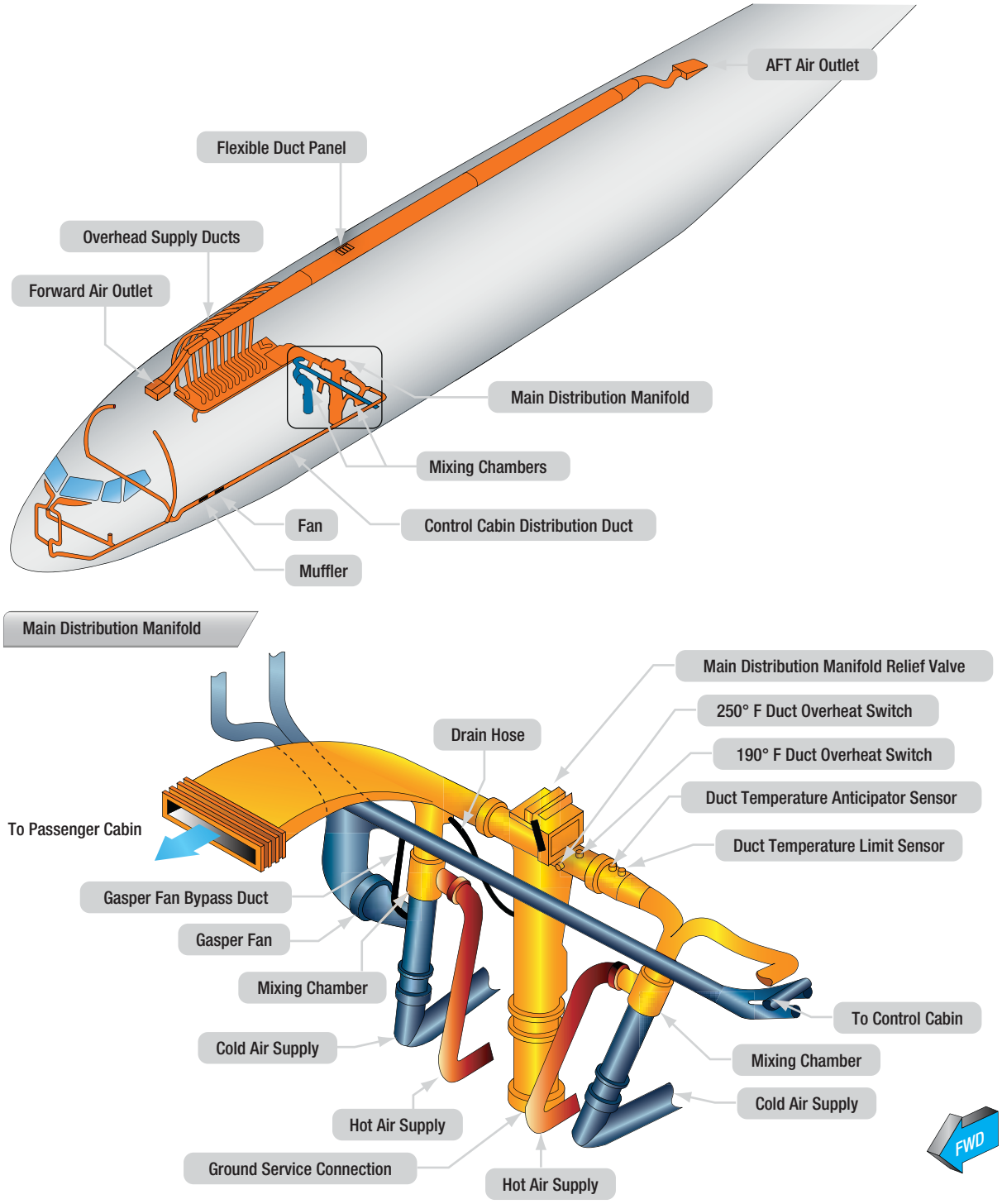


Figure 11-38. The conditioned air distribution system on a Boeing 737. The main distribution manifold is located under the cabin floor. Riser ducts run horizontally then vertically from the manifold to supply ducts, which follow the curvature of the fuselage carrying conditioned air to be released in the cabin.

extracted. Bleed ports at three stages of the compressor is common on turbofan engines.

Control of which bleed port delivers air to the pneumatic system is accomplished in a variety of ways. Typically, a high pressure shutoff valve closes when low-stage bleed air pressure rises with throttle setting increase. This causes bleed air to automatically be extracted from a higher stage port.

TEMPERATURE CONTROL

Most cabin temperature control systems operate in a similar manner. Temperature is monitored in the cabin, cockpit, conditioned air ducts, and distribution air ducts.

These values are input into a temperature controller, or temperature control regulator, normally located in the electronics bay. A temperature selector in the cockpit can be adjusted to input the desired temperature. (***Figure 11-39***)

The temperature controller compares the actual temperature signals received from the various sensors with the desired temperature input. Circuit logic for the selected mode processes these input signals. An output signal is sent to a valve in the air cycle air conditioning system.

This valve has different names depending on the aircraft manufacturer and design of the environmental control systems (i.e., mixing valve, temperature control valve, trim air valve). It mixes warm bleed air that bypassed the air cycle cooling process with the cold air produced by it.

By modulating the valve in response to the signal from the temperature controller, air of the selected temperature is sent to the cabin through the air distribution system.

Cabin temperature pickup units and duct temperature sensors used in the temperature control system are thermistors. Their resistance changes as temperature changes. The temperature selector is a rheostat that varies its resistance as the knob is turned. In the temperature controller, resistances are compared in a bridge circuit. The bridge output feeds a temperature regulating function. An electric signal output is prepared and sent to the valve that mixes hot and cold air. On large aircraft with separate temperature zones, trim air modulating valves for each zone are used. The valves modulate to provide the correct mix required to match the selected temperature.

Cabin, flight deck, and duct temperature sensors are strategically located to provide useful information to control cabin temperature. (***Figure 11-40***)

HUMIDITY CONTROL

Control of cabin humidity primarily occurs during the air cycle air conditioning process. Moisture condensed from the cool air exiting the expansion turbine is removed by the water separator. Most water separators remove water by swilling the air in some fashion so that the heavier water can be drawn off. However, at high altitudes, air from the air cycle air conditioners could be too dry for passenger and crew comfort. A water infiltration system can then be used to introduce moisture into the conditioned air.

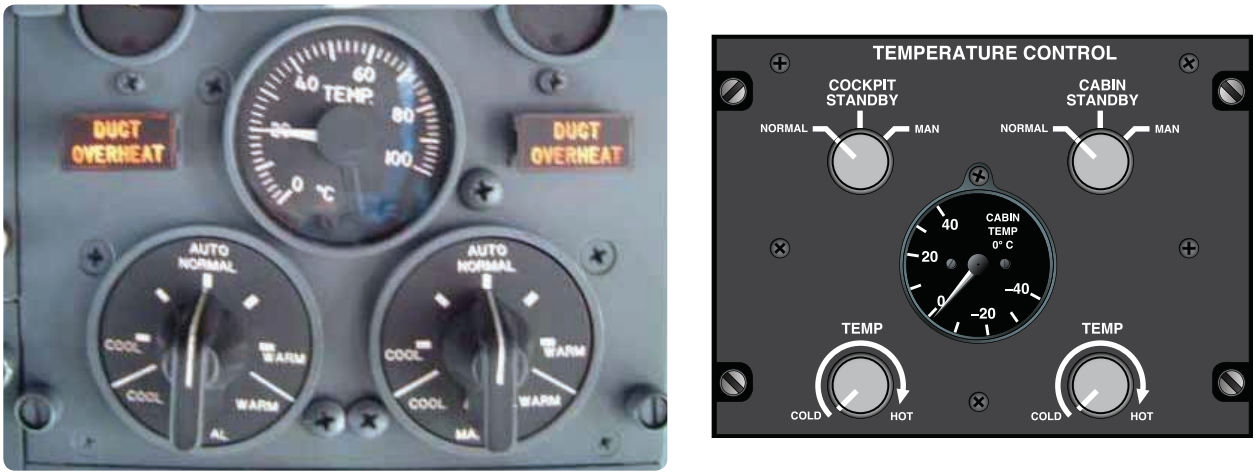


Figure 11-39. Typical temperature selectors on a transport category aircraft temperature control panel in the cockpit (left) and a business jet (right). On large aircraft, temperature selectors may be located on control panels located in a particular cabin air distribution zone.

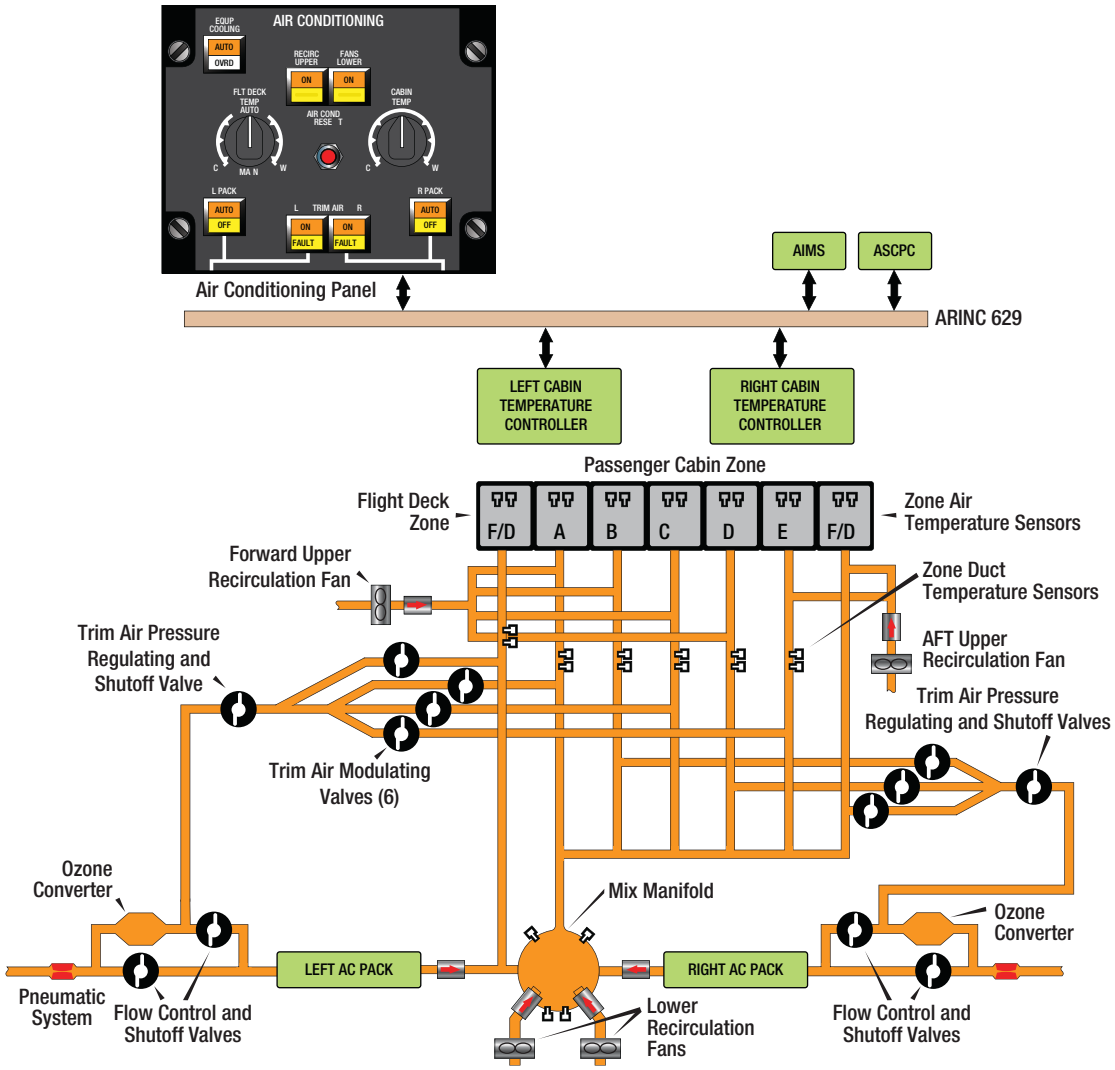


Figure 11-40. The temperature control system of a Boeing 777 combines the use of zone and duct temperature sensors with trim air modulating valves for each zone. Redundant digital left and right cabin temperature controllers process temperature input signals from the sensors and temperature selectors on the cockpit panel and throughout the aircraft to modulate the valves.

In a water infiltration system, the humidity level in the cabin is monitored. Water is pumped from a holding tank and sprayed into the conditioned air before it is directed into the cabin. An electronic humidity controller is used to signal the pump to operate.

PRESSURIZATION

PRESSURE OF THE ATMOSPHERE

The gases of the atmosphere (air), although invisible, have weight. A one square inch column of air stretching from sea level into space weighs 14.7 pounds. Therefore, it can be stated that the pressure of the atmosphere, or atmospheric pressure, at sea level is 14.7 psi. (***Figure 11-41***)

The weight exerted by a 1 square inch column of air stretching from sea level to the top of the atmosphere is what is measured when it is said that atmospheric pressure is equal to 14.7 pounds per square inch. Atmospheric pressure is also known as barometric pressure and is measured with a barometer. (***Figure 11-42***)

Expressed in various ways, such as in inches of mercury or millimeters of mercury, these measurements come from observing the height of mercury in a column when air pressure is exerted on a reservoir of mercury into which the column is set. The column must be evacuated so air inside does not act against the mercury rising. A

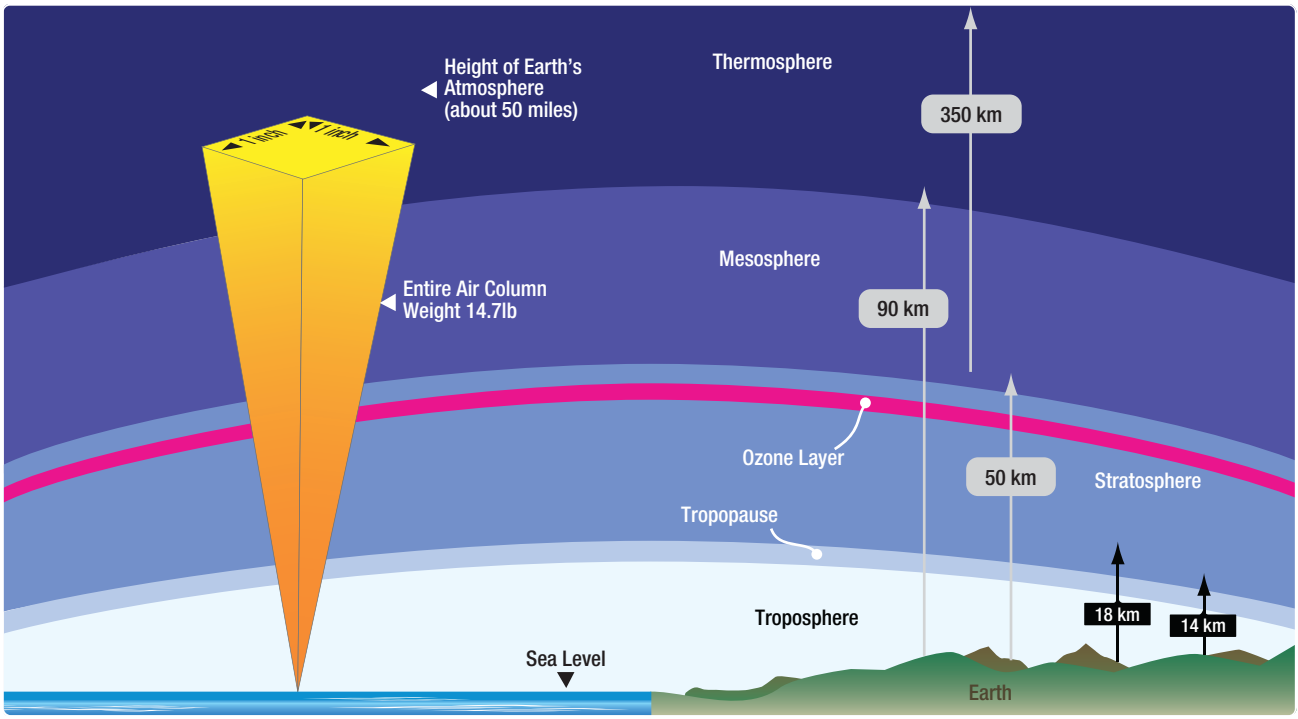


Figure 11-41. The weight exerted by a 1 square inch column of air stretching from sea level to the top of the atmosphere is what is measured when it is said that atmospheric pressure is equal to 14.7 pounds per square inch.

column of mercury 29.92 inches high weighs the same as a column of air that extends from sea level to the top of the atmosphere and has the same cross-section as the column of mercury. (***Figure 11-42***)

The weight of the atmosphere pushes down on the mercury in the reservoir of a barometer, which causes mercury to rise in the column. At sea level, mercury is forced up into the column approximately 29.92 inches. Therefore, it is said that barometric pressure is 29.92 inches of mercury at sea level. Aviators often interchange references to atmospheric pressure between linear displacement (e.g., inches of mercury) and units of force (e.g., psi). Over the years, meteorology has shifted its use of linear displacement representation of atmospheric pressure to units of force. However, the unit of force nearly universally used today to represent atmospheric pressure in meteorology is the hectopascal (hPa). A hectopascal is a metric (SI) unit that expresses force in newtons per square meter. 1 013.2 hPa is equal to 14.7 psi. (***Figure 11-43***)

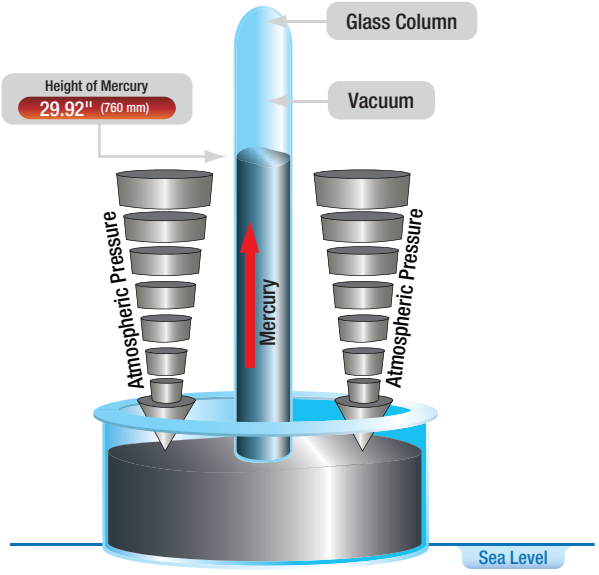


Figure 11-42. The weight of the atmosphere pushes down on the mercury in the reservoir of a barometer, which causes mercury to rise in the column. At sea level, mercury is forced up into the column approximately 29.92 inches. Therefore, it is said that barometric pressure is 29.92 inches of mercury at sea level.

Atmospheric pressure decreases with increasing altitude. The simplest explanation for this is that the column of air that is weighed is shorter. How the pressure changes for a given altitude is shown in ***Figure 11-44***.

The decrease in pressure is a rapid one and, at 50 000 feet, the atmospheric pressure has dropped to almost one-tenth of the sea level value.

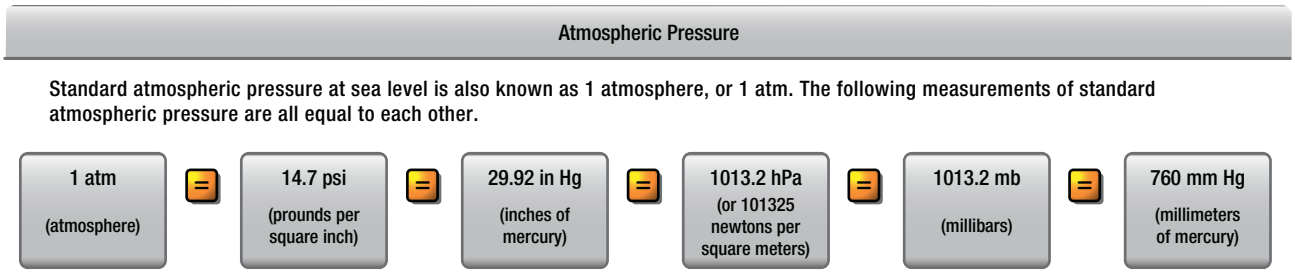


Figure 11-43. Various equivalent representations of atmospheric pressure at sea level.

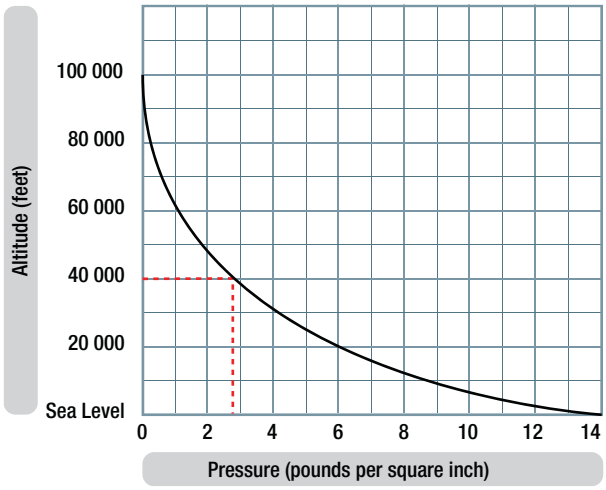


Figure 11-44. Atmospheric pressure decreasing with altitude. At sea level the pressure is 14.7 psi, while at 40 000 feet, as the dotted lines show, the pressure is only 2.72 psi.

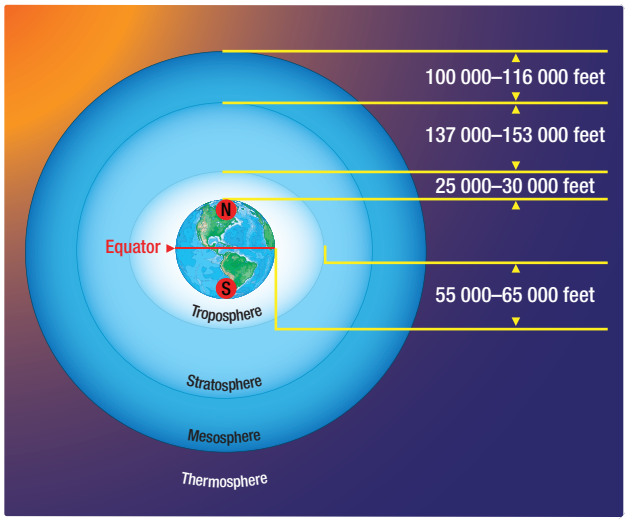


Figure 11-45. The troposphere extends higher above the earth's surface at the equator than it does at the poles.

TEMPERATURE AND ALTITUDE

Temperature variations in the atmosphere are of concern to aviators. Weather systems produce changes in temperature near the earth is surface. Temperature also changes as altitude is increased. The troposphere is the lowest layer of the atmosphere. On average, it ranges from the earth is surface to about 38 000 feet above it. Over the poles, the troposphere extends to only 25 000 - 30 000 feet and, at the equator, it may extend to around 60 000 feet. This oblong nature of the troposphere is illustrated in ***Figure 11-45***.

Most civilian aviation takes place in the troposphere in which temperature decreases as altitude increases. The rate of change is somewhat constant at about -2 °C or -3.5 °F for every 1 000 feet of increase in altitude. The upper boundary of the troposphere is the tropopause. It is characterized as a zone of relatively constant temperature of -57°C or -69°F. Above the tropopause lies the stratosphere. Temperature increases with altitude in the stratosphere to near 0 °C before decreasing again in the mesosphere, which lies above it. The stratosphere contains the ozone layer that protects the earth is inhabitants from harmful UV rays. Some civilian flights and numerous military flights occur in the stratosphere.

Figure 11-46 diagrams the temperature variations in different layers of the atmosphere. When an aircraft is flown at high altitude, it burns less fuel for a given airspeed than it does for the same speed at a lower altitude.

This is due to decreased drag that results from the reduction in air density. Bad weather and turbulence can also be avoided by flying in the relatively smooth air above storms and convective activity that occur in the lower troposphere. To take advantage of these efficiencies, aircraft are equipped with environmental systems to overcome extreme temperature and pressure levels. While supplemental oxygen and a means of staying warm suffice, aircraft pressurization and air conditioning systems have been developed to make high altitude flight more comfortable. ***Figure 11-47*** illustrates the temperatures and pressures at various altitudes in the atmosphere.

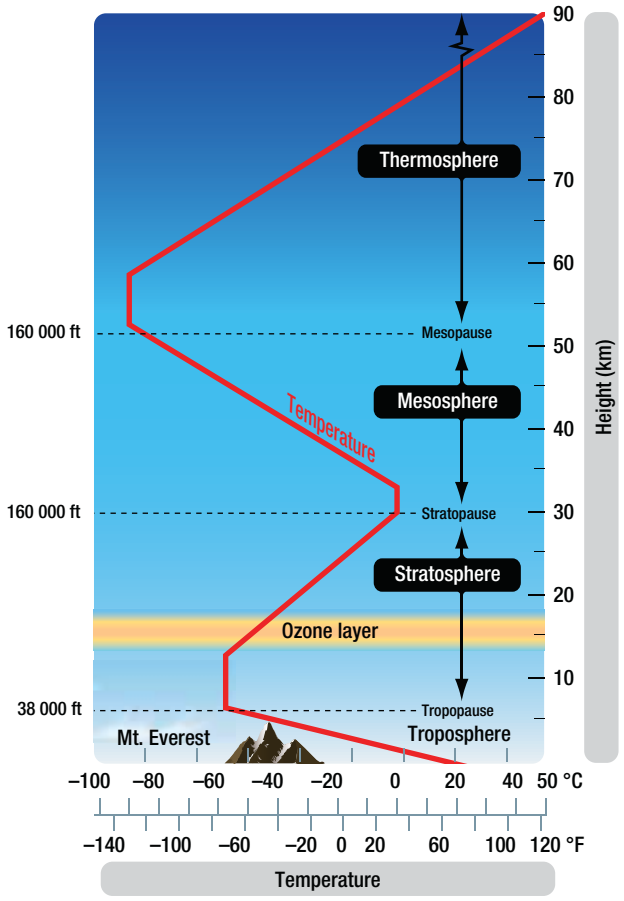


Figure 11-46. The atmospheric layers with temperature changes depicted by the red line.

Altitude	Pressure			Temperature	
feet	psi	hPa	in Hg	°F	°C
0	14.69	1013.2	29.92	59.0	15
1 000	14.18	977.2	28.86	55.4	13
2 000	13.66	942.1	27.82	51.9	11
3 000	13.17	908.1	26.82	48.3	9.1
4 000	12.69	875.1	25.84	44.7	7.1
5 000	12.23	843.1	24.90	41.2	5.1
6 000	11.77	812.0	23.98	37.6	3.1
7 000	11.34	781.8	23.09	34.0	1.1
8 000	10.92	752.6	22.23	30.5	–0.8
9 000	10.51	724.3	21.39	26.9	–2.8
10 000	10.10	696.8	20.58	23.3	–4.8
12 000	9.34	644.4	19.03	16.2	–8.8
14 000	8.63	595.2	17.58	9.1	–12.7
16 000	7.96	549.2	16.22	1.9	–16.7
18 000	7.34	506.0	14.94	–5.2	–29.7
20 000	6.76	465.6	13.75	–12.3	–24.6
22 000	6.21	427.9	12.64	–19.5	–28.6
24 000	5.70	392.7	11.60	–26.6	–32.5
26 000	5.22	359.9	10.63	–33.7	–36.5
28 000	4.78	329.3	9.72	–40.9	–40.5
30 000	4.37	300.9	8.89	–48.0	–44.4
32 000	3.99	274.5	8.11	–55.1	–48.4
34 000	3.63	250.0	7.38	–62.2	–52.4
36 000	3.30	227.3	6.71	–69.4	–56.3
38 000	3.00	206.5	6.10	–69.4	–56.5
40 000	2.73	187.5	5.54	–69.4	–56.5
45 000	2.14	147.5	4.35	–69.4	–56.5
50 000	1.70	116.0	3.42	–69.4	–56.5

Figure 11-47. Cabin environmental systems establish conditions quite different from these found outside the aircraft.

PRESSURIZATION TERMS

The following terms should be understood for the discussion of pressurization and cabin environmental systems that follows:

1. Cabin altitude - given the air pressure inside the cabin, the altitude on a standard day that has the same pressure as that in the cabin. Rather than saying the pressure inside the cabin is 10.92 psi, it can be said that the cabin altitude is 8 000 feet (MSL).
2. Cabin differential pressure - the difference between the air pressure inside the cabin and the air pressure outside the cabin. Cabin pressure (psi) - ambient pressure (psi) = cabin differential pressure (psid or ? psi).
3. Cabin rate of climb - the rate of change of air pressure inside the cabin, expressed in feet per minute (fpm) of cabin altitude change.

PRESSURIZATION SYSTEMS

Pressurizing an aircraft cabin assists in making flight possible in the hostile environment of the upper atmosphere. The degree of pressurization and the operating altitude of any aircraft are limited by critical design factors. A cabin pressurization system must accomplish several functions if it is to ensure adequate passenger comfort and safety. It must be capable of maintaining a cabin pressure altitude of approximately 8 000 feet or lower regardless of the cruising altitude of the aircraft. This is to ensure that passengers and crew have enough oxygen present at sufficient pressure to facilitate full blood saturation. A pressurization system must also be designed to prevent rapid changes of cabin pressure, which can be uncomfortable or injurious to passengers and crew. Additionally, a pressurization system should circulate air from inside the cabin to the outside at a rate that quickly eliminates odors and to remove stale air. Cabin air must also be heated or cooled on pressurized aircraft. Typically, these functions are incorporated into the pressurization source.

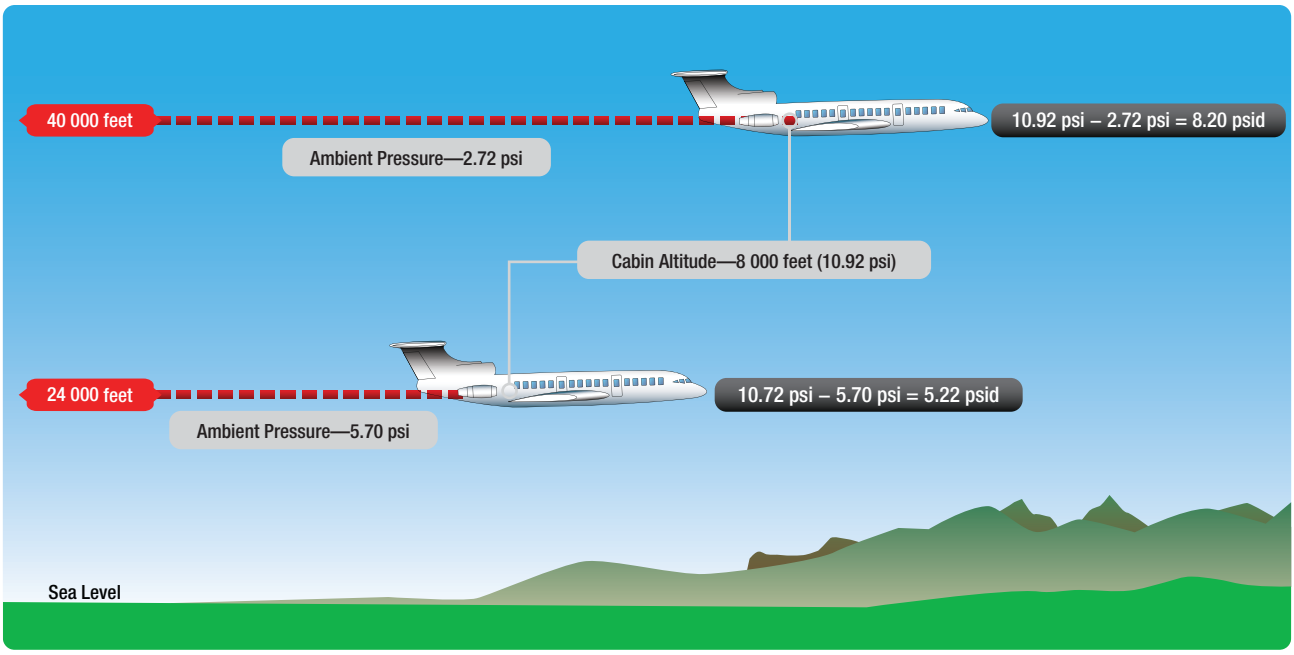


Figure 11-48. Differential pressure (psid) is calculated by subtracting the ambient air pressure from the cabin air pressure.

To pressurize, a portion of the aircraft designed to contain air at a pressure higher than outside atmospheric pressure must be sealed. A wide variety of materials facilitate this. Compressible seals around doors combine with various other seals, grommets, and sealants to essentially establish an air tight pressure vessel. This usually includes the cabin, flight compartment, and the baggage compartments. Air is then pumped into this area at a constant rate sufficient to raise the pressure slightly above that which is needed. Control is maintained by adjusting the rate at which the air is allowed to flow out of the aircraft.

A key factor in pressurization is the ability of the fuselage to withstand the forces associated with the increase in pressure inside the structure versus the ambient pressure outside. This differential pressure can range from 3.5 psi for a single engine reciprocating aircraft, to approximately 9 psi on high performance jet aircraft. (***Figure 11-48***)

If the weight of the aircraft structure were of no concern, this would not be a problem. Making an aircraft strong for pressurization, yet also light, has been an engineering challenge met over numerous years beginning in the 1930s.

The development of jet aircraft and their ability to exploit low drag flight at higher altitude made the problem even more pronounced. Today, the proliferation of composite materials in aircraft structure continues this engineering challenge. In addition to being strong enough to withstand the pressure differential between the air inside and the air outside the cabin, metal fatigue from repeated pressurization and depressurization weakens the airframe. Some early pressurized aircraft structures failed due to this and resulted in fatal accidents. The FAA's aging aircraft program was instituted to increase inspection scrutiny of older airframes that may show signs of fatigue due to the pressurization cycle. Aircraft of any size may be pressurized. Weight considerations when making the fuselage strong enough to endure pressurization usually limit pressurization to high performance light aircraft and larger aircraft. A few pressurized single-engine reciprocating aircraft exist, as well as many pressurized single-engine turboprop aircraft.

CONTROL OF CABIN PRESSURE

PRESSURIZATION MODES

Aircraft cabin pressurization can be controlled via two different modes of operation. The first is the isobaric mode, which works to maintain cabin altitude at a single pressure despite the changing altitude of the aircraft.

For example, the flight crew may select to maintain a cabin altitude of 8 000 feet (10.92 psi). In the isobaric mode, the cabin pressure is established at the 8 000 foot level and remains at this level, even as the

altitude of the aircraft fluctuates. The second mode of pressurization control is the constant differential mode, which controls cabin pressure to maintain a constant pressure difference between the air pressure inside the cabin and the ambient air pressure, regardless of aircraft altitude changes. The constant differential mode pressure differential is lower than the maximum differential pressure for which the airframe is designed, keeping the integrity of the pressure vessel intact.

When in isobaric mode, the pressurization system maintains the cabin altitude selected by the crew. This is the condition for normal operations. But when the aircraft climbs beyond a certain altitude, maintaining the selected cabin altitude may result in a differential pressure above that for which the airframe was designed. In this case, the mode of pressurization automatically switches from isobaric to constant differential mode. This occurs before the cabin's max differential pressure limit is reached. A constant differential pressure is then maintained, regardless of the selected cabin altitude. In addition to the modes of operation described above, the rate of change of the cabin pressure, also known as the cabin rate of climb or descent, is also controlled. This can be done automatically or manually by the flight crew. Typical rates of change for cabin pressure are 300 to 500 fpm. Also, note that modes of pressurization may also refer to automatic versus standby versus manual operation of the pressurization system.

CABIN PRESSURE CONTROLLERS

The cabin pressure controller is the device used to control the cabin air pressure. Older aircraft use strictly pneumatic means for controlling cabin pressure. Selections for the desired cabin altitude, rate of cabin altitude change, and barometric pressure setting are all made directly to the pressure controller from pressurization panel in the cockpit. (***Figure 11-49***)

Adjustments and settings on the pressure controller are the control input parameters for the cabin pressure regulator. The regulator controls the position of the outflow valve(s) normally located at the rear of the aircraft pressure vessel. Valve position determines the pressure level in the cabin. Modern aircraft often combine pneumatic, electric, and electronic control of pressurization. Cabin altitude, cabin rate of change, and barometric setting are made on the cabin pressure selector of the pressurization panel in the

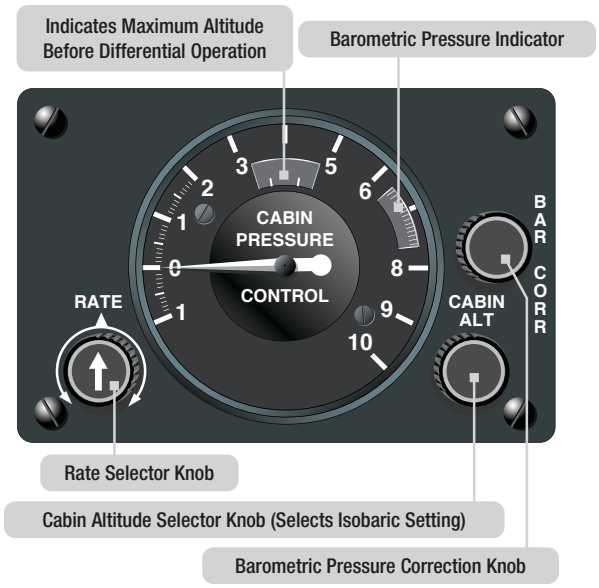


Figure 11-49. A pressure controller for an all pneumatic cabin pressure control system.

cockpit. Electric signals are sent from the selector to the cabin pressure controller, which functions as the pressure regulator. It is remotely located out of sight near the cockpit but inside the pressurized portion of the aircraft. The signals are converted from electric to digital and are used by the controller. Cabin pressure and ambient pressure are also input to the controller, as well as other inputs. (***Figure 11-50***)

Using this information, the controller, which is essentially a computer, supplies pressurization logic for various stages of a flight. On many small transport and business jets, the controller's electric output signal drives a torque motor in the primary outflow valve. This modulates pneumatic airflow through the valve, which positions the valve to maintain the pressurization schedule.

On many transport category aircraft, two cabin pressure controllers, or a single controller with redundant circuitry, are used. Located in the electronics equipment bay, they receive electric input from the panel selector, as well as ambient and cabin pressure input. Flight altitude and landing field altitude information are often the crew selection choices on the pressurization control panel. Cabin altitude, rate of climb, and barometric setting are automatic through built-in logic and communication with the ADC and the flight management system (FMS). The controllers process the information and send electric signals to motors that directly position the outflow valve(s). (***Figure 11-51***)

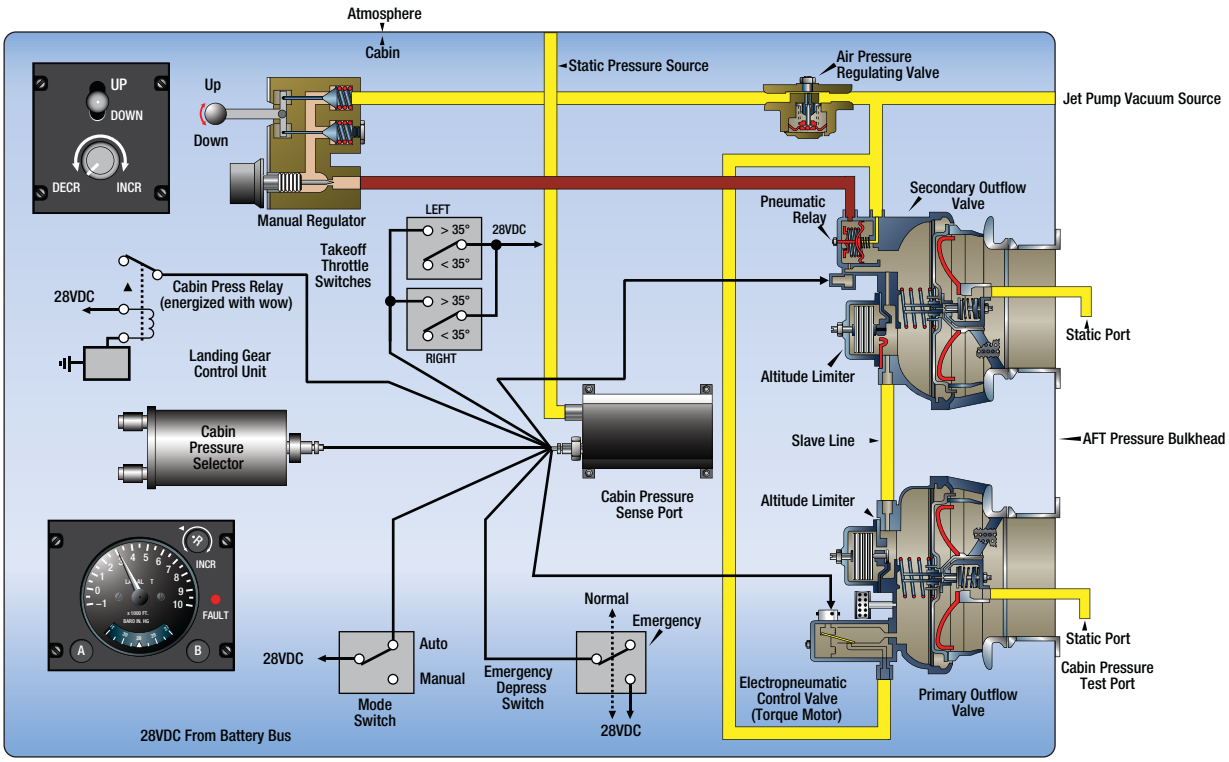


Figure 11-50. The pressurization control system on many small transports and business jets utilizes a combination of electronic, electric and pneumatic control elements.

Modern pressurization control is fully automatic once variable selections are made on the pressurization control panel if, in fact, there are any to be made. Entering or selecting a flight plan into the FMS of some aircraft automatically supplies the pressurization controller with the parameters needed to establish the pressurization schedule for the entire flight. No other input is needed from the crew.

All pressurization systems contain a manual mode that can override automatic control. This can be used in flight or on the ground during maintenance. The operator selects the manual mode on the pressurization control panel. A separate switch is used to position the outflow valve open or closed to control cabin pressure. The switch is visible in ***Figure 11-51***, as well as a small gauge that indicates the position of the valve.

Control Valve (Outflow Valve)

Controlling cabin pressurization is accomplished through regulating the amount of air that flows out of the cabin. A cabin outflow valve opens, closes, or modulates to establish the amount of air pressure maintained in the cabin. Some outflow valves contain the pressure regulating and the valve mechanism in a single unit. They operate pneumatically in response to the settings on the cockpit pressurization panel that influence the balance between cabin and ambient air pressure. (***Figure 11-52***)

Pneumatic operation of outflow valves is common. It is simple, reliable, and eliminates the need to convert air pressure operating variables into some other form. Diaphragms, springs, metered orifices, jet pumps, bellows, and poppet valves are used to sense and manipulate cabin and ambient air pressures to correctly position the outflow valve without the use of electricity. Outflow valves that combine the use of electricity with pneumatic operation have all-pneumatic standby and manual modes, as shown in ***Figure 11-50***.

The pressure regulating mechanism can also be found as a separate unit. Many air transport category aircraft have an outflow valve that operates electrically, using signals sent from a remotely located cabin air pressure controller that acts as the pressure regulator.

The controller positions the valve(s) to achieve the settings on the cockpit pressurization panel selectors according to predetermined pressurization schedules. Signals are sent to electric motors to move the valve as



Figure 11-51. This pressurization panel from an 800 series Boeing 737 has input selections of flight altitude and landing altitude.

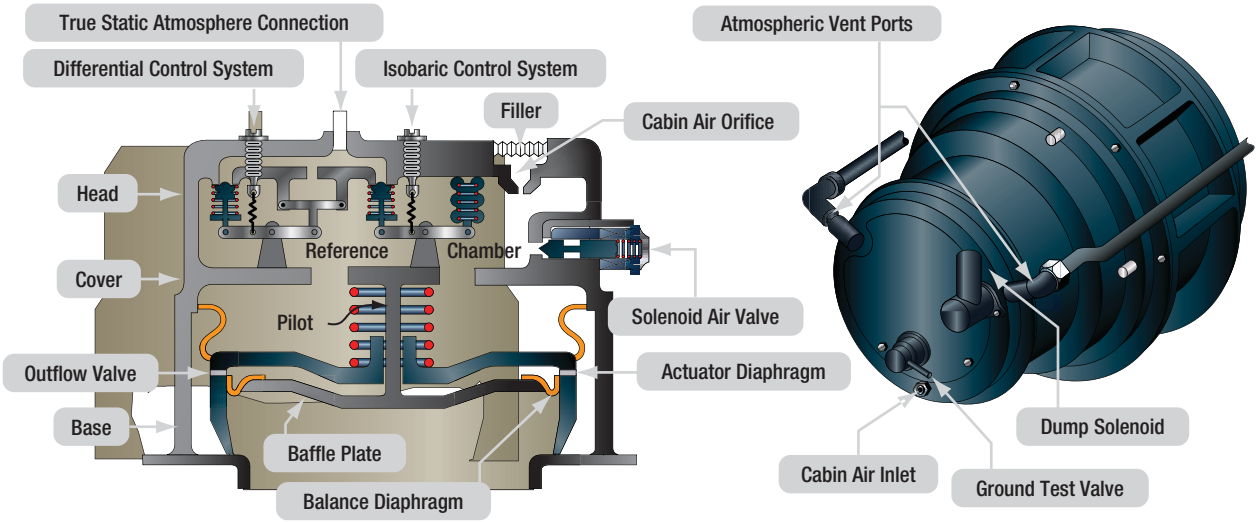


Figure 11-52. An all-pneumatic cabin pressure regulator and outflow valve.

needed. On transports, often AC motors are used with a redundant DC motor for standby or manual operations. (***Figure 11-53***)

CABIN AIR PRESSURE SAFETY VALVE OPERATION

Aircraft pressurization systems incorporate various features to limit human and structural damage should the system malfunction or become inoperative. A means for preventing over-pressurization is incorporated to ensure the structural integrity of the aircraft if control of the pressurization system is lost. A cabin air safety valve is a pressure relief valve set to open at a predetermined pressure differential. It allows air to flow from the cabin to prevent internal pressure from exceeding design limitations. ***Figure 11-54*** shows cabin air pressure safety valves on a large transport category aircraft. On most aircraft, safety valves are set to open between 8 and 10 psi.

Pressurization safety valves are used to prevent the over pressurization of the aircraft cabin. They open at a preset differential pressure and allow air to flow out of the cabin. Wide-body transport category aircraft cabins may have more than one cabin pressurization safety valve. Some outflow valves incorporate the safety valve function into their design. This is common on some corporate jets when two outflow valves are used. One



Figure 11-53. This outflow valve on a transport category aircraft is normally operated by an ac motor controlled by a pressure controller in the electronics equipment bay. A second ac motor on the valve is use when in standby mode. A DC motor also on the valve is used for manual operation.

outflow valve operates as the primary and the other as a secondary. Both contain a pilot valve that opens when the pressure differential increases to a preset value. This, in turn, opens the outflow valve(s) to prevent further pressurization. The outflow valves shown in ***Figure 11-50*** operate in this manner.

Cabin altitude limiters are also used. These close the outflow valves when the pressure in the cabin drops well below the normal cabin altitude range, preventing a further increase in cabin altitude. Some limiter functions are built into the outflow valve(s). An example of this can be seen in ***Figure 11-50***. Other limiters are independent bellows units that send input to the outflow valve or are part of the cabin pressurization controller logic. A negative pressure relief valve is included on pressurized aircraft to ensure that air pressure outside the aircraft does not exceed cabin air pressure. The spring-loaded relief valve opens inward to allow ambient air to enter the cabin when this situation arises. Too much negative pressure can cause difficulty when opening the cabin door. If high enough, it could cause structural damage since the pressure vessel is designed for cabin pressure to be greater than ambient.

Some aircraft are equipped with pressurization dump valves. These essentially are safety valves that are operated automatically or manually by a switch in the cockpit. They are used to quickly remove air and air pressure from the cabin, usually in an abnormal, maintenance, or emergency situation. Incorporation of an emergency pressurization mode is found on some aircraft. A valve opens when the air conditioning packs fail or emergency pressurization is selected from the cockpit. It directs a mixture of bleed air and ram air into the cabin. This combines with fully closed outflow valves to preserve some pressurization in the aircraft.

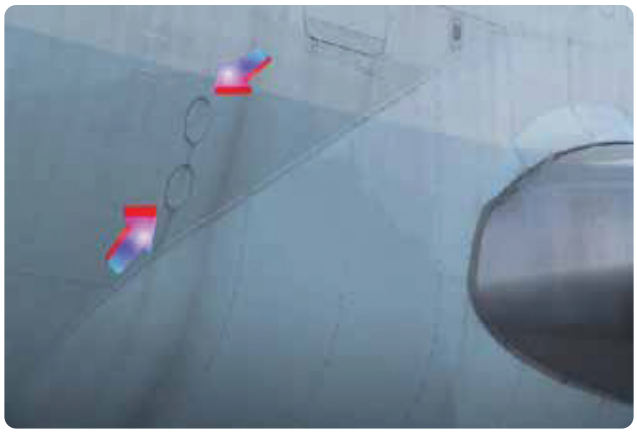


Figure 11-54. Two pressurization safety valves on a Boeing 747.

Pressurization Indication

While all pressurization systems differ slightly, usually three cockpit indications, in concert with various warning lights and alerts, advise the crew of pressurization variables. They are the cabin altimeter, the cabin rate of climb or vertical speed indicator, and the cabin differential pressure indicator. These can be separate gauges or combined into one or two gauges. All are typically located on the pressurization panel, although sometimes they are elsewhere on the instrument panel. Outflow valve position indicator(s) are also common. (***Figure 11-55***)

On modern aircraft equipped with digital aircraft monitoring systems with LCD displays, such as Engine Indicating and Crew Alerting System (EICAS) or Electronic Centralized Aircraft Monitor (ECAM), the pressurization panel may contain no gauges. The environmental control system (ECS) page of the monitoring system is selected to display similar information.

Increased use of automatic redundancy and advanced operating logic simplifies operation of the pressurization system. It is almost completely automatic. The cabin pressurization panel remains in the cockpit primarily for manual control. (***Figure 11-56***)



Figure 11-55. This cabin pressurization gauge is a triple combination gauge. The long pointer operates identically to a vertical speed indicator with the same familiar scale on the left side of the gauge. It indicates the rate of change of cabin pressure. The orange PSI pointer indicates the differential pressure on the right side scale. The ALT indicator uses the same scale as the PSI pointer, but it indicates cabin altitude when ALT indicator moves against it.

Pressurization Operation

The normal mode of operation for most pressurization control systems is the automatic mode. A standby mode can also be selected. This also provides automatic control of pressurization, usually with different inputs, a standby controller, or standby outflow valve operation. A manual mode is available should the automatic and standby modes fail. This allows the crew to directly position the outflow valve through pneumatic or electric control, depending on the system.

Coordination of all pressurization components during various flight segments is essential. A weight-on-wheels (WOW) switch attached to the landing gear and a throttle position switch are integral parts of many pressurization control systems. During ground operations and prior to takeoff, the WOW switch typically controls the position of the pressurization safety valve, which is held in the open position until the aircraft takes off.

In an advanced system, the WOW switch may give input to the pressurization controller, which in turn controls the positions and operation of all pressurization components. In other systems, the WOW switch may directly control the safety valve or a pneumatic source valve that causes the safety valve to be held open until the source is cut at takeoff when the WOW switch opens.

Throttle position switches can be used to cause a smooth transition from an unpressurized cabin to a pressurized cabin. A partial closing of the outflow valve(s) when the WOW switch is closed (on the ground) and the throttles are advanced gradually initiates pressurization during rollout. At takeoff, the rate of climb and the pressurization schedule require the outflow valve(s) to fully close. Passengers do not experience a harsh

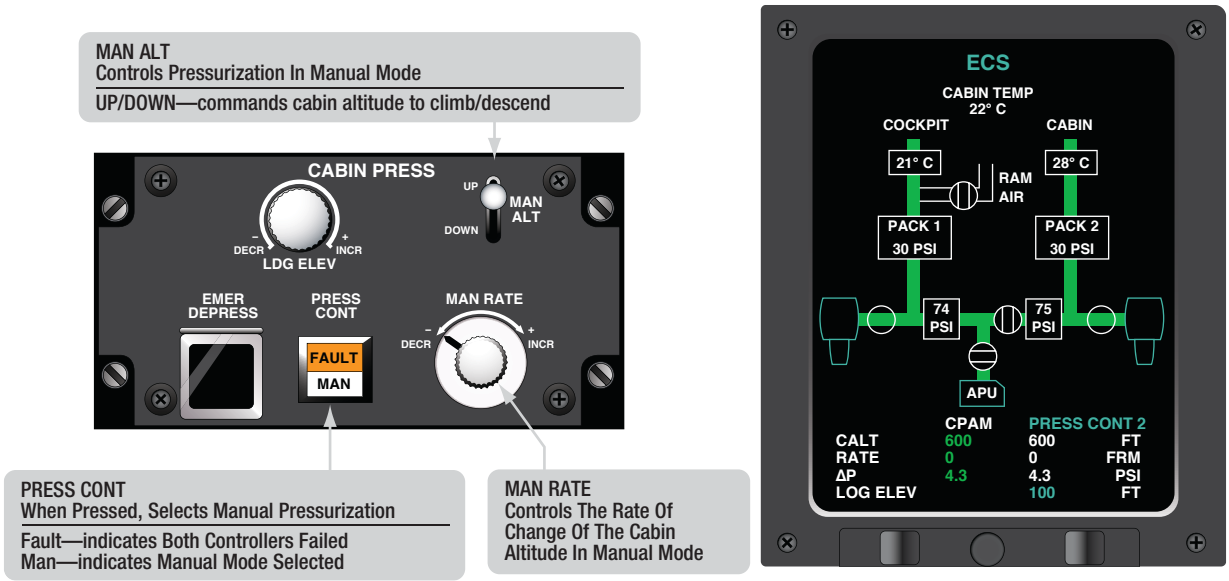


Figure 11-56. The pressurization panel and environmental control system page on a Bombardier CRJ200 50 passenger jet have no gauges. Traditional pressurization data is presented in digital format at the bottom of the page.

sensation from the fully closed valves because the cabin has already begun to pressurize slightly.

Once in flight, the pressurization controller automatically controls the sequence of operation of the pressurization components until the aircraft lands. When the WOW switch closes again at landing, it opens the safety valve(s) and, in some aircraft, the outflow valve(s) makes pressurizing impossible on the ground in the automatic pressurization mode. Maintenance testing of the system is done in manual mode. This allows the technician to control the position of all valves from the cockpit panel.

Cabin Pressurization Troubleshooting

While pressurization systems on different aircraft operate similarly with similar components, it cannot be assumed that they are the same. Even those systems constructed by a single manufacturer likely have differences when installed on different aircraft. It is important to check the aircraft manufacturer's service information when troubleshooting the pressurization system.

A fault, such as failure to pressurize or failure to maintain pressurization, can have many different causes. Adherence to the steps in a manufacturer's troubleshooting procedures is highly recommended to sequentially evaluate possible causes. Pressurization system test kits are available, or the aircraft can be pressurized by its normal sources during troubleshooting. A test flight may be required after maintenance.

SAFETY AND WARNING DEVICES

The correct functioning of cabin environmental systems is of obvious importance to passenger and crew safety. Various protection and warning devices are used to alert the crew of any malfunction that may require attention to insure safety.

PROTECTION AND WARNING DEVICES

One of the major concerns with pressurization and air conditioning systems is overheating. A leak anywhere in pneumatic system ducting can pose a fire hazard. Often a continuous loop fire detector will be run the length of pneumatic ducting or around a bay containing pneumatic lines such as the air conditioning and APU bays.

Warning annunciations occur in the cockpit on the central warning panel should an over heat caused by a pneumatic leak be detected. Typically the flow of pneumatic air in the indicated section of the pneumatic system is shut off. This is done by closing an isolation valve or shutoff valve. System redundancy permits safe operation with the remaining (functioning) portion of the pneumatic system.

Most air conditioning pack operation is protected by thermal switch monitoring. At the pack outlet before the air enters the distribution system, temperature detection switches shut down the pack should an overheat occur. The pack can typically be reset and operated again manually in case the overheat occurred due to a failure of the automatic temperature control system.

Smoke detection is sometimes used in the cabin, cargo bays, lavatories and other areas of the aircraft with controlled operating environments. Warnings are indicated on the central warning system. Cabin altitude must be maintained in the range of human survival. Failure of the pressurization system could put this in jeopardy. Most pressurized aircraft are fitted with excess cabin altitude sensors. When cabin air pressure becomes insufficient at around 10 000 feet cabin altitude, a warning is annunciated on the central warning system. The warning may be aural or visual. Flight crew may immediately utilize emergency oxygen to maintain control of the aircraft and to avoid hypoxia.

Question: 11-1
The volume of air for engine power production is _____ by bleeding air off of the compressor for air-conditioning and pressurization.

Question: 11-2
The heart of the air cycle air conditioning system is the refrigeration turbine unit, also known as the _____.

Question: 11-3
Vapor cycle air conditioning is a _____ system used solely for the transfer of heat from inside the cabin to outside of the cabin.

Question: 11-4
The low side of a vapor cycle air conditioning system has low _____ and low _____.

Question: 11-5
The thermostatic _____ has an adjustable orifice through which the correct amount of refrigerant is metered to obtain optimal cooling.

Question: 11-6
The outside air absorbs heat from the refrigerant flowing through the _____.

Question: 11-7
Servicing a vapor cycle air conditioning system is done with either a full service recovery, recycling, evacuation and recharging unit or a _____ set.

Question: 11-8
Water is removed from a vapor cycle air conditioning system by _____.

ANSWERS

Answer: 11-1
reduced.

Answer: 11-2
air cycle machine (ACM)

Answer: 11-3
closed.

Answer: 11-4
temperature.
pressure.

Answer: 11-5
expansion valve.

Answer: 11-6
condenser.

Answer: 11-7
manifold.

Answer: 11-8
evacuation (vacuuming down the system).

Question: 11-9
The temperature _____ compares the actual temperature signals received from the various sensors with the desired temperature input.

Question: 11-10
The unit of force nearly universally used today to represent atmospheric pressure in meteorology is the _____.

Question: 11-11
_____ is caused by repeated pressurization and depressurization of the fuselage pressure vessel.

Question: 11-12
Controlling cabin pressurization is accomplished through regulating the amount of air that flows out of the cabin through the _____ valve.

Question: 11-13
Three pressurization parameters monitored on the flight deck include: cabin altitude, cabin rate of climb, and _____ pressure.

Question: 11-14
A _____ is run the length of pneumatic ducting and/or around air conditioning bays to detect overheat.

ANSWERS

Answer: 11-9
controller.

Answer: 11-10
hectopascal (hPa).

Answer: 11-11
Metal fatigue.

Answer: 11-12
outflow.

Answer: 11-13
differential.

Answer: 11-14
continuous loop fire detector.



PART-66 SYLLABUS **LEVELS**

CERTIFICATION CATEGORY → **B2**

Sub-Module 12

FIRE PROTECTION (ATA 26)

Knowledge Requirements

13.12 - Fire Protection (ATA 26)

(a)	Fire and smoke detection and warning systems; Fire extinguishing systems, System tests.	3
(b)	Portable fire extinguisher.	1

INTRODUCTION

A complete fire protection system on modern aircraft, and on many older aircraft includes a fire detection system and a fire extinguishing system. Fire detection is accomplished in many different ways explained below. Fire extinguishing is accomplished with fixed and portable fire agent dispensing systems also explained in this sub-module.

REQUIREMENTS FOR FIRE TO OCCUR

Three things are required for a fire: (1) fuel - something that will, in the presence of heat combined with oxygen, thereby releasing more heat and as a result reduces itself to other chemical compounds; (2) heat - accelerates the combining of oxygen with fuel, in turn releasing more heat; and (3) oxygen - the element which combines chemically with another substance through the process of oxidation. Rapid oxidation, accompanied by a noticeable release of heat and light, is called combustion or burning. (***Figure 12-1***) Remove any one of these things and the fire extinguishes.



Figure 12-1. The fire triangle; all three elements shown are required for fire to occur.

CLASSES OF FIRES

The following classes of fires that are likely to occur onboard aircraft, as defined in the United States National Fire Protection Association (NFPA) Standard 10, Standard for Portable Fire Extinguishers, 2007 Edition, are:

1. Class A - fires involving ordinary combustible materials, such as wood, cloth, paper, rubber, and plastics.
2. Class B - fires involving flammable liquids, petroleum oils, greases, tars, oil-based paints, lacquers, solvents, alcohols, and flammable gases.
3. Class C - fires involving energized electrical equipment in which the use of an extinguishing media that is electrically nonconductive is important.
4. Class D - fires involving combustible metals, such as magnesium, titanium, zirconium, sodium, lithium, and potassium.

FIRE ZONES

Because fire is one of the most dangerous threats to an aircraft, the potential fire zones of modern multi-engine aircraft are protected by a fixed fire protection system.

A fire zone is an area, or region, of an aircraft designed by the manufacturer to require fire detection and/or fire extinguishing equipment and a high degree of inherent fire resistance. The term "fixed" describes a permanently installed extinguishing system in contrast to any type of portable fire extinguishing equipment, such as a hand-held Halon or water fire extinguisher.

Typical zones on aircraft that have a fixed fire detection and/or fire extinguisher system are:

1. Engines and auxiliary power unit (APU)
2. Cargo and baggage compartments
3. Lavatories on transport aircraft
4. Electronic bays
5. Wheel wells
6. Bleed air ducts

Note that fire zones are further classified by the airflow through them. The amount and flow characteristics through a zone greatly effect the characteristics of a fire and the methods used to detect and extinguish a fire. The zones are as follows:

1. Class A zone - area of heavy airflow past regular arrangements of similarly shaped obstructions. The power section of a reciprocating engine is usually of this type.
2. Class B zone - area of heavy airflow past aerodynamically clean obstructions. Included in this type are heat exchanger ducts, exhaust manifold shrouds, and areas where the inside of the enclosing cowling or other closure is smooth, free of pockets, and adequately drained so leaking flammables

cannot puddle. Turbine engine compartments may be considered in this class if engine surfaces are aerodynamically clean and all airframe structural formers are covered by a fireproof liner to produce an aerodynamically clean enclosure surface.

3. Class C zone - area of relatively low airflow. An engine accessory compartment separated from the power section is an example of this type of zone.
4. Class D zone - area of very little or no airflow. These include wing compartments and wheel wells where little ventilation is provided.
5. Class X zone - area of heavy airflow and of unusual construction, making uniform distribution of the extinguishing agent very difficult. Areas containing deeply recessed spaces and pockets between large structural formers are of this type. Tests indicate agent requirements to be double those for Class A zones.

FIRE PREVENTION

Leaking fuel, hydraulic, deicing, or lubricating fluids can be sources of fire in an aircraft. This condition should be noted and corrective action taken when inspecting aircraft systems. Minute pressure leaks of these fluids are particularly dangerous for they quickly produce an explosive atmospheric condition. Carefully inspect fuel tank installations for signs of external leaks. With integral fuel tanks, the external evidence may occur at some distance from where the fuel is actually escaping.

Many hydraulic fluids are flammable and should not be permitted to accumulate in the structure. Sound-proofing and lagging materials may become highly flammable if soaked with oil of any kind. Any leakage or spillage of flammable fluid in the vicinity of combustion heaters is a serious fire risk, particularly if any vapor is drawn into the heater and passes over the hot combustion chamber.

Oxygen system equipment must be kept absolutely free from traces of oil or grease, since these substances spontaneously ignite when in contact with oxygen under pressure.

Oxygen servicing cylinders should be clearly marked so they cannot be mistaken for cylinders containing air or nitrogen, as explosions have resulted from this error during maintenance operations.

FIRE DETECTION AND WARNING SYSTEMS

To detect fires or overheat conditions, detectors are placed in the various zones to be monitored. The complete aircraft fire protection systems of most large turbine engine and high performance aircraft incorporate several of these different detection methods.

1. Rate-of-temperature-rise detectors
2. Radiation sensing detectors
3. Smoke detectors
4. Overheat detectors
5. Carbon monoxide detectors
6. Combustible mixture detectors
7. Optical detectors
8. Observation of crew or passengers

The types of detectors most commonly used for fast detection of fires are the rate-of-rise, optical sensor, pneumatic loop, and electric resistance systems.

REQUIREMENTS FOR OVERHEAT AND FIRE DETECTION SYSTEMS

Fire protection systems on current-production aircraft do not rely solely on observation by crew members as a primary method of fire detection. Regardless of the type, an ideal fire detector system includes as many of the following features as possible:

1. No false warnings under any flight or ground condition.
2. Rapid indication of a fire and accurate location of the fire.
3. Accurate indication that a fire is out.
4. Indication that a fire has re-ignited.
5. Continuous indication for duration of a fire.
6. Means for electrically testing the detector system from the aircraft cockpit.
7. Resists damage from exposure to oil, water, vibration, extreme temperatures, or handling.
8. Light in weight and easily adaptable to any mounting position.
9. Circuitry that operates directly from the aircraft power system without inverters.
10. Minimum electrical current requirements when not indicating a fire.
11. Cockpit light that illuminates, indicating the location of the fire, and with an audible alarm system.
12. A separate detector system for each engine.

THERMAL SWITCH SYSTEMS

A number of detectors, or sensing devices, are available. Many older model aircraft still operating have some type of thermal switch system or thermocouple system. A thermal switch system has one or more lights energized by the aircraft power system and thermal switches that control operation of the light(s). These thermal switches are heat sensitive units that complete electrical circuits at a certain temperature. They are connected in parallel with each other but in series with the indicator lights. (***Figure 12-2***)

If the temperature rises above a set value in any one section of the circuit, the thermal switch closes, completing the light circuit to indicate a fire or overheat condition. No set number of thermal switches is required; the exact number is usually determined by the aircraft manufacturer. On some installations, all the thermal detectors are connected to one light; on others, there may be one thermal switch for each indicator light.

Some warning lights are push-to-test lights. The bulb is tested by pushing it in to check an auxiliary test circuit. The circuit shown in ***Figure 12-2*** includes a test relay. With the relay contact in the position shown, there are two possible paths for current flow from the switches to the light. This is an additional safety feature. Energizing the test relay completes a series circuit and checks all the wiring and the light bulb. Also included in the circuit shown in ***Figure 12-2*** is a dimming relay. By energizing the dimming relay, the circuit is altered to include a resistor in series with the light.

In some installations, several circuits are wired through the dimming relay, and all the warning lights may be dimmed at the same time.

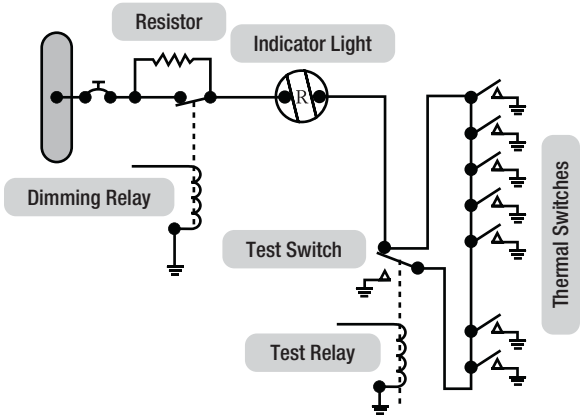


Figure 12-2. Thermal switch fire circuit.

THERMOCOUPLE SYSTEMS

The thermocouple fire warning system operates on an entirely different principle from the thermal switch system. A thermocouple depends on the rate of temperature rise and does not give a warning when an engine slowly overheats or a short circuit develops. The system consists of a relay box, warning lights, and thermocouples. The wiring system of these units may be divided into the following circuits:

1. Detector circuit
2. Alarm circuit
3. Test circuit

These circuits are shown in ***Figure 12-3***. The relay box contains two relays, the sensitive relay and the slave relay, and the thermal test unit. Such a box may contain from one to eight identical circuits, depending on the number of potential fire zones. The relays control the warning lights. In turn, the thermocouples control the operation of the relays. The circuit consists of several thermocouples in series with each other and with the sensitive relay.

The thermocouple is constructed of two dissimilar metals, such as chromel and constantan. The point at which these metals are joined and exposed to the heat of a fire is called a hot junction. There is also a reference junction enclosed in a dead air space between two insulation blocks. A metal cage surrounds the thermocouple to give mechanical protection without hindering the free movement of air to the hot junction. If the temperature rises rapidly, the thermocouple produces a voltage because of the temperature difference between the reference junction and the hot junction. If both junctions are heated at the same rate, no voltage results.

In an engine compartment, there is a normal, gradual rise in temperature from engine operation. Because it is gradual, both junctions heat at the same rate and no warning signal is given. If there is a fire, however, the hot junction heats more rapidly than the reference junction. The ensuing voltage causes a current to flow within the detector circuit. Any time the current is greater than 4 milliamperes (0.004 ampere), the sensitive relay closes. This completes a circuit from the aircraft power system to the coil of the slave relay.

The slave relay then closes and completes the circuit to the warning light to give a visual fire warning. The total

number of thermocouples used in individual detector circuits depends on the size of the fire zones and the total circuit resistance, which usually does not exceed 5 ohms. As shown in ***Figure 12-3***, the circuit has two resistors. The resistor connected across the slave relay terminals absorbs the coil's self-induced voltage to prevent arcing across the points of the sensitive relay. The contacts of the sensitive relay are so fragile that they burn or weld if arcing is permitted.

When the sensitive relay opens, the circuit to the slave relay is interrupted and the magnetic field around its coil collapses. The coil then gets a voltage through self-induction. However, with the resistor across the coil terminals, there is a path for any current flow from this voltage, eliminating arcing at the sensitive relay contacts.

CONTINUOUS-LOOP SYSTEMS

Transport aircraft almost exclusively use continuous thermal sensing elements for powerplant and wheel well protection. These systems offer superior detection performance and coverage, and they have the proven ruggedness to survive in the harsh environment of modern turbofan engines. A continuous-loop detector or sensing system permits more complete coverage of a fire hazard area than any of the spot-type temperature detectors. Two widely used types of continuous-loop systems are the thermistor type detectors such as the Kidde and the Fenwal systems and the pneumatic pressure detector such as the Lindberg system. (Lindberg system is also known as Systron-Donner and, more recently, Meggitt Safety Systems.)

FENWAL SYSTEM

The Fenwal system uses a slender Inconel tube packed with thermally sensitive eutectic salt and a nickel wire center conductor. (***Figure 12-4***) Lengths of these sensing elements are connected in series to a control unit. The elements may be of equal or varying length and of the same or different temperature settings.

The Fenwal system control unit, operating directly from the power source, applies a small voltage on the sensing elements. When an overheat condition occurs at any point along the element length, the resistance of the eutectic salt within the sensing element drops sharply, causing current to flow between the outer sheath and the center conductor. This current flow is sensed by the control unit, which produces a signal to actuate the output relay and activate the alarms. When the fire has been extinguished or the critical temperature lowered below the set point, the Fenwal system automatically returns to standby alert, ready to detect any subsequent fire or overheat condition. The Fenwal system may be wired to employ a loop circuit. In this case, should an open circuit occur, the system still signals fire or overheat. If multiple open circuits occur, only that section between breaks becomes inoperative.

KIDDE SYSTEM

In the Kidde continuous-loop system, two wires are imbedded in an Inconel tube filled with a thermistor core material. (***Figure 12-5***) The two electrical conductors go through the length of the core. One conductor has a ground connection to the tube, and the other conductor connects to the fire detection control unit. As the temperature of the core increases, electrical resistance to the ground decreases.

The fire detection control unit monitors this resistance. If the resistance decreases to the overheat set point, an overheat indication occurs in the flight deck. Typically, a 10-second time delay is incorporated for the overheat indication. If the resistance decreases more to the fire set point, a fire warning occurs.

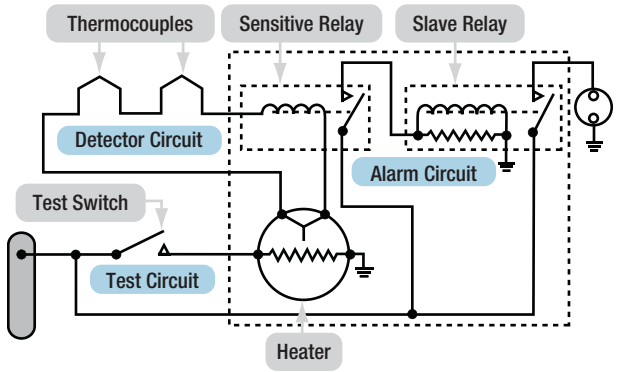


Figure 12-3. Thermocouple fire warning circuit.

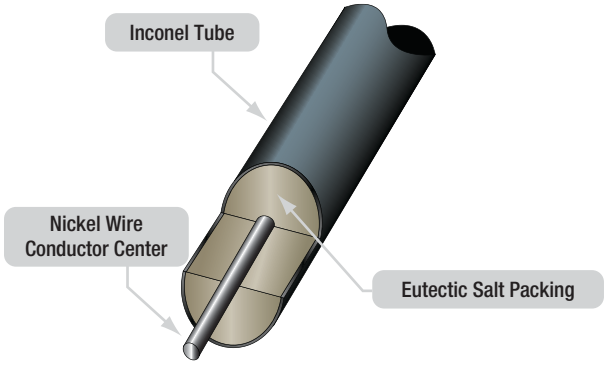


Figure 12-4. Fenwal sensing element.

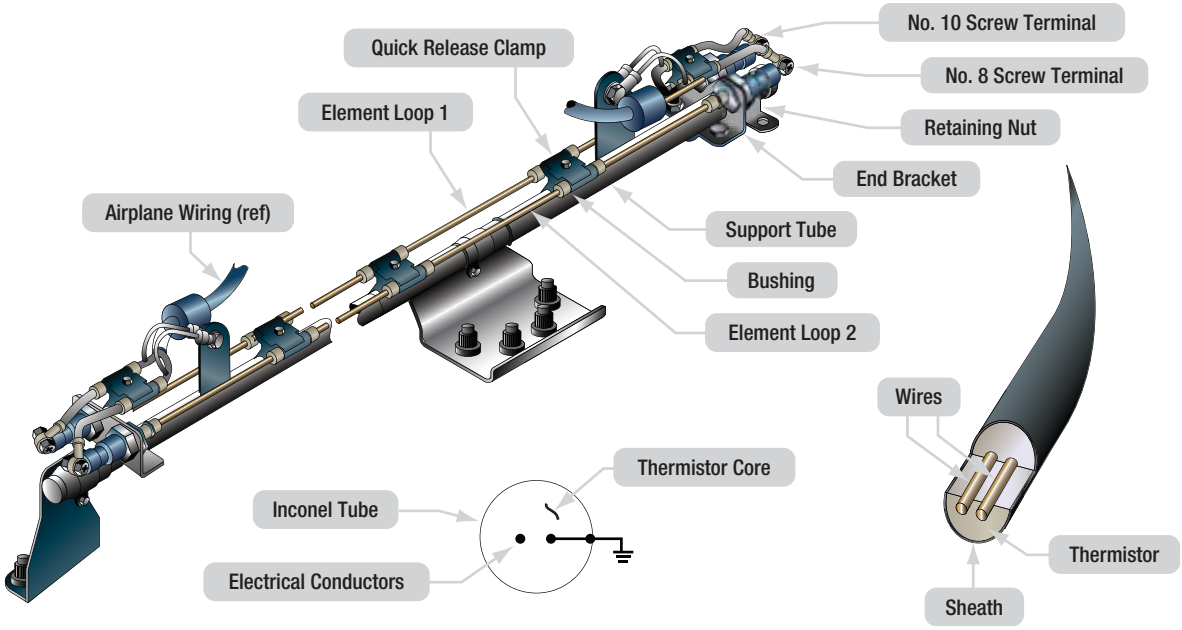


Figure 12-5. Kidde continuous-loop system.

When the fire or overheat condition is gone, the resistance of the core material in a Kidde detector system increases to the reset point and the flight deck indications disappear. The rate of change of resistance identifies an electrical short or a fire. The resistance decreases more quickly with an electrical short than with a fire. In some aircraft, in addition to fire and overheat detection, the Kidde continuous-loop system supplies nacelle temperature data to the airplane condition monitoring function of the aircraft in-flight monitoring system (AIMS).

Sensing Element

The resistance of a sensor varies inversely as it is heated. As sensor temperature is increased, its resistance decreases. Each sensor is composed of two wires embedded in thermistor material that is encased in a heavy wall Inconel tube for high strength at elevated temperatures. The electrical connectors at each end of the sensor are ceramic insulated. The Inconel tubes are shrouded in a perforated stainless steel tube and supported by Teflon-impregnated asbestos bushings at intervals. The shroud protects the sensor from breakage due to vibration, abrasion against airplane structure, and damage from maintenance activity.

The resistance of a sensor also varies inversely with its length, the increments of length being resistances in parallel. The heating of a short length of sensor out of a given length requires that the short length be heated above the temperature alarm point so the total resistance of the sensor decreases to the alarm point. This characteristic permits integration of all temperatures throughout the length of the installation rather than sensing only the highest local temperature. The two wires encased within the thermistor material of each Inconel tube form a variable resistance network between themselves, between the detector wire and the Inconel tube, and between each adjacent incremental length of sensor. These variable resistance networks are monitored by the application of 28 volts direct current (DC) to the detector wire from the detector control unit.

Combination Fire and Overheat Warning

The analog signal from the thermistor-sensing element permits the control circuits to be arranged to give a two-level response from the same sensing element loop. The first is an overheat warning at a temperature level below the fire warning indicating a general engine compartment temperature rise, such as would be caused by leakage of hot bleed air or combustion gas into the engine compartment. It could also be an early warning of fire and would alert the crew to appropriate action to reduce the engine compartment temperature. The second-level response is at a level above that attainable by a leaking hot gas and is the fire warning.

Temperature Trend Indication

The analog signal produced by the sensing element loop as its temperature changes is converted to signals

suitable for flight deck display to indicate engine bay temperature increases from normal. A comparison of the readings from each loop system also provides a check on the condition of the fire detection system, because the two loops should normally read alike.

System Test

The integrity of the continuous-loop fire detection system may be tested by actuating a test switch on the flight deck. This switches one end of the sensing element loop from its control circuit to a test circuit built into the control unit, which simulates the sensing element resistance change due to fire. (***Figure 12-6***)

If the sensing element loop is unbroken, the resistance detected by the control circuit is that of the simulated fire, and the alarm is activated. The test demonstrates, in addition to the continuity of the sensing element loop, the integrity of the alarm indicator circuit and the proper functioning of the control circuits. The thermistor properties of the sensing element remain unchanged for the life of the element (no irreversible changes take place when heated); the element functions properly as long as it is electrically connected to the control unit.

Fault Indication

Provision is made in the control unit to output a fault signal which activates a fault indicator whenever the short discriminator circuit detects a short in the sensing element loop. This is a requirement for transport category aircraft because such a short disables the fire detection system.

Dual-Loop Systems

Dual-loop systems are two complete basic fire detection systems with their output signals connected so that both must signal to result in a fire warning. This arrangement, called AND logic, results in greatly increased reliability against false fire warnings from any cause. Should one of the two loops be found inoperative at the preflight integrity test, a cockpit selector switch disconnects that loop and allows the signal from the other loop alone to activate the fire warning. Since the single operative loop meets all fire detector requirements, the aircraft can be safely dispatched and maintenance deferred to a more convenient time. However, should one of the two loops become inoperative in flight and a fire subsequently occur, the fire signaling loop activates a cockpit fault signal that alerts the flight crew to select single-loop operation to confirm the possible occurrence of fire.

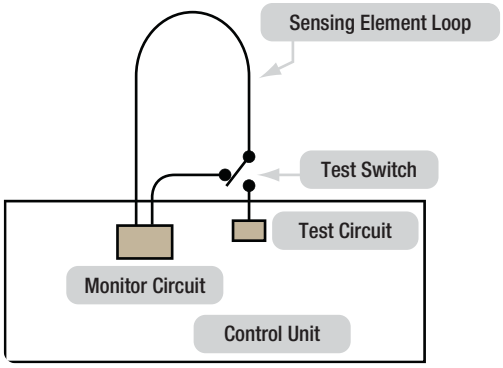


Figure 12-6. Continuously loop fire detection system test circuit.

Automatic Self-Interrogation

Dual-loop systems automatically perform the loop switching and decision-making function required of the flight crew upon appearance of the fault indication in the cockpit, a function called automatic self-interrogation. Automatic self-interrogation eliminates the fault indication and assures the immediate appearance of the fire indication should fire occur while at least one loop of the dual-loop system is operative. Should the control circuit from a single-loop signal fire, the self-interrogation circuit automatically tests the functioning of the other loop. If it tests operative, the circuit suppresses the fire signal because the operative loop would have signaled if a fire existed. If, however, the other loop tests inoperative, the circuit outputs a fire signal. The interrogation and decision takes place in milliseconds, so that no delay occurs if a fire actually exists.

Support Tube Mounted Sensing Elements

For those installations where it is desired to mount the sensing elements on the engine, and in some cases on the aircraft structure, the support tube mounted element solves the problem of providing sufficient element support points and greatly facilitates the removal and reinstallation of the sensing elements for engine or system maintenance.

Most modern installations use the support tube concept of mounting sensing elements for better maintainability as well as increased reliability. The sensing element is attached to a pre-bent stainless steel tube by closely spaced clamps and bushings where it is supported from vibration damage and protected from pinching and excessive bending. The support tube-mounted elements can be furnished with either single or dual sensing elements.

Being pre-bent to the designed configuration assures its installation in the aircraft precisely in its designed location, where it has the necessary clearance to be free from the possibility of the elements chafing against engine or aircraft structure. The assembly requires only a few attachment points and, should its removal for engine maintenance be necessary, it is quickly and easily accomplished. Should the assembly require repair or maintenance, it is easily replaced with another assembly, leaving the repair for the shop. Should a sensing element be damaged, it is easily replaced in the assembly.

Fire Detection Control Unit (Fire Detection Card)

The control unit for the simplest type of system typically contains the necessary electronic resistance monitoring and alarm output circuits housed in a hermetically sealed aluminum case fitted with a mounting bracket and electrical connector. For more sophisticated systems, control modules are employed that contain removable control cards with circuitry for individual hazard areas and/or unique functions. In the most advanced applications, the detection system circuitry controls all aircraft fire protection functions, including fire detection and extinguishing for engines, APUs, cargo bays, and bleed-air systems.

PRESSURE TYPE SENSOR RESPONDER SYSTEMS

Some smaller turboprop aircraft are outfitted with pneumatic single point detectors. The design of these detectors is based on the principles of gas laws. The sensing element consists of a closed, helium-filled tube connected at one end to a responder assembly. As the element is heated, the gas pressure inside the tube increases until the alarm threshold is reached. At this point, an internal switch closes and reports an alarm to the cockpit. Continuous fault monitoring is included. This type of sensor is designed as a single-sensor detection system and does not require a control unit.

PNEUMATIC CONTINUOUS-LOOP SYSTEMS

The pneumatic continuous-loop systems are also known by their manufacturers' names Lindberg, Systron-Donner, and Meggitt Safety Systems. These systems are used for engine fire detection of transport type aircraft and have the same function as the Kidde system; however, they work on a different principle. They are typically used in a dual-loop design to increase reliability of the system.

The pneumatic detector has two sensing functions. It responds to an overall average temperature threshold and to a localized discrete temperature increase caused by impinging flame or hot gasses. Both the average and discrete temperature are factory set and are not field adjustable. (***Figure 12-7***)

Averaging Function

The fire/overheat detector serves as a fixed-volume device filled with helium gas. The helium gas pressure inside the detector increases in proportion to the absolute temperature and operates a pressure diaphragm that closes an electrical contact, actuating the alarm circuit. The pressure diaphragm within the responder assembly serves as one side of the electrical alarm contact and is the only moving part in the detector. The alarm switch is preset at an average temperature. Typical temperature ranges for average temperature settings are 200 °F (93 °C) to 850 °F (454 °C).

Discrete Function

The fire/overheat detector's sensor tube also contains a hydrogen-filled core material. (***Figure 12-8***) Large quantities of hydrogen gas are released from the detector core whenever a small section of the tube is heated to the preset discrete temperature or higher. The core outgassing increases the pressure inside the detector and actuates the alarm switch.

Both the averaging and discrete functions are reversible. When the sensor tube is cooled, the average gas pressure is lowered and the discrete hydrogen gas returns to the core material. The reduction of internal pressure allows the alarm switch to return to its normal position, opening the electrical alarm circuit.

Figure 12-9 shows a typical aircraft fire detection system in which a control module monitors two loops of up to four pneumatic detectors each, connected in parallel. The



Figure 12-7. Pneumatic dual fire/overheat detector assembly.

control module responds directly to an alarm condition and continuously monitors the wiring and integrity of each loop.

The normally open alarm switch closes upon an overheat or fire condition, causing a short circuit between terminals A and C. During normal operation, a resistance value is maintained across the terminals by a normally closed integrity switch.

Loss of sensor gas pressure opens the integrity switch, creating an open circuit across the terminals of the faulted detector. In addition to the pressure-activated alarm switch, there is a second integrity switch in the detector that is held closed by the averaging gas pressure at all temperatures down to -65 °F (-54 °C).

If the detector should develop a leak, the loss of gas pressure would allow the integrity switch to open and signal a lack of detector integrity. The system then does not operate during test.

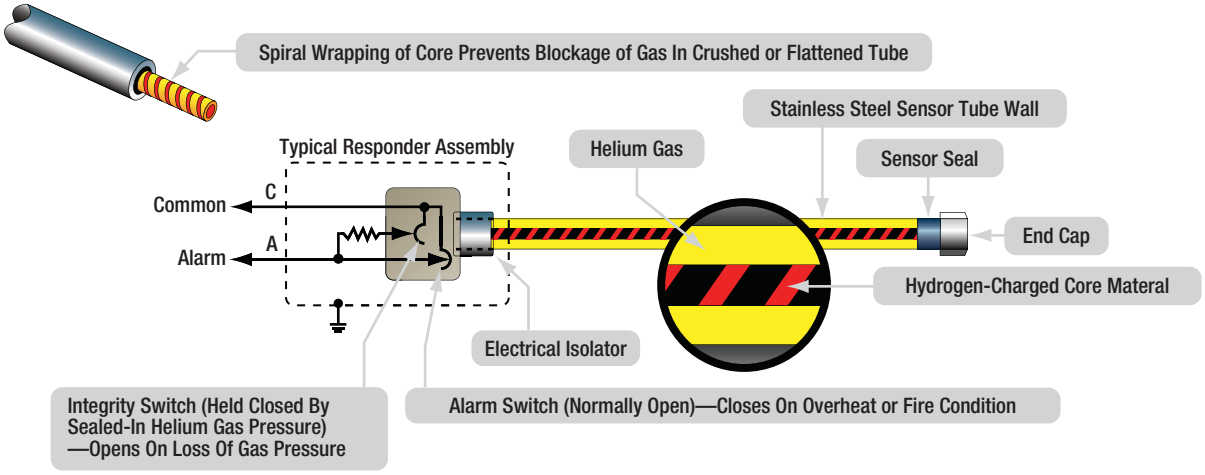


Figure 12-8. Pneumatic pressure loop detector system.

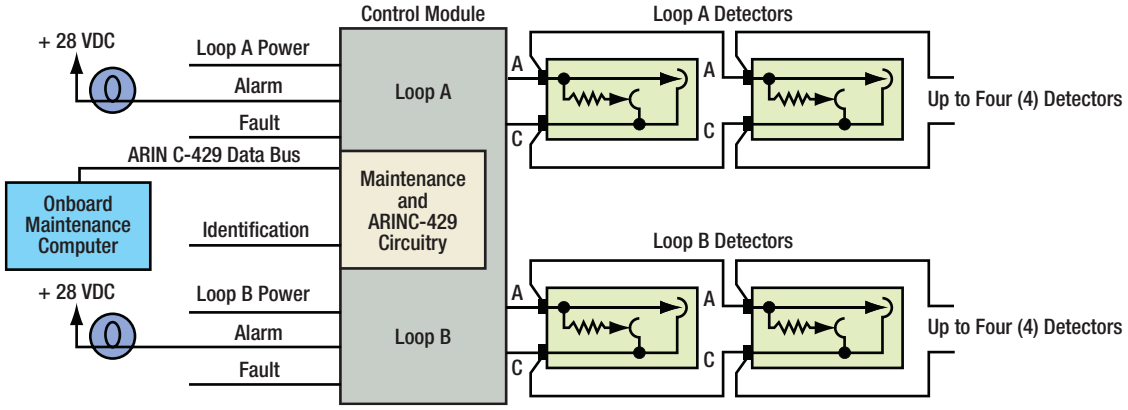


Figure 12-9. Aircraft detection system control module.

SMOKE, FLAME, AND CARBON MONOXIDE DETECTION SYSTEMS

SMOKE DETECTORS

A smoke detection system monitors the lavatories and cargo baggage compartments for the presence of smoke, which is indicative of a fire condition. Smoke detection instruments that collect air for sampling are mounted in the compartments in strategic locations. A smoke detection system is used where the type of fire anticipated is expected to generate a substantial amount of smoke before temperature changes are sufficient to actuate a heat detection system. Two common types used are light refraction and ionization.

LIGHT REFRACTION TYPE

The light refraction type of smoke detector contains a photoelectric cell that detects light refracted by smoke particles. Smoke particles refract the light to the photoelectric cell and, when it senses enough of this light, it creates an electrical current that sets off a light.

IONIZATION TYPE

Some aircraft use an ionization type smoke detector. The system generates an alarm signal (both horn and indicator) by detecting a change in ion density due to smoke in the cabin. The system is connected to the 28 volt DC electrical power supplied from the aircraft. Alarm output and sensor sensitive checks are performed simply with the test switch on the control panel.

FLAME DETECTORS

Optical sensors, referred to as flame detectors, are designed to alarm when they detect the presence of prominent specific radiation emissions from hydrocarbon flames.

The two types of optical sensors available are infrared (IR) and ultraviolet (UV), based on the specific emission wavelengths that they are designed to detect. IR-based optical flame detectors are used primarily on light turboprop aircraft and helicopter engines. These sensors have proven to be very dependable and economical for these applications.

When radiation emitted by the fire crosses the airspace between the fire and the detector, it impinges on the detector front face and window. The window allows a broad spectrum of radiation to pass into the detector where it strikes the sensing device filter. The filter allows only radiation in a tight waveband centered on 4.3 micrometers in the IR band to pass on to the radiation-sensitive surface of the sensing device. The radiation striking the sensing device minutely raises its temperature causing small thermoelectric voltages to be generated. These voltages are fed to an amplifier whose output is connected to various analytical electronic processing circuits. The processing electronics are tailored exactly to the time signature of all known hydrocarbon flame sources and ignores false alarm sources, such as incandescent lights and sunlight. Alarm sensitivity level is accurately controlled by a digital circuit. (***Figure 12-10***)

CARBON MONOXIDE DETECTORS

Carbon monoxide is a colorless, odorless gas that is a byproduct of incomplete combustion. Its presence in the breathing air of human beings can be deadly. To ensure crew and passenger safety, carbon monoxide detectors are used in aircraft cabins and cockpits. They are most often found on reciprocating engine aircraft with exhaust shroud heaters and on aircraft equipped with a combustion heater. Turbine bleed air, when used for heating the cabin, is tapped off of the engine upstream of the combustion chamber. Therefore, no threat of carbon monoxide presence is posed.

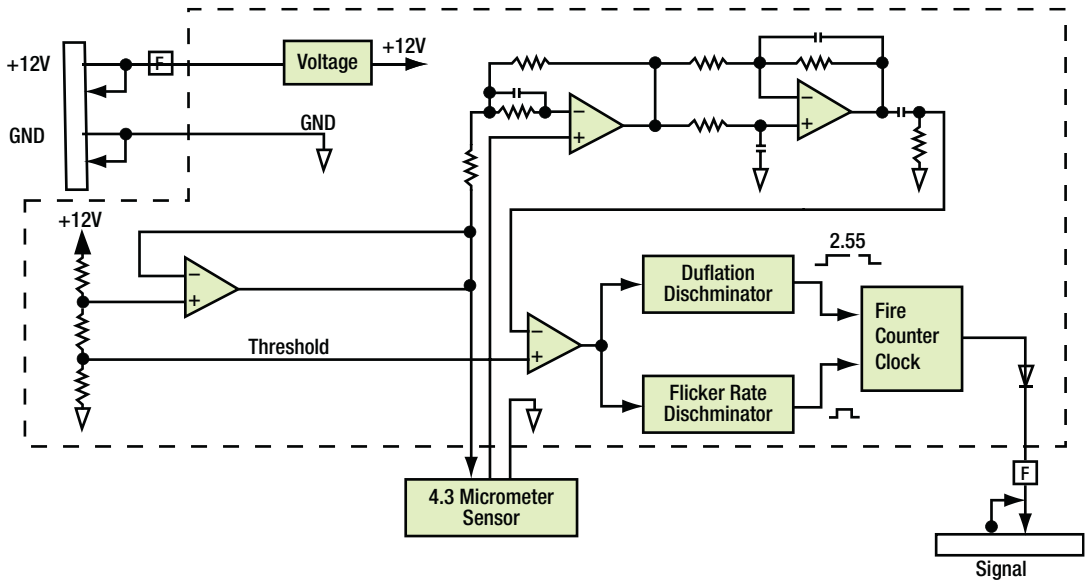


Figure 12-10. Infrared (IR) based optical flame detector.

Carbon monoxide gas is found in varying degrees in all smoke and fumes of burning carbonaceous substances. Exceedingly small amounts of the gas are dangerous if inhaled. A concentration of as little as 2 parts in 10 000 may produce headache, mental dullness, and physical lethargy within a few hours. Prolonged exposure or higher concentrations may cause death.

There are several types of carbon monoxide detectors. Electronic detectors are common. Some are panel mounted and others are portable. Chemical color-change types are also common. These are mostly portable. Some are simple buttons, cards, or badges that have a chemical applied to the surface. Normally, the color of the chemical is tan. In the presence of carbon monoxide, the chemical darkens to grey or even black. The transition time required to change color is inversely related to the concentration of CO present.

At 50 parts per million, the indication is apparent within 15 to 30 minutes. A concentration of 100 parts per million changes the color of the chemical in 2-5 minutes. As concentration increases or duration of exposure is prolonged, the color evolves from grey to dark grey to black.

FIRE EXTINGUISHING SYSTEMS

All types of aircraft store fire extinguishing agent on board to be accessed and deployed manually or automatically by the flight crew when a fire occurs. Storage of fire extinguishing agent is either in fixed containers, portable containers or both. Transport aircraft have both. Fixed containers are typically spherical in shape and are permanently installed in the aircraft. Apparatus to expel and direct the agent onto a fire is part of a fixed container system. Portable containers are also used. These are stored using quick release latches so that a user may quickly grab a container and hand carry it to the fire for deployment - usually by squeezing a trigger-type handle.

FIRE EXTINGUISHER AGENTS

Various agents are manufactured and used on aircraft. They are used in both fixed and portable systems. The following is a list of extinguishing agents and the class of fires for which each is appropriate.

1. Water - class A. Water deprives the fire of oxygen and cools the material below its ignition temperature. It soaks the burning material to prevent it from igniting again once the fire is extinguished. A water fire extinguisher should only be used on Class A fires. Transport aircraft use portable water fire extinguishers in the passenger cabin, however not on an electrical fire. Since water is conductive, spraying water on an electrical fire could cause electrocution and will certainly cause damage to the electrical equipment. Note that water fire extinguishers have antifreeze as well as water inside to ensure service should temperatures drop below freezing (32°F or 0°C).
2. Carbon dioxide (CO_2) - class B or C. Carbon dioxide acts as a blanketing agent. It smothers a fire and deprives it of oxygen. Caution must be exercised when using a CO_2 fire extinguisher in a confined area. The operator the CO_2 fire extinguisher may also be deprived of oxygen. Because of this, CO_2 is not recommended for hand-held fire extinguishers for internal aircraft use.

Carbon dioxide is an effective extinguishing agent. It is most often used in fire extinguishers that are available on the ramp to fight fires on the exterior of the aircraft, such as engine or APU fires. CO_2 has also been used for engine fire extinguishing on older transport aircraft. It can extinguish flammable fluid fires and fires involving electrical equipment although Halon is preferred for electrical fires. Carbon dioxide is noncombustible and does not react with most substances. It has a boiling point of -110 °F. As such, it provides its own (vapor) pressure for discharge from the storage vessel, except in extremely cold climates where a booster charge of nitrogen may be added to winterize the system.

Carbon dioxide is about 1½ times as heavy as air, which gives it the ability to replace air above burning surfaces and maintain a smothering atmosphere. CO_2 is effective as an extinguishing agent primarily because it dilutes the air and reduces the oxygen content so that combustion is no longer supported.

Under most conditions, some cooling effect is also realized. Carbon dioxide is considered only mildly toxic, but it can cause unconsciousness and death by

suffocation if the victim is allowed to breathe CO_2 in fire extinguishing concentrations for 20 to 30 minutes. CO_2 is not effective as an extinguishing agent on fires involving chemicals containing their own oxygen supply, such as cellulose nitrate (used in some aircraft paints). Also, fires involving magnesium and titanium cannot be extinguished by CO_2. Once used, a carbon dioxide fire extinguisher must be replaced.

3. Dry powder chemicals - class B, C or D. Dry powder extinguishers, while effective on Class B and C fires, are the best for use on Class D fires. The method of operation of dry powder fire extinguishers varies. Some containers use gas cartridge charges or store the agent under pressure within the container to force the powder charge out of the container onto the fire. Dry powder may also come in a large container or barrel from which it is applied by hand using a scoop or bucket. Examples of dry powder chemicals are sodium bicarbonate, potassium bicarbonate, and ammonium phosphate.

Dry powder is not recommended for use on aircraft fires except on metal fires. This is primarily because the leftover chemical residues and dust often make cleanup difficult, and can damage electronic or other delicate equipment. As such, dry powder is useful for Class D aircraft wheel and brake fires.

4. Halogenated hydrocarbons - class A, B, or C. Halogenated hydrocarbon (halon) fire extinguishing agents come in many chemical formulas. Halon 1211 (Bromochlorodifluoromethane, CBrClF2) and Halon 1301 (Bromotrifluoromethane, CBrF3) are commonly used in aviation depending on the application. Halon extinguishing agents smother a fire and deprive it of oxygen. They are volatile with part of their effect due to cooling of the burning materials through rapid expansion of the agent. Halon 1301 and Halon 1211 are less toxic than other halon formulas and are very effective. They are stored in pressurized containers. Halon 1301 creates satisfactory vapor pressure to expel itself. Halon 1211 has a higher boiling point and may require a nitrogen charge or a 1301 charge to pressurize adequately for effective discharge.

For over 45 years, halogenated hydrocarbons (Halons) have been practically the only fire extinguishing agents used in civil transport aircraft. However, Halon is an ozone depleting and global warming chemical, and its production has been banned by international agreement.

Although Halon usage has been banned in some parts of the world, aviation has been granted an exemption because of its unique operational and fire safety requirements. Halon replacement agents that have been found to be acceptable for environmental protection are available. To date, some of these are the halocarbons HCFC Blend B, HFC-227ea, and HFC-236fa. Halon is extremely effective on a per unit weight basis over a wide range of aircraft environmental conditions. It is electrically nonconducting, evaporate rapidly, leaves no residue, and requires no cleanup or neutralization.

NOTE: Do not use Halons on a class D fire. Halon agents may react vigorously with the burning metal.

FIXED CONTAINER FIRE EXTINGUISHING SYSTEMS

Transport aircraft have fixed fire extinguishing systems installed in:

1. Turbine engine compartments,
2. APU compartments,
3. Cargo and baggage compartments, and
4. Lavatories.

Older aircraft with reciprocating engines used CO_2 as an extinguishing agent, but all newer aircraft designs with turbine engines use Halon or equivalent extinguishing agent, such as halocarbon clean agents.

The fixed fire extinguisher systems used in most engine fire and cargo compartment fire protection systems are designed to dilute the atmosphere with an inert agent that does not support combustion. Many systems use perforated tubing or discharge nozzles to distribute the extinguishing agent.

High rate of discharge (HRD) systems use open-end tubes to deliver a large quantity of extinguishing agent in 1 to 2 seconds. The most common extinguishing agent still used today is Halon 1301

CONTAINERS

Most fixed fire extinguishing agent containers on high performance and transport aircraft are the high-rate-of-discharge (HRD) type. They typically store a liquid halogenated extinguishing agent (or other)

and a pressurized gas (typically nitrogen) to assist in the propulsion of the agent from the container. The containers are normally manufactured from stainless steel. Depending upon design considerations, alternate materials are available, including titanium.

Fixed fire agent containers are available in a wide range of capacities. They are produced under department of transportation specifications or exemptions of the manufacturer's country. Most aircraft containers are spherical in design, which provides the lightest weight possible. However, cylindrical shapes are available where space limitations are a factor. Each container incorporates a temperature/pressure sensitive safety relief diaphragm that prevents container pressure from exceeding container test pressure in the event of exposure to excessive temperatures. (***Figures 12-11 and 12-12***)



Figure 12-11. Built-in non-portable fire extinguisher containers (HRD bottles) on an airliner.

DISCHARGE VALVES

Discharge valves are installed on the containers. A cartridge (squib) and frangible disk type valve are installed in the outlet of the discharge valve assembly. Special assemblies having solenoid operated or manually operated seat type valves are also available. Two types of cartridge disk-release techniques are used. Standard release type uses a slug driven by explosive energy to rupture a segmented closure disc.

For high temperature or hermetically sealed units, a direct explosive impact type cartridge is used that applies fragmentation impact to rupture a prestressed corrosion resistant steel diaphragm. Most containers use conventional metallic gasket seals that facilitate refurbishment following discharge. (***Figure 12-13***)

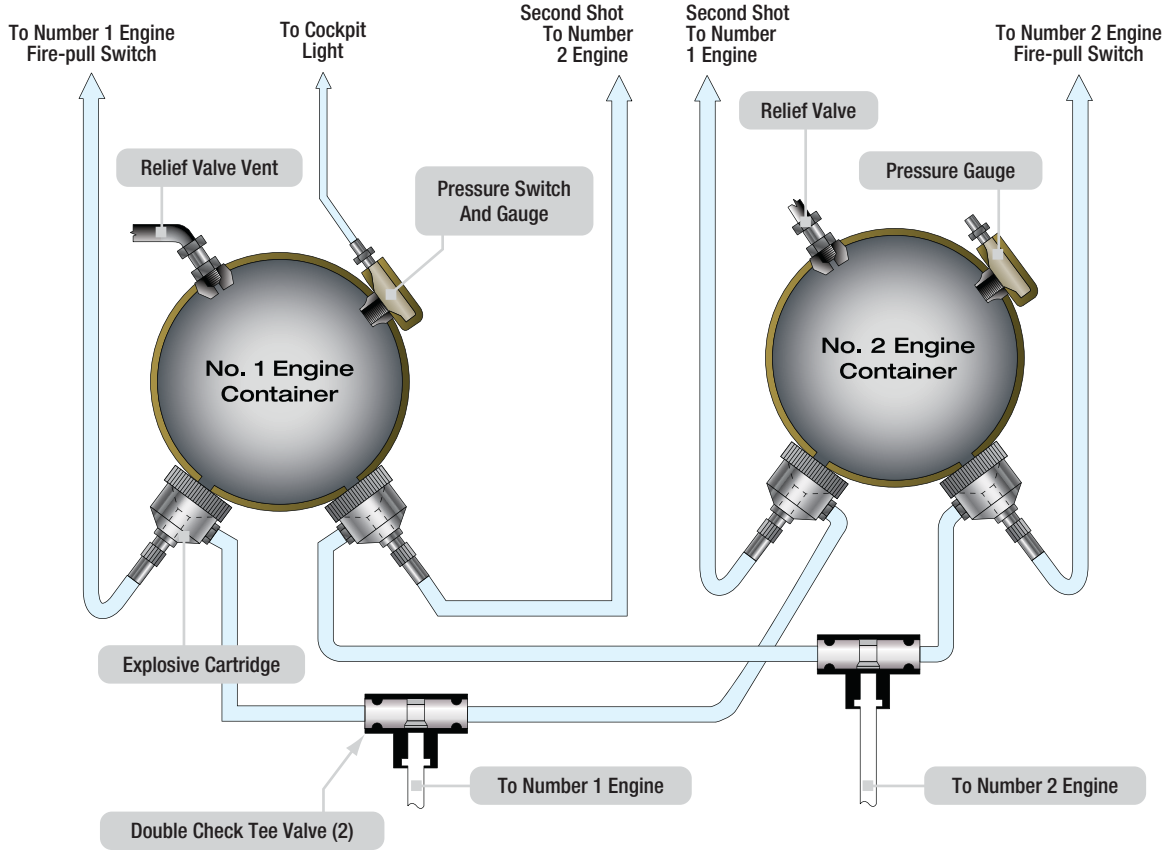


Figure 12-12. Diagram of fire extinguisher containers (HRD bottles).

PRESSURE INDICATION

A wide range of diagnostics is utilized to verify the fire extinguisher agent charge status. A simple visual indication gauge is available, typically a helical Bourdon-type indicator that is vibration resistant. (***Figure 12-12***)

A combination gauge switch visually indicates actual container pressure and also provides an electrical signal if container pressure is lost, precluding the need for discharge indicators.

A ground checkable diaphragm-type low-pressure switch is commonly used on hermetically sealed containers. The Kidde system has a temperature compensated pressure switch that tracks the container pressure variations with temperatures by using a hermetically sealed reference chamber.

TWO-WAY CHECK VALVE

Two-way check valves are required in a two-shot system to prevent the extinguisher agent from a reserve container from backing up into the previous emptied main container. Valves are supplied with either MS-33514 or MS-33656 fitting configurations.

DISCHARGE INDICATORS

Discharge indicators provide immediate visual evidence of container discharge on fire extinguishing systems. Two kinds of indicators can be furnished: thermal and discharge. Both types are designed for aircraft and skin mounting. (***Figure 12-14***)

Thermal Discharge Indicator (Red Disk)

The thermal discharge indicator is connected to the fire container relief fitting and ejects a red disk to show when container contents have dumped overboard due to excessive heat. The agent discharges through the opening left when the disk blows out. This gives the flight and maintenance crews an indication that the fire extinguisher container needs to be replaced before next flight.

Normal Discharge Indicator (Yellow Disk)

If the flight crew activates the fire extinguisher system, a yellow disk is ejected from the skin of the aircraft fuselage. This is an indication for the maintenance crew that the fire extinguishing system was activated by the flight crew, and the fire extinguishing container needs to be replaced before next flight.

FIRE SWITCH

The engine and APU fire switches are typically installed on the center overhead panel or center console on the flight deck. (***Figure 12-15***) When an engine fire switch is activated, the following happens: the engine stops because the fuel control shuts off, the engine is isolated from the aircraft systems, and the fire extinguishing system is activated. Some aircraft use fire switches that need to be pulled and turned to activate the system, while others use a push-type switch with a guard. To prevent accidental activation of the fire switch, a lock is installed that releases the fire switch only when a fire has been detected. This lock can be manually released by the flight crew if the fire detection system malfunctions. (***Figure 12-16***)



Figure 12-13. Discharge valve (left) and cartridge, or squib (right).

FIXED FIRE PROTECTION: CARGO COMPARTMENTS

Protection against fire in cargo compartments is mandatory. Transport aircraft need to have the following provisions for each cargo or baggage compartment:

1. The detection system must provide a visual indication to the flight crew within 1 minute after the start of a fire.
2. The system must be capable of detecting a fire at a temperature significantly below that at which the structural integrity of the airplane is substantially decreased.
3. There must be means to allow the crew to check, in flight, the functioning of each fire detector circuit.



Figure 12-14. Discharge indicators.

CARGO COMPARTMENT CLASSIFICATION

A Class A cargo or baggage compartment, is one in which the presence of a fire would be easily discovered

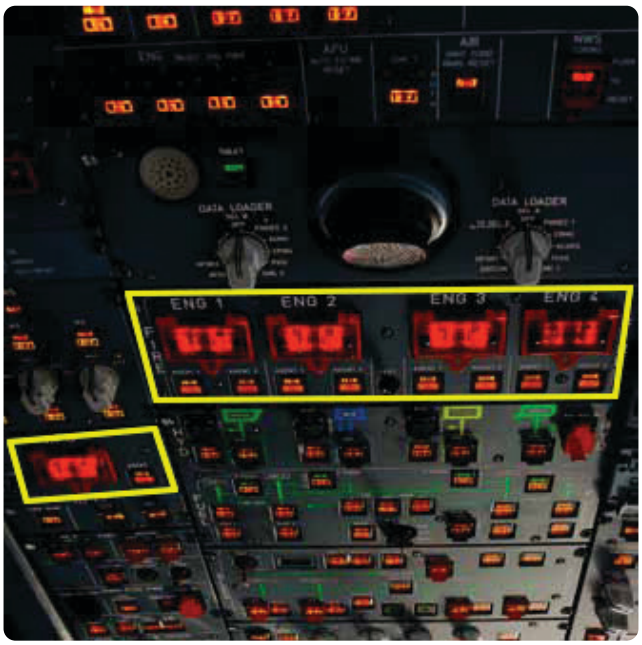


Figure 12-15. Engine and APU fire switches on the cockpit center overhead panel.

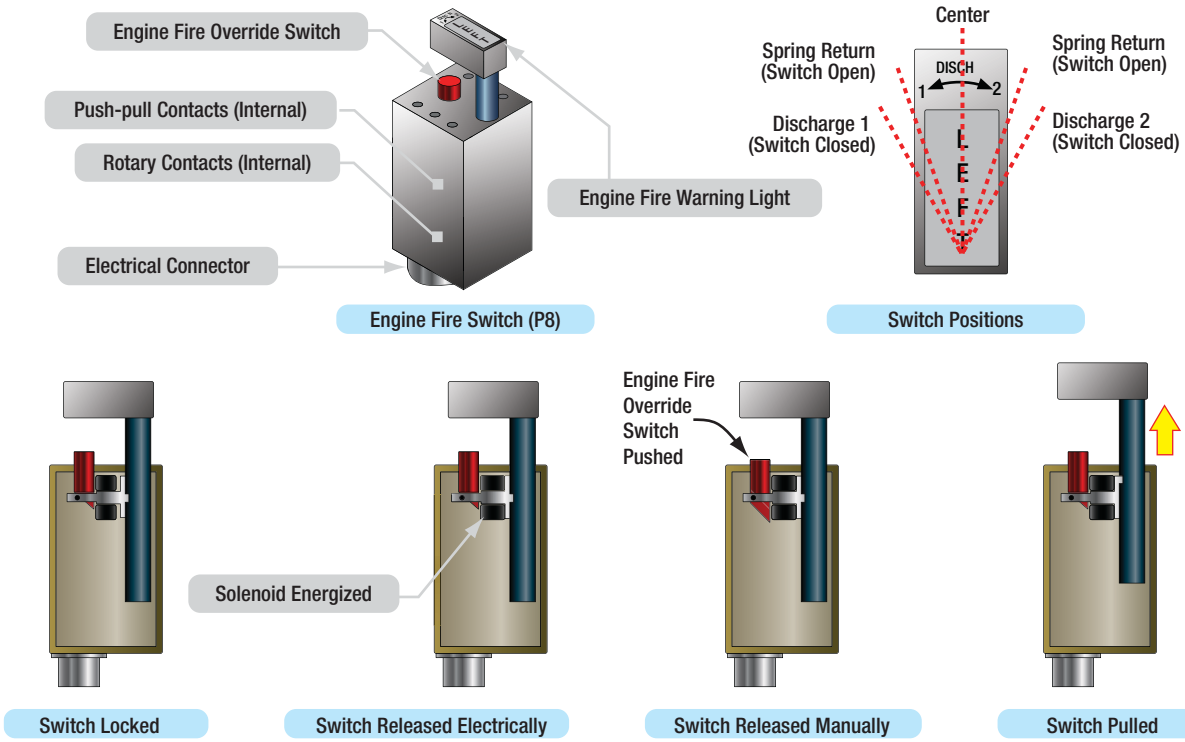


Figure 12-16. Engine fire switch operation.

by a crew member while at his or her station and each part of the compartment is easily accessible in flight.

A Class B cargo, or baggage compartment, is one in which there is sufficient access in flight to enable a crew member to effectively reach any part of the compartment with the contents of a hand fire extinguisher. When the access provisions are being used, no hazardous quantity of smoke, flames, or extinguishing agent enters any compartment occupied by the crew or passengers. There is a separate approved smoke detector or fire detector system to give warning at the pilot or flight engineer station.

A Class C cargo, or baggage compartment, is one not meeting the requirements for either a Class A or B compartment but in which:

1. There is a separate approved smoke detector or fire detector system to give warning at the pilot or flight engineer station.
2. There is an approved built-in fire extinguishing or suppression system controllable from the cockpit.
3. There are means to exclude hazardous quantities of smoke, flames, or extinguishing agent from any compartment occupied by the crew or passengers.
4. There are means to control ventilation and drafts within the compartment so that the extinguishing agent used can control any fire that may start within the compartment.

A Class E cargo compartment is one on airplanes used only for the carriage of cargo and in which:

1. There is a separate approved smoke or fire detector system to give warning at the pilot or flight engineer station.
2. The controls for shutting off the ventilating airflow to, or within, the compartment are accessible to the flight crew in the crew compartment.
3. There are means to exclude hazardous quantities of smoke, flames, or noxious gases from the flight crew compartment.
4. The required crew emergency exits are accessible under any cargo loading condition.

CARGO COMPARTMENT FIRE PROTECTION AND WARNING

Cargo compartments typically combine a smoke-type detection system with a fixed Halon extinguishing system. A smoke detector behind the sidewall in each compartment examines the air drawn through it by a circulation fan(s). If smoke is present, the system gives warnings on the flight deck. (***Figure 12-17***)

The following indications occur in the cockpit if there is smoke in a cargo compartment:

1. Master warning lights come on.
2. Fire warning aural operates.
3. A cargo fire warning message shows.
4. Cargo fire warning light comes on.

The master warning lights and fire warning aural are prevented from operating during part of the takeoff operation. (***Figure 12-18***)

Smoke Detector Systems

The optical smoke detector consists of source light emitting diodes (LEDs), intensity monitor photodiodes, and scatter detector photodiodes. Inside the smoke detection chamber, air flows between a source (LED) and a scatter detector photodiode. Usually, only a small amount of light from the LED gets to the scatter detector. If the air has smoke in it, the smoke particles reflect more light on the scatter detector. This causes an alarm signal.

The intensity monitor photodiode makes sure that the source LED is on and keeps the output of the source LED constant. This configuration also finds contamination of the LED and photodiodes. A defective diode, or contamination, causes the detector to change to the other set of diodes. The detector sends a fault message if this occurs. The smoke detector has multiple sampling ports. The fans draw air from the sampling ports through a water separator and a heater unit to the smoke detector. (***Figure 12-19***)

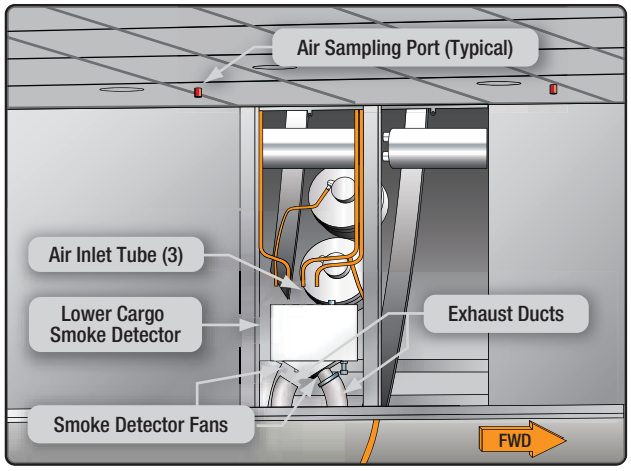


Figure 12-17. Smoke detector installation.



Figure 12-18. Cargo fire detection warning.

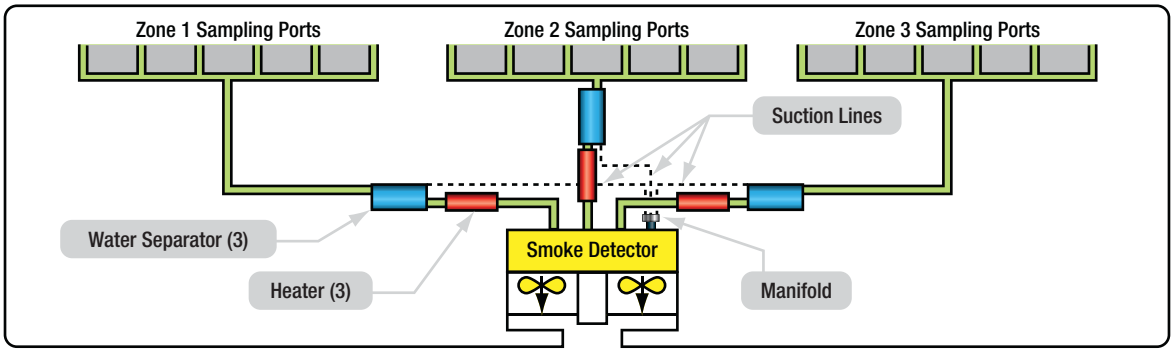


Figure 12-19. Cargo compartment smoke detector system.

Cargo Compartment Extinguishing System

The cargo compartment extinguishing system is activated by the flight crew if the smoke detectors detect smoke in the cargo compartment. Some aircraft are outfitted with two types of fire extinguisher containers. The first system is the dump system that releases the extinguishing agent directly when the cargo fire discharge switch is activated. This action extinguishes the fire.

The second system is the metered system. After a time delay, the metered bottles discharge slowly and at a controlled rate through the filter regulator. Halon from the metered bottles replaces the extinguishing agent leakage. This keeps the correct concentration of extinguishing agent in the cargo compartment to keep the fire extinguished for 180 minutes. The fire extinguishing bottles contain Halon 1301 or equivalent fire extinguishing agent pressurized with nitrogen. Tubing connects the bottles to discharge nozzles in the cargo compartment ceilings.

The extinguishing bottles are outfitted with squibs. The squib is an electrically operated explosive device. It is adjacent to a bottle diaphragm that can break. The diaphragm normally seals the pressurized bottle. When the cargo discharge switch is activated, the squib fires and the explosion breaks the diaphragm. Nitrogen pressure inside the bottle pushes the Halon through the discharge port into the cargo compartment. When the bottle discharges, a pressure switch is activated that sends an indication to the flight deck that a bottle has been discharged. Flow control valves are incorporated if the bottles can be discharged in multiple compartments. The flow control valves direct the extinguishing agent to the selected cargo compartment. (***Figure 12-20***)

FIXED FIRE PROTECTION: LAVATORIES

Airplanes that have a passenger capacity of 20 or more are equipped with a smoke detector system that monitors the lavatories for smoke. Smoke indications provide a warning light in the cockpit or provide a warning light or audible warning at the lavatory and at flight attendant stations that would be readily detected by a flight attendant. Each lavatory must have a built-in fire extinguisher that discharges automatically. The smoke detector is located in the ceiling of the lavatory. (***Figure 12-21***)

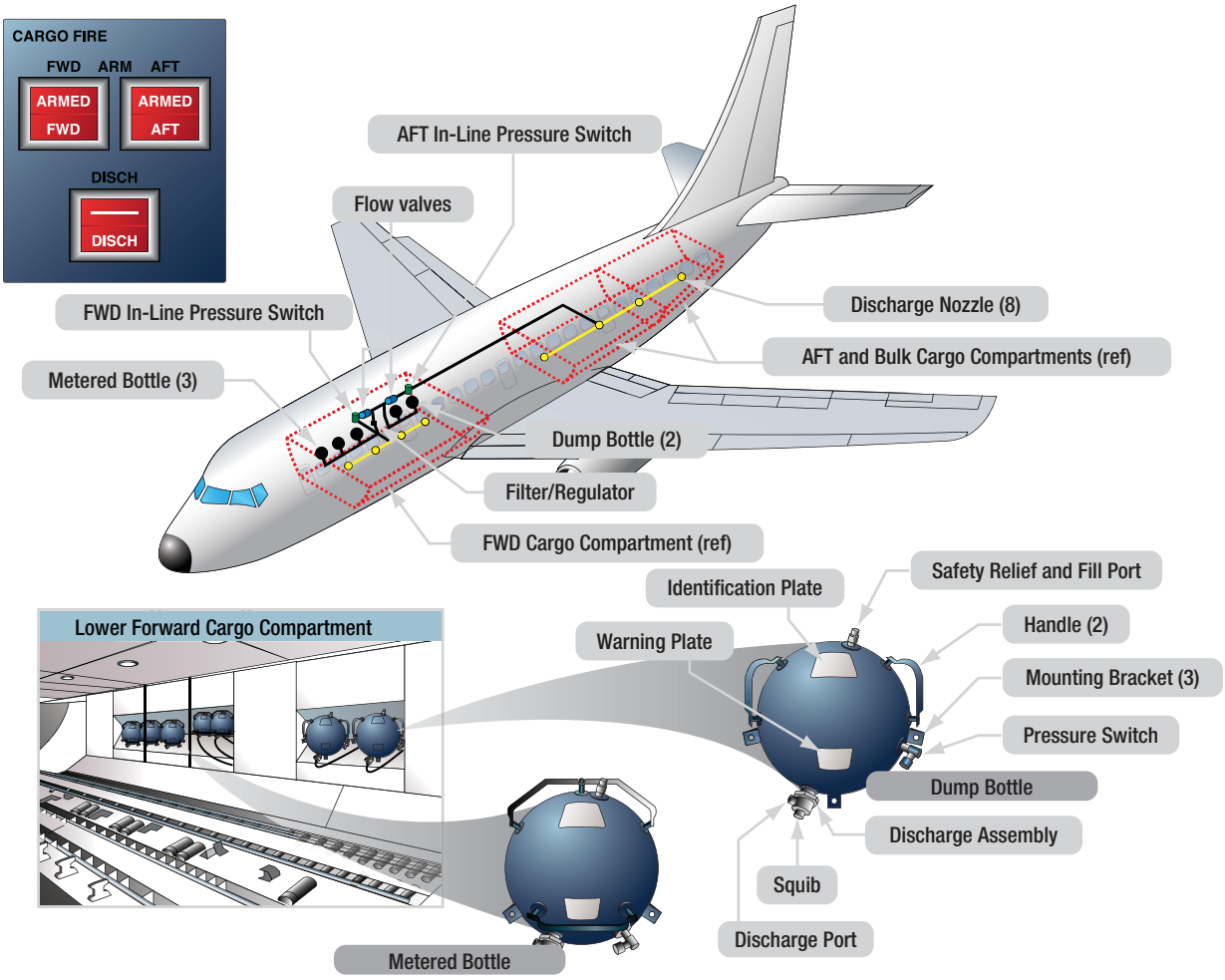


Figure 12-20. Cargo and baggage compartment extinguishing system.

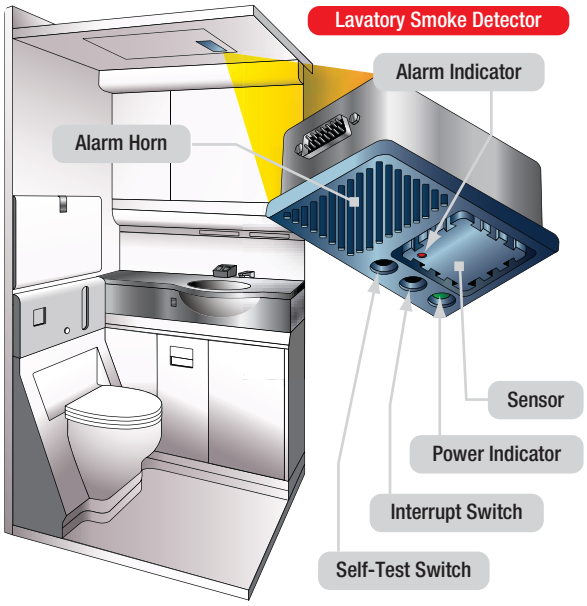


Figure 12-21. Lavatory smoke detector.

LAVATORY SMOKE DETECTOR AND WARNING SYSTEMS

Refer to ***Figure 12-22***. The lavatory smoke detector is powered by the 28-volt DC left/right main DC bus. If there is smoke in the sensing chamber of the smoke detector, the alarm LED (red) comes on. The timing circuit makes an intermittent ground. The warning horn and lavatory call light operate intermittently. The smoke detection circuit makes a ground for the relay. The energized relay makes a ground signal for the overhead electronics unit (OEU) in the central monitoring systems (CMS).

This interface gives these indications: lavatory master call light flashes, cabin system control panel (CSCP) and cabin area control panel (CACP) pop-up window shows, and the lavatory call chime operates. Push the lavatory call reset switch or the smoke detector interrupt switch to cancel the smoke indications. If there is still smoke in the lavatory, the alarm LED (red) stays on. Smoke indications go away automatically when the smoke is gone.

LAVATORY FIRE EXTINGUISHER SYSTEMS

The lavatory compartment is outfitted with a fire extinguisher bottle to extinguish fires in the waste compartment. The fire extinguisher is a bottle with two nozzles. The bottle contains pressurized Halon 1301 or equivalent fire extinguishing agent. When the temperature in the waste compartment reaches approximately 170 °F, the solder that seals the nozzles melt and the Halon is discharged. Weighing the bottle is often the only way to determine if the bottle is empty or full. (***Figure 12-23***)

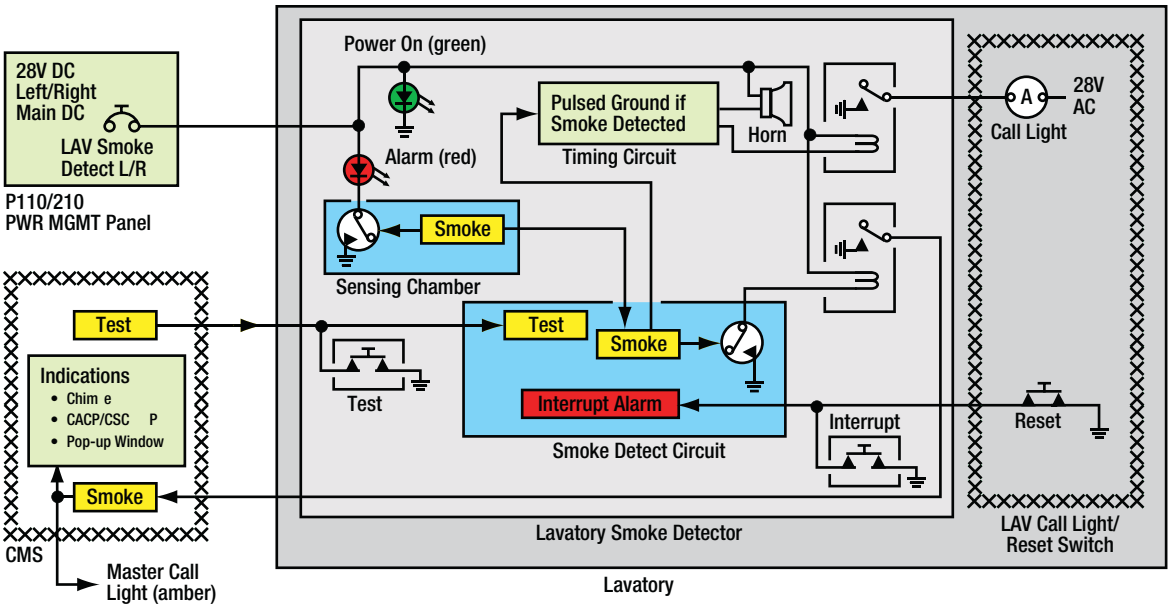


Figure 12-22. Lavatory smoke detector diagram.

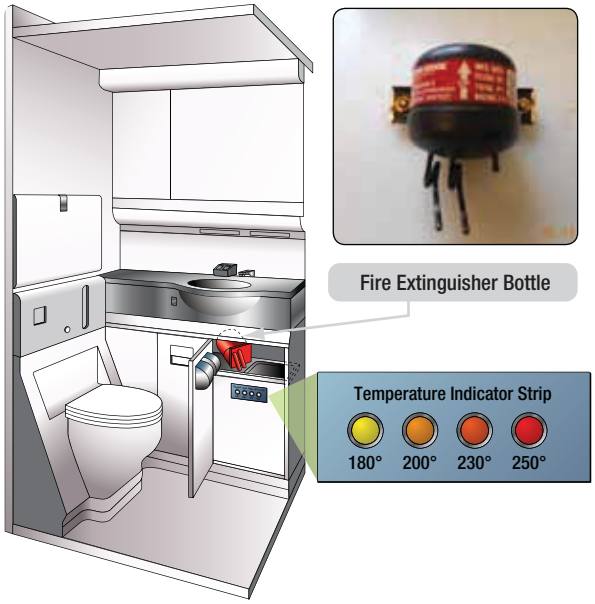


Figure 12-23. Lavatory fire extinguishing bottle.

FIRE DETECTION SYSTEM MAINTENANCE

Fire detector sensing elements are located in many high activity areas around aircraft engines. Their location, together with their small size, increases the chance of damage to the sensing elements during maintenance. An inspection and maintenance program for all types of continuous-loop systems should include the following

visual checks. NOTE: These procedures are examples and should not be used to replace the applicable manufacturer's instructions.

Sensing elements of a continuous-loop system should be inspected for the following:

1. Cracked or broken sections caused by crushing or squeezing between inspection plates, cowl panels, or engine components.
2. Abrasion caused by rubbing of the element on cowling, accessories, or structural members.
3. Pieces of safety wire, or other metal particles, that may short the spot-detector terminals.
4. Condition of rubber grommets in mounting clamps that may be softened from exposure to oils or hardened from excessive heat.
5. Dents and kinks in sensing element sections. Limits on the element diameter, acceptable dents and kinks, and degree of smoothness of tubing contour are specified by manufacturers. No attempt should be made to straighten any acceptable dent or kink, since stresses may be set up that could cause tubing failure. (***Figure 12-24***)
6. Nuts at the end of the sensing elements should be inspected for tightness and safety wire. (***Figure 12-25***) Loose nuts should be re-torqued to the value specified by the manufacturer's instructions. Some types of sensing element connection joints require the use of copper crush gaskets. These should be replaced any time a connection is separated.
7. If shielded flexible leads are used, they should be inspected for fraying of the outer braid. The braided sheath is made up of many fine metal strands woven into a protective covering surrounding the inner insulated wire. Continuous bending of the cable or rough treatment can break these fine wires, especially those near the connectors.
8. Sensing element routing and clamping should be inspected carefully. (***Figure 12-26***) Long, unsupported sections may permit excessive vibration that can cause breakage. The distance between clamps on straight runs, usually about 8 to 10 inches, is specified by each manufacturer. At end connectors, the first support clamp usually is located about 4 to 6 inches from the end connector fittings. In most cases, a straight run of one inch is maintained from all connectors before a bend is started, and an optimum bend radius of 3 inches is normally adhered to.

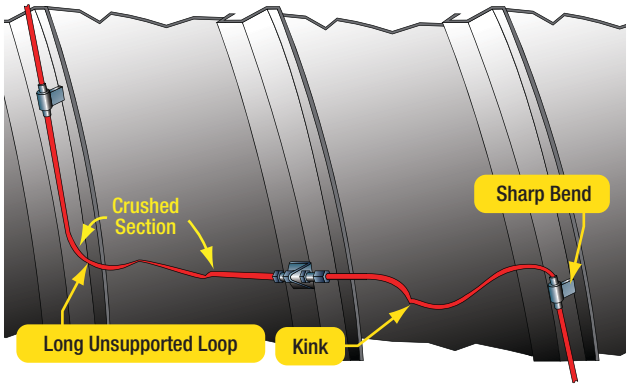


Figure 12-24. Sensing element defects.

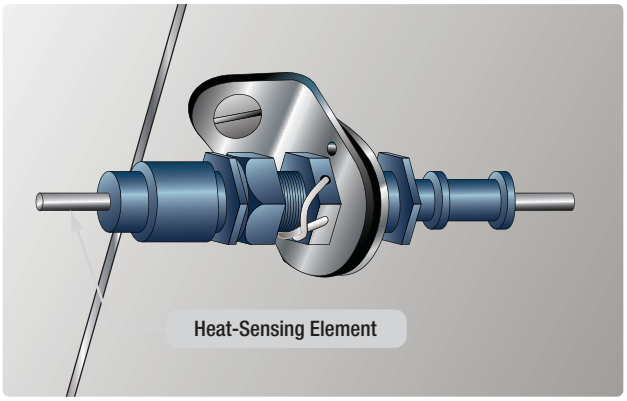


Figure 12-25. Connector joint fitting attached to the structure.

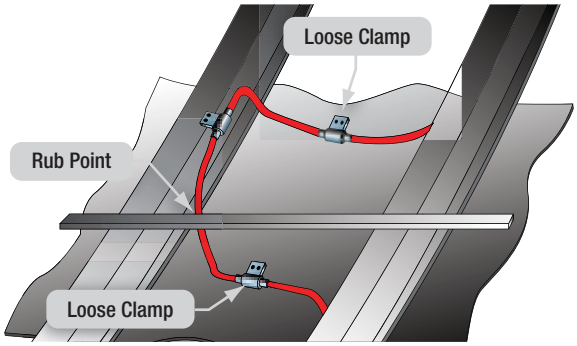


Figure 12-26. Rubbing interference.

9. Interference between a cowl brace and a sensing element can cause rubbing. This interference may cause wear and short the sensing element.
10. Grommets should be installed on the sensing element so that both ends are centered on its clamp. The split end of the grommet should face the outside of the nearest bend. Clamps and grommets should fit the element snugly. (***Figure 12-27***)

FIRE DETECTION SYSTEM TROUBLESHOOTING

The following troubleshooting procedures represent the most common difficulties encountered in engine fire detection systems:

1. Intermittent alarms are most often caused by an intermittent short in the detector system wiring. Such shorts may be caused by a loose wire that occasionally touches a nearby terminal, a frayed wire brushing against a structure, or a sensing element rubbing against a structural member long enough to wear through the insulation. Intermittent faults often can be located by moving wires to recreate the short.
2. Fire alarms and warning lights can occur when no engine fire or overheat condition exists. Such false alarms can be most easily located by disconnecting the engine sensing loop connections from the control unit. If the false alarm ceases when the engine sensing loop is disconnected, the fault is in the disconnected sensing loop, which should be examined for areas that have been bent into contact with hot parts of the engine. If no bent element can be found, the shorted section can be located by isolating the connecting elements consecutively around the entire loop.
3. Kinks and sharp bends in the sensing element can cause an internal wire to short intermittently to the outer tubing. The fault can be located by checking the sensing element with an ohm meter while tapping the element in the suspected areas to produce the short.
4. Moisture in the detection system seldom causes a false fire alarm. If, however, moisture does cause an alarm, the warning persists until the contamination is removed, or boils away, and the resistance of the loop returns to its normal value.
5. Failure to obtain an alarm signal when the test switch is actuated may be caused by a defective test switch or control unit, the lack of electrical power, inoperative indicator light, or an opening in the sensing element or connecting wiring. When the test switch fails to provide an alarm, the continuity of a two-wire sensing loop can be determined by opening the loop and measuring the resistance. In a single-wire, continuous loop system, the center conductor should be grounded.

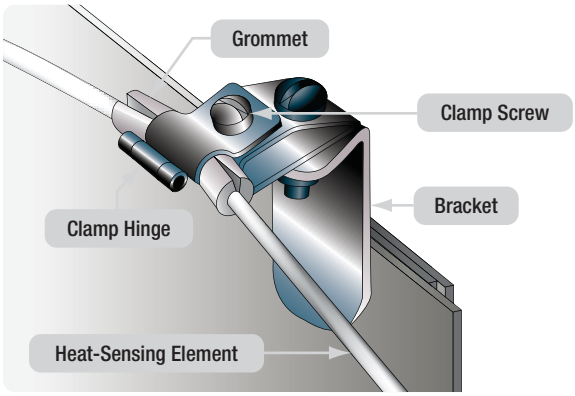


Figure 12-27. Inspection of fire detector loop clamp.

FIRE EXTINGUISHER SYSTEM MAINTENANCE

Regular maintenance of fire extinguisher systems typically includes such items as the inspection and servicing of fire extinguisher bottles (containers), removal and reinstallation of cartridge and discharge valves, testing of discharge tubing for leakage, and electrical wiring continuity tests. The following paragraphs contain details of some of the most typical maintenance procedures.

CONTAINER PRESSURE CHECK

Fire extinguisher containers are checked periodically to determine that the pressure is between the prescribed minimum and maximum limits. Changes of pressure with ambient temperatures must also fall within prescribed limits. The graph shown in ***Figure 12-28*** is typical of the pressure-temperature curve graphs that provide maximum and minimum gauge readings. If the pressure does not fall within the graph limits, the extinguisher container is replaced.

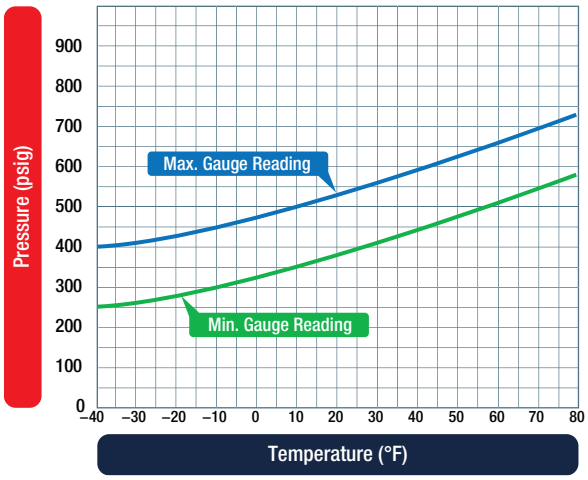


Figure 12-28. Fire extinguisher container pressure-temperature chart.

DISCHARGE CARTRIDGES

The service life of fire extinguisher discharge cartridges is calculated from the manufacturer's date stamp, which is usually placed on the face of the cartridge. The cartridge service life recommended by the manufacturer is usually in terms of years. Cartridges are available with a service life of 5 years or more. To determine the unexpired service life of a discharge cartridge, it is usually necessary to remove the electrical leads and discharge line from the plug body, which can then be removed from the extinguisher container.

AGENT CONTAINERS

Care must be taken in the replacement of cartridge and discharge valves. Most new extinguisher containers are supplied with their cartridge and discharge valve disassembled. Before installation on the aircraft, the cartridge must be assembled properly in the discharge valve and the valve connected to the container, usually by means of a swivel nut that tightens against a packing ring gasket. (***Figure 12-29***)

If a cartridge is removed from a discharge valve for any reason, it should not be used in another discharge valve assembly, since the distance the contact point

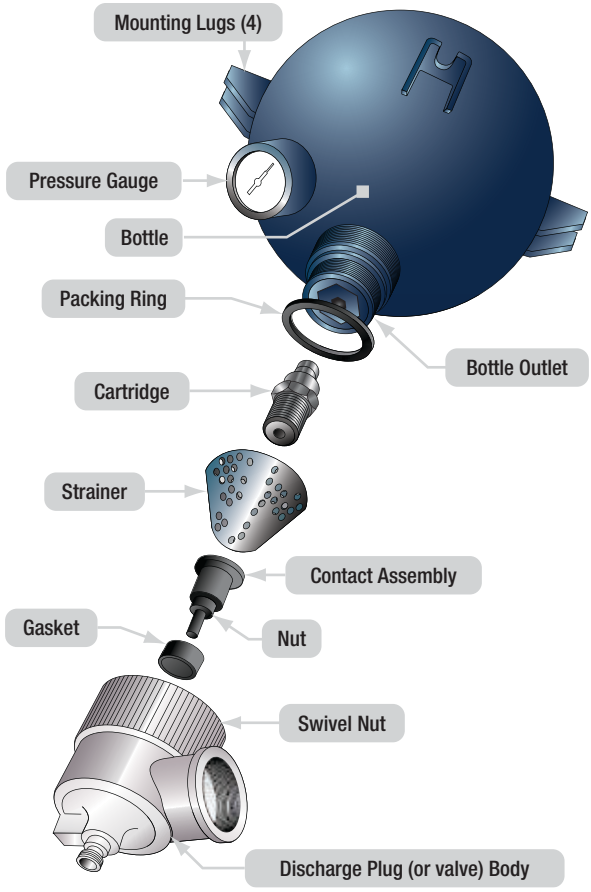


Figure 12-29. Components of fire extinguisher container.

protrudes may vary with each unit. Thus, continuity might not exist if a used plug that had been indented with a long contact point were installed in a discharge valve with a shorter contact point.

NOTE: The preceding material in this chapter has been largely of a general nature dealing with the principles involved and general procedures to be followed.

When actually performing maintenance, always refer to the applicable maintenance manuals and other related publications pertaining to a particular aircraft.

PORTABLE FIRE EXTINGUISHERS

There must be at least one hand held, portable fire extinguisher for use in the pilot compartment that is located within easy access of the pilot while seated. There must be at least one hand held fire extinguisher located conveniently in the passenger compartment of each airplane accommodating more than 6 and less than 30 passengers. Each extinguisher for use in a personnel compartment must be designed to minimize the hazard of toxic gas concentrations. The number of portable, hand held fire extinguishers for transport aircraft is shown in ***Figure 12-30***.

Passenger Capacity	No. of Extinguishers
7 through 30	1
31 through 60	2
61 through 200	3
201 through 300	4
301 through 400	5
401 through 500	6
501 through 600	7
601 through 700	8

Figure 12-30. Hand held fire extinguisher requirement for transport aircraft.

PORTABLE EXTINGUISHER TYPES

All materials used in the cockpit and cabin must conform to strict standards to prevent fire. In case of a fire, several types of portable fire extinguishers are available to fight the fire. The most common types are Halon 1211 and water.

Portable fire extinguishers are used to extinguish fires in the passenger cabin and on the flight deck. ***Figure 12-31*** shows a Halon fire extinguisher used in a general aviation aircraft. The Halon extinguishers are used on electrical and flammable liquid fires. Some transport aircraft also use water fire extinguisher for use on non-electrical fires.

The following portable, hand-held fire extinguishers are unsuitable as cabin or cockpit equipment:

- CO_2
- Dry chemicals (due to the potential for corrosion damage to electronic equipment, the possibility of visual obscuration if the agent were discharged into the flight deck area, and the cleanup problems after their use)
- Specialized dry powder, however, it is suitable for use in ground operations.

Instructions for use of portable firer extinguishers are located on the container. Usually, this involves removal of a safetying device (pin or break-away wire), direction of the extinguisher at the fire and squeezing a trigger handle. Other information on the label of a portable fire extinguisher includes the container approval number, weight and date of last service.



Figure 12-31. Portable fire extinguisher.

QUESTIONS

Question: 12-1
A complete fire protection system on modern aircraft, and on many older aircraft, includes a fire _____ system and a fire _____ system.

Question: 12-2
Name 8 means of fire detection used on large turbine-engine and high performance aircraft.

Question: 12-3
A _____ depends on the rate of temperature rise and does not give a warning when an engine slowly overheats or a short circuit develops.

Question: 12-4
A _____ permits more complete coverage of a fire hazard area than any of the spot-type temperature detectors.

Question: 12-5
In a Kidde continuous loop fire detection system, as the temperature of the detector loop core increases, electrical resistance to the ground _____.

Question: 12-6
The test circuit for a continuous loop fire detection element is located in the _____.

Question: 12-7
Most modern engine installations use a _____ to mount fire detection sensing elements for better maintainability, as well as increased reliability.

Question: 12-8
A pneumatic fire detector system has two sensing functions. Name them.

ANSWERS

Answer: 12-1
detection, extinguishing.

Answer: 12-2
1. Rate-of-temperature-rise detectors.
2. Radiation sensing detectors.
3. Smoke detectors.
4. Overheat detectors.
5. Carbon monoxide detectors.
6. Combustible mixture detectors.
7. Optical detectors.
8. Observation of crew or passengers.

Answer: 12-3
thermocouple.

Answer: 12-4
continuous-loop detector or sensing system.

Answer: 12-5
decreases.

Answer: 12-6
control unit.

Answer: 12-7
support tube.

Answer: 12-8
Overall average temperature threshold.
Localized discrete temperature increase.

Question: 12-9
Storage of fire extinguishing agent is either in fixed containers or _____ containers or both.

Question: 12-10
The best fire extinguishing agent for a Class D fire is _____.

Question: 12-11
The most commonly used Halon fire extinguishing agents are Halon _____ and Halon _____.

Question: 12-12
Most fixed fire extinguishing agent containers on high performance and transport aircraft are the _____ type.

Question: 12-13
Standard release-type discharge valves use a slug driven by _____ to rupture a segmented closure disc.

Question: 12-14
A normal discharge of fire extinguishing agent will cause a _____ disk to be ejected from the skin of the aircraft fuselage.

Question: 12-15
Cargo compartments typically combine a _____ detection system with a fixed Halon extinguishing system.

Question: 12-16
In addition to a dump extinguisher, some cargo compartments have a _____ extinguishing system that discharge agent art a controlled rate.

ANSWERS

Answer: 12-9
portable.

Answer: 12-10
dry powder chemical.

Answer: 12-11
1301.
1211.

Answer: 12-12
high-rate-of-discharge (HRD).

Answer: 12-13
explosive energy.

Answer: 12-14
yellow.

Answer: 12-15
smoke-type.

Answer: 12-16
metered.



PART-66 SYLLABUS **LEVELS**

CERTIFICATION CATEGORY → **B2**

Sub-Module 13

FUEL SYSTEMS (ATA 28)

Knowledge Requirements

13.13 - Fuel Systems (ATA 28)	
System lay-out;	1
Fuel tanks;	1
Supply systems;	1
Dumping, venting and draining;	1
Cross-feed and transfer;	2
Indications and warnings;	3
Refuelling and defuelling;	2
Longitudinal balance fuel systems.	3

BASIC FUEL SYSTEMS REQUIREMENTS

All powered aircraft require fuel on board to operate the engine(s). A fuel system consisting of storage tanks, pumps, filters, valves, fuel lines, metering devices, and monitoring devices is designed and certified under strict guidelines of the civil authority of the manufacturing company. Each system must provide an uninterrupted flow of contaminant-free fuel regardless of the aircraft's attitude. Since fuel load can be a significant portion of the aircraft's weight, a sufficiently strong airframe must be designed. Varying fuel loads and shifts in weight during maneuvers must not negatively affect control of the aircraft in flight. Although the technician is rarely involved with designing fuel systems, a review of fuel system design criteria gives insight into how an aircraft fuel system operates.

Each fuel system must be constructed and arranged to ensure fuel flow at a rate and pressure established for proper engine and auxiliary power unit (APU) functioning under each likely operating condition. This includes any maneuver for which certification is requested during which the engine or APU may be in operation. (***Figure 13-1***) Each fuel system must be arranged so that no fuel pump can draw fuel from more than one tank at a time. There must also be a means to prevent the introduction of air into the system.

FUEL SYSTEM INDEPENDENCE

Each fuel system for a multi-engine airplane must be arranged so that, in at least one system configuration, the failure of any one component (other than a fuel tank) does not result in the loss of power of more than one engine or require immediate action by the pilot to prevent the loss of power of more than one engine. If a single fuel tank (or series of fuel tanks inter connected to function as a single fuel tank) is used on a multi-engine airplane, independent tank outlets for each engine, each incorporating a shut-off valve at the tank, must be provided. The shutoff valves may serve as firewall shutoff valves, which are also required. Lines and any components from each tank outlet to each engine must be completely independent of each other.

FUEL SYSTEM LIGHTNING PROTECTION

The fuel system must be designed and arranged to prevent the ignition of fuel vapor within the system by direct lightning strikes or swept lightning strokes (where highly probable). Swept strokes occur when the lightning strike is deformed by interaction with aerodynamic forces and propagates in a unique manner due to the material and shape of the airframe surfaces. Corona and streamering must also be inhibited at fuel vent outlets since they may ignite the fuel-air mixture. A corona is a luminous discharge that occurs as a result of an electrical potential difference between the aircraft and the surrounding area.

Streamering is a branch-like ionized path that occurs in the presence of a direct stroke or under conditions when lightning strokes are imminent. (***Figure 13-2***)

FUEL FLOW

The ability of the fuel system to provide fuel at a rate of flow and pressure sufficient for proper engine operation is vital in aircraft. Moreover, the fuel system must deliver the fuel at the aircraft attitude that is most critical with respect to fuel feed and quantity of unusable fuel. Tests are performed to demonstrate this performance. Fuel flowmeters are installed on most aircraft. During testing, the flowmeter is blocked and fuel must flow through or bypass the meter and still supply the engine at sufficient rate and pressure.

FLOW BETWEEN INTERCONNECTED TANKS

If fuel can be pumped from one tank to another in flight, the fuel tank vents and the fuel transfer system must be designed so that no structural damage to any airplane component can occur because of overfilling of any tank.

UNUSABLE FUEL SUPPLY

The unusable fuel supply for each tank must be established. It cannot be less than that quantity at which



Figure 13-1. Aircraft fuel systems must deliver fuel during any maneuver for which the aircraft is certified.

the first evidence of malfunctioning appears under the most adverse fuel feed condition occurring under each intended operation and flight maneuver involving that tank. The effect on the usable fuel quantity as a result of a failure of any pump is also determined.

FUEL SYSTEM HOT WEATHER OPERATION

Each fuel system must be free from vapor lock when using fuel at its critical temperature, with respect to vapor formation, when operating the airplane in all critical operating and environmental conditions for which approval is requested.

FUEL TANKS

Each fuel tank must be able to withstand, without failure, the vibration, inertia, fluid, and structural loads to which it may be subjected in operation. Aircraft fuel tanks must be able to withstand the forces that are encountered throughout the entire spectrum of operation. Various tank testing standards exist. A main focus is to ensure that tanks are strong enough to remain fully operational and not deform when under various loads. The total usable capacity of any tank(s) must be enough for at least 30 minutes of operation at maximum continuous power. Each integral fuel tank must have adequate facilities for interior inspection and repair. Additionally, each fuel quantity indicator must be adjusted to account for the unusable fuel supply. Each fuel tank must have an expansion space of not less than two percent of the tank capacity. To allow proper fuel flow, each fuel tank must be vented from the top part of the expansion space.

Keeping contaminants out of the fuel delivered to the engine begins with the proper construction and installation of the fuel tank(s). Each tank must have a



Figure 13-2. Lightning streamering at the wingtips of a jet fighter.

drainable sump. Each fuel tank must allow drainage of any hazardous quantity of waterfrom any part of the tank to its sump.

Each fuel tank filler connection must be specifically marked. Different filler openings are used for aircraft with engines that use gasoline fuel and turbine fuel. Fuel filling points must have a provision for electrically bonding the airplane to ground fueling equipment (except pressure fueling connection points).

PRESSURE FUELING SYSTEMS

Pressure fueling systems are used on many large, high performance, and air carrier aircraft. Each pressure fueling system fuel manifold connection must have means to prevent the escape of hazardous quantities of fuel from the system if the fuel entry valve fails. A means for automatic shutoff must be provided to prevent the quantity of fuel in each tank from exceeding the maximum quantity approved for that tank.

FUEL PUMPS

Fuel pumps are part of most aircraft fuel systems. Standards exist for main pumps and emergency pumps. Operation of any fuel pump may not affect engine operation by creating a hazard, regardless of the engine power or thrust setting or the functional status of any other fuel pump.

Emergency pumps are used and must be immediately available to supply fuel to the engine if any main pump fails. The power supply for each emergency pump must be independent of the power supply for each corresponding main pump. If both the main fuel pump and the emergency pump operate continuously, there must be a means to indicate a malfunction of either pump to the appropriate flight crew member.

FUEL SYSTEM LINES AND FITTINGS

Aircraft fuel system fluid lines and fittings have standards to ensure proper fuel system operation. Each fuel line must be installed and supported to prevent excessive vibration and to withstand loads due to fuel pressure and accelerated flight conditions. Lines connected to components of the airplane, between which relative motion could exist, must have provisions for flexibility. Any hose that is used must be shown to be suitable for a particular application.

AIRCRAFT FUEL SYSTEM LAYOUT

While each manufacturer designs its own fuel system, the basic fuel system requirements referenced at the beginning of this sub-module yield fuel systems of similar design and function in the field. In the following sections, representative examples of various fuel systems are given. The fuel systems of other aircraft are similar but certainly not identical.

Consult the manufacturer's maintenance manual for complete fuel system information on the aircraft upon which maintenance is being performed. Each aircraft fuel system must store and deliver clean fuel to the engine(s) at a pressure and flow rate able to sustain operations regardless of the operating conditions of the aircraft.

Fuel systems on high performance and large transport category jet aircraft are complex with some features and components not found in reciprocating-engine aircraft fuel systems. They typically contain more redundancy and facilitate numerous options from which the crew can choose while managing the aircraft's fuel load. Features like an onboard APU, single point pressure refueling, and fuel jettison systems (not needed on smaller aircraft)add to the complexity of an airliner fuel system. Jet transport fuel systems can be regarded as the fuel subsystems as follows:

1. Storage
2. Vent
3. Distribution
4. Feed
5. Indicating

FUEL TANKS

Most transport category aircraft fuel systems are very much alike. On many aircraft, especially transport category and high performance aircraft, part of the structure of the wings or fuselage is sealed with a fuel resistant two-part sealant to form a fuel tank. The sealed skin and structural members provide the highest volume of space available with the lowest weight. This type of tank is called an integral fuel tank since it forms a tank as a unit within the airframe structure. Integral fuel tanks in the otherwise unused space inside the wings are most common. Aircraft with integral fuel tanks in the wings are said to have wet wings. Center wing section or fuselage tanks are also common. These may be sealed structure or bladder type.

For fuel management purposes, sometimes a wing is sealed into separate tanks and may include a surge tank or an overflow tank, which is normally empty but sealed to hold fuel when needed. When an aircraft maneuvers, the long horizontal nature of an integral wing tank requires baffling to keep the fuel from sloshing. The wing ribs and box beam structural members serve as baffles and others may be added specifically for that purpose. Integral fuel tanks must have access panels for inspection and repairs of the tanks and other fuel system components. On large aircraft, technicians physically enter the tank for maintenance. Transport category aircraft often have more than a dozen oval access panels or tank plates on the bottom surface of the wing for this purpose. (***Figure 13-3***).

When entering and performing maintenance on an integral fuel tank, all fuel must be emptied from the tank and strict safety procedures must be followed.

Fuel vapors must be purged from the tank and respiratory equipment must be used by the technician. A full-time spotter must be positioned just outside of the tank to assist if needed.

Jet transport aircraft carry tens of thousands of pounds of fuel on board. ***Figure 13-4*** shows a diagram of a Boeing 777 fuel tank configuration with tank capacities. Note that there are optional fuel storage configurations available on the same model airliner. For example, airlines expecting to use an aircraft on transoceanic flights may order the aircraft with long-range auxiliary tanks. These additional tanks, usually located in the fuselage section of the aircraft, can alter fuel management logistics in addition to complicating the fuel system.

In addition to main and auxiliary fuel tanks, surge tanks may also be found on jet transports. These normally empty tanks located in the wing structure outboard of the main wing tanks are used for fuel overflow. A check valve allows the one-way drainage of fuel back into the main tanks. Surge tanks are also used for fuel system venting. Aircraft using integral fuel tanks normally have sophisticated fuel systems that include in-tank boost pumps. On various aircraft, these in-tank boost pumps are also used to transfer fuel to other tanks, jettison fuel, and defuel the aircraft.

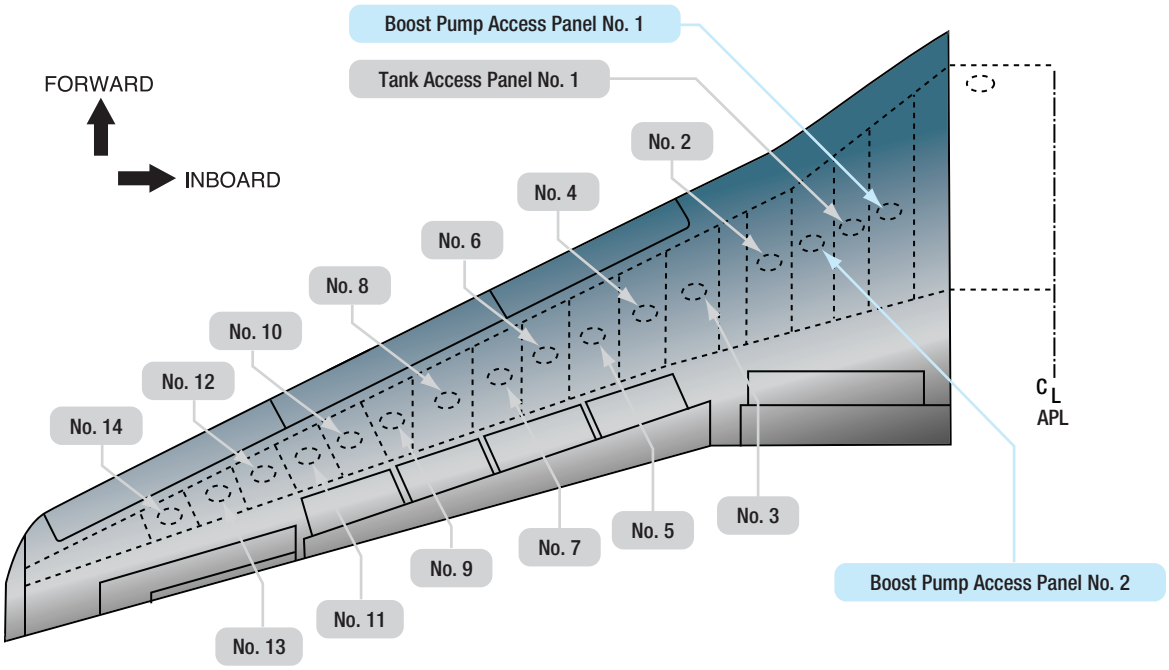


Figure 13-3. Fuel tank access panel locations on a Boeing 737 (A), and typical fuel tank access panel seals (B).

Tank	Gallons	Pounds *
Left Main Tank	9 560	64 000
Right Main Tank	9 560	64 000
Center Tank	26 100	174 900
Total	45 200	302 900

* Usable Fuel at Level Attitude
Fuel Density = 6.7 Pounds Per U.S. Gallon

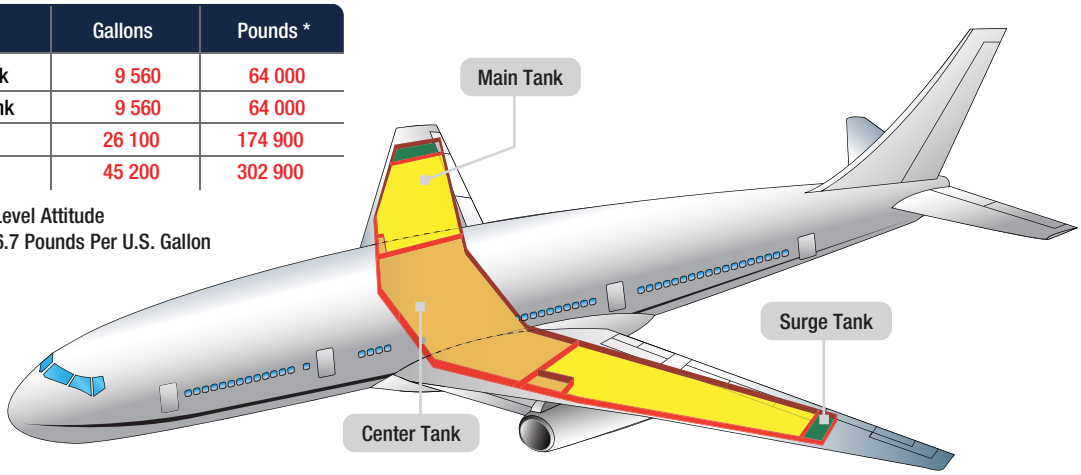


Figure 13-4. Boeing 777 fuel tank locations and capacities.

A fuel tank made out of a reinforced flexible material called a bladder tank can be used instead of a rigid tank (***Figure 13-5***). A bladder tank contains most of the features and components of a rigid tank but does not require as large an opening in the aircraft skin to install. Bladder fuel tanks are used on aircraft of all size. Bladder tanks are usually required to remain wetted with fuel at all times to prevent drying and cracking of the bladder material.

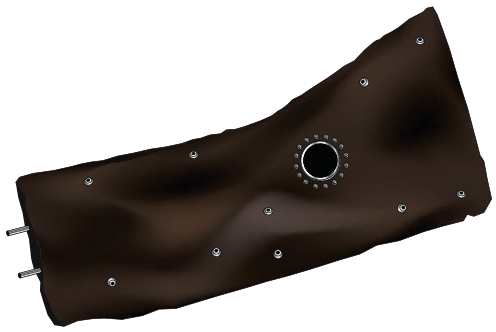


Figure 13-5. An aircraft bladder fuel tank.

FUEL SUPPLY SYSTEMS

Fuel is supplied from the fuel tanks of transport category aircraft to the engines and auxiliary power unit through a distribution system. This consists of fuel lines that connect various valves, pumps, heater exchangers and indication system components. The fuel feed, crossfeed and transfer systems are all part of the distribution system. Pressure fueling and refueling components may also be considered part of the fuel supply/distribution system. They are covered in a separate section of this sub-module.

FUEL FEED

The fuel feed system is the heart of the fuel supply system since it delivers fuel to the engines. Jet transport aircraft supply fuel to the engines via in-tank fuel boost pumps, usually two per tank. They pump fuel under pressure through a shutoff valve for each engine. A manifold or connecting tubing typically allows any tank to supply any engine through the use of cross-feed valves.

Boost pump bypass valves allow fuel flow should a pump fail and check valves allow fuel flow only in the proper direction towards the engines. Note that the engines are designed to be able to run without any fuel boost pumps operating. However, each engine's fuel shutoff valve must be open to allow any flow to the engines from the tanks.

CROSS-FEED AND TRANSFER SYSTEMS

It is important for the flight crew as well as technicians to manage the location of fuel between tanks. The fuel transfer system is the series of fuel lines and valves that permits movement of fuel from one tank to another on board the aircraft. Fuel boost pumps or dedicated transfer pumps move fuel from the tank in which the pump is located into a manifold containing valves for the other tanks. By opening the fuel valve (or refuelling valve) to the tank into which fuel is desired, the fuel is transferred from one tank, into the manifold, and then into to the desired tank.

Figure 13-6 shows the fuel system diagram for a DC-10. Dedicated transfer boost pumps move fuel into the transfer manifold. Opening the fuel valve on one of the tanks transfers the fuel into that tank. Note that the transfer manifold and boost pumps are also used to jettison fuel overboard by opening the proper dump valves with a transfer boost pump(s) operating. Additionally, the transfer system can function to supply the engines if the normal engine fuel feed malfunctions.

Not all jet transports have such fuel transfer capability. Through the use of a fuel feed manifold and cross-feed valves, some aircraft simply allow engines to be run off fuel from any tank as a means for managing fuel location. This is seen in ***Figure 13-7***. The 737 does not have transfer pumps and a dedicated transfer manifold. It does have a fuel manifold made up of interconnect fuel lines from each boost pump. A cross feed valve separates the manifold into two halves.

Normally, the left fuel manifold is used to operate the left engine and the right fuel manifold is used to operate the right engine. The boost pumps on one side of the aircraft pump fuel to the engine on the opposite side when the cross-feed valve is open. Essentially, by opening the cross-feed valve, the fuel manifold can be

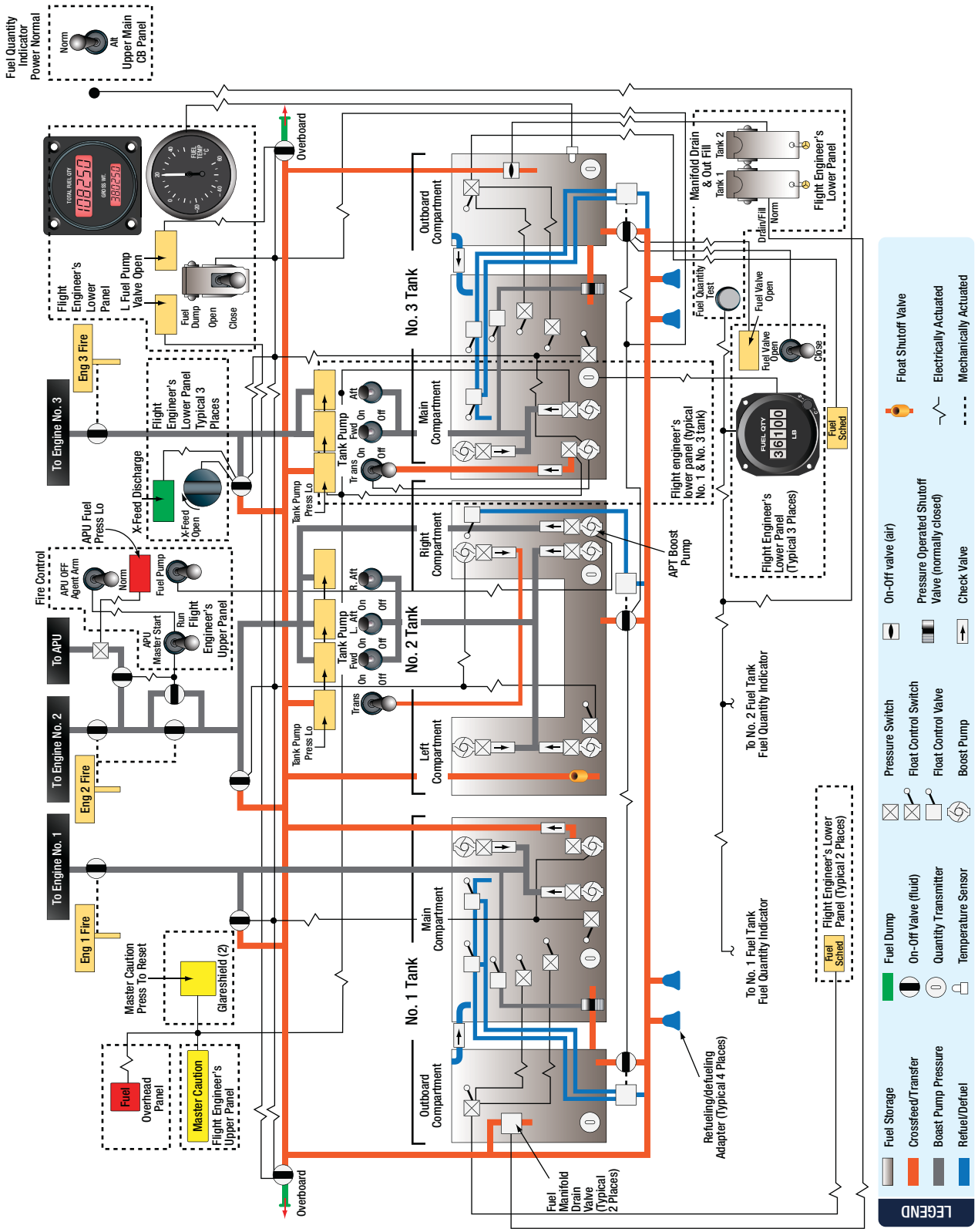


Figure 13-6. The fuel distribution systems, components, and cockpit controls of a DC-10 airliner. NOTE: Fuel transfer system components and lines are used to complete the fuel dump system, the refuel/defuel system, back-up fuel delivery system, and the fuel storage system.

pressurized and supplied with fuel from any boost pump on the aircraft. Thus, by opening the engine fuel shutoff valve to a particular engine, the engine can operate off of any boost pump in any tank.

FUEL DUMPING, VENTING AND DRAINING

FUEL SYSTEM DRAINS

Aircraft fuel systems must be fitted with at least one drain to allow safe drainage of the entire fuel system with the airplane in its normal ground attitude. The drain must discharge the fuel clear of all parts of the aircraft. A readily accessible drain valve that can easily be opened and closed is required (either manually or automatically). Fuel should be collectible from the system drain valve so it can be examined.

FUEL VENT SYSTEMS

Transport category fuel systems require venting similar to reciprocating engine aircraft fuel systems. A series of vent tubing and channels exists that connects all tanks to vent space in the surge tanks (if present) or vent overboard. Venting must be configured to ensure the fuel is vented regardless of the attitude of the aircraft or the quantity of fuel on board. This sometimes requires the installation of various check valves, float valves, and multiple vent locations in the same tank.

FUEL JETTISONING SYSTEM

If an aircraft's design landing weight is less than that of the maximum takeoff weight, a situation could occur in which a landing is desired before sufficient fuel has burned off to lighten the aircraft. Fuel jettisoning systems are required on these aircraft so that fuel can be jettisoned in flight to avoid structural damage cause by landing the aircraft when it is too heavy. Fuel jettisoning systems are also referred to as fuel dump systems. The system must be designed so that any reasonably probable single malfunction in the system does not result in a hazardous condition due to unsymmetrical jettisoning of, or inability to jettison, fuel.

FUEL SYSTEM COMPONENTS

To better understand aircraft fuel systems and their operation, the following discussion of various components of aircraft fuel systems is included. Fuel tanks are a key component that have already been discussed above.

FUEL LINES AND FITTINGS

Aircraft fuel lines can be rigid or flexible depending on location and application. Flexible hoses are used in areas where vibration exists between components, such as between the engine and the aircraft structure. Sometimes manufacturers wrap either flexible or rigid fuel lines to provide even further protection from abrasion and especially from fire.

Metal fuel lines and all aircraft fuel system components need to be electrically bonded and grounded to the aircraft structure. This is important because fuel flowing through the fuel system generates static electricity that must have a place to flow to ground rather than build up.

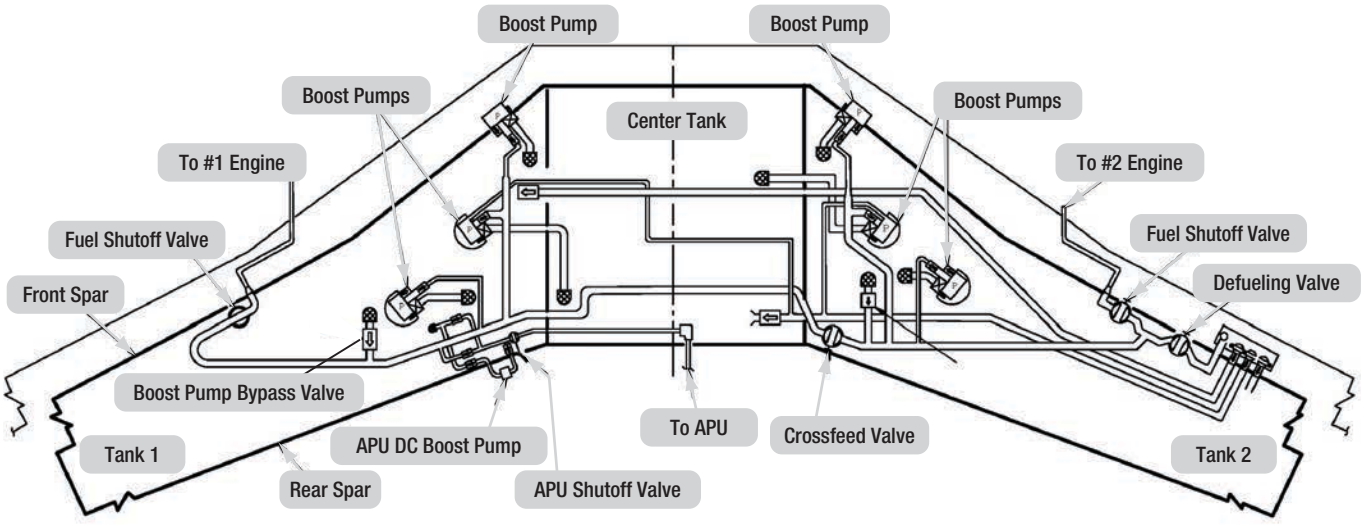


Figure 13-7. The fuel feed system of a Boeing 737.

FUEL VALVES

There are many fuel valves used in aircraft fuel systems. They are used to shut off fuel flow or to route the fuel to a desired location. Large aircraft fuel systems have numerous valves. Most simply open and close and are known by different names related to their location and function in the fuel system (e.g., shutoff valve, transfer valve, cross-feed valve). Fuel valves can be manually operated, solenoid operated, or operated by electric motor. A feature of all aircraft fuel valves is a means for positively identifying the position of the valve at all times. Motor and solenoid operated valves use position annunciator lights to indicate valve position in addition to the switch position.

Flight management system (FMS) fuel pages also display the position of the fuel valves graphically in diagrams called up on the flat screen monitors. (***Figure 13-8***).

Note that many valves have an exterior position handle, or lever, that indicates valve position. When maintenance personnel directly observe the valve, it can be manually positioned by the technician using this same lever. (***Figure 13-9***)

MANUALLY-OPERATED GATE VALVES

In complex fuel systems of transport category aircraft, fuel flow is controlled with a series of ON/OFF, or shutoff, type valves that are plumbed between system components. Hand operated gate valves can be used, especially as fire control valves, requiring no electrical power to shutoff fuel flow when the emergency fire handle is pulled.

MOTOR-OPERATED VALVES

The use of electric motors to operate fuel system valves is common on large aircraft due to the remote location from the cockpit of fuel system components. The types of valves used are basically the same as the manually operated valves, but electric motors are used to actuate the units.

SOLENOID-OPERATED VALVES

An additional way to operate a remotely located fuel valve is through the use of electric solenoids.

FUEL PUMPS

Other than aircraft with gravity-feed fuel systems, all aircraft have at least one fuel pump to deliver clean fuel under pressure to the fuel metering device for each engine. Engine driven pumps are the primary delivery device. Auxiliary pumps are used on many aircraft as well. Sometimes known as booster pumps or boost pumps, auxiliary pumps are used to provide fuel under positive pressure to the engine-driven pump and during starting when the engine-driven pump is not yet up to speed for sufficient fuel delivery.

They are also used to back up the engine-driven pump during takeoff and at high altitude to guard against vapor lock. On many large aircraft, boost pumps are used to move fuel from one tank to another. There are many different types of auxiliary fuel pumps in use. Most are electrically operated.



Figure 13-8. The graphic depiction of the fuel system on this electronic centralized aircraft monitor (ECAM) fuel page includes valve position information.



Figure 13-9. This motor-operated gate valve has a red position indicating lever that can be used by maintenance personnel to identify the position of the valve. The lever can be moved by the technician to position the valve.

FUEL FILTERS

Two main types of fuel cleaning device are utilized on aircraft. Fuel strainers are usually constructed of relatively coarse wire mesh. They are designed to trap large pieces of debris and prevent their passage through the fuel system. Micronic filters are commonly used on turbine powered aircraft. This is a type of filter that captures extremely fine particles in the range of 10 - 25 microns. A micron is 1/1 000 of a millimeter.

FUEL INDICATION AND WARNINGS

Fuel indicating systems on jet transport aircraft monitor a variety of parameters, some not normally found on general aviation aircraft. Business jet aircraft share many of these features. True fuel flow indicators for each engine are used as the primary means for monitoring fuel delivery to the engines. A fuel temperature gauge is common as are fuel filter bypass warning lights.

The temperature sensor is usually located in a main fuel tank. The indicator is located on the instrument panel or is displayed on a multifunction display (MFD). These allow the crew to monitor the fuel temperature during high altitude flight in extremely frigid conditions. The fuel filters have bypasses that permit fuel flow around the filters if clogged. Indicator light(s) illuminate in the cockpit when this occurs. Valve position indicators and various warning lights and annunciations are also used.

Low fuel pressure warning lights are also common on jet transport aircraft. The sensors for these are located in the boost pump outlet line. They give an indication of possible boost pump failure. Fuel quantity gauges are important features on all aircraft. Indications exist for all tanks on a transport category aircraft. Often, these use a capacitance type fuel quantity indication system and a fuel totalizer as discussed below. The location of fuel instrumentation varies depending on the type of cockpit displays utilized on the aircraft.

FUEL QUANTITY INDICATING SYSTEMS

All aircraft fuel systems must have some form of fuel quantity indicator. These devices vary widely depending on the complexity of the fuel system and the aircraft on which they are installed.

RATIOMETER TYPE

Electric fuel quantity indicators are common in aircraft. Most of these units operate with direct current (DC) and use variable resistance in a circuit to drive a ratiometertype indicator. The movement of a float in the tank moves a connecting arm to the wiper on a variable resistor in the tank unit. This resistor is wired in series with one of the coils of the ratiometer type fuel gauge in the instrument panel. Changes to the current flowing through the tank unit resistor change the current flowing through one of the coils in the indicator. This alters the magnetic field in which the indicating pointer pivots. The calibrated dial indicates the corresponding fuel quantity. (***Figure 13-10***)

Digital indicators are available that work with the same variable resistance signal from the tank unit. They convert the variable resistance into a digital display in the cockpit instrument head. (***Figure 13-11***) Fully digital instrumentation systems, such as those found in a glass cockpit aircraft, convert the variable resistance into a digital signal to be processed in a computer and displayed on a flat screen panel.

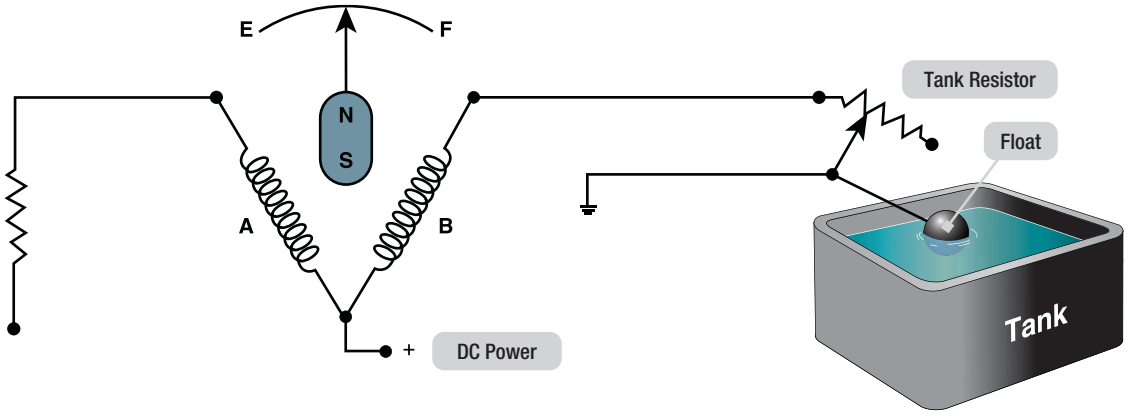


Figure 13-10. A DC electric fuel quantity indicator uses a variable resistor in the tank unit, which is moved by a float arm.



Figure 13-11. Digital fuel quantity gauges that work off of variable resistance from the tank unit are shown in A and B. The fuel quantity indication of a Garmin G-1000 flat screen display is shown in C.

CAPACITANCE TYPE

Large and high performance aircraft typically utilize electronic fuel quantity systems. These more costly systems have the advantage of having no moving parts in the tank sending units. Variable capacitance transmitters are installed in the fuel tanks extending from the top to the bottom of each tank in the usable fuel. Several of these tank units, or fuel probes as they are sometimes called, may be installed in a large tank. (***Figure 13-12***)

They are wired in parallel. As the level of the fuel changes, the capacitance of each unit changes. The capacitance transmitted by all of the probes in a tank is totalled and compared in a bridge circuit by a microchip computer in the tank's digital fuel quantity indicator in the cockpit.

As the aircraft maneuvers, some probes are in more fuel than others due to the attitude of the aircraft. The indication remains steady, because the total capacitance transmitted by all of the probes remains the same. A trimmer is used to match the capacitance output with the pre-calibrated quantity indicator.

A capacitor is a device that stores electricity. The amount it can store depends on three factors: the area of its plates, the distance between the plates, and the dielectric constant of the material separating the plates. A fuel tank unit contains two concentric plates that are a fixed distance apart. Therefore, the capacitance of a unit can change if the dielectric constant of the material separating the plates varies. The units are open at the top and bottom so they can assume the same level of fuel as is in the tanks. Therefore, the material between the plates is either fuel (if the tank is full), air (if the tank is empty), or some ratio of fuel and air depending on how much fuel remains in the tank. ***Figure 13-13*** shows a simplified illustration of this construction.

The bridge circuit that measures the capacitance of the tank units uses a reference capacitor for comparison.

When voltage is induced into the bridge, the capacitive reactance of the tank probes and the reference capacitor can be equal or different. The magnitude of the difference is translated into an indication of the fuel quantity in the tank calibrated in pounds. ***Figure 13-14*** represents the nature of this comparison bridge circuit.

The use of tank unit capacitors, a reference capacitor, and a microchip bridge circuit in the fuel quantity indicators is complicated by the fact that temperature affects the dielectric constant of the fuel. A compensator unit

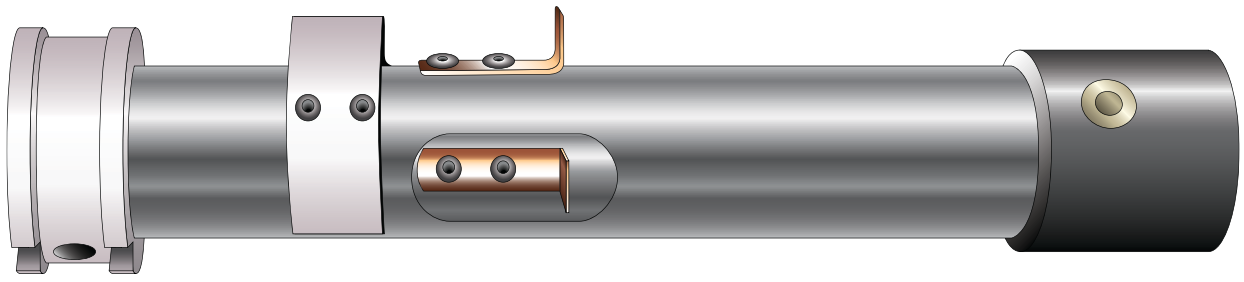


Figure 13-12. A fuel tank transmitter for a capacitance-type fuel quantity indicating system.

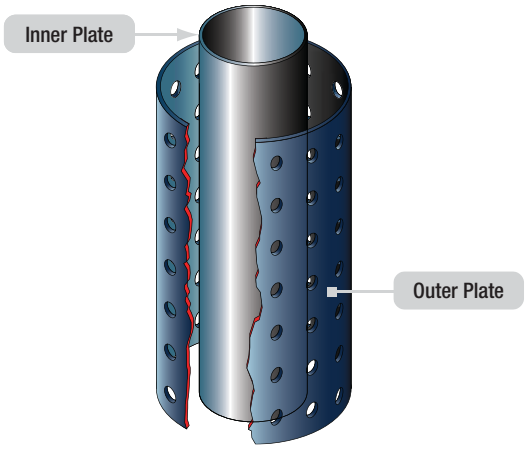


Figure 13-13. The capacitance of tank probes varies in a capacitance- type fuel tank indicator system as the space between the inner and outer plates is filled with varying quantities of fuel and air depending on the amount of fuel in the tank.

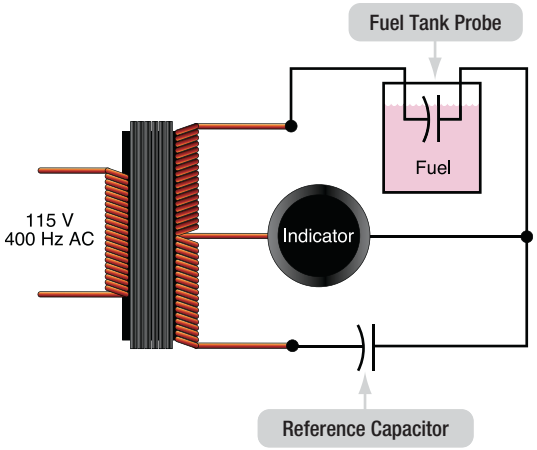


Figure 13-14. A simplified capacitance bridge for a fuel quantity system.

(mounted low in the tank so it is always covered with fuel) is wired into the bridge circuit. It modifies current flow to reflect temperature variations of the fuel, which affect fuel density and thus capacitance of the tank units. (***Figure 13-15***)

An amplifier is also needed in older systems. The amplitude of the electric signals must be increased to move the servo motor in the analog indicator. Additionally, the dielectric constant of different turbine engine fuels approved for a particular aircraft may also vary. Calibration is required to overcome this. A fuel summation unit is part of the capacitance fuel quantity indication system. It is used to add the tank quantities from all indicators. This total aircraft fuel quantity can be used by the crew and by flight management computers for calculating optimum airspeed and engine performance limits for climb, cruise, descent, etc. Capacitance fuel quantity system test units are available for troubleshooting and ensuring proper functioning and calibration of the indicating system components.

MECHANICAL TYPE

Many aircraft with capacitance fuel indicating systems also use a mechanical indication system to cross-check fuel quantity indications and to ascertain the amount of fuel onboard the aircraft when electrical power is not available. A handful of fuel measuring sticks, or drip sticks, are mounted throughout each tank. When pushed and rotated, the drip stick can be lowered until fuel begins to exit the hole on the bottom of each stick. This is the point at which the top of the stick is equal to the height of the fuel. The sticks have a calibrated scale on them. By adding the indications of all of the drip sticks and converting to pounds or gallons via a chart supplied by the manufacturer, the quantity of the fuel in the tank can be ascertained. (***Figure 13-16***)

FUEL FLOWMETERS

A fuel flowmeter indicates an engine's fuel use in real time. This can be useful to the pilot for ascertaining engine performance and for flight planning calculations. The types of fuel flow meter used on an aircraft depends primarily on the powerplant being used and the associated fuel system. Measuring fuel flow accurately is complicated by the fact that the fuel mass changes with temperature or with the type of fuel used in turbine engines.

Turbine engine aircraft experience the greatest range of fuel density from temperature variation and fuel composition. An elaborate fuel flow device is used on these aircraft. It measures fuel mass for accurate fuel flow indication in the cockpit. The mass flow indicator takes advantage of the direct relationship between fuel mass and viscosity. Fuel is swirled by a cylindrical impeller that rotates at a fixed speed. The outflow deflects a turbine just downstream of the impeller. The turbine is held with calibrated springs. Since the impeller motor swirls the fuel at a fixed rate, any variation of the turbine deflection is caused by the volume and viscosity of the fuel. The viscosity component represents the mass of the fuel. (***Figure 13-17***)

An alternating current (AC) synchro system is part of the mass fuel flowmeter. It is used to position a pointer against the cockpit indicator scale calibrated in pounds per hour. With accurate fuel flow knowledge, numerous calculations can be performed to aid the pilot's situational awareness and flight planning. Most high performance aircraft have a fuel totalizer that electronically calculates and displays information, such as total fuel used, total fuel remaining onboard the aircraft, total range and flight time remaining at the present airspeed, rate of fuel consumption, etc.

Relatively new types of fuel flow sensors/transmitters are available in new aircraft and for retrofit to older aircraft. Increasing use of microprocessors and computers on

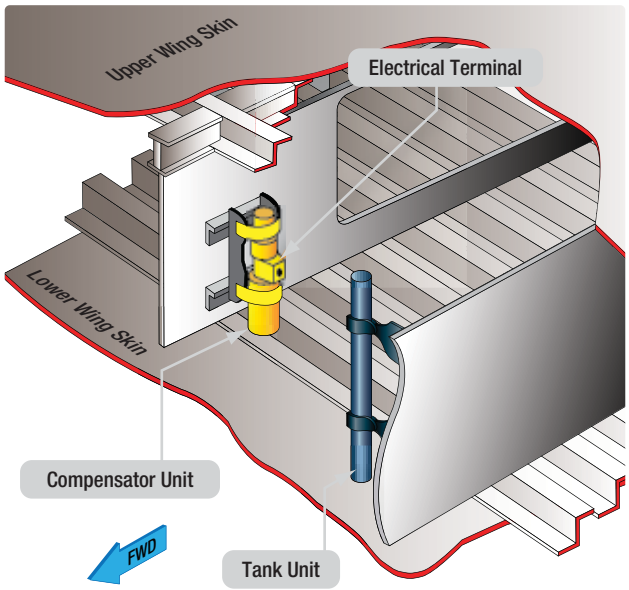


Figure 13-15. A fuel quantity tank unit and compensator unit installed inside a wing tank.

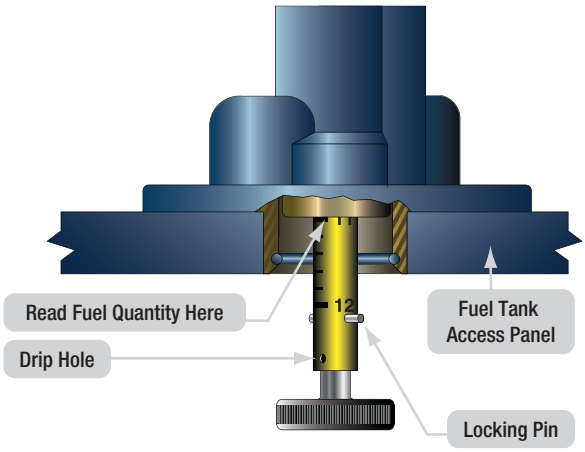


Figure 13-16. A fuel drip stick is lowered from the fuel tank bottom until fuel drips out the hole at the bottom. By reading the calibrated scale and adding readings from all tank drip sticks, a chart can be consulted to arrive at the total fuel quantity on the aircraft by weight or by volume.

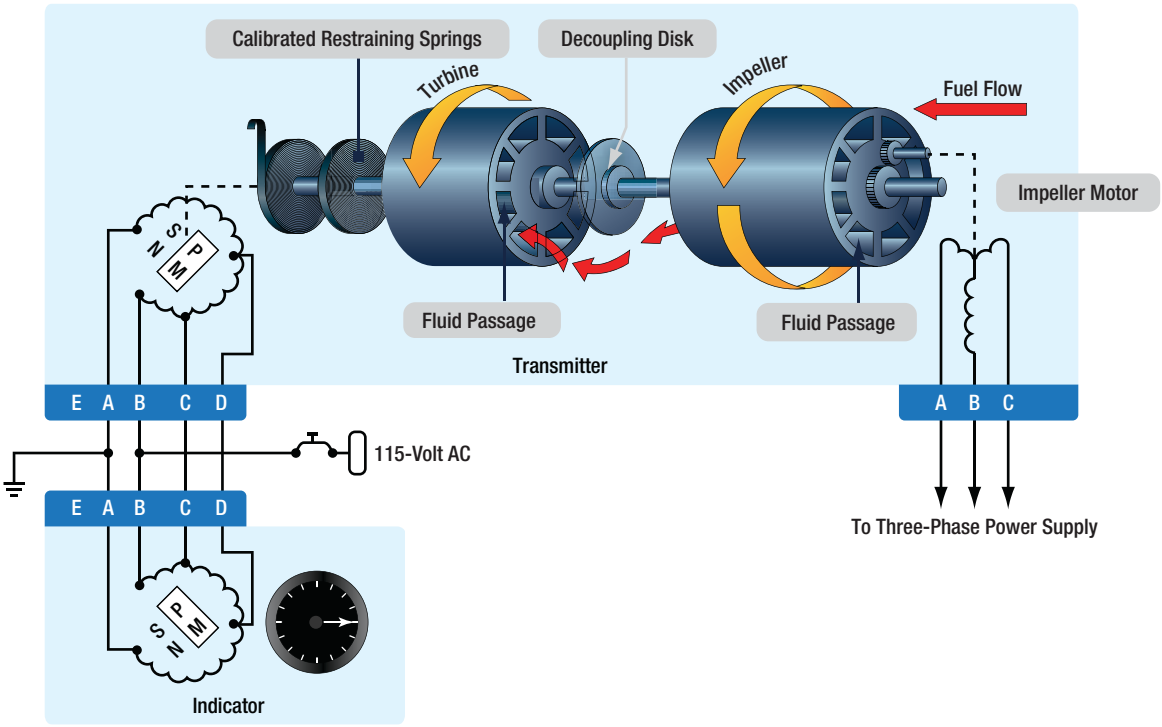


Figure 13-17. A mass flow fuel flow indicating system used on turbine-engine aircraft uses the direct relationship between viscosity and mass to display fuel flow in pounds per hour.

aircraft enable the integration of fuel temperature and other compensating factors to produce highly accurate fuel flow information. Fuel flow sensing with digital output facilitates this with a high degree of reliability. Thermal dispersion technology provides flow sensing with no moving parts and digital output signals. The sensor consists of two resistance temperature detectors (RTDs). One is a reference RTD that measures the temperature of the fuel. The other is the active RTD. It is heated by an adjacent element to a temperature higher than the fuel.

As the fuel flows, the active element cools proportionally to the fuel flow. The temperature difference between the two RTDs is highest at no flow. The RTDs are connected to an electronic assembly that supplies power to the heater and uses sensing circuitry and a microprocessor to control a constant temperature difference between the heated and unheated RTDs. The electrical current to the heater is proportional to the mass flow of the fuel. As mentioned, the reference RTD is used as a temperature sensor to provide a temperature output and allow for temperature compensation of the flow measurement. (***Figure 13-18***)



Figure 13-18. Fuel flow sensing units using thermal dispersion technology have no moving parts and output digital signals.

FUEL TEMPERATURE GAUGES

Monitoring fuel temperature can inform the pilot when fuel temperature approaches that which could cause ice to form in the fuel system, especially at the fuel filter. Many large and high performance turbine aircraft use a resistance type electric fuel temperature sender in a main fuel tank for this purpose. It can display on a traditional ratiometer gauge (***Figure 13-19***) or can be input into a computer for processing and digital display. A low fuel temperature can be corrected with the use of a fuel heater if the aircraft is so equipped. Also as mentioned, fuel temperature can be integrated into fuel flow processing calculations. Viscosity differences at varying fuel temperatures that affect fuel flow sensing accuracy can be corrected via microprocessors and computers.

FUEL HEATERS

Turbine powered aircraft operate at high altitude where the temperature is very low. As the fuel in the fuel tanks cools, water in the fuel condenses and freezes. It may form ice crystals in the tank or as the fuel/water solution slows and contacts the cool filter element on its way through fuel filter to the engine(s). The formation of ice on the filter element blocks the flow of fuel through the filter. A valve in the filter unit bypasses unfiltered fuel when this occurs. Fuel heaters are used to warm the fuel so that ice does not form. These heat exchanger units also heat the fuel sufficiently to melt any ice that has already formed. The most common types of fuel heaters are air/fuel heaters and oil/fuel heaters. An air/fuel heater uses warm compressor bleed air to heat the fuel. An oil/fuel exchanger heats the fuel with hot engine oil. This latter type is often referred to as a fuel cooled oil cooler (FCOC). It not only heats the fuel but also cools the engine oil. (***Figure 13-20***)

Fuel heaters often operate intermittently as needed. A switch in the cockpit can direct the hot air or oil through the unit or block it. The flight crew uses the information supplied by the filter bypass indicating lights and fuel temperature gauge as seen in ***Figure 13-19*** to know when to heat the fuel. Fuel heaters can also be automatic.

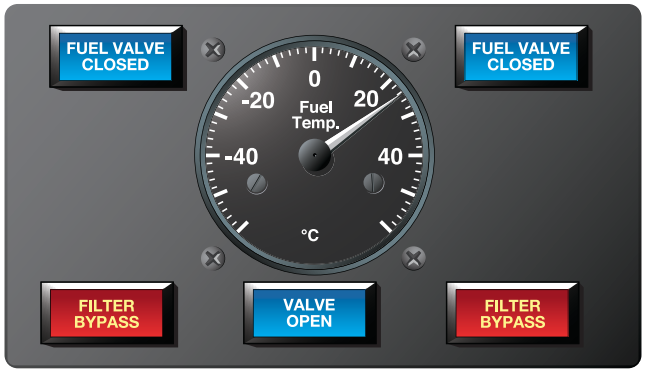


Figure 13-19. A Boeing 737 cockpit fuel panel showing illuminated valve position indicators and fuel filter bypass lights. The fuel temperature in tank No.1 is also indicated.



Figure 13-20. Jet transport aircraft fly at high altitudes where temperatures can reach -50 °F. Most have fuel heaters somewhere in the fuel system to help prevent fuel icing. This fuel-cooled oil cooler on an RB211 turbofan engine simultaneously heats the fuel while cooling the oil.

A built-in thermostatic device opens or closes a valve that permits the hot air or hot oil to flow into the unit to cool the fuel. (***Figure 13-21***)

Note that some aircraft have a hydraulic fluid cooler in one of the aircraft fuel tanks. The fluid helps warm the fuel as it cools in this type of full time heat exchanger.

FUEL PRESSURE GAUGES

Monitoring fuel pressure can give the pilot early warning of a fuel system related malfunction. Verification that the fuel system is delivering fuel to the fuel metering device can be critical. Simple light reciprocating engine aircraft typically utilize a direct reading Bourdon tube pressure gauge. It is connected into the fuel inlet of the fuel metering device with a line extending to the back of the gauge in the cockpit instrument panel. A more complex aircraft may have a sensor with a transducer located at the fuel inlet to the metering device that sends electrical signals to a cockpit gauge. (***Figure 13-22***)

In aircraft equipped with an auxiliary pump for starting and to backup the engine driven pump, the fuel pressure gauge indicates the auxiliary pump pressure until the engine is started. When the auxiliary pump is switched off, the gauge indicates the pressure developed by the engine driven pump. Modern aircraft may use a variety of sensors including solid state types and those with digital output signals or signals that are converted to digital output. These can be processed in the instrument gauge microprocessor, if so equipped, or in a computer and sent to the display unit. (***Figure 13-23***)

PRESSURE WARNING SIGNAL

On aircraft of any size, visual and audible warning devices are used in conjunction with gauge indications to draw the pilot's attention to certain conditions. Fuel pressure is an important parameter that merits the use of a warning signal when it falls outside of the normal operating range. Low fuel pressure warning lights can be illuminated through the use of simple pressure sensing switches. (***Figure 13-24***)

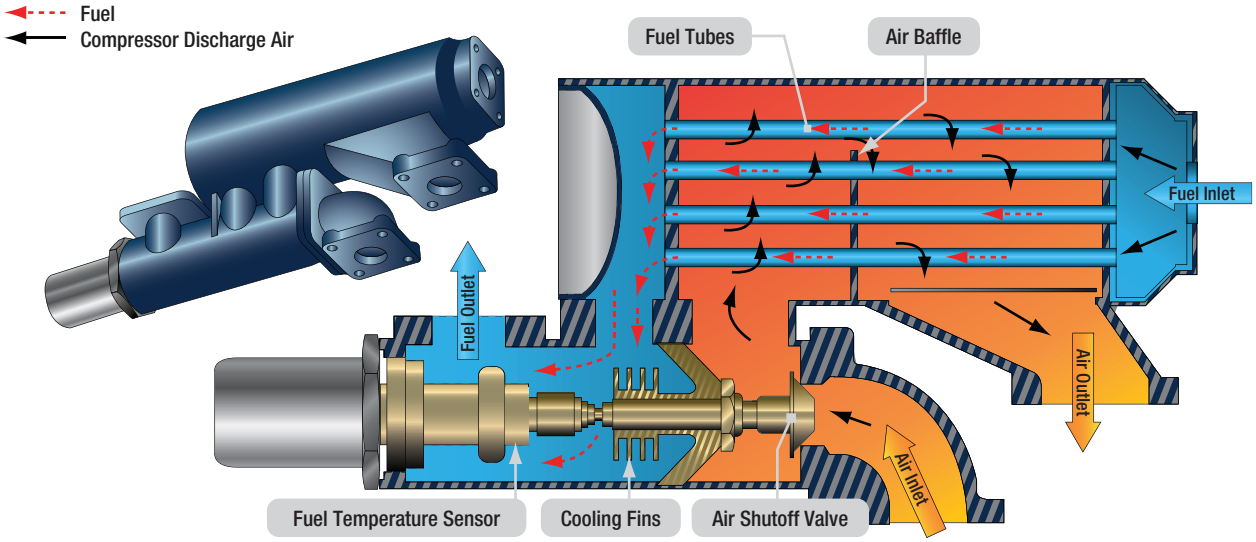


Figure 13-21. An air-fuel heat exchanger uses engine compressor bleed air to warm the fuel on many turbine engine powered aircraft.

The contacts of the switch will close when fuel pressure against the diaphragm is insufficient to hold them open. This allows current to flow to the annunciator or warning light in the cockpit. Most turbine powered aircraft utilize a low pressure warning switch at the outlet of each fuel boost pump. The annunciator for each is typically positioned adjacent to the boost pump ON/OFF switch on the fuel panel in the cockpit. (***Figure 13-25***)

VALVE-IN-TRANSIT INDICATOR LIGHTS

Aircraft with multiple fuel tanks use valves and pumps to move fuel and to have it flow to desired locations, such as the engines, a certain tank, or overboard during fuel jettison. The functioning of the valves in the fuel system is critical. Some aircraft indicate to the crew when the valve is opening or closing with the use of valve-intransit lights. Contacts in the valve control the lights that go out when the valve is fully open or when it is fully closed. Alternately, annunciator lights that show the valve position as OPEN or CLOSED are also used. Valve-in-transit and valve position indicators, or lights, are located on the fuel panel in the cockpit adjacent to the valve ON/OFF switches. (***Figure 13-26***) Sometimes the switch mechanism has the annunciator light built into it. Digital display systems graphically depict valve positions on screen.

REFUELING AND DEFUELING

Single point pressure fueling at a fueling station accessible by ramp refueling trucks allows all aircraft fuel tanks to be filled with one connection of the fuel hose. Leading and trailing edge wing locations are common for these stations. ***Figure 13-27*** shows an airliner fueling station with the fueling rig attached.

To fuel with pressure refueling, a hose nozzle is attached at the fueling station and valves to the tanks required to be filled are opened. These valves are called fueling valves or refueling valves depending upon the manufacturer's preference.

Various automatic shutoff systems have been designed to close tank fueling valves before the tanks overfill or are damaged. Gauges on the refueling panel allow refueling personnel to monitor progress. Occasionally, defueling the aircraft is required for an inspection or repair. The same fueling station is used, and the hose from the fuel truck is connected to same receptacle used to fuel the aircraft. To allow fuel to exit the aircraft, a defueling valve is opened. Fuel can either be pumped out of the aircraft using the boost pumps located in the tanks that need to be emptied, or the pump in the refueling truck can be used to draw the fuel out of the tanks.

Control over the operation is maintained by positioning various shutoff and crossfeed valves, as well as the defuel valve so that fuel travels from the tank to the fueling station and into the truck. Maintenance technicians are often asked to fuel or defuel aircraft. Fueling procedure

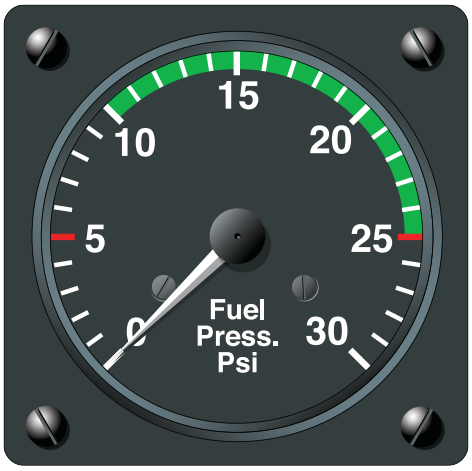


Figure 13-22. A typical fuel gauge that uses a signal from a sensing transducer to display fuel inlet pressure at the metering device.

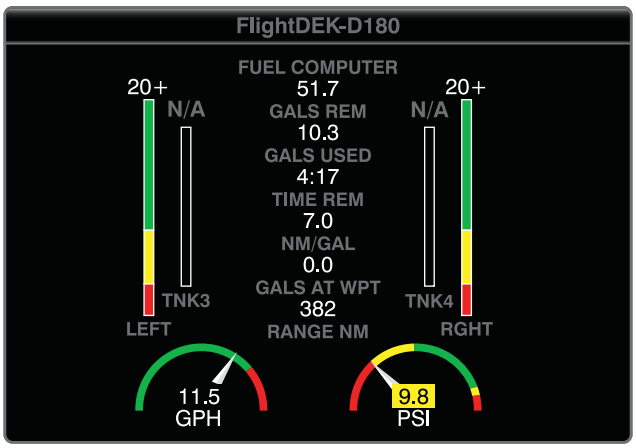


Figure 13-23. An electronic display of fuel parameters, including fuel pressure.

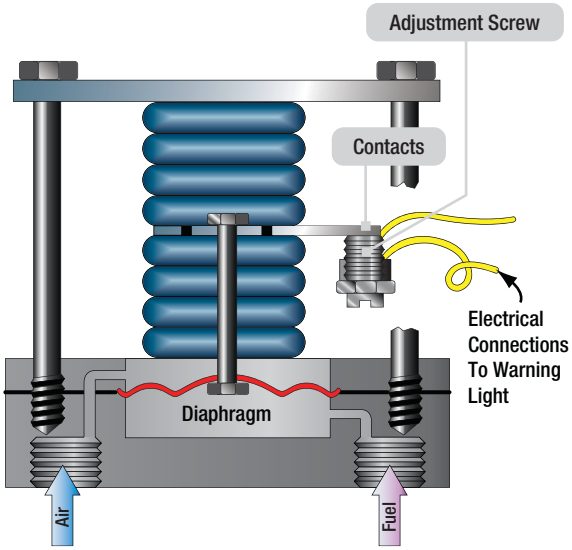


Figure 13-24. A fuel pressure warning signal is controlled by a switch that closes when fuel pressure is low.



Figure 13-25. A transport category aircraft fuel panel with low pressure warning lights for each fuel boost pump.

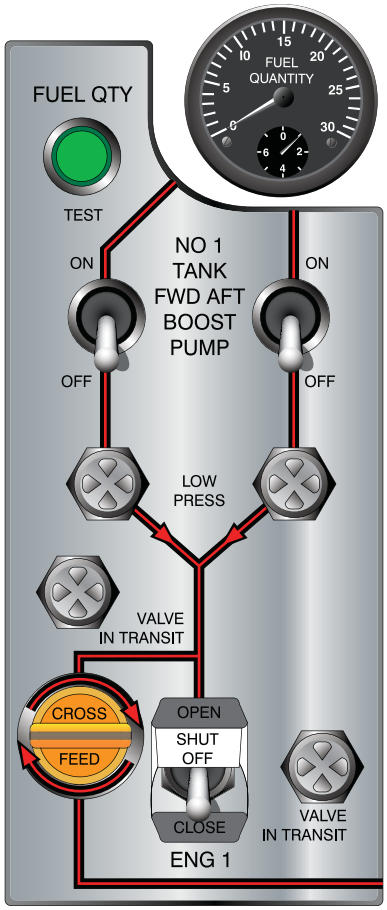


Figure 13-26. Valve-in-transit lights are used on this section of a transport category aircraft fuel panel. Low boost pump pressure lights that look the same are also on the panel.

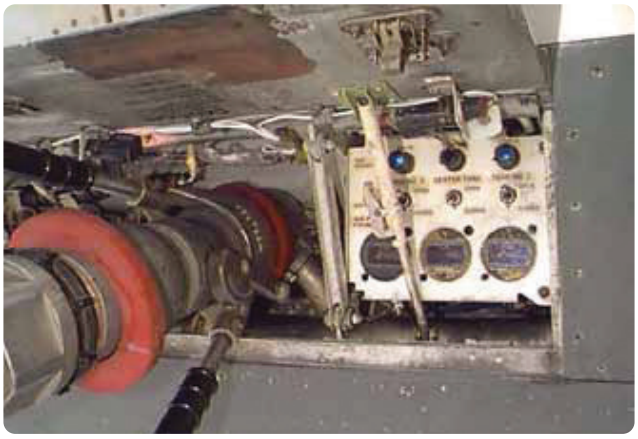


Figure 13-27. A central pressure refueling station on a transport category aircraft allows all fuel tanks to be filled from one position.

can vary from aircraft to aircraft. Tanks may need to be fueled in a prescribed sequence to prevent structural damage to the airframe. The proper procedure should be confirmed before fueling an unfamiliar aircraft.

FUELING

Always fuel aircraft outside, not in a hangar where fuel vapors may accumulate and increase the risk and severity of an accident. Generally, there are two types of fueling process: over-the-wing refueling and pressure refueling.

OVER THE WING REFUELING

Over the wing refueling is accomplished by opening the fuel tank cap on the upper surface of the wing or fuselage, if equipped with fuselage tanks. The fueling nozzle is carefully inserted into the fill opening and fuel is pumped into the tank. This process is similar to the

process used to refuel an automobile gas tank. Aviation fuel nozzles are equipped with static bonding wires that must be attached to the aircraft before the fuel cap is opened. (***Figure 13-28***)

PRESSURE REFUELING

When pressure refueling, the aircraft receptacle is part of a fueling valve assembly. When the fueling nozzle is properly connected and locked, a plunger unlocks the aircraft valve so fuel can be pumped through it. Normally, all tanks can be fueled from a single point. Valves in the aircraft fuel system are controlled at the fueling station to direct the fuel into the proper tank. (***Figure 13-29***)

Note that, while similar, pressure fueling panels and their operation are different on different aircraft. Refueling personnel should be guided through the correct use of each panel.

DEFUELING

Removing the fuel contained in aircraft fuel tanks is sometimes required. This can occur for maintenance, inspection, or due to contamination. Occasionally, a change in flight plan may require defueling. Safety procedures for defueling are the same as those for fueling. Always defuel outside. Fire extinguishers should be on hand. Bonding cables should be attached to guard against static electricity buildup. Defueling should be performed by experienced personnel, and inexperienced personnel must be checked out before doing so without assistance.

Remember that there may be a sequence in defueling an aircraft's fuel tanks just as there is when fueling to avoid structural damage. Consult the manufacturer's maintenance/operations manual(s) if in doubt. Pressure fueled aircraft normally defuel through the pressure fueling port. The aircraft's in-tank boost pumps can be used to pump the fuel out. The pump on a fuel truck can also be used to draw fuel out. These tanks can also be drained through the tank sump drains, but the large size of the tanks usually makes this impractical. Aircraft fueled over the wing are normally drained through the tank sump drains. Follow the manufacturer's procedure for defueling the aircraft.

FIRE HAZARDS WHEN FUELING OR DEFUELING

Due to the combustible nature of AVGAS and turbine engine fuel, the potential for fire while fueling and defueling aircraft must be addressed. Always fuel and defuel outside, not in a hangar that serves as an enclosed area for vapors to build up to a combustible level.

Clothing worn by refueling personnel should not promote static electricity buildup. Synthetics, such as nylon should be avoided. Cotton has proved to be safe for fuel handling attire.

As previously mentioned, the most controllable of the three ingredients required for fire is the source of ignition. It is absolutely necessary to prevent a source of ignition anywhere near the aircraft during fueling or refueling. Any open flame, such as a lit cigarette, must be extinguished. Operation of any electrical devices must be avoided. Radio and radar use is prohibited. It is important to note that fuel vapors proliferate well beyond the actual fuel tank opening and a simple spark,

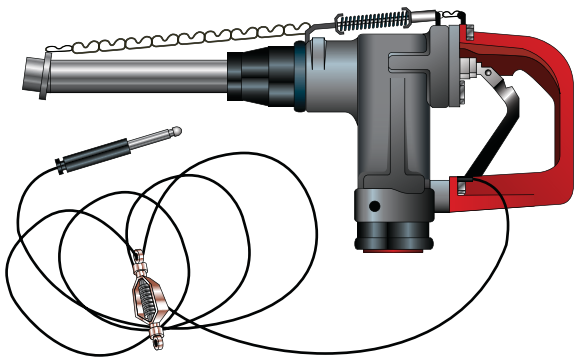


Figure 13-28. An AVGAS fueling nozzle with static bonding grounding wire.



Figure 13-29. This panel at the pressure refueling station has valve position switches and quantity gauges to be used during refueling. Valve open position lights are adjacent to the switches for each tank.

even one caused by static electricity, could be enough for ignition. Any potential for sparks must be nullified.

Spilled fuel poses an additional fire hazard. A thin layer of fuel vaporizes quickly. Small spills should be wiped up immediately. Larger spills can be flooded with water to dissipate the fuel and the potential for ignition. Do not sweep fuel that has spilled onto the ramp.

Class B fire extinguishers need to be charged and accessible nearby during the fueling and defueling processes. Fueling personnel must know exactly where they are and how to use them. In case of an emergency, the fuel truck, if used, may need to be quickly driven away from the area. For this reason alone, it should be positioned correctly on the ramp relative to the aircraft.

LONGITUDINAL BALANCE SYSTEMS

As large modern aircraft burn off fuel in flight, the center of gravity of the aircraft can change. This may cause the nose of the aircraft to pitch up or down. Use of elevator trim to compensate for the nose-down attitude is possible. However, when the amount of trim is increased, the amount of drag is also increased. The aircraft is designed to fly most efficiently without trim. Transferring fuel to maintain the desired aircraft center of gravity so that trim is not required for cruise flight is an option on some aircraft. Fuel tanks designated as trim tanks are used to carry the fuel that is transferred to maintain longitudinal balance.

To relieve the flight crew of the task of continually monitoring and transferring fuel to maintain a trim-free center of gravity, computerized center of gravity control systems are used. Fuel flow, fuel usage and tank volumes are monitored by the system computer so that valves and pumps are automatically operated to transfer fuel to the proper tank locations to maintain the desired CG. Pitch attitude and stabilizer position are also monitored by the CGCC (center of gravity control computer). The CG is generally given as a range of the percentage of the mean aerodynamic chord (MAC).

The flight crew initiates system operation by entering variables such as passenger and cargo loads before flight. Once airborne, as fuel is consumed, fuel management system and tank volume inputs are used by the CG control system to calculate and perform the fuel transfers that keep the CG within the desired range. Flight deck display of the aircraft weight and CG location allow crew surveillance of the automated transfers. Should the computerized CG control system fail, the system automatically switches to an alternate mode. It may also switch to a fault mode which automatically empties the trim tank.

Question: 13-1
Each aircraft fuel system must be arranged so that no fuel pump can draw fuel from more than _____ tank(s) at a time.

Question: 13-2
Name three types of fuel quantity indicator systems.

Question: 13-3
Name five fuel sub-systems found on jet transport aircraft.

Question: 13-4
Measuring _____ accurately is complicated by the fact that the fuel mass changes with temperature or with the type of fuel used in turbine engines.

Question: 13-5
On what type of aircraft might a bladder fuel tank be found?

Question: 13-6
To fuel w ith pressure refuel ing, a hose nozzle is attached at the fueling station and _____ to the tanks required to be filled are opened.

Question: 13-7
Fuel flowing through the fuel system generates _____ that must have a place to flow to ground rather than build up.

Question: 13-8
The two most common electric motor operated fuel valves are the _____ and the plug type valve.

ANSWERS

Answer: 13-1
One.

Answer: 13-2
Ratiometer-type.
Capacitance-type.
Mechanical type (drip stick).

Answer: 13-3
Storage.
Vent.
Distribution.
Supply.
Feed.
Indicating.
Warning.

Answer: 13-4
fuel flow.

Answer: 13-5
All sizes of aircraft.

Answer: 13-6
valves.

Answer: 13-7
static electricity

Answer: 13-8
gate valve.



PART-66 SYLLABUS **LEVELS**

CERTIFICATION CATEGORY → **B2**

Sub-Module 14

HYDRAULIC POWER (ATA 29)

Knowledge Requirements

13.14 - Hydraulic Power (ATA 29)

System lay-out;	1
Hydraulic fluids;	1
Hydraulic reservoirs and accumulators;	1
Pressure generation: electrical, mechanical, pneumatic;	3
Emergency pressure generation;	3
Filters;	1
Pressure control;	3
Power distribution;	1
Indication and warning systems;	3
Interface with other systems.	3

AIRCRAFT HYDRAULIC POWER

The word "hydraulics" is based on the Greek word for water and originally meant the study of the physical behavior of water at rest and in motion. Today, the meaning has been expanded to include the physical behavior of all liquids, including hydraulic fluid. Hydraulic systems are not new to aviation. Early aircraft had hydraulic brake systems. As aircraft became more sophisticated, newer systems with hydraulic power were developed.

Hydraulic systems in aircraft provide a means for the operation of aircraft components. The operation of landing gear, flaps, flight control surfaces, and brakes is largely accomplished with hydraulic power systems. Hydraulic system complexity varies from small aircraft that require fluid only for manual operation of the wheel brakes to large transport aircraft where complex systems power flight controls, landing gear retraction, nose wheel steering, brakes and more. To achieve the necessary redundancy and reliability, the system may consist of several subsystems. Each subsystem has a power generating device (pump), reservoir, accumulator, heat exchanger, filtering system, etc. System operating pressure may vary from a couple hundred pounds per square inch (psi) in small aircraft and rotorcraft to 5 000 psi in large transports.

Hydraulic systems have many advantages as power sources for operating various aircraft units; they combine the advantages of light weight, ease of installation, simplification of inspection, and minimum maintenance requirements. Hydraulic operations are also almost 100 percent efficient, with only negligible loss due to fluid friction.

SYSTEM LAYOUT

Regardless of its function and design, every hydraulic system has a minimum number of basic components in addition to a means through which the fluid is transmitted. A basic system consists of a pump, reservoir, directional valve, check valve, pressure relieve valve, selector valve, actuator, and filter. (***Figure 14-1***)

OPEN-CENTER HYDRAULIC SYSTEMS

An open-center system is one having fluid flow, but no pressure in the system when the actuating mechanisms are idle. The pump circulates the fluid from the reservoir, through the selector valves, and back to the reservoir. (***Figure 14-2A***)

The open-center system may employ any number of subsystems, with a selector valve for each subsystem. Unlike the closed center system, the selector valves of the open center system are always connected in series with each other. In this arrangement, the system pressure line goes through each selector valve. Fluid is always allowed free passage through each selector valve and back to the reservoir until one of the selector valves is positioned to operate a mechanism.

When one of the selector valves is positioned to operate an actuating device, fluid is directed from the pump through one of the working lines to the actuator. (***Figure 14-2B***) With the selector valve in this position, the flow of fluid through the valve to the reservoir is blocked. The pressure builds up in the system to overcome the resistance and moves the piston of the actuating cylinder; fluid from the opposite end of the actuator returns to the selector valve and flows back to the reservoir. Operation of the system following actuation of the component depends on the type of selector valve being used. Several types of selector valves are used in conjunction with the open center system.

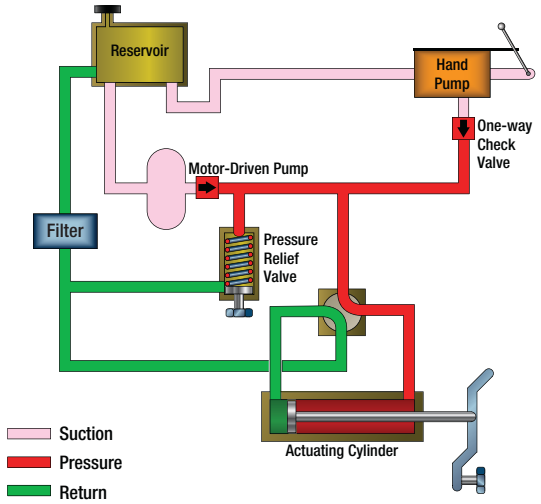


Figure 14-1. Basic hydraulic system.

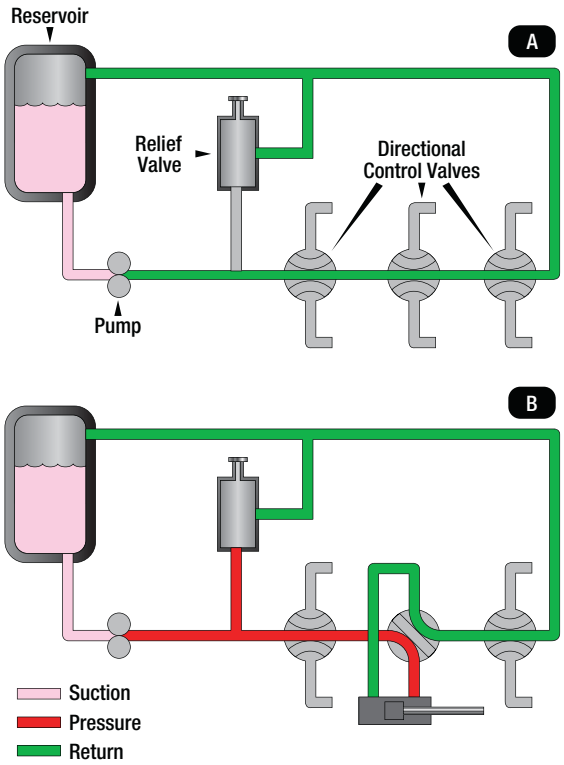


Figure 14-2. Open center hydraulic system.

CLOSED-CENTER HYDRAULIC SYSTEMS

In the closed-center system (***Figure 14-3***), the fluid is under pressure whenever the power pump is operating. The three actuators are arranged in parallel and actuating units B and C are operating at the same time, while actuating unit A is not operating. This system differs from the open-center system in that the selector or directional control valves are arranged in parallel and not in series. The means of controlling pump pressure varies in the closed-center system. If a constant delivery pump is used, the system pressure is regulated by a pressure regulator. A relief valve acts as a backup safety device in case the regulator fails. If a variable displacement pump is used, system pressure is controlled by the pump's integral pressure mechanism compensator. The compensator automatically varies the volume output. An advantage of the open-center system over the closed-center system is that the continuous pressurization of the system is eliminated. Closed-center systems are the most widely used.

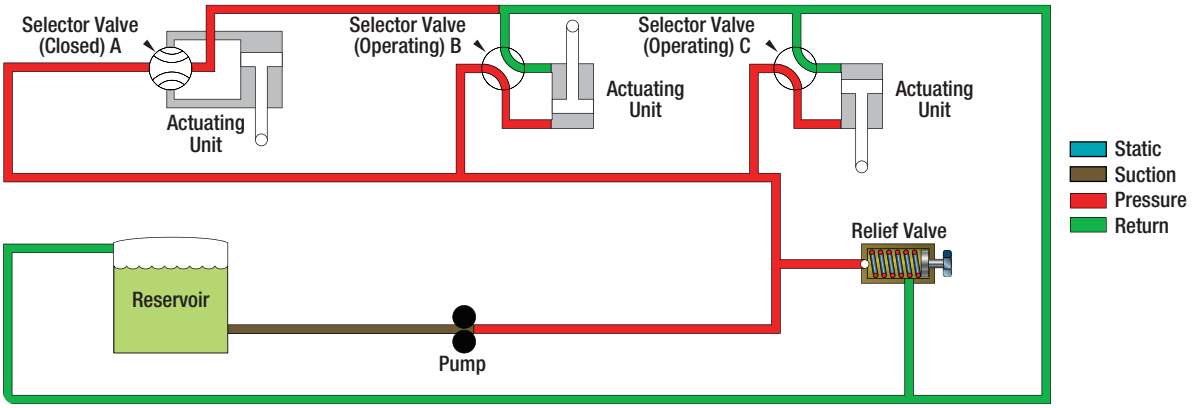


Figure 14-3. A basic closed-center hydraulic system with a variable displacement pump.

EVOLUTION OF HYDRAULIC SYSTEMS

Smaller aircraft have relatively low flight control surface loads, and the pilot can operate the flight controls by hand. Hydraulic systems were utilized for brake systems on early aircraft. When aircraft started to fly faster and got larger in size, the pilot was not able to move the control surfaces by hand anymore, and hydraulic power boost systems were introduced. Power boost systems assist the pilot in overcoming high control forces, but the pilot still actuates the flight controls by cable or push rod. Small power packs are the latest evolution of the simple hydraulic system. They reduce weight by eliminating hydraulic lines and large quantities of hydraulic fluid.

HYDRAULIC POWER PACK SYSTEM

A hydraulic power pack is a compact unit that consists of an electric pump, a reservoir, valves, filters, and pressure relief valve all in one assembly. (***Figure 14-4***)

The advantage of the power pack is that there is no need for a centralized hydraulic power supply system

and long stretches of hydraulic lines. This reduces weight. Power packs are driven by either an engine gearbox or electric motor.

These systems are used to control the stabilizer trim, landing gear, or flight control surfaces directly, thus eliminating the need for a centralized hydraulic system.

MODERN HIGH PERFORMANCE SYSTEMS

Many modern aircraft use a power supply system and fly-by-wire flight control. The pilot input is electronically sent to the flight control servos. The servos use hydraulic pressure to move the control surfaces. Cables or push rods are not used. Some manufacturers are reducing the use of hydraulic systems in their aircraft in favor of electrically controlled systems.

The Boeing 787 is the first aircraft designed with more electrical systems than hydraulic systems. Large aircraft hydraulic systems are complexed close-center systems with a wide variety of components and typically triple redundancy. ***Figure 14-5*** is a typical example of a hydraulic system in a large commercial aircraft. After a discussion of hydraulic fluid and various hydraulic system components, large aircraft hydraulic systems will be discussed later in this sub-module.



Figure 14-4. Hydraulic power pack.

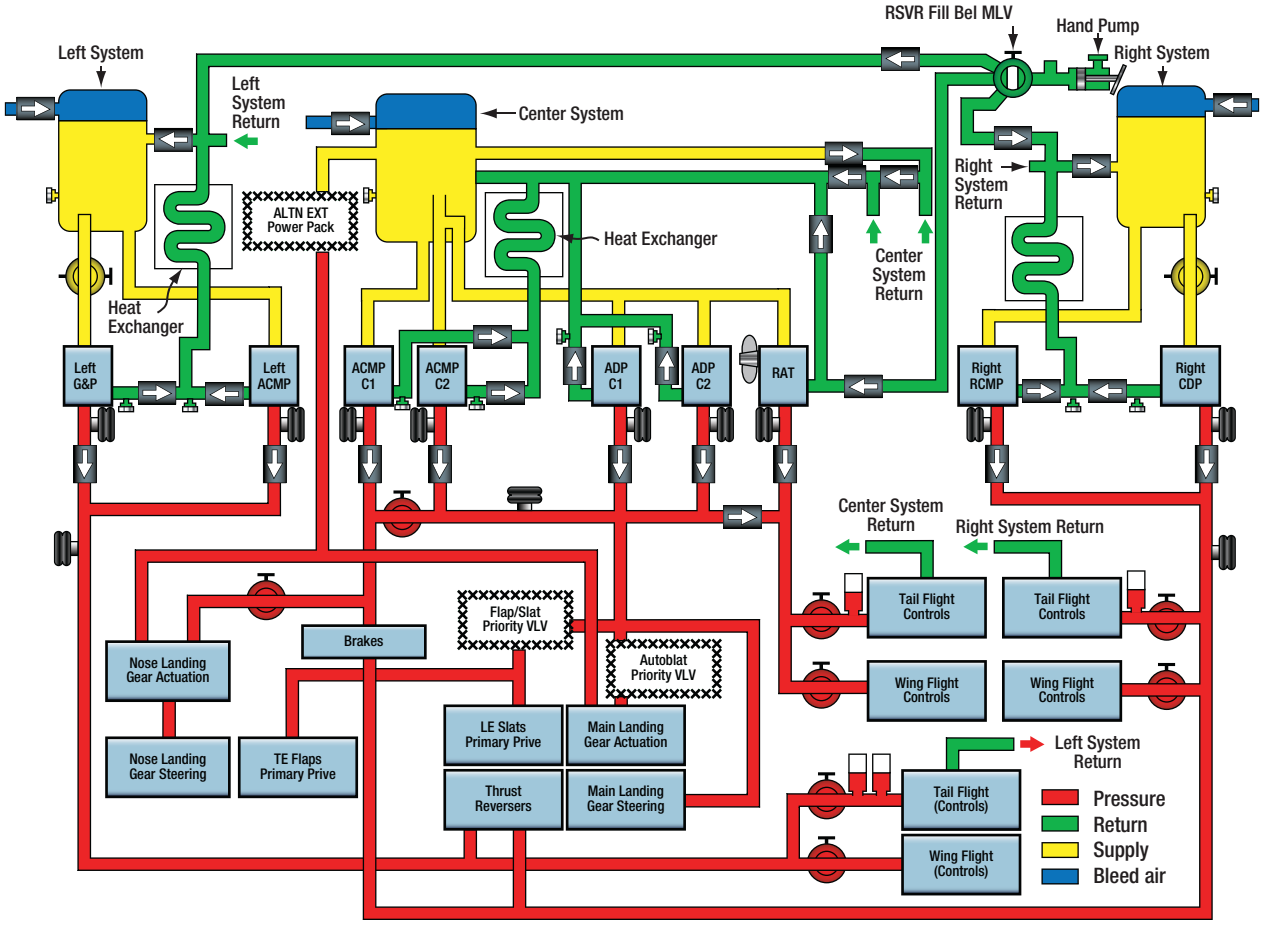


Figure 14-5. Large commercial aircraft hydraulic system.

HYDRAULIC FLUID

Hydraulic system liquids are used primarily to transmit and distribute forces to various units to be actuated. Liquids are able to do this because they are almost incompressible. Pascal's Law states that pressure applied to any part of a confined liquid is transmitted with undiminished intensity to every other part.

Thus, if a number of passages exist in a system, pressure can be distributed through all of them by means of the liquid. Manufacturers of hydraulic devices usually specify the type of liquid best suited for use with their equipment in view of the working conditions, the service required, temperatures expected inside and outside the systems, pressures the liquid must withstand, the possibilities of corrosion, and other conditions that must be considered. If incompressibility and fluidity were the only qualities required, any liquid that is not too thick could be used in a hydraulic system. But a satisfactory liquid for a particular installation must possess a number of other properties.

TYPES OF HYDRAULIC FLUIDS

To assure proper system operation and to avoid damage to nonmetallic components of the hydraulic system, the correct fluid must be used. When adding fluid to a system, use the type specified in the aircraft manufacturer's maintenance manual or on the instruction plate affixed to the reservoir or unit being serviced.

The three principal categories of hydraulic fluids are:

1. Minerals: Mineral oil-based hydraulic fluid (MIL-H-5606) is the oldest, dating back to the 1940s. It is used in many systems, especially where the fire hazard is comparatively low.
2. Polyalphaolefins: MIL-H-83282 is a fire-resistant hydrogenated polyalphaolefin-based fluid developed in the 1960s to overcome the flammability characteristics of MIL-H-5606. MIL-H-83282 is significantly more flame resistant than MIL-H-5606, but a disadvantage is the high viscosity at low temperature.
3. Phosphate esters: These fluids are used in most commercial transport category aircraft and are extremely fire-resistant. However, they are not fireproof and under certain conditions, they burn.

When servicing a hydraulic system, the technician must be certain to use the correct category of replacement fluid. Hydraulic fluids are not necessarily compatible. For example, contamination of the fire-resistant fluid MIL-H-83282 with MIL-H-5606 may render the MIL-H-83282 non fire-resistant.

HEALTH AND HANDLING

Skydrol® fluids are phosphate ester-based fluids blended with performance additives. Phosphate esters are good solvents and dissolve away some of the fatty materials of the skin. Repeated or prolonged exposure may cause drying of the skin, which if unattended, could result in complications, such as dermatitis or even secondary infection from bacteria. Skydrol® fluids could cause itching of the skin but have not been known to cause allergic type skin rashes. Always use the proper gloves and eye protection when handling any type of hydraulic fluid. When Skydrol®/Hyjet mist or vapor exposure is possible, a respirator capable of removing organic vapors and mists must be worn. Ingestion of any hydraulic fluid should be avoided. Although small amounts do not appear to be highly hazardous, any significant amount should be tested in accordance with manufacturer's direction, followed with hospital supervised stomach treatment.

HYDRAULIC RESERVOIRS AND ACCUMULATORS

RESERVOIRS

The reservoir is a tank in which an adequate supply of fluid for the system is stored. Fluid flows from the reservoir to the pump, where it is forced through the system and eventually returned to the reservoir. The reservoir not only supplies the operating needs of the system, but it also replenishes fluid lost through leakage. Furthermore, the reservoir serves as an overflow basin

for excess fluid forced out of the system by thermal expansion (the increase of fluid volume caused by temperature changes), the accumulators, and by piston and rod displacement. The reservoir also furnishes a place for the fluid to purge itself of air bubbles that may enter the system. Foreign matter picked up in the system may also be separated from the fluid in the reservoir or as it flows through line filters. Reservoirs are either pressurized or non-pressurized. Baffles and/or fins are incorporated in most reservoirs to keep the fluid within the reservoir from having random movement, such as vortexing (swirling) and surging. These conditions can cause fluid to foam and air to enter the pump along with the fluid. Many reservoirs incorporate strainers in the filler neck to prevent the entry of foreign matter during servicing.

NON-PRESSURIZED RESERVOIRS

Non-pressurized reservoirs are used in aircraft that are not designed for violent maneuvers, do not fly at high altitudes, or in which the reservoir is located in the pressurized area of the aircraft. High altitude in this situation means an altitude where atmospheric pressure is inadequate to maintain sufficient flow of fluid to the hydraulic pumps. Most non-pressurized reservoirs are constructed in a cylindrical shape. (***Figure 14-6***)

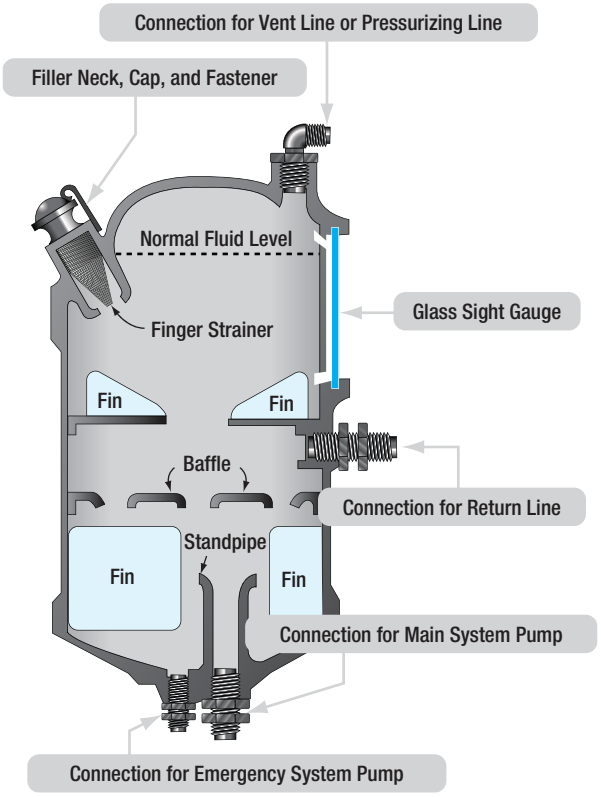


Figure 14-6. Nonpressurized reservoir.

PRESSURIZED RESERVOIRS

Reservoirs on aircraft designed for high-altitude flight are usually pressurized. Pressurizing assures a positive flow of fluid to the pump at high altitudes when low atmospheric pressures are encountered. On some aircraft, the reservoir is pressurized by bleed air taken from the compressor section of the engine. (***Figure 14-7***) On others, the reservoir may be pressurized by hydraulic system pressure.

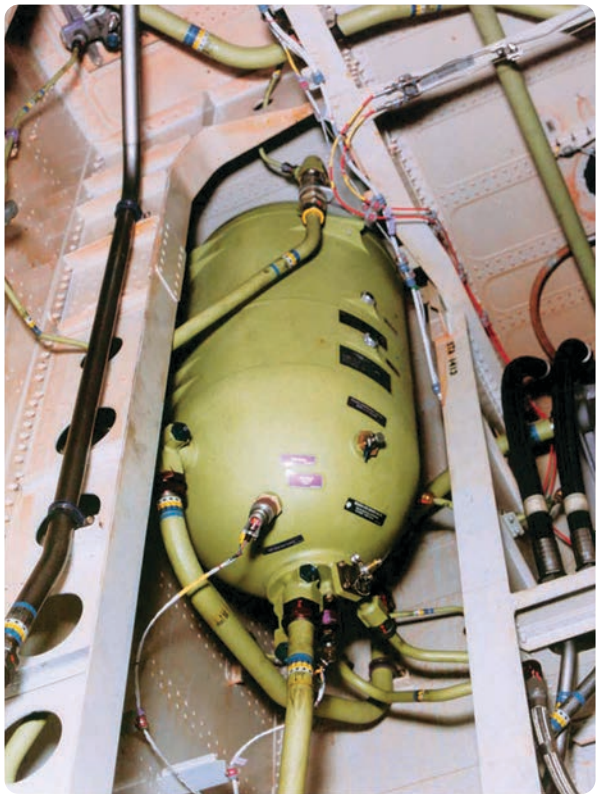


Figure 14-7. Air-pressurized reservoir.

ACCUMULATORS

The accumulator is a steel sphere divided into two chambers by a synthetic rubber diaphragm. The upper chamber contains fluid at system pressure, while the lower chamber is charged with nitrogen or air. Cylindrical types are also used in high-pressure hydraulic systems. Many aircraft have several accumulators in the hydraulic system. There may be a main system accumulator and an emergency system accumulator. There may also be auxiliary accumulators located in various sub-systems.

The functions of an accumulator are to:

1. Dampen pressure surges in the hydraulic system caused by actuation of a unit and the effort of the pump to maintain pressure at a preset level.
2. Aid or supplement the power pump when several units are operating at once by supplying extra power from its accumulated, or stored, power.
3. Store power for the limited operation of a hydraulic unit when the pump is not operating.
4. Supply fluid under pressure to compensate for small internal or external (not desired) leaks that would cause the system to cycle continuously by action of the pressure switches continually kicking in.

There are two general types of accumulators used in aircraft hydraulic systems:

1. Spherical: It is constructed in two halves that are fastened and threaded, or welded, together. Two threaded openings exist. The top port accepts fittings to connect to the pressurized hydraulic system to the accumulator. The bottom port is fitted with a gas servicing valve. A synthetic rubber diaphragm, or bladder, is installed in the sphere to create two chambers. Pressurized hydraulic fluid occupies the upper chamber and nitrogen or air charges the lower chamber. (***Figure 14-8***)
2. Cylindrical: consist of a cylinder and piston assembly. End caps are attached to both ends of the cylinder. The internal piston separates the fluid and air/nitrogen chambers. The end caps and piston are sealed with gaskets and packings to prevent external leakage around the end caps and internal leakage between the chambers. (***Figure 14-9***).

Maintenance consists of inspections, minor repairs, replacement of component parts, and testing. There is an element of danger in maintaining accumulators. Therefore, proper precautions must be strictly observed to prevent injury and damage.

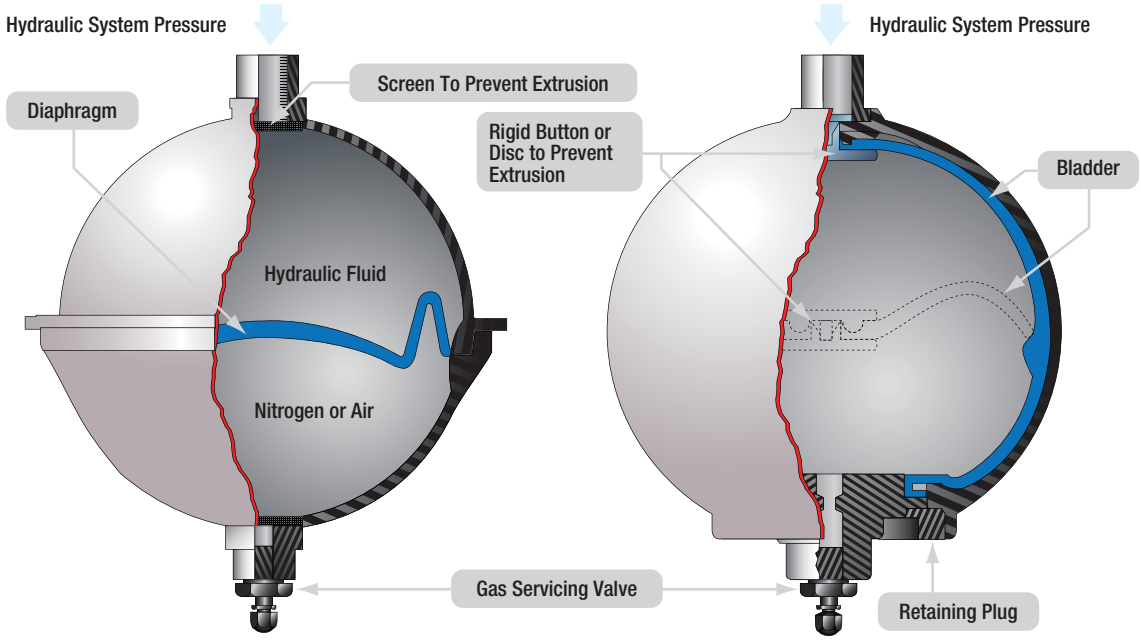


Figure 14-8. A spherical accumulator with diaphragm (left) and bladder (right). The dotted lines in the right drawing depict the bladder when the accumulator is charged with both hydraulic system fluid and nitrogen preload.

HYDRAULIC PRESSURE GENERATION

All aircraft hydraulic systems have at least one power driven pump and may include a hand pump as an additional unit when the power-driven pump is inoperative. The pump is the source of fluid flow, which when restricted, generates pressure in the hydraulic system. A hydraulic pump can be driven mechanically, electrically or with pneumatic air.



Figure 14-9. Cylindrical accumulator.

MECHANICAL, ELECTRICAL AND PNEUMATIC-DRIVEN PUMPS

Mechanically driven pumps are the primary source of pressure generation on most aircraft. Typically, the pump is mounted on the accessory gearbox of the main engine and is rotated by a shaft. When the engine is operating, the pump supplies ample fluid flow to generate pressure within the hydraulic system. Electrical motor driven pumps also exist.

Often, these are the same pumps as the mechanically driven pumps but the drive shaft is turned by an electrical motor. As such, electrically driven pumps can be mounted away from the engine(s). On large aircraft, they are usually mounted in the wheel well or in a hydraulics bay near the root of the wings. Electrically driven pumps are installed for use in emergencies or during ground operation when engines are not running.

Pneumatically driven pumps also exist on aircraft. Typically these pumps are used as demand pumps to supplement the primary pumps and are driven by air from the pneumatic system or by ram air in the case of a ram air turbine (RAT). Modern transport aircraft use a combination of engine-driven power pumps, electrically-driven power pumps, pneumatically-driven power pumps, power transfer units (PTU), and a RAT-driven pump in a highly effective, fully redundant aircraft hydraulic system. For example, the Airbus A380 has two hydraulic systems, eight engine driven pumps, and three electrical driven pumps. The Boeing 777 has three hydraulic systems with two engine driven pumps, four electrical driven pumps, two air driven pumps, and a hydraulic pump motor driven by the RAT. (***Figures 14-10 and Figure 14-11***)

Many of the power driven hydraulic pumps of current aircraft are of variable delivery, compensator-controlled type. Constant delivery pumps are also in use. Principles of operation are the same for both types of pumps and are described in further detail below. A discussion of various types of pumps and how they work begins with hand pumps.

HAND PUMPS

The hydraulic hand pump is used in some older aircraft for the operation of hydraulic subsystems and in a few newer aircraft systems as a backup unit. Hand pumps are generally installed for testing purposes, as well as for use in emergencies. Hand pumps are also installed to service the reservoirs from a single refilling station. The single refilling station reduces the chances for the introduction of fluid contamination.

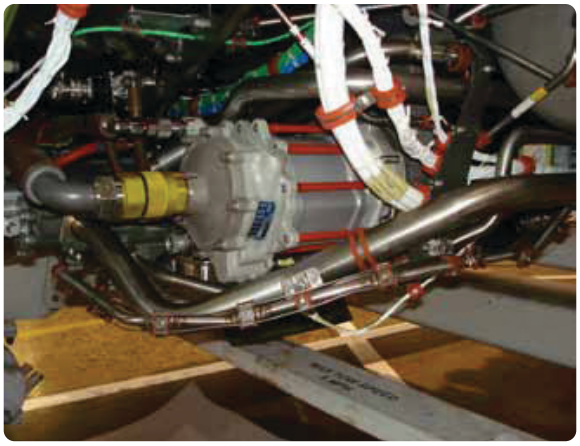


Figure 14-10. Engine driven pump.



Figure 14-11. Electrically driven pump.

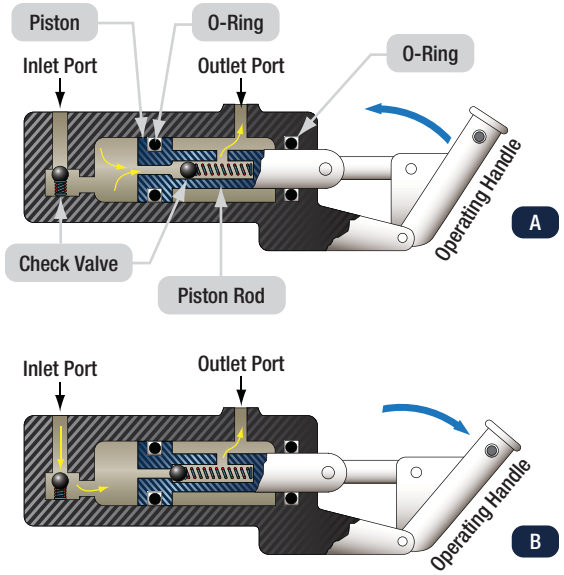


Figure 14-12. Double action hand pump.

Several types of hand pumps are used: single action, double action, and rotary. A single action hand pump draws fluid into the pump on one stroke and pumps that fluid out on the next stroke. It is rarely used in aircraft due to this inefficiency.

Double-action hand pumps produce fluid flow and pressure on each stroke of the handle. (***Figure 14-12***). The double action hand pump consists essentially of a housing that has a cylinder bore and two ports, a piston, two spring-loaded check valves, and an operating handle. An O-ring on the piston seals against leakage between the two chambers of the piston cylinder bore. An O-ring in a groove in the end of the pump housing seals against leakage between the piston rod and housing.

When the piston is moved to the right, the pressure in the chamber left of the piston is lowered. The inlet port ball check valve opens and hydraulic fluid is drawn into the chamber. At the same time, the rightward movement of the piston forces the piston ball check valve against its seat. Fluid in the chamber to the right of the piston is forced out of the outlet port into the hydraulic system. When the piston is moved to the left, the inlet port ball check valve seats. Pressure in the chamber left of the piston rises, forcing the piston ball check valve off of its seat. Fluid flows from the left chamber through the piston to the right chamber. The volume in the chamber right of the piston is smaller than that of the left chamber due to the displacement created by the piston rod. As the fluid from the left chamber flows into the smaller right chamber, the excess volume of fluid is forced out of the outlet port to the hydraulic system.

A rotary hand pump may also be employed. It produces continuous output while the handle is in motion. ***Figure 14-13*** shows a rotary hand pump in a hydraulic system.

CLASSIFICATION OF PUMPS

All pumps may be classified as either positive displacement or non-positive displacement. Most pumps used in hydraulic systems are positive displacement. A non-positive displacement pump produces a continuous flow. However, because it does not provide a positive internal seal against slippage, its output varies considerably as pressure varies. Centrifugal and propeller pumps are examples of non-positive-displacement pumps. If the



Figure 14-13. Rotary hand pump.

output port of a non-positive-displacement pump was blocked off, the pressure would rise and output would decrease to zero. Although the pumping element would continue moving, flow would stop because of slippage inside the pump. In a positive-displacement pump, slippage is negligible compared to the pumpís volumetric output flow. If the output port were plugged, pressure would increase instantaneously to the point that the pump pressure relief valve opens.

CONSTANT-DISPLACEMENT PUMPS

A constant-displacement pump, regardless of pump rotations per minute, forces a fixed or unvaryingquantity of fluid through the outlet port during each revolution of the pump. Constant-displacement pumps are sometimes called constant-volume or constant delivery pumps. They deliver a fixed quantity of fluid per revolution, regardless of the pressure demands. Since the constant-delivery pump provides a fixed quantity of fluid during each revolution of the pump, the quantity of fluid delivered per minute depends upon pump rotations per minute. When a constant displacement pump is used in a hydraulic system in which the pressure must be kept at a constant value, a pressure regulator is required.

GEAR-TYPE POWER PUMP

A gear-type power pump is a constant-displacement pump. It consists of two meshed gears that revolve in

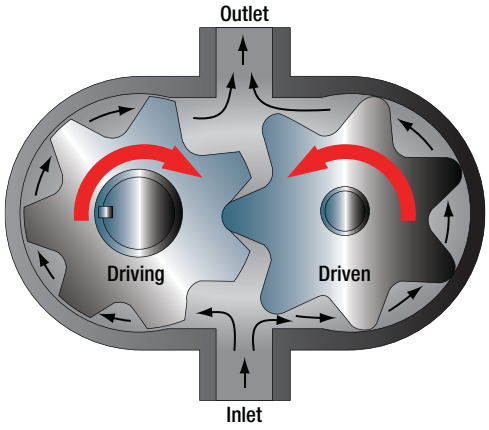


Figure 14-14. Gear-type power pump.

a housing. (***Figure 14-14***) The driving gear is driven by the aircraft engine or some other power unit. The driven gear meshes with, and is driven by, the driving gear. Clearance between the teeth as they mesh and between the teeth and the housing is very small. The inlet port of the pump is connected to the reservoir, and the outlet port is connected to the pressure line. When the driving gear turns, as shown in ***Figure 14-14***, it turns the driven gear. Fluid is captured by the teeth as they pass the inlet, and it travels around the housing and exits at the outlet.

GEROTOR PUMP

A gerotor-type power pump consists essentially of a housing containing an eccentric-shaped stationary liner, an internal gear rotor having seven wide teeth of short height, a spur driving gear having six narrow teeth, and a pump cover that contains two crescent-shaped openings. (***Figure 14-15***) One opening extends into an inlet port and the other extends into an outlet port. During the operation of the pump, the gears turn clockwise together. As the pockets between the gears on the left side of the pump move from a lowermost position toward a topmost position, the pockets increase in size, resulting in the production of a partial vacuum within these pockets. Since the pockets enlarge while over the inlet port crescent, fluid is drawn into them. As these same pockets (now full of fluid) rotate over to the right side of the pump, moving from the topmost position toward the lowermost position, they decrease in size. This results in the fluid being expelled from the pockets through the outlet port crescent.

PISTON PUMPS

Piston pumps can be constant-displacement or variable displacement pumps. The common features of design

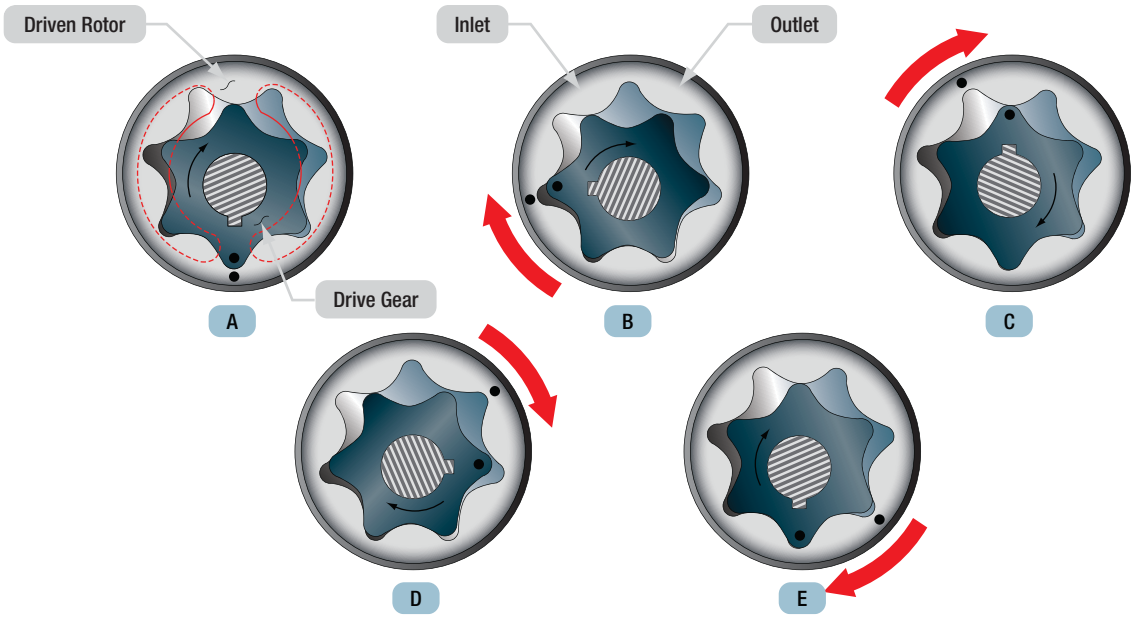


Figure 14-15. Gerotor pump.

and operation that are applicable to all piston-type hydraulic pumps are described in the following paragraphs. Piston-type power-driven pumps have flanged mounting bases for the purpose of mounting the pumps on the accessory drive cases of aircraft engines. A pump drive shaft, which turns the mechanism, extends through the pump housing slightly beyond the mounting base. Torque from the driving unit is transmitted to the pump drive shaft by a drive coupling. The drive coupling is a short shaft with a set of male splines on both ends. The splines on one end engage with female splines in a driving gear; the splines on the other end engage with female splines in the pump drive shaft. Pump drive couplings are designed to serve as safety devices. The shear section of the drive coupling, located midway between the two sets of splines, is smaller in diameter than the splines.

If the pump becomes unusually hard to turn or becomes jammed, this section shears, preventing damage to the pump or driving unit. (***Figure 14-16***)

The basic pumping cylinder block, a piston for each bore, and a valve plate with inlet and outlet slots. The purpose of the valve plate slots is to let fluid into and out of the bores as the pump operates. The cylinder

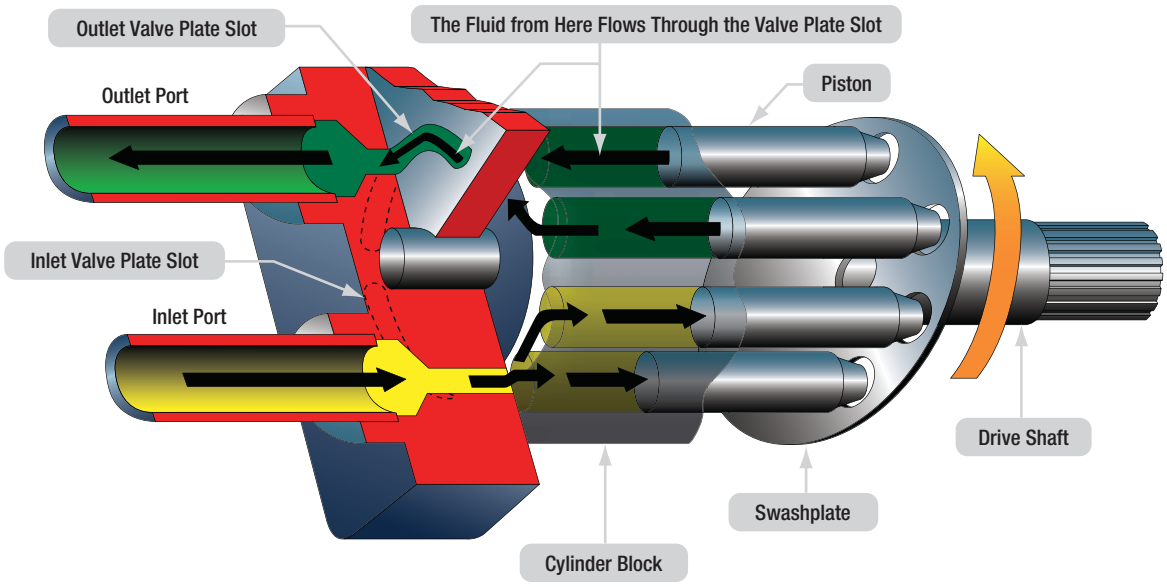


Figure 14-16. Axial inline piston pump.

bores lie parallel to and symmetrically around the pump axis. All aircraft axial-piston pumps have an odd number of pistons. (***Figure 14-17***)

BENT AXIS PISTON PUMP

A typical constant-displacement axial-type pump is shown in ***Figure 14-18***. The angular housing of the pump causes a corresponding angle to exist between the cylinder block and the drive shaft plate to which the pistons are attached. It is this angular configuration of the pump that causes the pistons to stroke as the pump shaft is turned.

When the pump operates, all parts within the pump (except the outer races of the bearings that support the drive shaft, the cylinder bearing pin on which the cylinder block turns, and the oil seal) turn together as a rotating group. At one point of rotation of the rotating group, a minimum distance exists between the top of the cylinder block and the upper face of the drive shaft plate. Because of the angled housing at a point of rotation 180¬ away, the distance between the top of the cylinder block and the upper face of the drive shaft plate is at a maximum. At any given moment of operation, three of the pistons are moving away from the top face of the cylinder block, producing a partial vacuum in the bores in which these pistons operate.

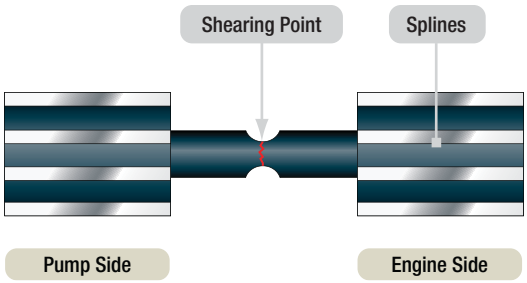


Figure 14-17. Hydraulic pump shear shaft.

This occurs over the inlet port, so fluid is drawn into these bores at this time. On the opposite side of the cylinder block, three different pistons are moving toward the top face of the block. This occurs while the rotating group is passing over the outlet port causing fluid to be expelled from the pump by these pistons. The continuous and rapid action of the pistons is overlapping in nature and results in a practically non-pulsating pump output.

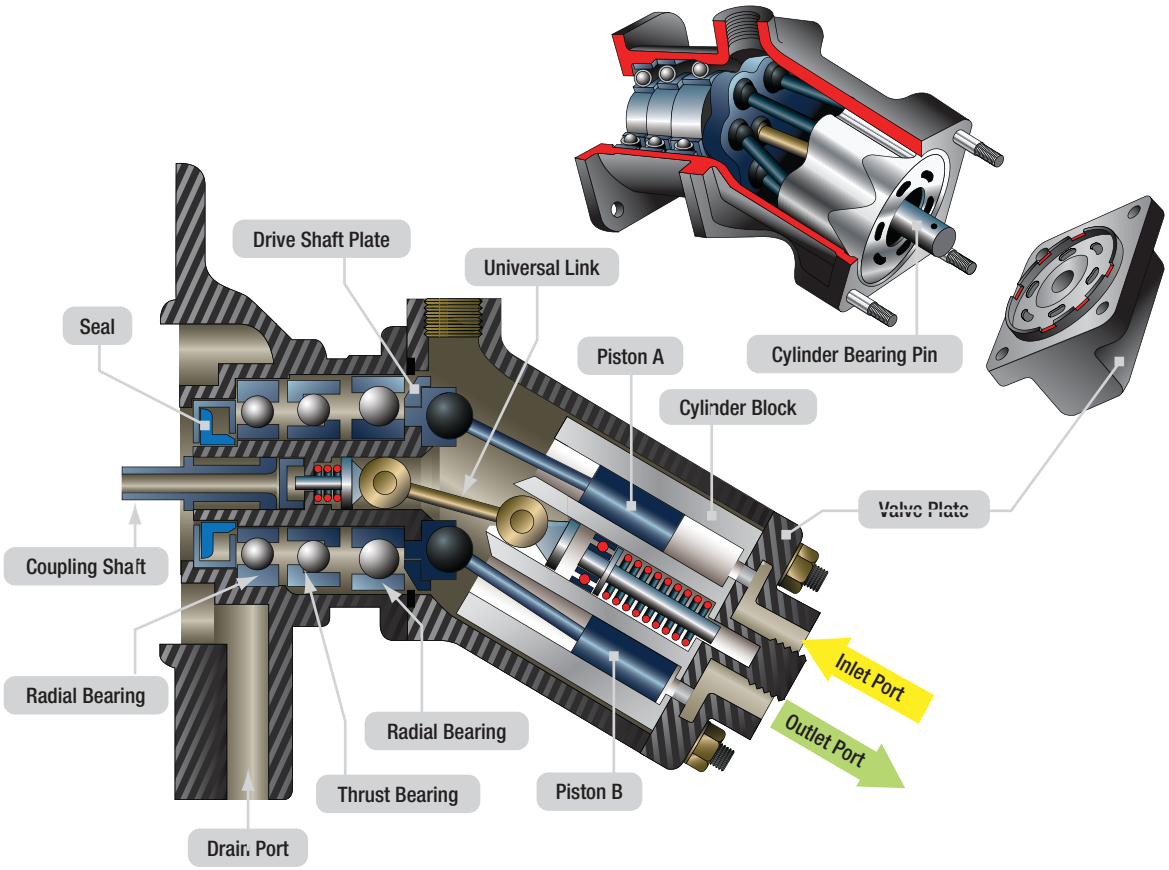


Figure 14-18. Bent axis piston pump.

INLINE PISTON PUMP

The simplest type of axial piston pump is the swash plate design in which a cylinder block is turned by the drive shaft. Pistons fitted to bores in the cylinder block are connected through piston shoes and a retracting ring so that the shoes bear against an angled swash plate. As the block turns, the piston shoes follow the swash plate, causing the pistons to reciprocate. The ports are arranged in the valve plate so that the pistons pass the inlet as they are pulled out, and pass the outlet as they are forced back in. In these pumps, displacement is determined by the size and number of pistons, as well as their stroke length, which varies with the swash plate angle. This constant-displacement pump is illustrated in ***Figure 14-17***.

VANE PUMP

The vane type power pump is also a constant-displacement pump. It consists of a housing containing four vanes (blades), a hollow steel rotor with slots for the vanes, and a coupling t to turn the rotor. (***Figure 14-19***)

The rotor is positioned off center within the sleeve. The vanes, which are mounted in the slots in the rotor, together with the rotor, divide the bore of the sleeve into four sections. As the rotor turns, each section passes one point where its volume is at a minimum and another point where its volume is at a maximum. The volume gradually increases from minimum to maximum during the first half of a revolution and gradually decreases from maximum to minimum during the second half of the revolution. As the volume of a given section increases, that section is connected to the pump inlet port through a slot in the sleeve. Since a partial vacuum is produced by the increase in volume of the section, fluid is drawn into the section through the pump inlet port and the slot in the sleeve. As the rotor turns through the second half of the revolution and the volume of the given section is decreasing, fluid is displaced out of the section, through the slot in the sleeve aligned with the outlet port, and out of the pump.

VARIABLE-DISPLACEMENT PUMPS

A variable-displacement pump has a fluid output that is varied to meet the pressure demands of the system. The pump output is changed automatically by a pump compensator within the pump. The following paragraph discusses a two-stage Vickers variable displacement pump. The first stage of the pump consists

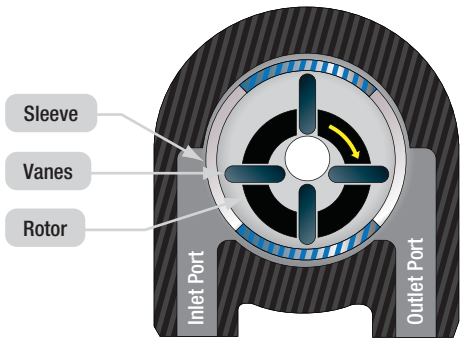


Figure 14-19. Vane-type power pump.

of a centrifugal pump that boosts the pressure before the fluid enters the piston pump. (***Figure 14-20***)

BASIC PUMPING OPERATION

The aircraft's engine rotates the pump drive shaft, cylinder block, and pistons via a gearbox. Pumping action is generated by piston shoes that are restrained and slide on the shoe bearing plate in the yoke assembly. Because the yoke is at an angle to the drive shaft, the rotary motion of the shaft is converted to piston reciprocating motion. As the piston begins to withdraw from the cylinder block, system inlet pressure forces fluid through a porting arrangement in the valve plate into the cylinder bore. The piston shoes are restrained in the yoke by a piston shoe retaining plate and a shoe plate during the intake stroke. As the drive shaft continues to turn the cylinder block, the piston shoe continues following the yoke bearing surface. This begins to return the piston into its bore (i.e., toward the valve block).

The fluid contained in the bore is pre-compressed, then expelled through the outlet port. Discharge pressure holds the piston shoe against the yoke bearing surface during the discharge stroke and also provides the shoe pressure balance and fluid film through an orifice in the piston and shoe subassembly. With each revolution of the drive shaft and cylinder block, each piston goes through the pumping cycle described above, completing one intake and one discharge stroke. High-pressure fluid is ported out through the valve plate, past the blocking valve, to the pump outlet. The blocking valve is designed to remain open during normal pump operation. Internal leakage keeps the pump housing filled with fluid for lubrication of rotating parts and cooling. The leakage is returned to the system through a case drain port. The case valve relief valve protects the pump against excessive case pressure, relieving it to the pump inlet.

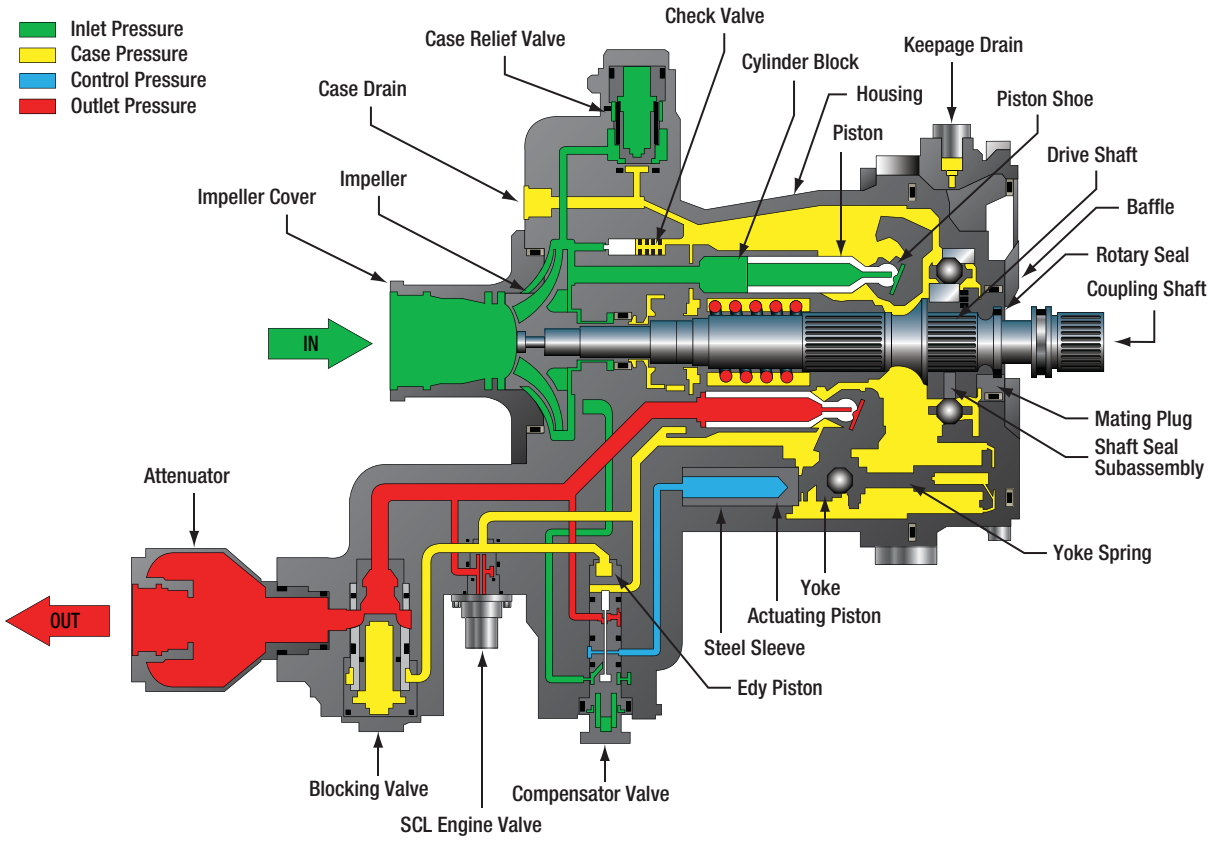


Figure 14-20. Variable displacement pump.

NORMAL PUMPING MODE

The pressure compensator is a spool valve that is held in the closed position by an adjustable spring load. (***Figure 14-21***) When pump outlet pressure (system pressure) exceeds the pressure setting (2 850 psi for full flow), the spool moves to admit fluid from the pump outlet against the yoke actuator piston. In ***Figure 14-21***, the pressure compensator is shown at cracking pressure; the pump outlet pressure is just high enough to move the spool to begin admitting fluid to the actuator piston.

The yoke is supported inside the pump housing on two bearings. At pump outlet pressures below 2 850 psi, the yoke is held at its maximum angle relative to the drive shaft centerline by the force of the yoke return spring. Decreasing system flow demand causes outlet pressure to become high enough to crack the compensator valve open and admit fluid to the actuator piston.

This control pressure overcomes the yoke return spring force and strokes the pump yoke to a reduced angle. The reduced angle of the yoke results in a shorter stroke for the pistons and reduced displacement. (***Figure 14-22***)

The lower displacement results in a corresponding reduction in pump flow. The pump delivers only that flow required to maintain the desired pressure in the system. When there is no demand for flow from the system, the yoke angle decreases to nearly zero degrees stroke angle. In this mode, the unit pumps only its own internal leakage. Thus, at pump outlet pressures above 2 850 psi, pump displacement decreases as outlet pressure rises. At system pressures below this level, no fluid is admitted through the pressure compensator valve to the actuator piston and the pump remains at full displacement, delivering full flow. Pressure is then determined by the system demand. The unit maintains zero flow at system pressure of 3 025 psi.

DEPRESSURIZED MODE

When the solenoid valve is energized, the EDV solenoid valve moves up against the spring force and the outlet fluid is ported to the EDV control piston on the top of the compensator (depressurizing piston). (***Figure 14-23***) The high-pressure fluid pushes the compensator spool beyond its normal metering position. This removes the compensator valve from the circuit and connects the actuator piston directly to the pump outlet. Outlet fluid

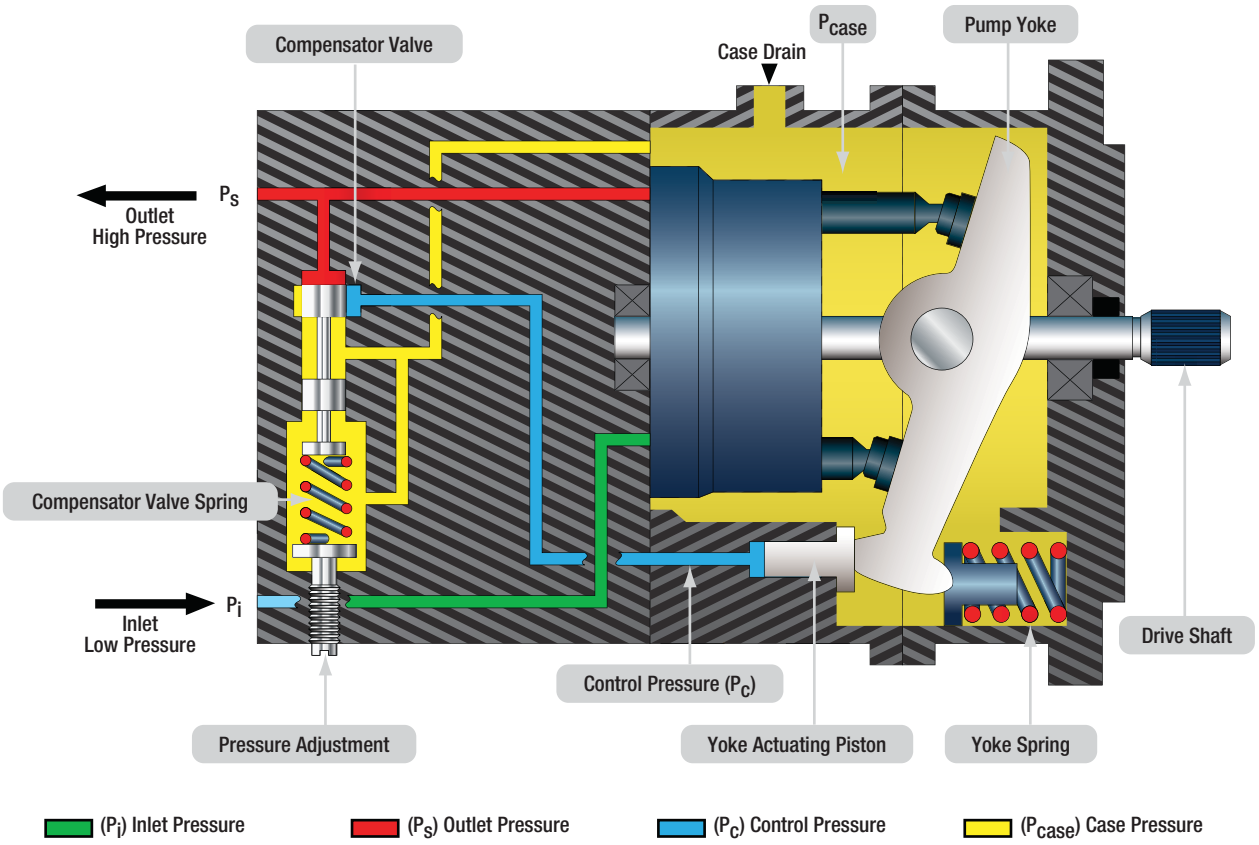


Figure 14-21. Normal pumping mode.

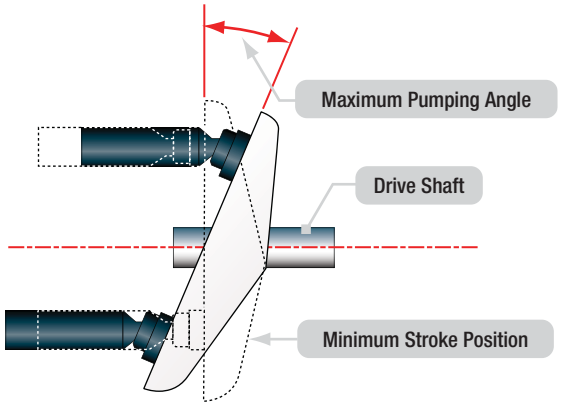


Figure 14-22. Yoke angle.

is also ported to the blocking valve spring chamber, which equalizes pressure on both sides of its plunger.

The blocking valve closes due to the force of the blocking valve spring and isolates the pump from the external hydraulic system. The pump strokes itself to zero delivery at an outlet pressure that is equal to the pressure required on the actuator piston to reduce the yoke angle to nearly zero, approximately 1 100 psi.

This depressurization and blocking feature can be used to reduce the load on the engine during startup and, in a multiple pump system, to isolate one pump at a time and check for proper system pressure output.

EMERGENCY PRESSURE GENERATION

Generation of hydraulic pressure in emergency situations varies. Large aircraft with multiple hydraulic systems are designed to ensure hydraulic pressure to critical components even in the event of a complete system failure or loss of engines. Generally, electrically-driven pumps are used when engine driven pumps fail. Should there be no working engines, not only would the engine driven hydraulic pump be inoperative but electrical generator output would cease as well. This may leave only the aircraft batteries to provide electrical power to the electrically driven pumps. However, it is common for large transport aircraft to be fitted with a ram air turbine (RAT) for yet again an additional source of hydraulic and electric power.

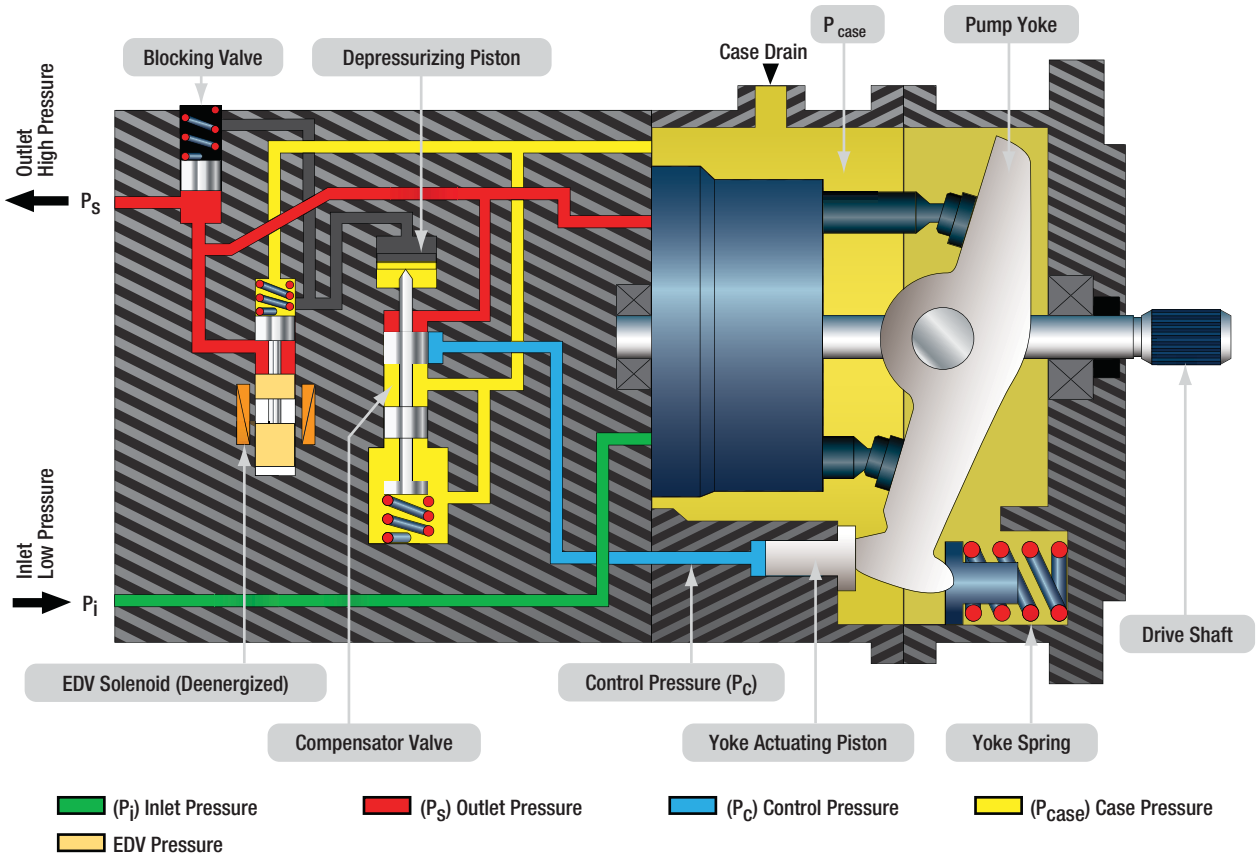


Figure 14-23. Depressurized mode.

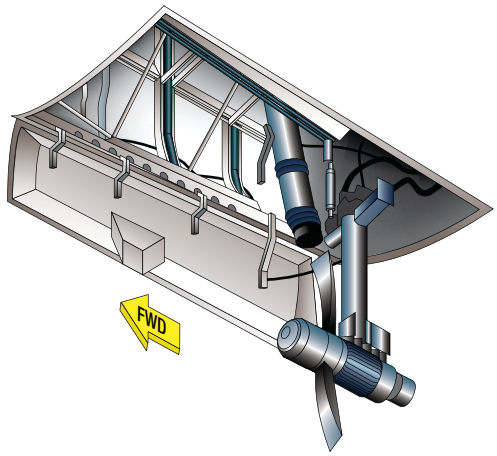


Figure 14-24. Ram air turbine.

RAM AIR TURBINE (RAT)

A RAT is installed in the aircraft to provide electrical and hydraulic power if the primary sources of aircraft power are lost. Ram air is used to turn the blades of a turbine that, in turn, operates a hydraulic pump and generator. The turbine and pump assembly is generally installed on the inner surface of a door installed in the fuselage. The door is hinged, allowing the assembly to be extended into the slipstream by pulling a manual release in the flight deck. In some aircraft, the RAT automatically deploys when the main hydraulic pressure system fails and/or electrical system malfunction occurs. (***Figure 14-24***)

HYDRAULIC MOTORS

Just as a rotating shaft drives a hydraulic pump workings to move fluid, fluid forced through the pump can rotate the attached shaft. This is the principle behind a hydraulic motor. Hydraulic fluid forced through the pump rotates the shaft of the pump, which as a result, makes the pump a motor. The motion of the shaft is then used to drive something to which it is attached. Piston-type motors are the most commonly used in hydraulic systems. (***Figure 14-25***)

They are basically the same as hydraulic pumps except they are used to convert hydraulic energy into mechanical (rotary) energy. Hydraulic motors are either of the axial inline or bent axis type. The most commonly used hydraulic motor is the fixed displacement bent axis type. These types of motors are used for the activation

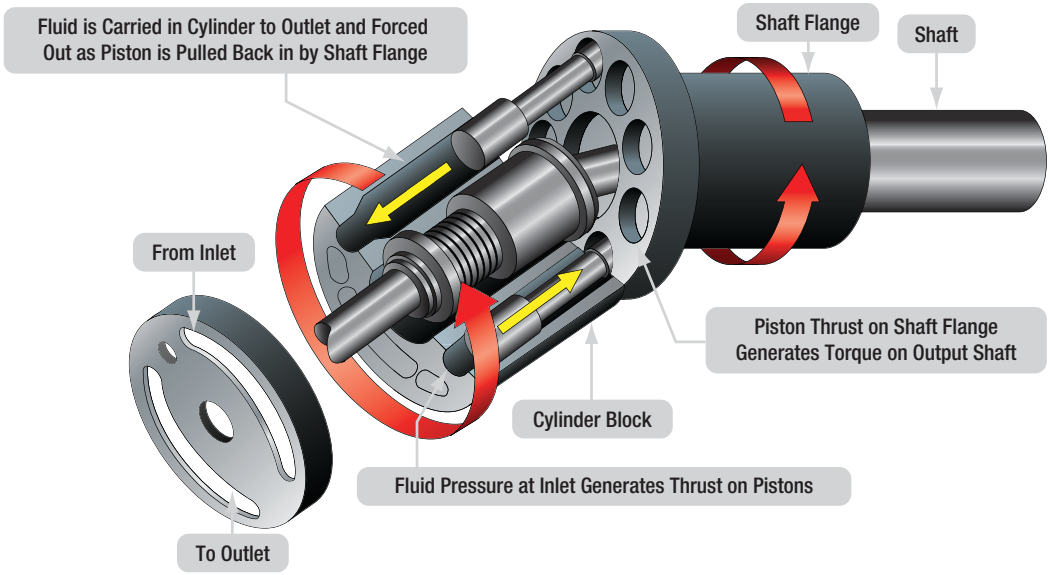


Figure 14-25. Bent axis pistion motor.

of trailing edge flaps, leading edge slats, and stabilizer trim. Some equipment uses a variable-displacement piston motor where very wide speed ranges are desired.

Although some piston-type hydraulic motors are controlled by directional control valves, they are often used in combination with variable displacement pumps. This pump-motor combination is used to provide a transfer of power between a driving element and a driven element. Some applications for which hydraulic transmissions may be used are speed reducers, variable speed drives, constant speed or constant torque drives, and torque converters. Some advantages of hydraulic transmission of power over mechanical transmission of power are as follows:

- Quick, easy speed adjustment over a wide range while the power source is operating at a constant (most efficient) speed
- Rapid, smooth acceleration or deceleration
- Control over maximum torque and power
- Cushioning effect to reduce shock loads moother reversal of motion

POWER TRANSFER UNITS (PTUS)

Hydraulic motors are also used in power transfer units (PTUs). In a PTU, two units, a hydraulic pump and hydraulic motor, are connected via a single drive shaft so that power can be transferred between two hydraulic systems. Depending on the direction of power transfer, each unit works as either a motor or a pump. The pressurized hydraulic system forces fluid through the motor which turns the shaft of the pump that moves fluid through the second hydraulic system. Thus, power is transferred from one system to the other. While the PTU transfers power, it does not transfer any fluid from one system to the other. (***Figure 14-26***)

Different types of PTUs are in use; some can only transfer power in one direction while others can transfer power both ways. Some PTUs have a fixed displacement, while others use a variable displacement hydraulic pump. Regardless, the application of PTU's in aircraft allow component operation in a hydraulic system in which the pump has failed. The system with a working pump drives the motor of the PTU so that the pump shaft rotates in the system with the failed pump. Activation can be manual or automatic depending on the aircraft. In an automatically activated setup, a pressure switch is used to detect pump failure which open a valve to divert fluid from the healthy system to the PTU.

HYDRAULIC MOTOR DRIVEN GENERATORS (HMDGS)

In case of an electrical failure, a hydraulic motor driven generator can be employed. An HMDG provides an alternative source of electrical power. The servo controlled variable displacement HMDG is an AC generator driven by the hydraulic motor portion of the unit. The generator part is typically designed to maintain the desired system output frequency of 400 Hz. Thus, an aircraft with an HMDG can maintain electrical power should a primary generator fail through the use of the hydraulic system. Conversely, a hydraulic pump failure is backed up by an electrically driven hydraulic pump.

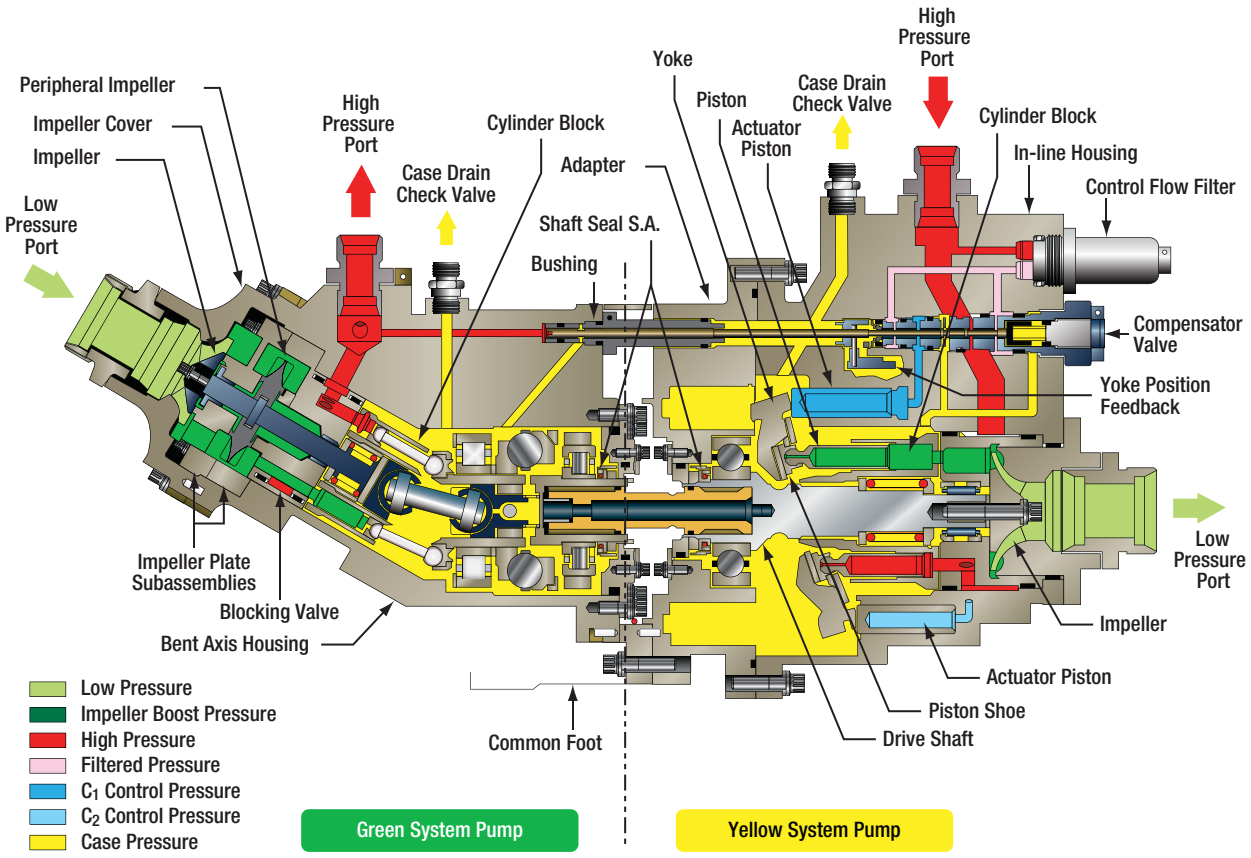


Figure 14-26. Power transfer unit.

PRESSURE CONTROL

The safe and efficient operation of fluid power systems, system components, and related equipment requires a means of controlling pressure. There are many types of automatic pressure control valves designed for this purpose. Some of them are an escape for pressure that exceeds a set pressure; some only reduce the pressure to a lower pressure system or subsystem; and some keep the pressure in a system within a required range.

RELIEF VALVES

Hydraulic pressure must be regulated in order to use it to perform the desired tasks. A pressure relief valve is used to limit the amount of pressure being exerted on a confined liquid. This is necessary to prevent failure of components or rupture of hydraulic lines under excessive pressures. The pressure relief valve is, in effect, a system safety valve. The design of pressure relief valves incorporates adjustable spring loaded valves. They are installed in such a manner as to discharge fluid from the pressure line into a reservoir return line when the pressure exceeds the predetermined maximum for which the valve is adjusted.

Various makes and designs of pressure relief valves are in use, but, in general, they all employ a spring-loaded valving device operated by hydraulic pressure and spring tension. (***Figure 14-27***)

Pressure relief valves are adjusted by increasing or decreasing the tension on the spring to determine the pressure required to open the valve. They may be classified by type of construction or uses in the system. The most common types of valve are:

1. Ball type - in pressure relief valves with a ball-type valving device, the ball rests on a contoured seat. Pressure acting on the bottom of the ball pushes it off its seat, allowing the fluid to bypass.
2. Sleeve type - in pressure relief valves with a sleeve type valving device, the ball remains stationary and a sleeve-type seat is moved up by the fluid pressure. This allows the fluid to bypass between the ball and the sliding sleeve-type seat.
3. Poppet type - in pressure relief valves with a poppet type valving device, a cone-shaped poppet may have any of several design configurations;

however, it is basically a cone and seat machined at matched angles to prevent leakage. As the pressure rises to its predetermined setting, the poppet is lifted off its seat, as in the ball-type device. This allows the fluid to pass through the opening created and out the return port.

Pressure relief valves cannot be used as pressure regulators in large hydraulic systems that depend on engine driven pumps for the primary source of pressure because the pump is constantly under load and the energy expended in holding the pressure relief valve off its seat is changed into heat. This heat is transferred to the fluid and, in turn, to the packing rings, causing them to deteriorate rapidly. Pressure relief valves, however, may be used as pressure regulators in small, low-pressure systems or when the pump is electrically driven and is used intermittently.

Pressure relief valves may be used as:

1. System Relief Valve - the most common use of the pressure relief valve is as a safety device against the possible failure of a pump compensator or other pressure regulating device. All hydraulic systems that have hydraulic pumps incorporate pressure relief valves as safety devices.
2. Thermal Relief Valve - the pressure relief valve is used to relieve excessive pressures that may exist due to thermal expansion of the fluid. They are used where a check valve or selector valve prevents pressure from being relieved through the main system relief valve. Thermal relief valves are usually smaller than system relief valves. As pressurized fluid in the line in which it is installed builds to an excessive amount, the valve poppet is forced off its seat. This allows excessive pressurized fluid to flow through the relief valve to the reservoir return line. When system pressure decreases to a predetermined pressure, spring tension overcomes system pressure and forces the valve poppet to the closed position.

PRESSURE REGULATORS

The term pressure regulator is applied to a device used in hydraulic systems that are pressurized by constant-delivery pumps. One purpose of the pressure regulator is to manage the output of the pump to maintain system operating pressure within a predetermined range. The other purpose is to permit the pump to turn without resistance (termed unloading the pump) at times when

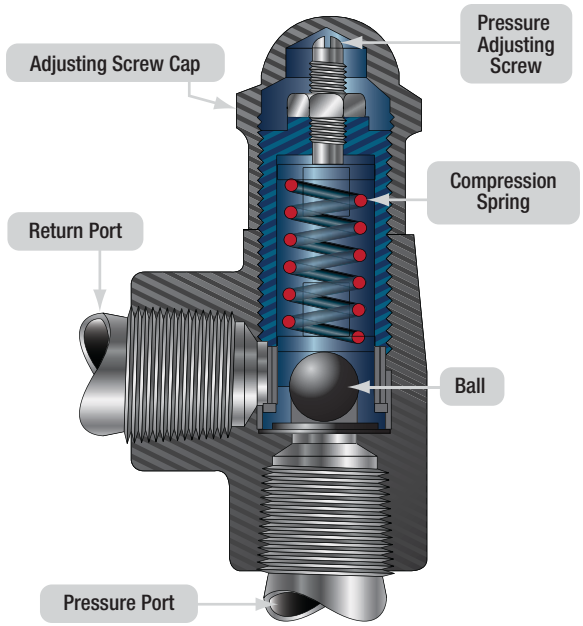


Figure 14-27. Pressure relief valve.

pressure in the system is within normal operating range. The pressure regulator is located in the system so that pump output can get into the system pressure circuit only by passing through the regulator. The combination of a constant-delivery pump and the pressure regulator is virtually the equivalent of a compensator controlled, variable-delivery-type pump. (***Figure 14-28***)

PRESSURE REDUCERS

Pressure reducing valves are used in hydraulic systems where it is necessary to lower the normal system operating pressure by a specified amount. Pressure reducing valves provide a steady pressure into a system that operates at a lower pressure than the supply system. A reducing valve can normally be set for any desired downstream pressure within the design limits of the valve. Once the valve is set, the reduced pressure is maintained regardless of changes in supply pressure (as long as the supply pressure is at least as high as the reduced pressure desired) and regardless of the system load, if the load does not exceed the designed capacity of the reducer. (***Figure 14-29***)

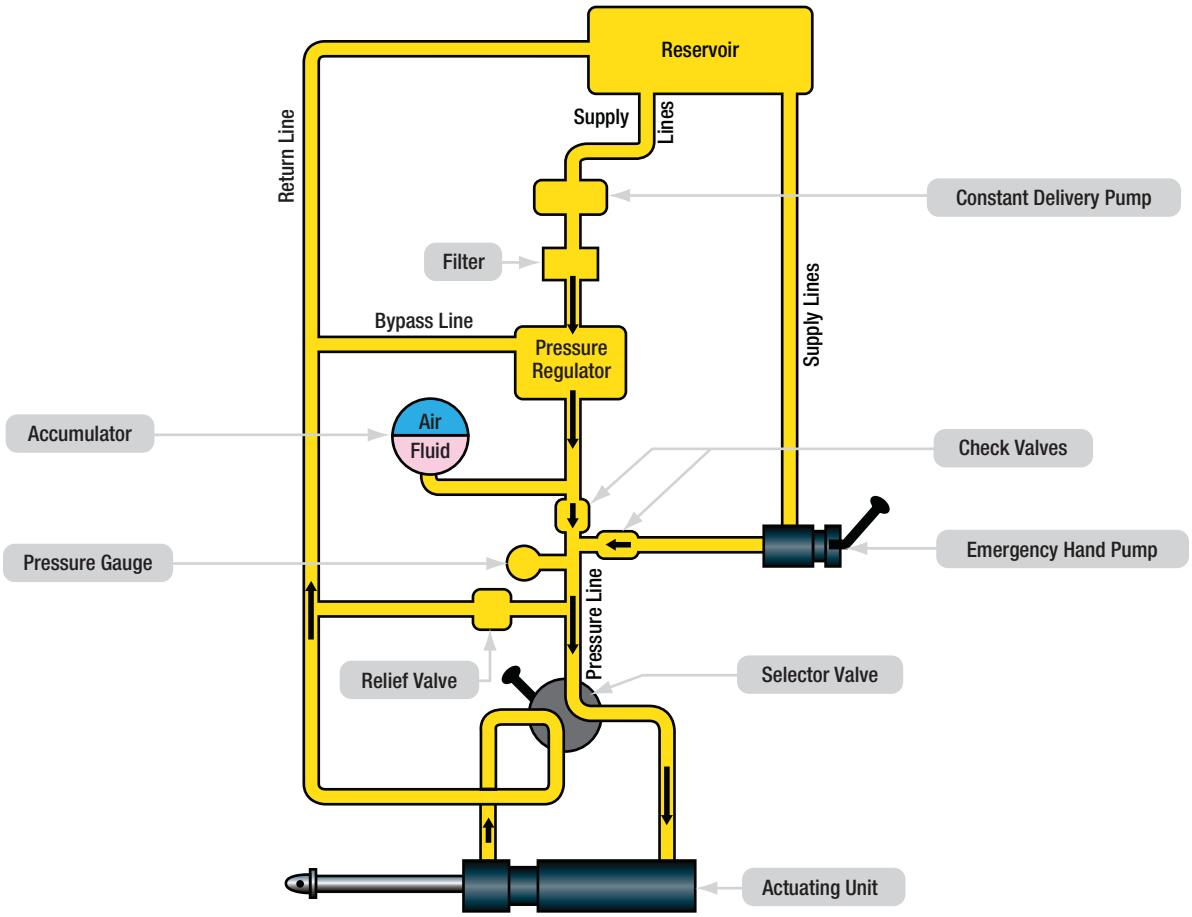


Figure 14-28. The location of a pressure regulator in a basic hydraulic system. The regulator unloads the constant delivery pump by bypassing fluid to the return line when the predetermined system pressure is reached.

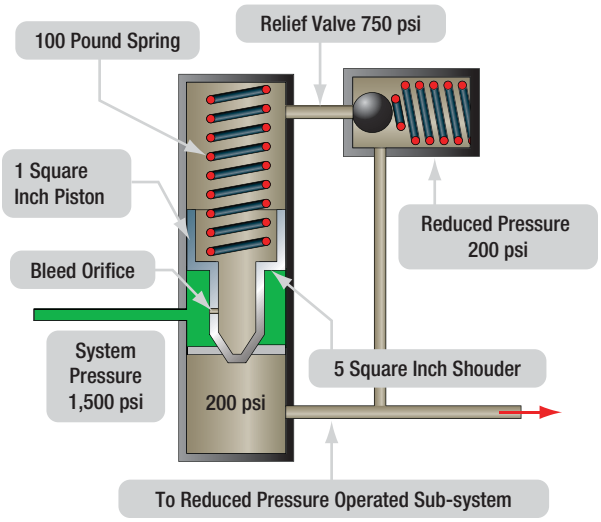


Figure 14-29. Operating mechanism of a pressure reducing valve.

POWER DISTRIBUTION

Power distribution in a hydraulic system is controlled through the use of variety of flow control valves. These valves control the speed and/or direction of fluid flow in the hydraulic system. They provide for the operation of various components when desired and the speed at which the component operates. Examples of flow

control valves include: selector valves, check valves, sequence valves, priority valves, shuttle valves, quick disconnect valves, hydraulic fuses and shutoff valves.

SHUTOFF VALVES

Shutoff valves are used to shutoff the flow of fluid to a particular system, sub-system or component. In general, these types of valves are electrically powered. They are used to distribute hydraulic power to various components in the system. A shutoff valve may be used to create a priority in a hydraulic system. (***Figure 14-30***)



Figure 14-30. Shutoff valves.

SELECTOR VALVES

A selector valve is used to control the direction of movement of a hydraulic actuating cylinder or similar device. It provides for the simultaneous flow of hydraulic fluid both into and out of the unit. Hydraulic system pressure can be routed with the selector valve to operate the unit in either direction and a corresponding return path for the fluid to the reservoir is provided. There are two main types of selector valves: open-center and closed-center. Selector valves may be poppet-type, spool-type, piston-type, rotary-type, or plug-type. (***Figure 14-31***)

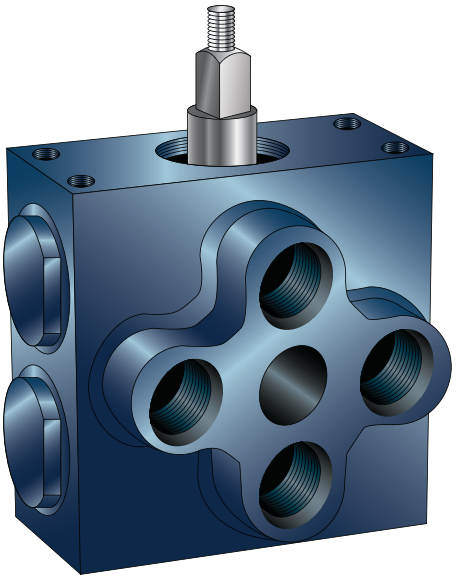


Figure 14-31. A poppet-type four-way selector valve.

Most selector valves are mechanically controlled by a lever or electrically controlled by solenoid or servo. (***Figure 14-32***)

CHECK VALVES

Another common flow control valve in aircraft hydraulic systems is the check valve. A check valve allows fluid to flow unimpeded in one direction, but prevents or restricts fluid flow in the opposite direction.

A check valve may be an independent component situated in-line somewhere in the hydraulic system or it may be built-in to a component. When part of a component, the check valve is said to be an integral check valve.

SEQUENCE VALVES

Sequence valves control the sequence of operation between two branches in a circuit; they enable one unit to automatically set another unit into motion. An example of the use of a sequence valve is in an aircraft landing gear actuating system. In a landing gear actuating system, the landing gear doors must open before the landing gear starts to extend. Conversely, the landing gear must be completely retracted before the



Figure 14-32. Four-way servo control valve.

doors close. A sequence valve installed in each landing gear actuating line performs this function. A sequence valve is somewhat similar to a relief valve except that, after the set pressure has been reached, the sequence valve diverts the fluid to a second actuator or motor to do work in another part of the system. There are various types of sequence valves. Some are controlled by pressure, some are controlled mechanically, and some are controlled by electric switches.

PRIORITY VALVES

A priority valve gives priority to the critical hydraulic subsystems over noncritical systems when system pressure is low. For instance, if the pressure of the priority valve is set for 2 200 psi, all systems receive pressure when the pressure is above 2 200 psi. If the pressure drops below 2 200 psi, the priority valve closes and no fluid pressure flows to the noncritical systems. Some hydraulic designs use pressure switches and electric shutoff valves to assure that the critical systems have priority over noncritical systems when system pressure is low.

SHUTTLE VALVES

In certain fluid power systems, the supply of fluid to a subsystem must be from more than one source to meet system requirements. In some systems, an emergency system is provided as a source of pressure in the event of normal system failure. The emergency system usually actuates only essential components. The main purpose of the shuttle valve is to isolate the normal system from an alternate or emergency system. It is small and simple; yet, it is a very important component.

QUICK DISCONNECT VALVES

Quick disconnect valves are installed in hydraulic lines to prevent loss of fluid when units are removed. Such valves are installed in the pressure and suction lines of the system immediately upstream and downstream of the power pump. In addition to pump removal, a power pump can be disconnected from the system and a hydraulic test stand connected in its place.

HYDRAULIC FUSES

A hydraulic fuse is a safety device. Fuses may be installed at strategic locations throughout a hydraulic system. They detect a sudden increase in flow, such as a burst downstream, and shut off the fluid flow. By closing, a fuse preserves hydraulic fluid for the rest of the system. Hydraulic fuses are fitted to the brake system, leading edge flap and slat extend and retract lines, nose landing gear up and down lines, and the thrust reverser pressure and return lines.

OTHER HYDRAULIC SYSTEM COMPONENTS

HYDRAULIC ACTUATORS

A hydraulic actuating cylinder transforms energy in the form of fluid pressure into mechanical force, or action, to perform work. It is used to impart powered linear motion to some movable object or mechanism. A typical actuating cylinder consists of a cylinder housing, one or more pistons and piston rods, and some seals. Actuating cylinders are of two major types: single action and double action. The single-action (single port) actuating cylinder is capable of producing powered movement in one direction only. The double-action (two ports) actuating cylinder is capable of producing powered movement in two directions.

FILTERS

A filter is a screening or straining device used to clean the hydraulic fluid, preventing foreign particles and contaminating substances from remaining in the system. (***Figure 14-33***).

If such objectionable material were not removed, the entire hydraulic system of the aircraft could fail through the breakdown or malfunctioning of a single unit of the system.

The hydraulic fluid holds in suspension tiny particles of metal that are deposited during the normal wear of selector valves, pumps, and other system components. Such minute particles of metal may damage the units and parts through which they pass if they are not removed by a filter. Since tolerances within the hydraulic system components are quite small, it is apparent that the reliability and efficiency of the entire system depends upon adequate filtering.

Filters may be located within the reservoir, in the pressure line, in the return line, or in any other location the designer of the system decides that they are needed to safeguard the hydraulic system against impurities.

Modern design often uses a filter module that contains several filters and other components. There are many models and styles of filters. Their position in the aircraft and design requirements determine their shape and size.

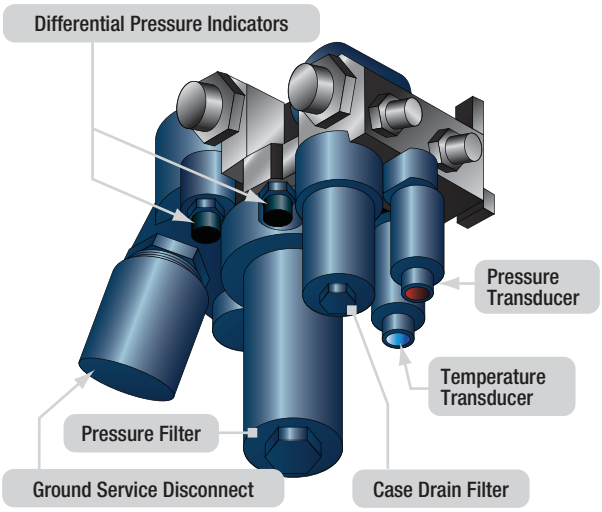


Figure 14-33. Filter module components.

HYDRAULIC INDICATING AND WARNING

There are just a few hydraulic system indications on the flight deck. Fluid pressure and temperature are the primary parameters monitored as well as fluid quantity. Reservoir pressurization air pressure may also be monitored. Typically, electro-hydraulic transducers are mounted in the system in key locations so that hydraulic pressure and temperature can be displayed on a gauge or LCD screen. A separate transmitter and indication is used for brake pressure. For servicing and maintenance, direct reading indicators are installed so that maintenance technicians can observe system status while on the ramp.

System pressure sensors are often located at the hydraulic system pressure filter modules. Low pressure warning switches are located down-stream of the pump outlet and may also be at the module. A hydraulic panel on the flight deck incorporates pump switches and temperature and pressure indications in older aircraft. Warnings and system indications are displayed on system status screens away from the switches on class cockpit aircraft.

Figure 14-34 illustrates a typical hydraulic control panel on the flight deck of an older Boeing aircraft.

Figure 14-35 illustrates system status, synoptic and maintenance page displays for the hydraulic system on a glass cockpit aircraft.

Hydraulic fluid quantity is monitored at the reservoir through the use of a float gauge, sight glass, or other sensing mechanism which sends a signal to the flight deck for gauge or LCD display. A low quantity warning switch may be included in the system. Hydraulic system fluid temperature indication is usually limited to an OVERHEAT annunciation for each pump/system. Temperature switches, often located in the return line as the fluid enters the reservoir, trip when a preset temperature is reached. A signal is sent to the flight deck for annunciation. Temperature sensors for hydraulic systems with electrically driven pumps serve as motor temperature monitoring devices as well. Motor driven pumps are more likely to get hot than engine driven pumps. The hot motor transfers some of its heat to the fluid as it circulates.



Figure 14-34. Hydraulic panel on an older Boeing transport aircraft.

Hydraulic system warnings include low pressure annunciations for each hydraulic system. Typically a lamp will illuminate, flash or change color on the flight deck when a pressure sensor sends an electric or electronic signal that a low pressure condition exists.

Many indicators display a low pressure warning when the hydraulic pumps are OFF which goes away when the pumps are switched ON and operate normally.

INTERFACE WITH OTHER SYSTEMS

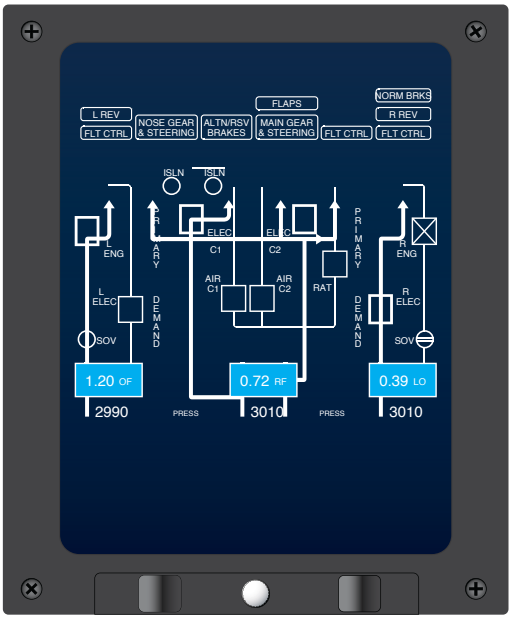
Many aircraft systems use hydraulic power such as landing gear extension and retraction, flight controls, and auto pilot. In most cases, the operational logic for these advanced systems are controlled by computer. To integrate the mechanical power of the hydraulic system, hydraulic system parameters and status condition must be input into the controlling computer. In the absence of any malfunction, the computer controller activates the correct hydraulic system components when needed.

Vital systems control logic can also dictate operation in alternate modes should the hydraulic system parameters be out of the normal operating range. For example, if the hydraulic pump used for normal operations is not maintaining acceptable system pressure, logic circuits reconfigure the operational mode from 'normal' to an alternate mode that utilizes the back-up hydraulic pump.

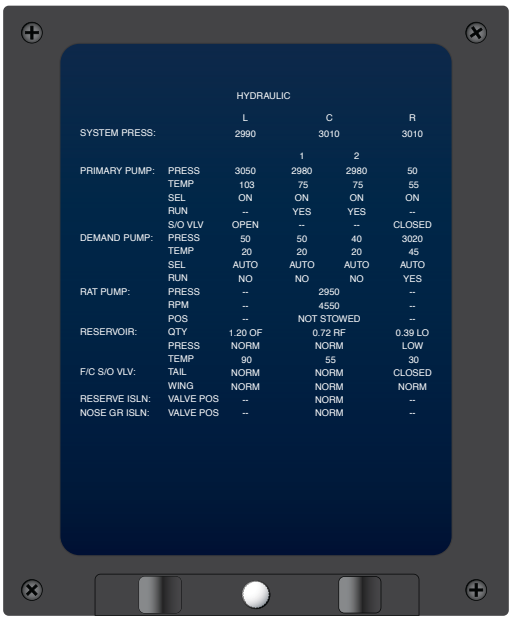
Hydraulic system parameters that are captured in analog format are converted to digital format for use in the control system logic.



Status Display



Hydraulic Synoptic Display



HYDRAULIC

		L	C		R
SYSTEM PRESS:		2990	3010		3010
			1	2	
PRIMARY PUMP:	PRESS	3050	2980	2980	50
	TEMP	103	75	75	55
	SEL	ON	ON	ON	ON
	RUN	--	YES	YES	--
	S/O VLV	OPEN	--	--	CLOSED
DEMAND PUMP:	PRESS	50	50	40	3020
	TEMP	20	20	20	45
	SEL	AUTO	AUTO	AUTO	AUTO
	RUN	NO	NO	NO	YES
RAT PUMP:	PRESS	--	2950		--
	RPM	--	4550		--
	POS	--	NOT STOWED		--
RESERVOIR:	QTY	1.20 OF	0.72 RF		0.39 LO
	PRESS	NORM	NORM		LOW
	TEMP	90	55		30
F/C S/O VLV:	TAIL	NORM	NORM		CLOSED
	WING	NORM	NORM		NORM
RESERVE ISLN:	VALVE POS	--	NORM		--
NOSE GR ISLN:	VALVE POS	--	NORM		--

Hydraulic Maintenance Display

Figure 14-35. Hydraulic system information displayed on a digital fight deck flat panel screen.

Question: 14-1
Hydraulic operations are nearly _____% efficient.

Question: 14-2
Selector valves in an open-center hydraulic system are arranged in _____ with each other.

Question: 14-3
The most widely used type of hydraulic system used in aviation is the _____ - center system.

Question: 14-4
Liquids are used to transmit and distribute force in a hydraulic system because they are basically _____.

Question: 14-5
Hydraulic reservoirs may be pressurized by bleed air or by _____.

Question: 14-6
Generally, _____ are used when engine-driven pumps fail.

Question: 14-7
Variable displacement hydraulic pump output is changed automatically by a pump _____ within the pump.

Question: 14-8
Name 3 hydraulic system parameters that may be monitored on the flight deck.

ANSWERS

Answer: 14-1
100.

Answer: 14-2
series.

Answer: 14-3
Closed.

Answer: 14-4
incompressible.

Answer: 14-5
hydraulic system pressure.

Answer: 14-6
electrically-driven pumps.

Answer: 14-7
compensator.

Answer: 14-8
Pressure.
Temperature.
Fluid quantity.



PART-66 SYLLABUS **LEVELS**

CERTIFICATION CATEGORY → **B2**

Sub-Module 15

ICE AND RAIN PROTECTION (ATA 30)

Knowledge Requirements

13.15 - Ice and Rain Protection (ATA 30)

Ice formation, classification and detection;	2
Anti-icing systems: electrical, hot air and chemical;	2
De-icing systems: electrical, hot air, pneumatic, chemical;	3
Rain repellent;	1
Probe and drain heating;	3
Wiper Systems.	1

ICE FORMATION, CLASSIFICATION, AND EFFECTS

Rain, snow, and ice, are longtime enemies of transportation. Flying has added a new dimension, particularly with respect to ice. Under certain atmospheric conditions, ice can build rapidly on airfoils and air inlets. On days when there is visible moisture in the air, ice can form on aircraft leading edge surfaces at altitudes where freezing temperatures start. Water droplets in the air can be supercooled to below freezing without actually turning into ice unless they are disturbed in some manner. This unusual occurrence is partly due to the surface tension of the water droplet not allowing the droplet to expand and freeze. However, when aircraft surfaces disturb these droplets, they immediately turn to ice on the aircraft surfaces.

In order to perform as designed, aircraft airfoils must be completely smooth and free of any irregularities or contamination in the form of ice, snow or frost. Even a small amount of surface contamination can reduce lift and raise the stall speed. Accidents have occurred due to airfoil surface roughness caused by frost. The additional weight caused by ice accumulation is also problematic. All aviators must be diligent to only dispatch an aircraft that is free from any type of ice, snow, or frost contamination.

There are two types of ice are encountered during flight: clear ice and rime ice. Clear ice forms when the remaining liquid portion of the water drop flows out over the aircraft surface, gradually freezing as a smooth sheet of solid ice. Formation occurs when droplets are large, such as in rain or in cumuliform clouds. Clear ice is hard, heavy, and tenacious. Its removal by deicing equipment is especially difficult. (***Figure 15-1***)

Rime ice forms when water drops are small, such as those in stratified clouds or light drizzle. The liquid portion remaining after initial impact freezes rapidly before the drop has time to spread over the aircraft surface. The small frozen droplets trap air giving the ice a white appearance. Rime ice is lighter in weight than clear ice, however its weight is of little significance. The irregular shape and rough surface of rime ice decrease the effectiveness and efficiency of the aerodynamic airfoils. This reduces lift and increases drag. Rime ice is brittle and more easily removed than clear ice.



Figure 15-1. Formation of ice on aircraft leading edge.

Mixed clear and rime icing can form rapidly when water drops vary in size or when liquid drops intermingle with snow or ice particles. Ice particles become embedded in clear ice, building a very rough accumulation sometimes in a mushroom shape on leading edges. Ice may be expected to form whenever there is visible moisture in the air and temperature is near or below freezing. An exception is carburetor icing, which can occur during warm weather with no visible moisture present.

ICING EFFECTS

Ice or frost forming on aircraft creates hazards detrimental to safe flight. The resulting malformation of the airfoil when ice adheres to it decreases the amount of lift. The additional weight and unequal formation of the ice may also cause unbalancing of the aircraft, making it hard to control. Enough ice to cause an unsafe flight condition can form in a very short period of time, thus some method of ice prevention or removal is necessary.

Ice buildup increases drag and reduces lift. It causes destructive vibration and hampers true instrument readings. Control surfaces become unbalanced or frozen. Fixed slots are filled and movable slots jammed. Radio reception is hampered and engine performance is affected. Ice, snow, and slush have a direct impact on the safety of flight. Not only because of degraded lift, reduced takeoff performance, and/ or maneuverability of the aircraft, but when chunks break off, they can also cause engine failures and structural damage. Fuselage aft-mounted engines are particularly susceptible to this foreign object damage (FOD) phenomenon. Wing mounted engines are not excluded however.

Ice can be present on any part of the aircraft and, when it breaks off, there is some probability that it could go into an engine. The worst case is that ice on the wing breaks off during takeoff due to the flexing of the wing and goes directly into the engine, leading to surge, vibration, and complete thrust loss. Light snow that is loose on the wing surfaces and the fuselage can also cause engine damage leading to surge, vibration, and thrust loss. Whenever icing conditions are encountered, the performance characteristics of the airplane deteriorate. (***Figure 15-2***)

Increased aerodynamic drag increases fuel consumption, reducing the airplane's range and making it more difficult to maintain speed. Decreased rate of climb must be anticipated, not only because of the decrease in wing and empennage efficiency but also because of the possible reduced efficiency of the propellers and increase in gross weight. Abrupt maneuvering and steep turns at low speeds must be avoided because the airplane stalls at higher-than-published speeds with ice accumulation. On final approach for landing, increased airspeed must be maintained to compensate for this increased stall speed. After touchdown with heavy ice accumulation, landing distances may be as much as twice the normal distance due to the increased landing speeds. In this sub-module, ice prevention and ice elimination using electric, pneumatic and chemical systems is discussed.

The ice and rain protection systems used on aircraft keep ice from forming or remove ice on the following airplane components:

- Wing leading edges
- Horizontal and vertical stabilizer leading edges
- Engine cowl leading edges
- Propellers
- Propeller spinner
- Air data probes
- Flight deck windows
- Water and waste system lines and drains
- Antenna

Figure 15-3 gives an overview of ice and rain protection systems installed in a large transport category aircraft. In modern aircraft, many of these systems are automatically controlled by the ice detection system and onboard computers.

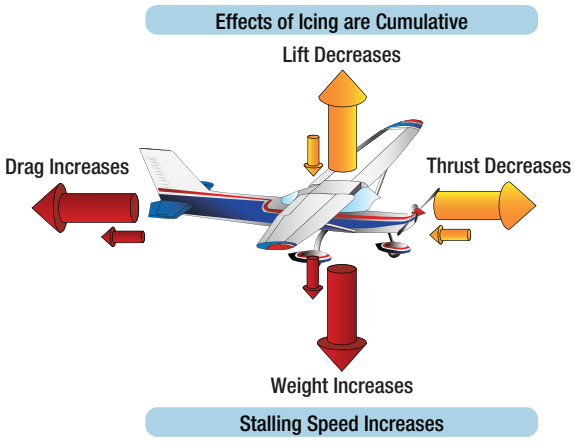


Figure 15-2. Effects of structural icing.

ICE DETECTION

Ice can be detected visually, but most modern aircraft have one or more ice detector sensors that warn the flight crew of icing conditions. Generally, an annunciator light comes on to alert the flight crew when ice is detected. In some aircraft models, multiple ice detectors are used, and the ice detection system automatically turns on certain anti-ice systems when icing is detected.

As can be seen in ***Figure 15-4***, ice detectors are mounted on the sides of the forward fuselage to receive impact air as the aircraft moves forward. Two independent detectors and detector systems are normal. The typical anti-ice detector contains a probe which is vibrated at an established rate. For detectors on the Boeing 777, the vibration rate is 40 000 Hz. Inside the probe housing are circuit cards and a microprocessor. The probe is electrically connected to the ice protection control unit and the aircraft data buses.

Should ice collect on the ice detector probe, the established frequency of vibration decreases. A slight lowering of the frequency causes the integral electric probe heater to come ON. The heater quickly melts the ice in 5-7 seconds and shuts OFF. If ice reforms on the probe, the heater will cycle ON again, melt the ice, and turn OFF again. The control logic inside the detector unit monitors the heater cycles. When the probe heat cycles two or more times, the detector automatically sends an icing signal to the airfoil and cowl ice protection system and engine inlet anti-ice is automatically turned

ON. An annunciation is also made to alert the crew. Note that the aircraft must be in the air with the anti-ice switch on the flight deck set to AUTO.

If the probe heat cycles ten or more times, an icing signal is sent and the wing anti-ice system (WIA) is also automatically turned ON. Note that if the probe heat is on for more than 15 seconds, the EAI comes ON. After 25 seconds, if the probe vibration does not return to its normal rate, a fault is annunciated and the heater is turned OFF. ***Figure 15-5*** illustrates the ice detector time cycles on a Boeing 777. Ice detection system logic and operation receives input from the air/ground sensing system. It does not function on the ground. However, the aircraft information management system displays the system information on the ice protection maintenance page and ground tests of the ice detectors can be initiated from there.

Detector probes get extremely hot when heated and cause serious burns. Also note that electrostatic handling precautions should be exercised when handling ice detectors. ***Figure 15-6*** illustrates the ice detection system on a Boeing 777.

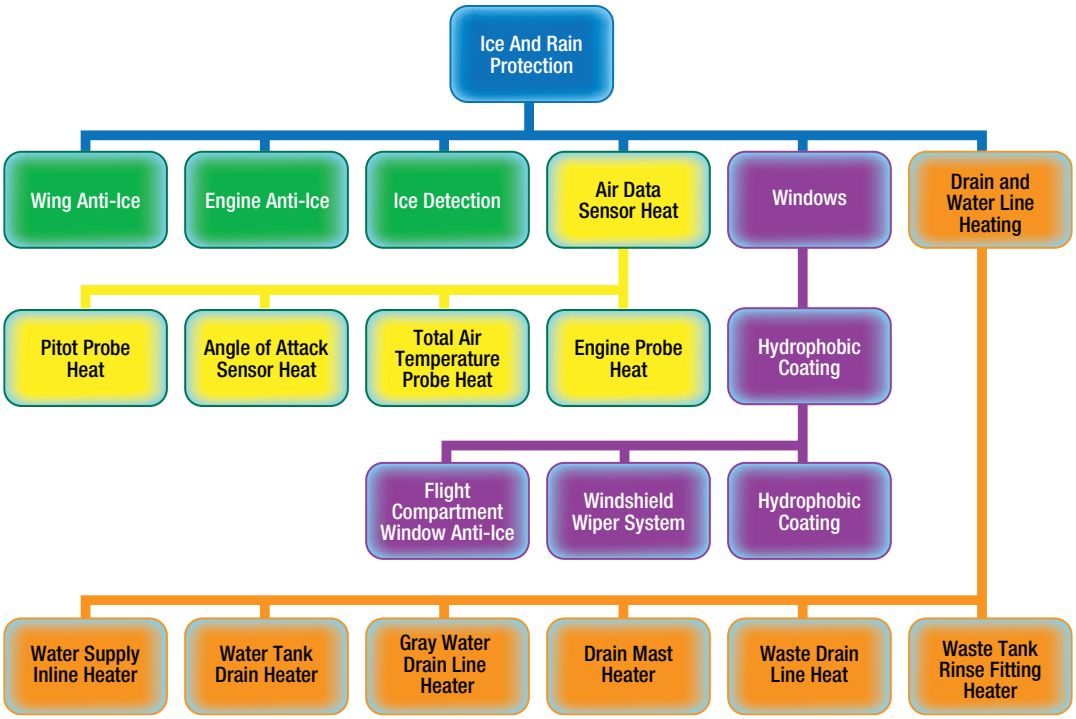


Figure 15-3. Ice and rain protection systems.

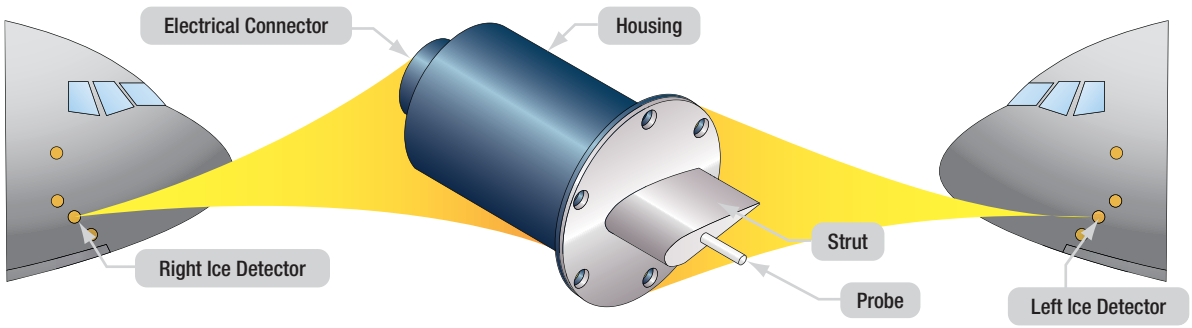


Figure 15-4. An ice detector alerts the flight crew of icing conditions and, on some aircraft, automatically activates ice protection systems. One or more detectors are located on the forward fuselage.

ANTI-ICE VERSUS DEICE

Ice control systems are designed for anti-icing or for deicing. Anti-icing equipment is turned on before entering icing conditions and is designed to prevent ice from forming. A surface may be anti-iced by keeping it dry, by heating to a temperature that evaporates water upon impingement, or by heating the surface just enough to prevent freezing. Deicing equipment is designed to remove ice after it begins to accumulate, typically on the wings and stabilizer leading edges. Ice may be controlled on aircraft structure, by the methods described in ***Figure 15-7***.

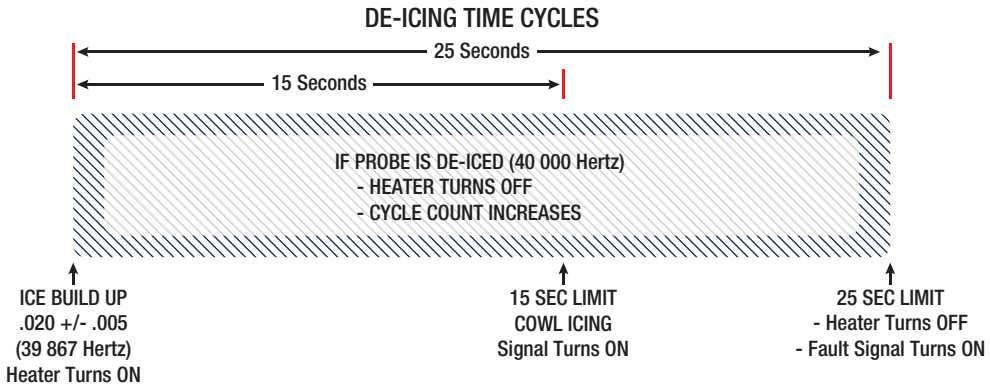


Figure 15-5. Deicing time cycle.

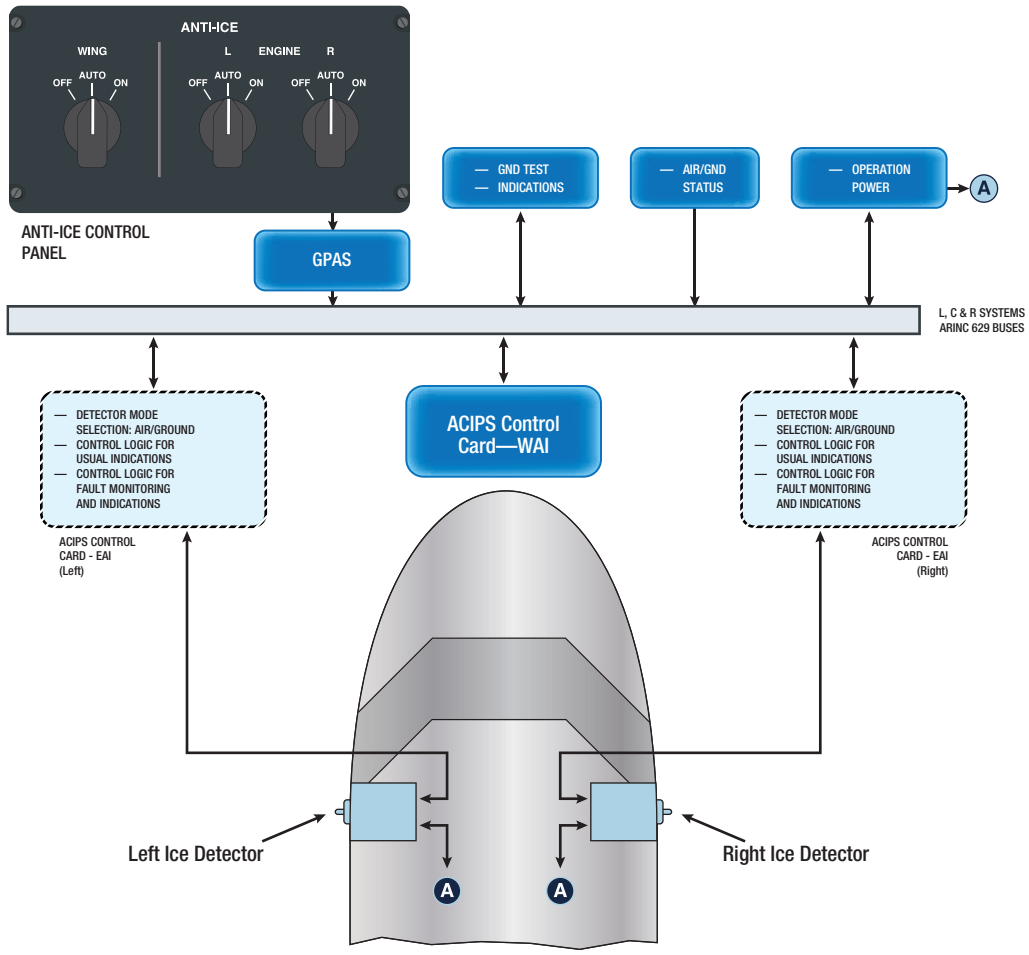


Figure 15-6. Forward fuselage ice detection system on a Boeing 777.

Location of Ice	Method of Control	
	Anti-Ice	De-Ice
Leading Edge of The Wing	Thermal Pneumatic	Pneumatic Boots
	Thermal Electric	Electric Boots
	Chemical	Chemical (Minor)
Engine Inlets	Thermal Pneumatic	
	Thermal Electric	
Leading Edge of Horizontal and Vertical Stabilizers	Thermal Pneumatic	Pneumatic Boots
	Thermal Electric	Electric Boots
	Chemical	
Pitot Tube, Static Ports, Air Data Sensors,	Thermal Electric	
Water Drains, Tanks and Lines		
Propeller		Electric Element Boots
		Chemical
Windshield/Flight Deck Windows	Thermal Pneumatic	Thermal Pneumatic
	Thermal Electric	(Primarily for De-fogging)
	Chemical	

Figure 15-7. Typical ice control methods.

Several means of preventing or removing ice are used depending on the type and location of the component that requires ice control.

They basically include they following:

1. Heating the surfaces with hot air.
2. Heating by electrical elements.
3. Breaking up ice that has formed, usually with inflatable boots.
4. Application of chemicals to prevent adhesion of ice.

ANTI-ICING SYSTEMS

On large, high performance turbine powered aircraft, anti-ice is the preferred method of ice control. Thermal pneumatic anti-ice using engine bleed air is most common for large surfaces such as the leading edges of the wing, empennage and engine inlet cowling. Thermal electric anti-ice is most common on probes, drain, tanks and windshields. Chemical anti-icing is used on smaller turbine powered aircraft. Once activated, most systems are automatically operated.

THERMAL PNEUMATIC ANTI-ICING

Thermal pneumatic systems used for the purpose of preventing the formation of ice on airfoil leading edges usually use heated air ducted span-wise along the inside of the leading edge of the airfoil and distributed around its inner surface. These thermal pneumatic anti-icing systems are used for wings, leading edge slats, horizontal and vertical stabilizers, engine inlets, and more. As stated, the most common source of the heated air is the turbine engine compressor bleed air.

WING ANTI-ICE (WAI) SYSTEM

Thermal wing anti-ice (WAI or TAI) systems for business jet and large-transport category aircraft take advantage of the relatively large amounts of very hot air that can be bled off of turbine engine compressors to provide a satisfactory source of anti-icing heat. The hot air is routed through ducting, manifolds, and valves to the leading edges of the wings. (***Figure 15-8***)



Figure 15-8. Aircraft with thermal WAI system.

Figure 15-9 shows a typical WAI system schematic for a business jet. The bleed air is routed to each wing leading edge by an ejector in each wing inboard area. The ejector discharges the bleed air into piccolo tubes for distribution along the leading edge. Fresh ambient air is introduced into the wing leading edge by two flush-mounted ram air scoops in each wing leading edge, one at the wing root and one near the wingtip. The ejectors entrain ambient air, reduce the temperature of the bleed air, and increase the mass airflow in the piccolo tubes. The wing leading edge is constructed of two skin layers separated by a narrow passageway. (***Figure 15-10***) The air directed against the leading edge can only escape through the passageway, after which it is vented overboard through a vent in the bottom of the wingtip. When the WAI switch is turned on, the pressure regulator is energized and the shutoff valve opens. When the wing leading edge temperature reaches approximately +140 ˚F, temperature switches turn ON the operation light above the switch. If the temperature in the wing leading edge exceeds approximately +212 ˚F (outboard) or +350 ˚F (inboard), the red WING OV HT warning light on the annunciator panel illuminates.

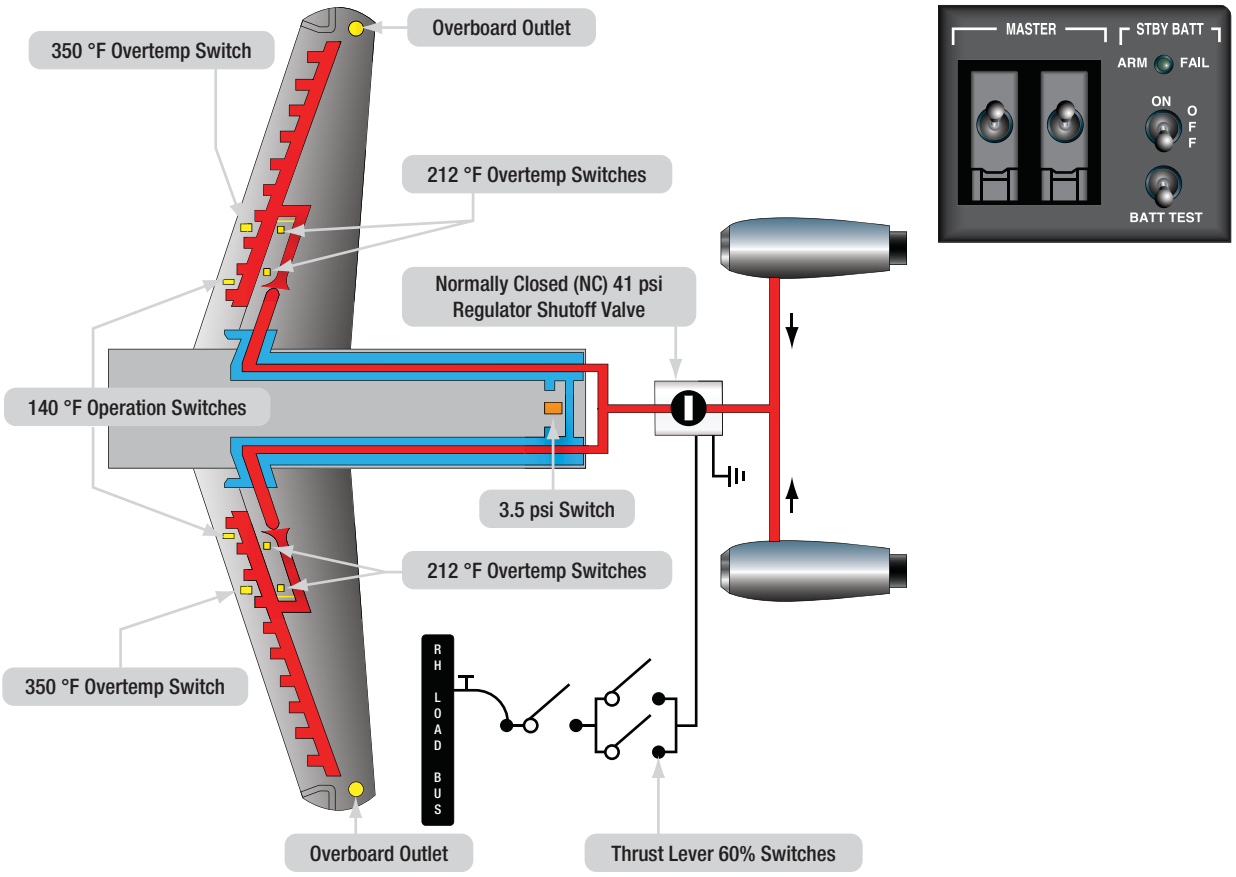


Figure 15-9. Thermal WAI system.

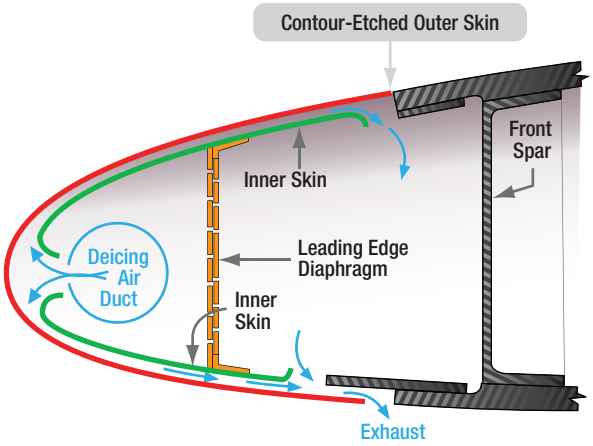


Figure 15-10. Heated wing leading edge.

Large aircraft WAI systems are similar to the business jet system just described. However, the larger engines with multiple stage compressor bleed air taps usually do not require that ambient air is mixed with the bleed air for temperature control.

The ducting of WAI systems on large aircraft usually consists of aluminum alloy, titanium, stainless steel, or molded fiberglass tubes. The tube, or duct sections are attached to each other by bolted end flanges or by band-type V-clamps. In some locations, the ducting is covered with a fire-resistant, heat-insulating material such as fiberglass. Thin stainless steel expansion bellows are used at strategic positions in the ducting to absorb any distortion or expansion of the ducting that may occur due to temperature variations. The joined sections of ducting are hermetically sealed by sealing rings. These seals are fitted into annular recesses in the duct joint faces.

When installing a section of duct, make certain that the seal bears evenly against and is compressed by the adjacent duct's flange. When specified, the ducts should be pressure tested at the pressure recommended by the manufacturer of the aircraft concerned. Leak checks are made to detect defects in the duct that would permit the escape of heated air. The rate of leakage at a given pressure should not exceed that recommended in the aircraft maintenance manual.

Air leaks can often be detected audibly and are sometimes revealed by holes in the lagging or thermal insulation material. However, if difficulty arises in locating leaks, a soap-and-water solution may be used. All ducting should be inspected for security, general condition, or distortion. Lagging or insulating blankets must be checked for security and must be free of flammable fluids, such as oil or hydraulic fluid.

Slat Leading Edges

Most large turbine aircraft are fitted with leading edge slats. This means that it may be the slats that actually receive thermal pneumatic wing anti-ice, not the wing proper. The WAI ducts move air from the pneumatic system through the wing leading edge to the leading edge slats. The ducting warms the cavities through which it is routed. Combined with telescoping ducts that direct air into the slats when extended, retracted or in transit, this is sufficient to keep ice from forming on the entire wing leading edge. Holes in the bottom of each slat allow the pneumatic air to exit the slats. The telescoping duct sections that direct air into the slats attached to the slat on one end, and slide over a narrow diameter "T" section that is connected into the WAI duct. A seal prevents any loss of air. (***Figure 15-11***)

WAI Operation

Large turbine powered aircraft wing anti ice system have automatic anti-ice control systems. The following paragraph explains the operation of the Boeing 777. Other Boeings and Airbus are similar. The primary components of the WAI system are the Airfoil And Cowl Ice Protection System (ACIPS) computer logic cards, the WAI valve and the WAI duct pressure sensor. Inputs include the ice detection system and aircraft status inputs. A switch on the flight deck turns the system ON, AUTO, or OFF. In the auto mode, the system turns on when the ice detection system detects ice. The OFF and ON positions are used for manual control of the WAI system. The WAI system is only used in the air, except for ground tests. The weight on wheels system (WOW) and/or airspeed data disarms the system when the aircraft is on the ground. ***Figure 15-12*** illustrates the WAI system on a modern airliner.

The WAI valve controls the flow of bleed air from the pneumatic system to the WAI ducts. The valve is electrically controlled and pneumatically actuated. The torque motor controls operation of the valve. With no electrical power to the torque motor, air pressure on one side of the actuator holds the valve closed. Electrical current through the torque motor allows air pressure to open the valve. As the torque motor current increases, the valve opening increases. (***Figure 15-13***)

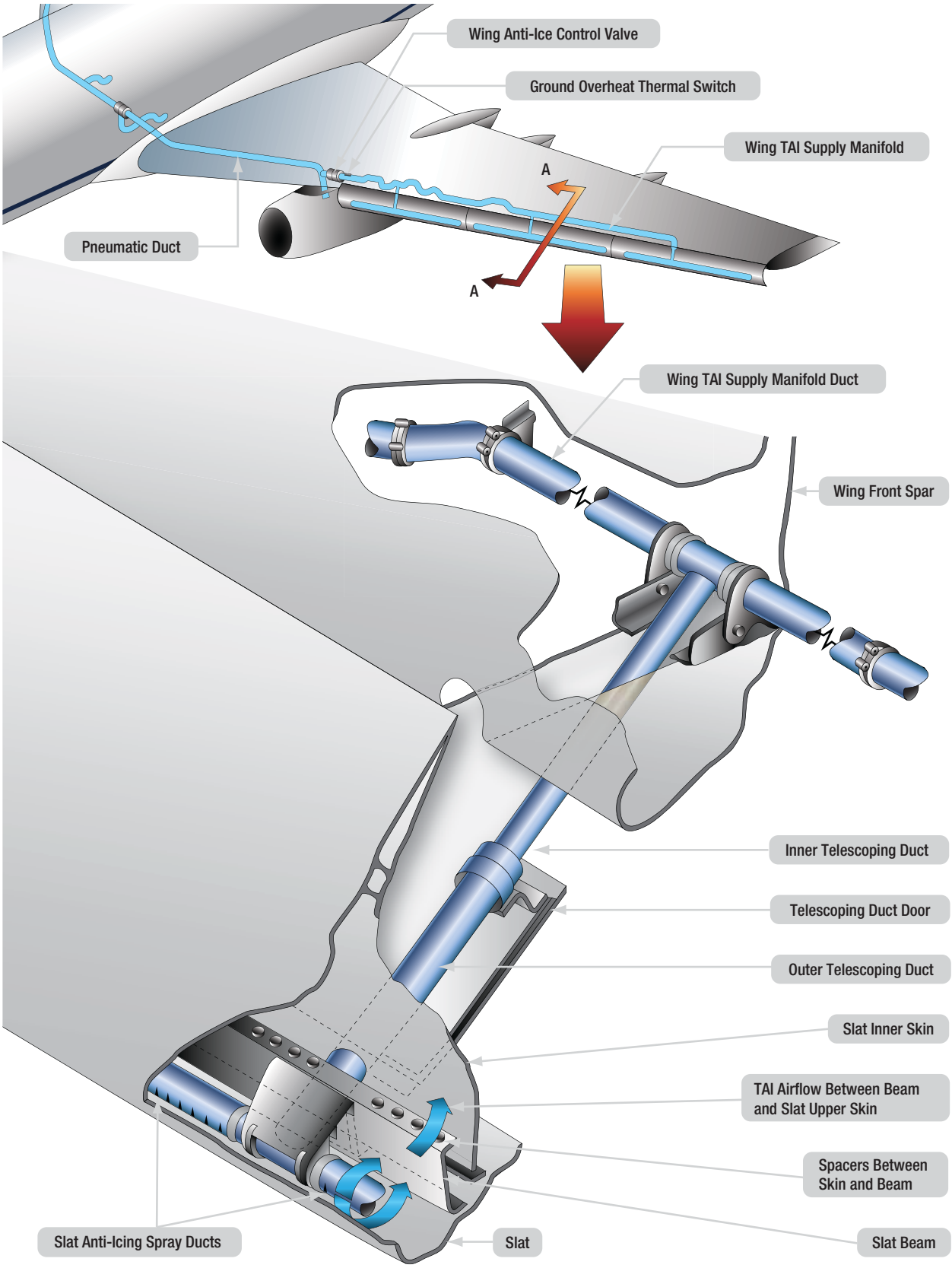


Figure 15-11. WAI ducting.

The WAI pressure sensor senses the air pressure in the WAI duct after the WAI valve. The airfoil and cowl ice protection system (ACIPS) uses integrated circuit system logic cards in a card file system. An ACIPS control card uses the WAI pressure information to control the WAI system.

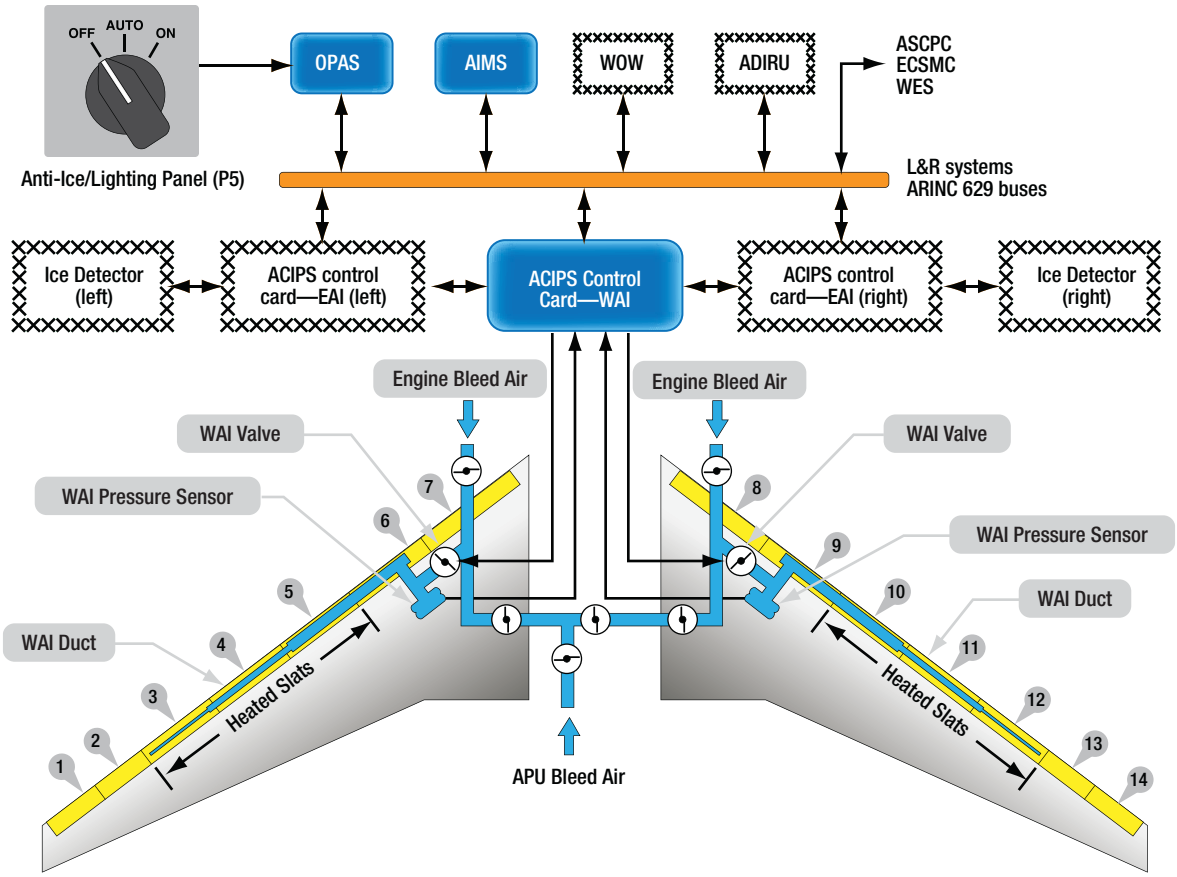


Figure 15-12. Wing leading edge slat anti-ice system.

WAI Control

The WAI system is controlled by the ACIPS computer card. The ACIPS computer card controls both WAI valves. The required positions of the WAI valves change as bleed air temperature and altitude change. The left and right valves operate at the same time to heat both wings equally. This keeps the airplane aerodynamically stable in icing conditions. The WAI pressure sensors supply feedback information to the WAI ACIPS computer card for WAI valve control and position indication. If either pressure sensor fails, the WAI ACIPS computer card sets the related WAI valve to either fully open or fully closed. If either valve fails closed, the WAI computer card keeps the other valve closed.

As stated, there is one selector for the WAI system. The selector has three positions: AUTO, ON, and OFF. With the selector in AUTO and no operational mode inhibits, the WAI ACIPS computer card sends a signal to open the WAI valves when either ice detector detects ice. The valves close after a 3-minute delay when the ice detector no longer detects ice.

The time delay prevents frequent ON/OFF cycles during intermittent icing conditions. With the selector ON and no operational mode inhibits, the WAI valves open. With the selector OFF, the WAI valves close. The operational mode for the WAI valves can be inhibited by many different sets of conditions. (***Figure 15-14***)

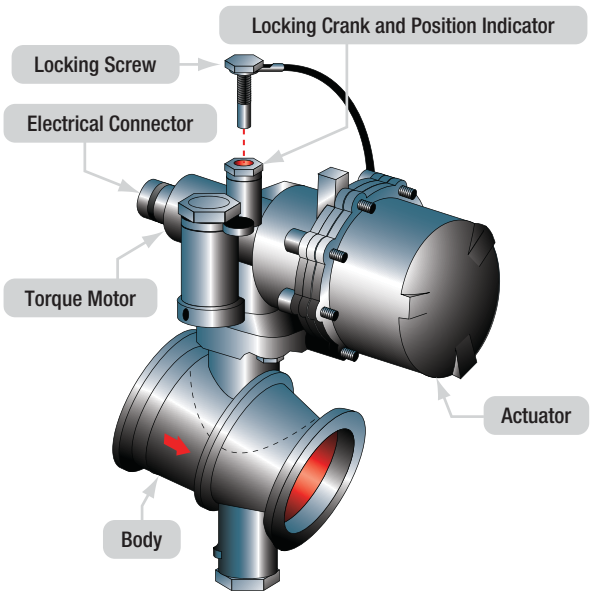


Figure 15-13. A wing anti-ice valve.

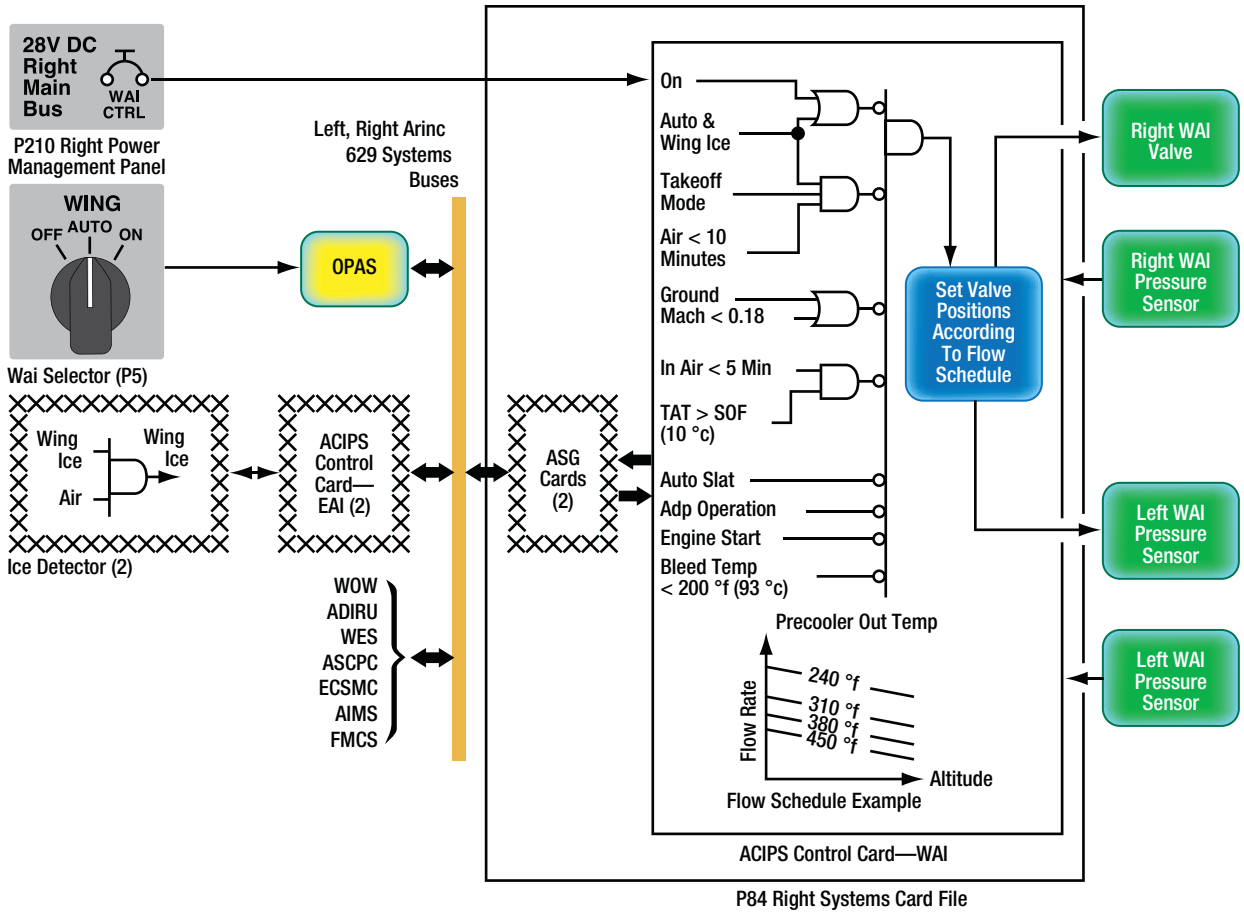


Figure 15-14. WAI inhibit logic schematic.

Operational mode is inhibited if all of these conditions occur:

- Auto mode is selected.
- Takeoff mode is selected.
- Airplane has been in the air less than 10 minutes.

With AUTO or ON selected, the operational mode is inhibited if any of these conditions occur:

- Airplane on the ground (except during an initiated or periodic built-in test equipment (BITE) test).
- Total Air Temperature (TAT) is more than 50 °F (10 °C) and the time since takeoff is less than 5 minutes.
- Auto slat operation.
- Air-driven hydraulic pump operation.
- Engine start.
- Bleed air temperature less than 200 °F (93 °C).

The WAI valves stay closed as long as the operational mode inhibit is active. If the valves are already open, the operational mode inhibit causes the valves to close.

WAI Indication System

The aircrew can monitor the WAI system on the onboard computer maintenance page. (***Figure 15-15***)

The following information is shown:

- WING MANIFOLD PRESS - pneumatic duct pressure in PSIG
- VALVE - WAI valve open, closed, or regulating
- AIR PRESS - pressure downstream of the WAI valves in PSIG
- AIR FLOW - air flow through the WAI valves in pounds per minute

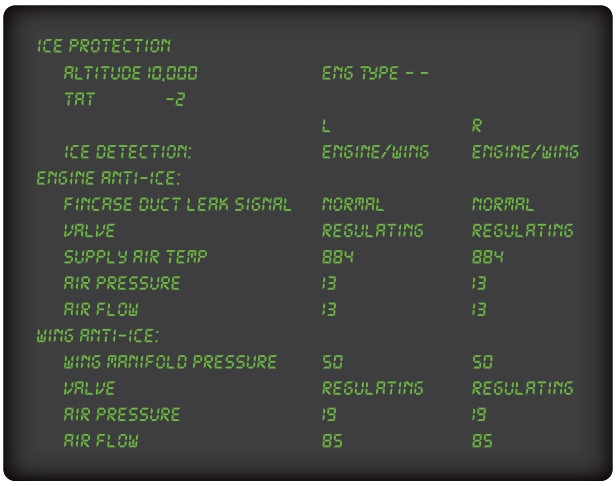


Figure 15-15. Ice protection onboard computer maintenance page.

WAI System BITE Test

Built In Test Equipment (BITE) circuits in the WAI ACIPS computer card continuously monitor the WAI system. Faults that affect the dispatch of the aircraft cause status messages. Other faults cause Central Maintenance Computer System (CMCS) maintenance messages. The BITE in the WAI ACIPS computer card also performs automatic power-up and periodic tests. Faults found during these tests that affect dispatch cause status messages. Other faults cause CMCS maintenance messages. The power-up test occurs when the card gets power. BITE does a test of the card hardware and software functions and the valve and pressure sensor interfaces. The valves do not move during this test. The periodic test occurs when all these conditions are true:

- The airplane has been on the ground between 1 and 5 minutes.
- The WAI selector is set to AUTO or ON.
- Air-driven hydraulic pumps are not in intermittent operation.
- Bleed pressure is sufficient to open the WAI valves.
- The time since the last periodic test is more than 24 hours.

During this test, the WAI valves cycle open and closed. This test makes sure that valve malfunctions are detected.

ENGINE ANTI-ICE (EAI)

In addition to thermal pneumatic wing anti-ice, large turbine powered aircraft also have thermal pneumatic Engine Inlet Anti-Ice (EAI). It is extremely important that ice not be allowed to build on the engine inlet cowl. Should ice form and then break off, it is ingested by the engine and could cause engine damage. This is why Boeing engine anti-ice is automatically turned on before wing anti-ice when the ice detection system begins to cycle the probe heat. The EAI operates similarly to the WAI. Bleed air supplied from a high stage compressor bleed port is ducted to the leading edge of the engine inlet cowls. It exits the cowl through overboard vents. (***Figure 15-16***)

A pneumatically actuated EAI valve controls the flow of the warm bleed air to the inlet cowl. The valve is supplied control pressure from an EAI controller. The controller has a torque motor that moves in response to ACIPS - EAI logic card signals. It regulates activation pressure to the EAI valve. (***Figure 15-17***)

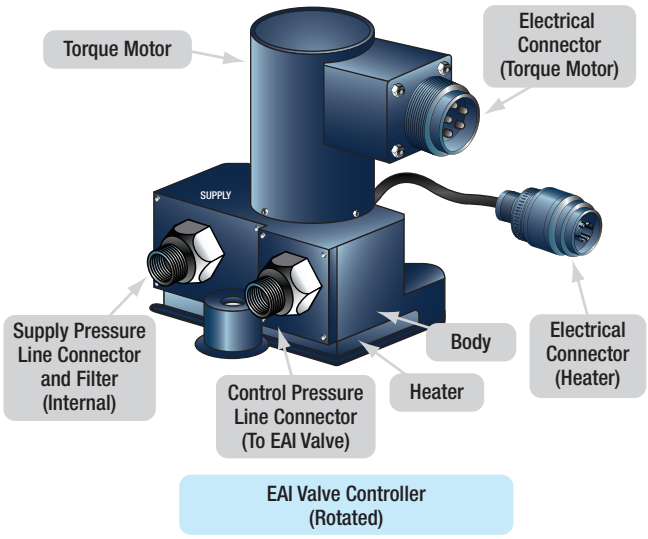


Figure 15-16. The EAI controller regulates air that operates the EAI anti-ice valve.

Signals from the ice detection system are delivered to the EAI ACIPS logic control card along with EAI duct pressure information from sensors downstream of the EAI valve. The logic card circuits control the operation of the EAI controller which positions the EAI valve.

An air to air heat exchanger cools the bleed air used by the EAI controller, the pressure regulating and shutoff valve (PRSOV) and the high pressure and fan air controller (HPFAC). ***Figure 15-18*** illustrates the Engine anti-ice system.

EAI is automatic when the anti ice switch is set to AUTO. To operate EAI on the ground, ON must be selected. Built in Test Equipment BITE is active with the switch position ON or AUTO on the ground and in the air. EAI indications and warnings are displayed on the flight deck EICAS displays.

THERMAL ELECTRIC ANTI-ICING

Electricity is used to heat various components on an aircraft so that ice does not form. This type of anti-ice is typically limited to small components due to high amperage draw. Effective thermal electric anti-ice is used on most air data probes, such as pitot tubes, static air ports, TAT and AOA probes, ice detectors, and engine P2/T2 sensors. Water lines, waste water drains, and some turboprop inlet cowls are also heated with electricity to prevent ice from forming. Transport category and high performance aircraft use thermal electric anti-icing in windshields.

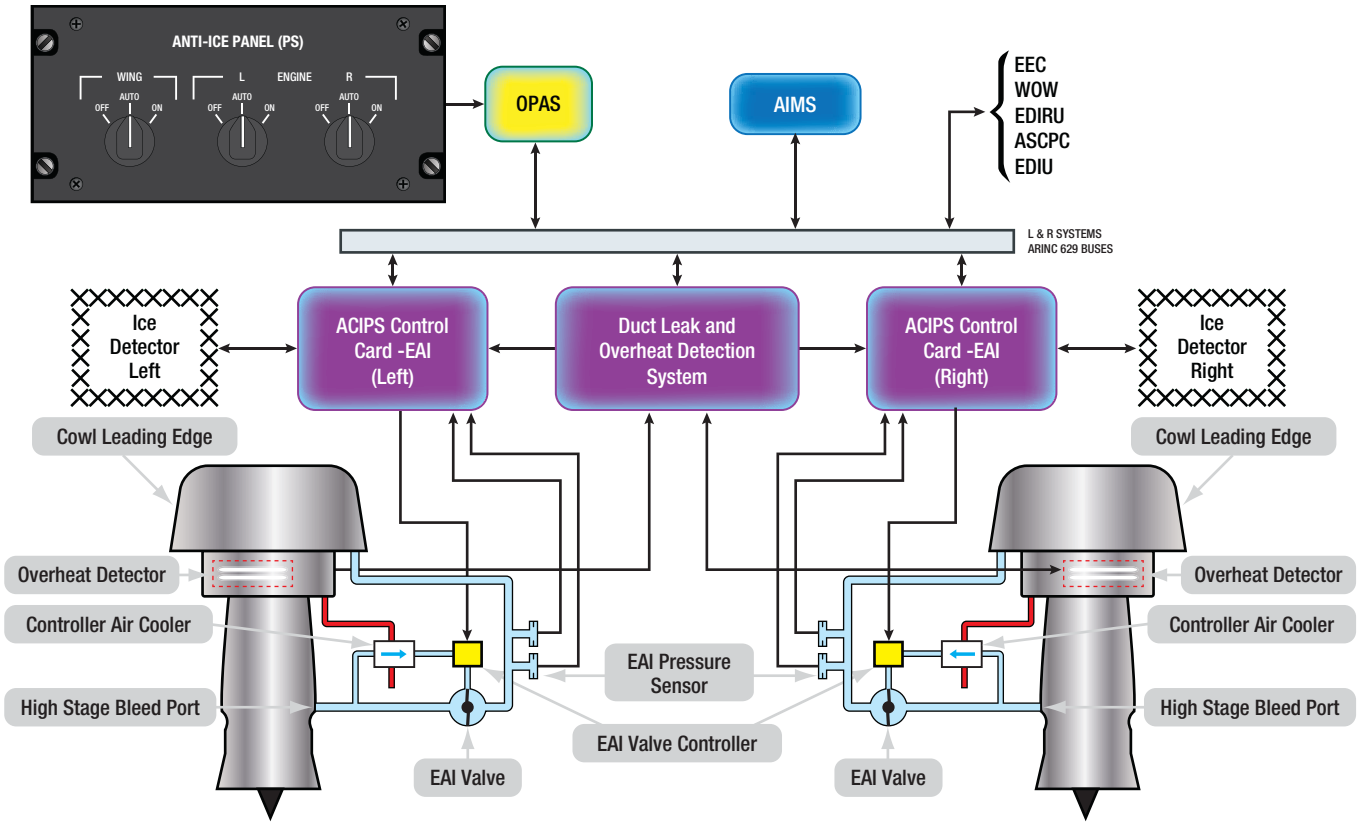


Figure 15-17. Thermal pneumatic engine anti-ice system on a Boeing 777.

PROBE ANTI-ICE

In devices that use thermal electric anti-ice, current flows through an integral conductive element that produces heat. The temperature of the component is elevated above the freezing point of water so ice cannot form. Various schemes are used, such as an internal coil wire, externally wrapped blankets or tapes, as well as conductive films and heated gaskets. A basic discussion of probe heat follows.

Data probes that protrude into the ambient airstream are particularly susceptible to ice formation in flight. ***Figure 15-18*** illustrates the types and location of probes that use thermal electric heat on one airliner. A pitot tube, for example, contains an internal electric element that is controlled by a switch in the cockpit. Use caution checking the function of the pitot heat when the aircraft is on the ground. The tube gets extremely hot since it must keep ice from forming at altitude in temperatures near -50 °F at speeds possibly over 500 miles per hour. An ammeter or load meter in the circuit can be used as a substitute to touching the probe, if so equipped. Simple probe heat circuits exist on most with a switch and a circuit breaker to activate and protect the device.

Advanced aircraft may have more complex circuitry in which control is by computer and flight status condition of the aircraft is considered before thermal electric heaters are activated automatically. ***Figure 15-19*** shows such a circuit for a pitot tube. The primary flight computer (PFC) supplies signals for the air data card (ADC) to energize ground and air heat control relays to activate probe heat. Information concerning speed of the aircraft, whether it is in the air or on the ground, and if the engines are running are factors considered by the ADC logic. Similar control is use for other probe heaters.

WATER SYSTEM AND DRAIN ANTI-ICE

Transport type aircraft have water and waste systems on board, and electrical heaters are often used to prevent the formation of ice in the water lines of these systems. Water lines carry water from the potable tanks to the lavatories and galleys. The waste water tanks collect the gray water from the galleys and lavatories. Heater blankets, in-line heaters, or heater boots are often used to heat the water supply lines, water tank drain hoses, waste drain lines, waste tank rinse fittings, and drain masts. Thermostats in the water lines supply temperature data to the control unit that turns the electrical heaters on and off. When the temperature falls below freezing, the electrical heaters

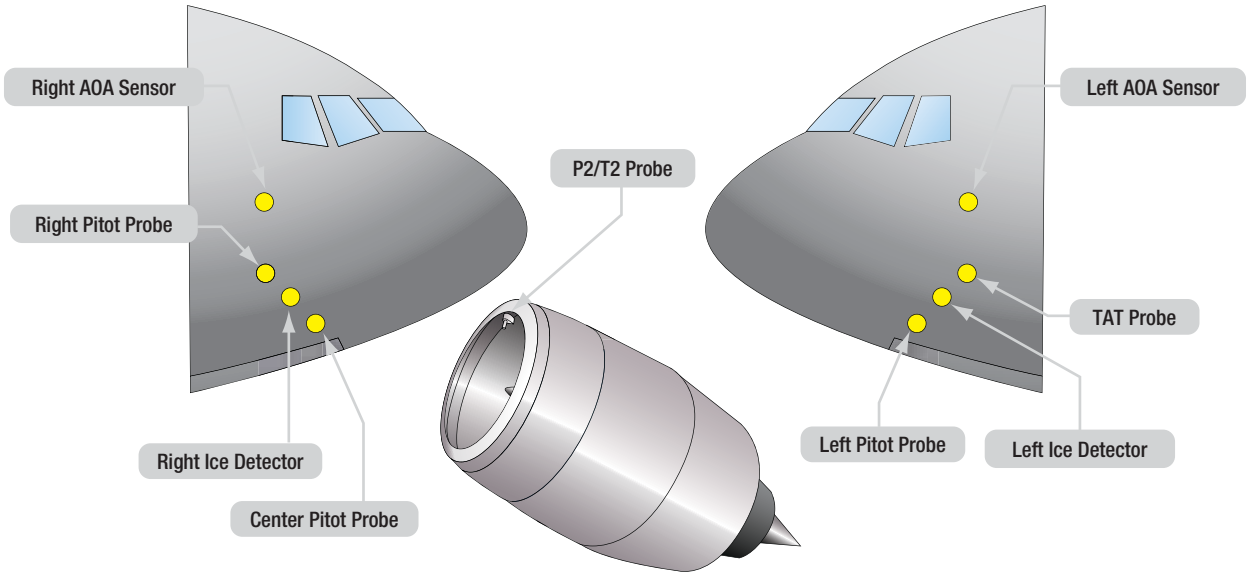


Figure 15-18. Probes with thermal electric anti-icing on one commercial airliner.

turn on and stay on until the temperature reaches a safe temperature. ***Figure 15-20*** is a schematic of a water supply line heater system, and ***Figure 15-21*** shows the location of the waste water tanks and heater blanket.

On modern aircraft, the particular heating device and the thermostat that controls it are line replaceable units and easily changed by the technician if inoperative. Drain mast electric heating elements are integral and require that the mast be replaced. Drain line heating elements are either flexible wrap type or integral. Consult the manufacturer's maintenance and parts manual for replacement information.

WINDSHIELD ANTI-ICE

High performance and transport category aircraft windshields are typically made of laminated glass, polycarbonate, or similar ply material. Typically clear vinyl plies are also included to improve performance characteristics. The laminations create the strength and impact resistance of the windshield assembly. These are critical feature for windshields as they are subject to a wide range of temperatures and pressures. They must also withstand the force of a 4 pound bird strike at cruising speed to be certified. The laminated construction facilitates the inclusion of electric heating elements into the glass layers, which are used to keep the windshield clear of ice, frost, and fog. The elements can be in the form of resistance wires or a transparent conductive material may be used as one of the window plies. To ensure enough heating is applied to the outside of the windshield, heating elements are placed on the inside of the outer glass ply. Windshields are typically bonded together by the application of pressure and heat without the use of cement. ***Figure 15-22*** illustrates the plies in one transport category aircraft windshield.

Whether resistance wires or a laminated conductive film is used, aircraft window heat systems have window heat control units to supply power and feedback mechanisms, such as thermistors, to provide the window heat control units with information used to keep operating temperature within acceptable limits. Most systems are automatic once switched with switches on the flight deck. Separate circuits for pilot and co-pilot are common to ensure visibility in case of a malfunction. Consult the manufacturer's maintenance information for details on the particular window heat system in question. Some windshield heating systems can be operated at two heat levels. On these aircraft, NORMAL heating supplied heat to the broadest area of windshield. HIGH heating supplies a higher intensity of heat to a smaller but more essential viewing area. Typically, this window heating system is always on and set in the NORMAL position.

Figure 15-23 illustrates a simplified windshield heat system of this type. Window heat anti-ice systems typically BITE test automatically. They are connected to an aircraft data bus for communication with the aircraft information management system. ***Figure 15-24*** is a synoptic diagram of a Boeing 777 window heat system. On this aircraft, the back-up window heat is always ON. When selecting window heat, the crew is actually asking for more heat from the window heat control units.

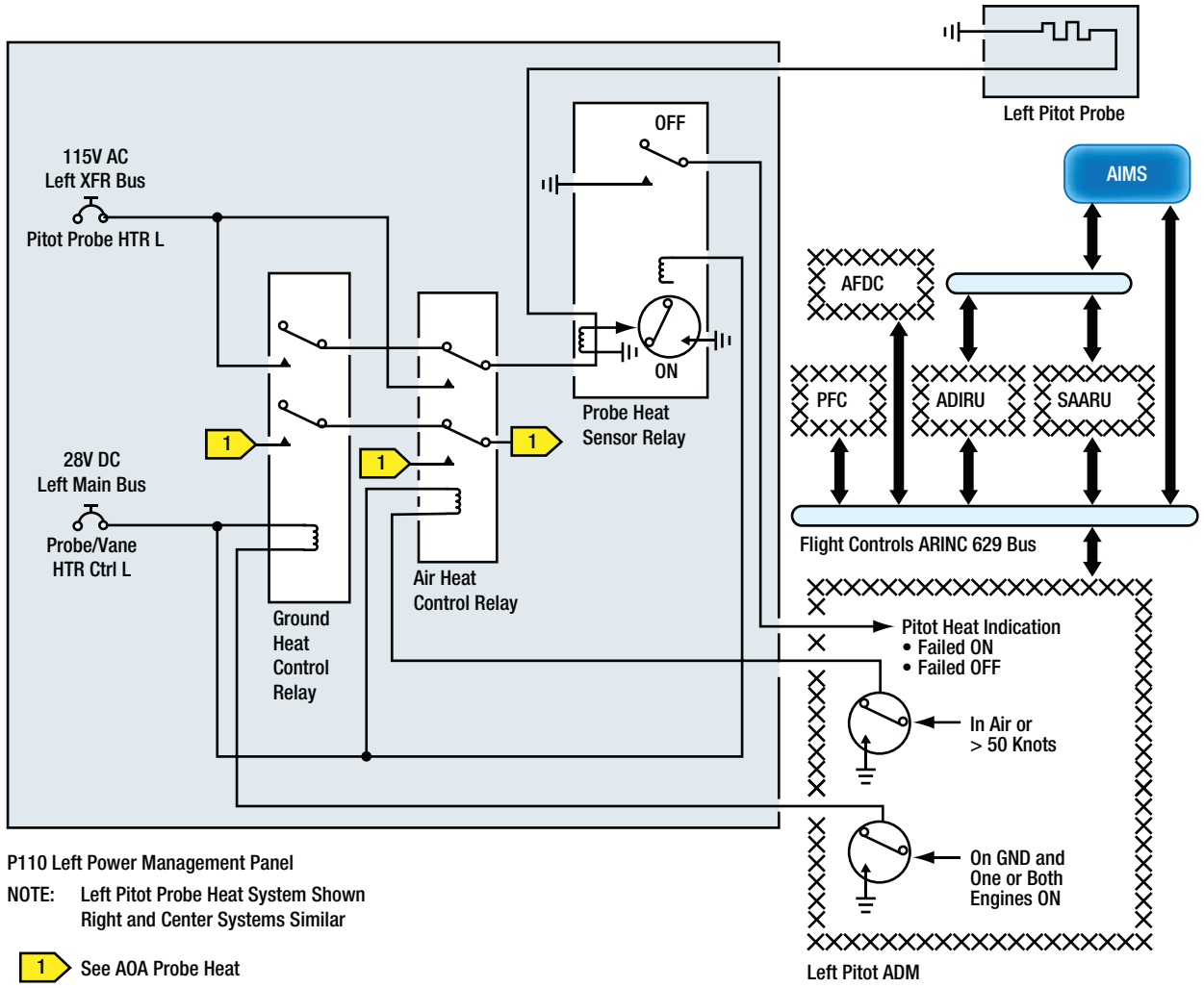


Figure 15-19. Pitot probe heat system.

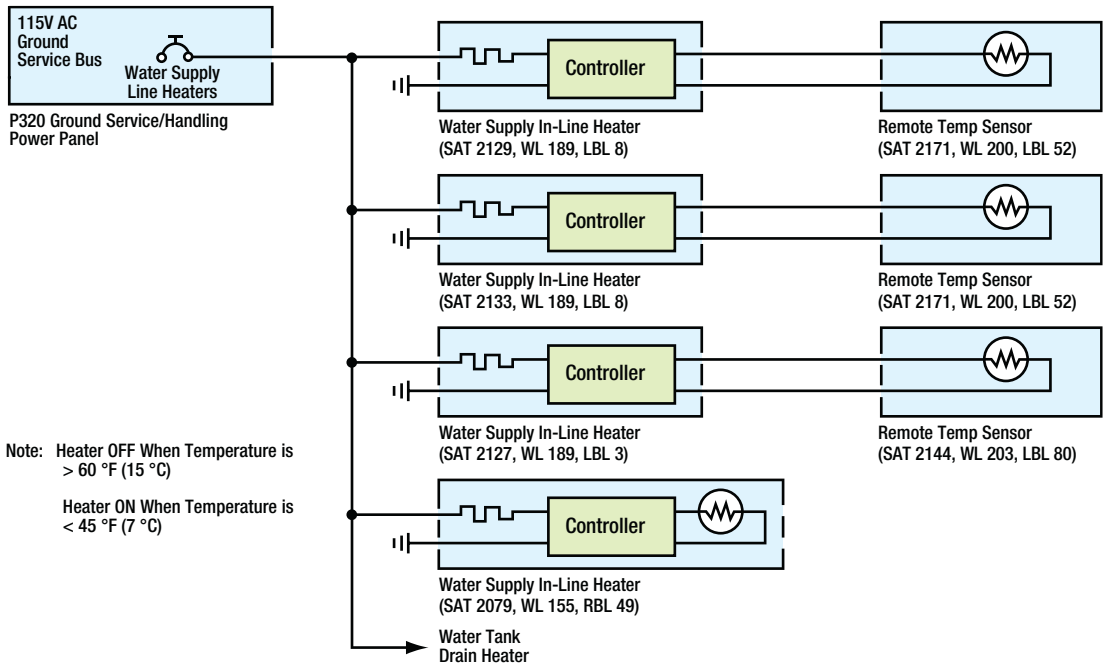


Figure 15-20. Water supply line heater system.

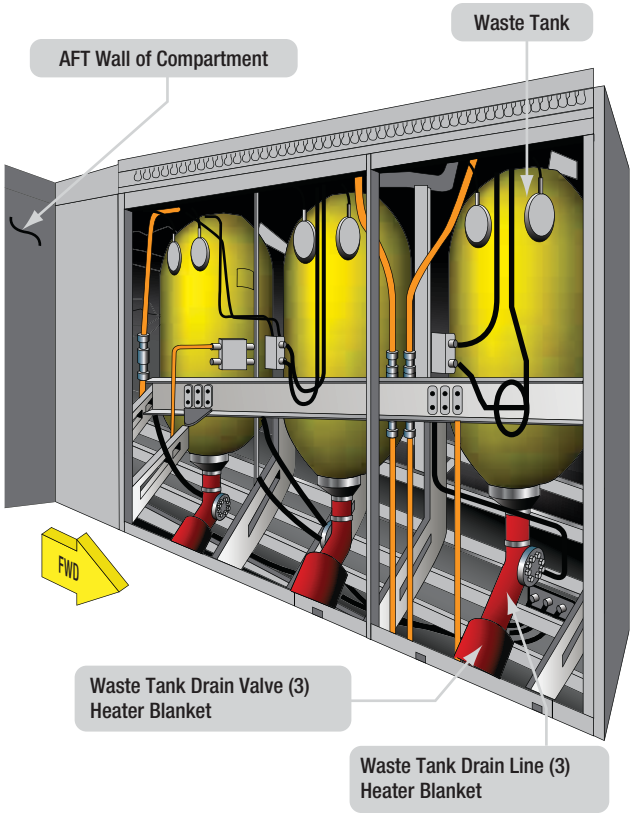


Figure 15-21. Waste water tanks and tank drain valve heater blankets.

PROPELLER ANTI-ICE

Many propellers use thermal electric boots to remove any ice that forms on the blades (de-ice). However some aircraft permit operation of the electric heating element boots to prevent ice from forming. In this case, the boots perform an anti-icing function. More information on this type of propeller ice control is given in the de-ice section of this sub-module.

CHEMICAL ANTI-ICING

Chemical anti-icing is used in some aircraft to anti-ice the leading edges of the wing, stabilizers, windshields, and propellers. The wing and stabilizer systems are often called weeping wing systems or are known by

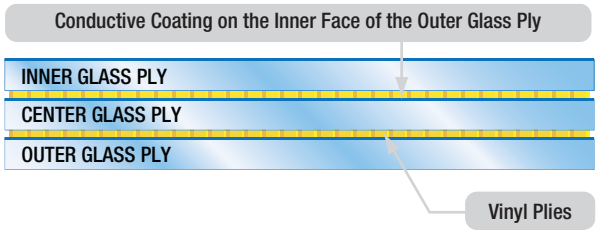


Figure 15-22. Cross-section of a transport category windshield.

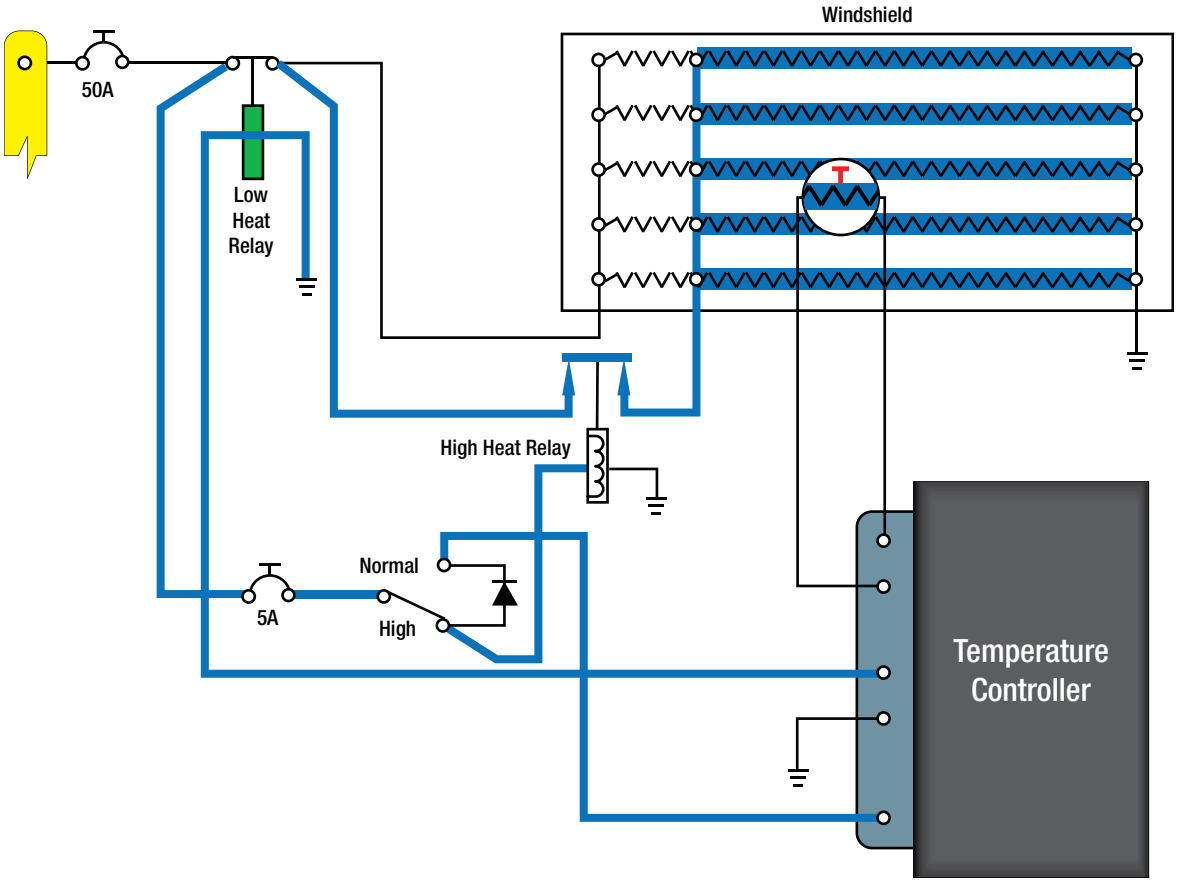


Figure 15-23. Electric windshield heat schematic.

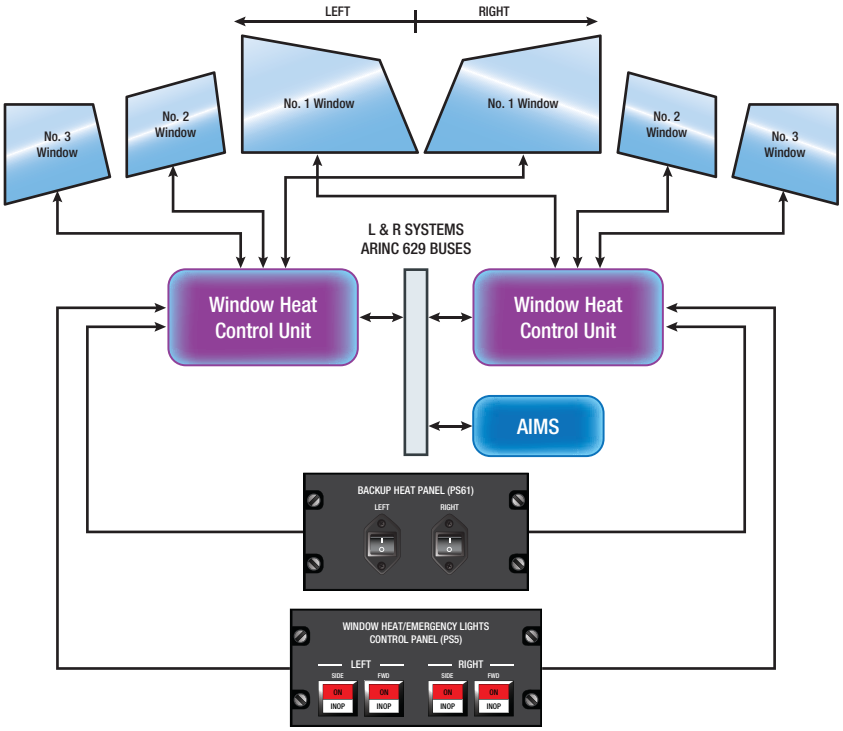


Figure 15-24. Electric window heat synoptic diagram for a Boeing 777.

their trade name of TKS™ systems. Ice protection is based upon the freezing point depressant concept. An antifreeze solution is pumped from a reservoir through a mesh screen embedded in the leading edges of the wings and stabilizers. Activated by a switch in the cockpit, the liquid flows over the wing and tail surfaces, preventing the formation of ice as it flows. The solution mixes with the supercooled water in the cloud, depresses its freezing point, and allows the mixture to flow off of the aircraft without freezing.

The system is designed to anti-ice, but it is also capable of deicing an aircraft as well. When ice has accumulated on the leading edges, the antifreeze solution chemically breaks down the bond between the ice and airframe. This allows aerodynamic forces to carry the ice away. Thus, the system clears the airframe of accumulated ice before transitioning to anti-ice protection. ***Figure 15-25*** shows a chemical anti-ice system. The TKS™ weeping wing system contains formed titanium panels that are laser drilled with over 800 tiny holes (.002 5 inch diameter) per square inch.

These are mated with non-perforated stainless steel rear panels and bonded to wing and stabilizer leading edges. As fluid is delivered from a central reservoir and pump, it seeps through the holes. Aerodynamic forces cause the fluid to coat the upper and lower surfaces of the airfoil. The glycol based fluid prevents ice from adhering to the aircraft structure. Some aircraft with weeping wing systems are certified to fly into known icing conditions.

Others use it as a hedge against unexpected ice encountered in flight. The systems are basically the same. Reservoir capacity permits 1-2 hours of operation. TKS™ weeping wings are used primarily on reciprocating aircraft that lack a supply of warm bleed air for the installation of a thermal anti-ice system. However, the system is simple and effective leading to its use on some turbine powered corporate aircraft as well.

GROUND APPLIED ANTI-ICE

When aircraft surfaces are contaminated by frozen moisture, they must be deiced prior to dispatch. When freezing precipitation exists and there is a risk of recontamination of the surface before takeoff, aircraft surfaces must be anti-iced.

Ground deicing of an aircraft is performed by a crew outside the aircraft equipped with spray equipment and often a boom truck to facilitate access. There are many formulas for deicing fluid. An ethylene glycol or propylene glycol based liquid is typical. It is mixed with hot water and sprayed on the aircraft being careful to avoid spraying it into critical areas of the aircraft such as:

- Engine inlets.
- Probes and ports.
- Air conditioning inlets and exits.
- APU inlet.
- Cooler and heat exchanger inlets.
- Fuel tank vents.

Type I de-ice fluid has few additives. It is an effective de-icing agent depending on the ratio of water to glycol and the temperature of the fluid when applied as well as the ambient conditions. A smooth thin layer of Type I fluid remains on the aircraft after the de-icing application which acts as an anti-icing agent. However, Type I fluid has low viscosity and its anti-ice capacity lasts only a few minutes.

Type II and Type IV de-ice fluids are commonly used on large turbine powered aircraft. Type II fluid is propylene glycol based fluid with molecular polymers added as a thickener. It becomes thixotropic fluid which becomes less viscous as stress is applied. As such, when applied to de-ice the aircraft, Type II becomes a thicker coat adhering to the airfoil surfaces and protects agains new ice, snow or frost from forming. As the aircraft airspeed increases during takeoff rollout and flight, the force of the air against the fluid layer decreases its viscosity and it is blown off.

Type IV fluid is similar to Type II fluid in that it has significant additives and leaves a thixotropic coating once applied. The anti-icing capabilities are greater for Type IV fluid which is evident by longer holdover times. A holdover time is the time between when the anti-ice fluid is applied and when it is no longer effective. This is discussed further in the section on ground deicing.

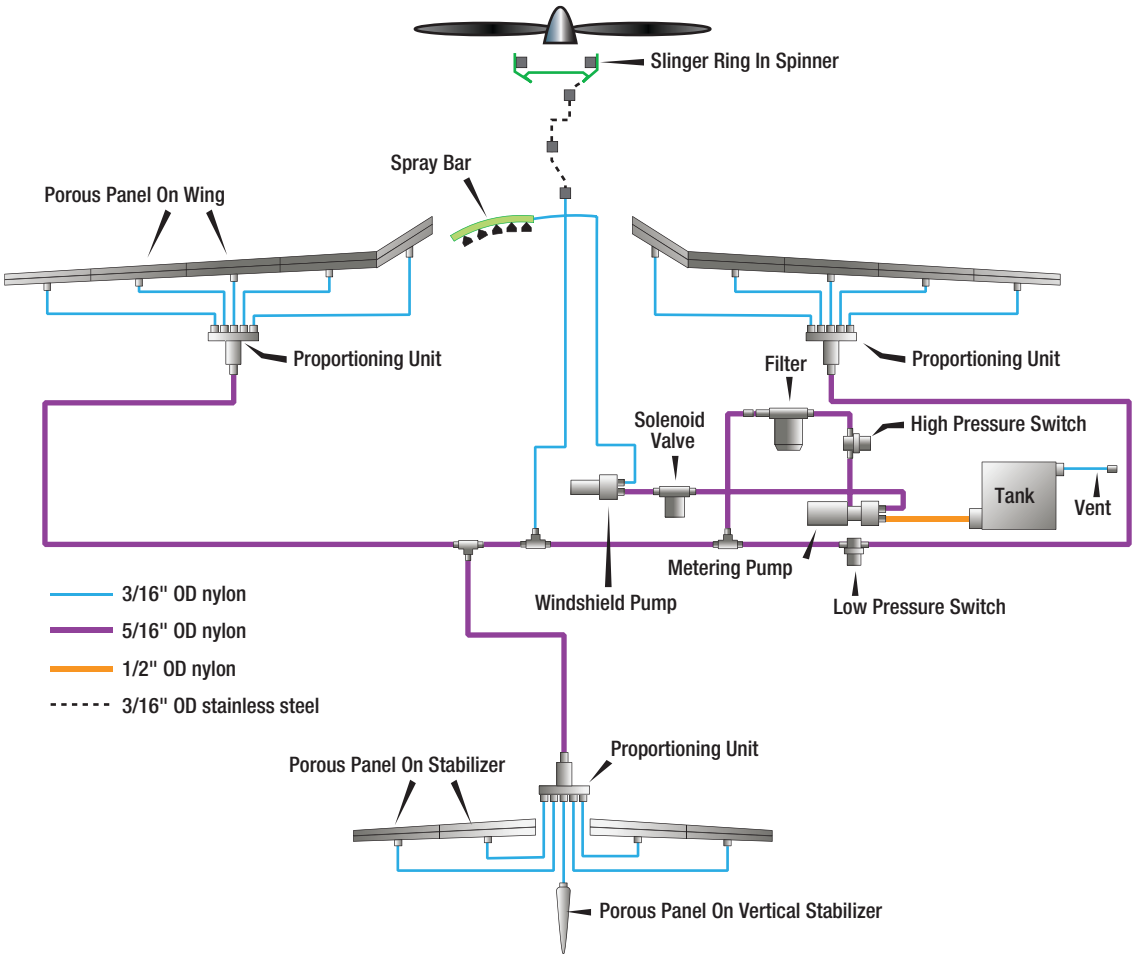


Figure 15-25. Chemical anti-ice system.

DE-ICING SYSTEMS

When ice, snow or frost are allowed to accumulate on aircraft surfaces and then are removed, the process is known as de-icing. Smaller turbine powered aircraft and reciprocating aircraft often incorporate de-ice systems rather anti-ice system although some aircraft my use a combination of de-ice and anti-ice for overall ice protection.

PNEUMATIC DE-ICE BOOTS

The most common means for de-icing wings and stabilizers on small turbine powered aircraft and reciprocating engine aircraft is with pneumatic de-ice boots. The leading edges of the wings and stabilizers have inflatable boots attached to them. The boots expand when inflated by pneumatic pressure, which breaks away ice accumulated on the boot. Most boots are inflated for

6 to 8 seconds. They are deflated by vacuum suction. The vacuum is continuously applied to hold the boots tightly against the aircraft while not in use.

CONSTRUCTION AND INSTALLATION OF DEICE BOOTS

Deicer boots are made of soft, pliable rubber, or rubberized fabric, and contain tubular air cells. The outer ply of the deicer boot is of conductive neoprene to provide resistance to deterioration by the elements and many chemicals. The neoprene also provides a conductive surface to dissipate static electricity charges. These charges, if allowed to accumulate, would eventually discharge through the boot to the metal skin beneath, causing static interference with the radio equipment. (***Figure 15-26***)

On modern aircraft, the deicer boots are bonded with an adhesive to the leading edge of the wing and tail surfaces. The trailing edges of this type boot are tapered to provide a smooth airfoil. Elimination of fairing strips, screws, and rivnuts used on older types of deicing boots reduces the weight of the deice system. The deicer boot air cells are connected to system pressure and vacuum lines by non-kinking flexible hose.

When gluing the deice boots to the leading edge of wings and stabilizers, the manufacturer's instruction must be strictly followed. The glue is typically a contact cement normally spread on both the airfoil and the boot and allowed to become tacky before mating the surfaces. Clean, paint free surfaces are required for the glue to adhere properly. Removal of old boots is performed by re-softening the cement with solvent.

SOURCES OF OPERATING AIR

The source of operating air for deice boot systems varies with the type of powerplant installed on the aircraft. Reciprocating engine aircraft typically use a dedicated engine driven air pump mounted on the accessory drive gear box of the engine.

The suction side of the pump is used to operate the gyroscopic instruments if installed on the aircraft. It is also used to hold the deice boots tight to the aircraft when they are not inflated. The pressure side of the pump supplies air to inflate the deice boots, which breaks up ice that has formed on the wing and stabilizer leading edges. The pump operates continuously. Valves, regulators, and switches in the cockpit are used to control the flow of source air to the system.

Turbine Engine Bleed Air

The source of deice boot operating air on turbine engine aircraft is typically bleed air from the engine compressor(s). A relatively low volume of air on an intermittent basis is required to operate the boots. This has little effect on engine power enabling use of bleed air instead of adding a separate engine driven air pump. Valves controlled by switches in the cockpit deliver air to the boots when requested.

RECIPROCATING ENGINE AIRCRAFT DE-ICE SYSTEMS

General aviation reciprocating engine aircraft, especially twin engine models, are commonly equipped with pneumatic deicer systems. Rubber boots are attached with glue to the leading edges of the wings and stabilizers. These boots have a series of inflatable

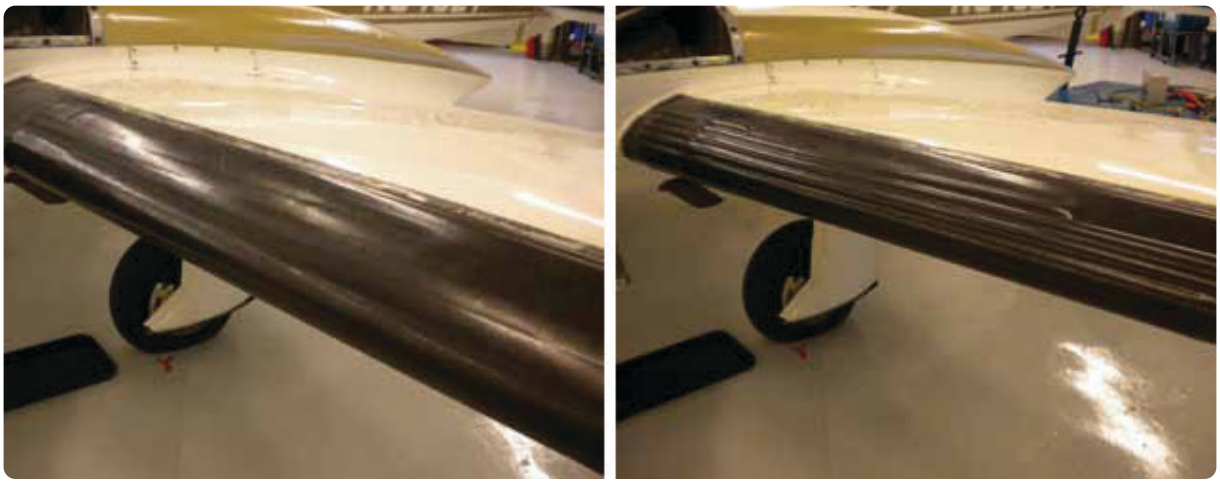


Figure 15-26. Deicing boots inflated (left) and deflated (right).

tubes. During operation, the tubes are inflated and deflated in an alternating cycle. (***Figure 15-27***) This inflation and deflation causes the ice to crack and break off. The ice is then carried away by the airstream. Boots used in GA aircraft typically inflate and deflate along the length of the wing.

In larger turbo prop aircraft, the boots are installed in sections along the wing with the different sections operating alternately and symmetrically about the fuselage. This is done so that any disturbance to airflow caused by an inflated tube is kept to a minimum by inflating only short sections on each wing at a time.

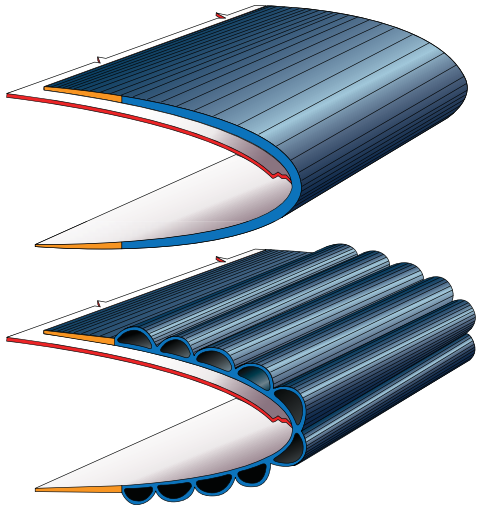


Figure 15-27. Pneumatic deicing system for a twin engine GA aircraft.

System Operation

Figure 15-28 shows a deice system used on a GA twin-engine aircraft with reciprocating engines. In normal flight, all of the components in the deice system are de-energized. Discharge air from the dry air pumps is dumped overboard through the deice control valves. The deflate valve is open connecting the deice boots to the suction side of the pump through the check valve manifold and the vacuum regulator. The gyroscopic instruments are also connected to the vacuum side of the dry air pump. The vacuum regulator is set to supply the optimum suction for the gyros, which is sufficient to hold the boots tightly against the airfoil surfaces.

When the switch shown in ***Figure 15-29*** is pushed ON, the solenoid operated deice control valves in each nacelle open and the deflate valve energizes and closes. Pressurized air from the discharge side of the pumps is routed through the control valves to the deice boot. When the system reaches 17 psi, pressure switches located on the deflate valve de-energize the deice control valve solenoids. The valves close and route pump air

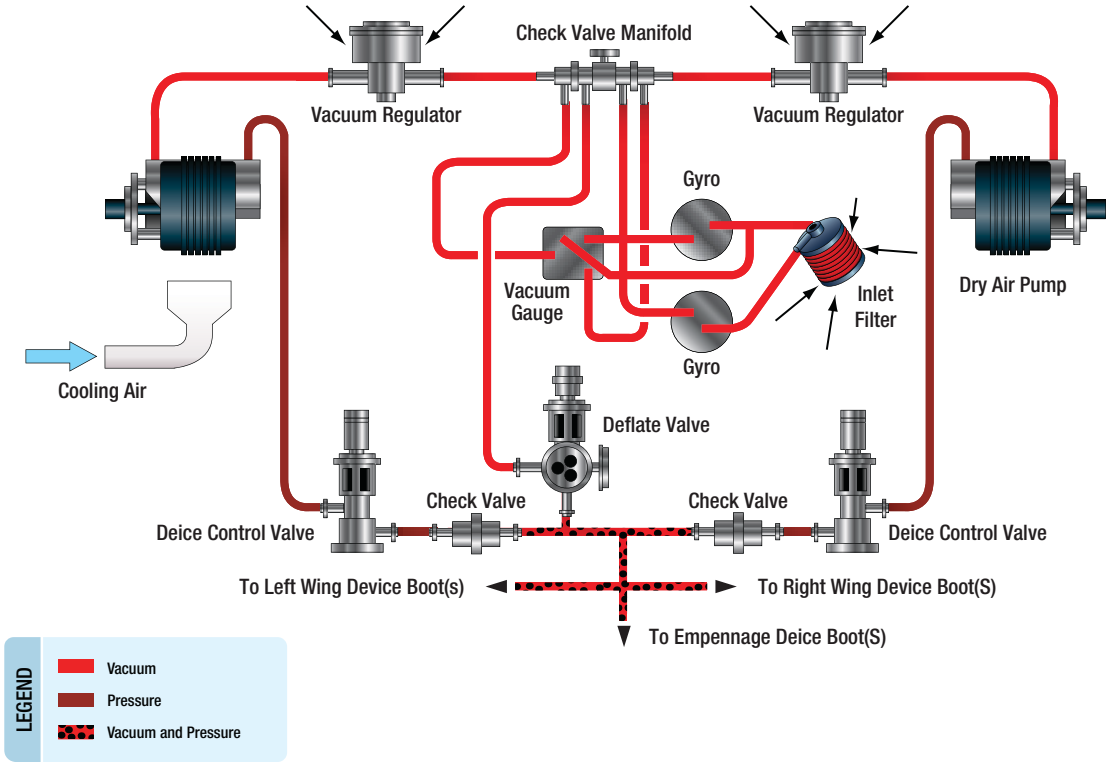


Figure 15-28. Pneumatic deicing system for a twin engine GA aircraft.



Figure 15-29. Wing deice switch.

output overboard. The deflate valve opens and the boots are again connected to vacuum. On this simple system, the pilot must manually start this inflation/deflation cycle by pushing the switch each time deice is required.

Larger aircraft with more complex systems may include a timer, which will cycle the system automatically until turned OFF. The use of distributor valves is also common. A distributor valve is a multi-position control valve controlled by the timer.

It routes air to different deice boots in a sequence that minimizes aerodynamic disturbances as the ice breaks of the aircraft. Boots are inflated symmetrically on each side of the fuselage to maintain control in flight while deicing occurs. Distributor valves are solenoid operated and incorporate the deflate valve function to reconnect the deice boots with the vacuum side of the pump after all have been inflated. Combining functional components of a deice system into a single unit is fairly common.

Figure 15-30 illustrates the right side of a large recip aircraft deice boot system. The left side is the same. In addition to the distributor valves, which combine functions of a control valve and deflate valve, the system also uses a combination unit. This unit combines the functions of a shutoff control valve for all pump supply air, as well as a pressure regulator for the system. It also contains a secondary air filter.

DEICE SYSTEM FOR TURBOPROP AIRCRAFT

Figure 15-31 shows a pneumatic deice system used on a turboprop aircraft. The source of pneumatic air is engine bleed air, which is used to inflate two inboard wing boots, two outboard boots, and horizontal stabilizer boots. Additional bleed air is routed through the brake de-ice valve to the brakes. A three-position switch controls the operation of the boots. This switch is spring loaded to the center OFF position. When ice has accumulated, the switch should be selected to the single-cycle (up) position and released. (***Figure 15-32***)

Pressure regulated bleed air from the engine compressors supply air through bleed air flow control units and pneumatic shutoff valves to a pneumatic control assembly that inflates the wing boots. After an inflation period of 6 seconds, an electronic timer switches the distributor in the control assembly to deflate the wing boots, and a 4-second inflation begins in the horizontal stabilizer boots. After these boots have been inflated and deflated, the cycle is complete, and all boots are again held down tightly against the wings and horizontal stabilizer by vacuum.

The spring loaded switch must be selected up again for another cycle to occur. Each engine supplies a common bleed air manifold. To ensure the operation of the system, if one engine is inoperative, a flow control unit with check valve is incorporated in the bleed air line from each engine to prevent the loss of pressure through the compressor of the inoperative engine. If the boots fail to function sequentially, they may be operated manually by selecting the DOWN position of the same de-ice cycle switch.

Depressing and holding it in the manual DOWN position inflates all the boots simultaneously. When the switch is released, it returns to the (spring-loaded) OFF position, and each boot is deflated and held by vacuum. When operated manually, the boot should not be left inflated for more than 7 to 10 seconds, as a new layer of ice may begin to form on the expanded boots and become unremovable. If one engine is inoperative, the loss of its pneumatic pressure does not affect boot operation. Electric power to the boot system is required to inflate the boots in either single cycle or manual operation. When electric power is lost, the vacuum holds the boots tightly against the leading edge.

DEICING SYSTEM COMPONENTS

Several components are used to construct all de-ice boot systems. The components may differ slightly in name and location within the system depending on the aircraft. Components may also combine functions to save space and weight. The basic functions of filtering, pressure regulation, distribution, and attachment to a vacuum when boots are not in use must all be present.

Check valves must also be installed to prevent back flow in the system. Manifolds are common on multi-engine aircraft to allow sourcing of low pressure air from both engine pumps. Note that air-pump pressure is typically expelled overboard when not needed. Bleed air is shut off by a valve when not needed for deice boot operation on turbine engine aircraft. A timer, or control unit with an automatic mode, exists on many aircraft to repeat the deice cycle periodically.

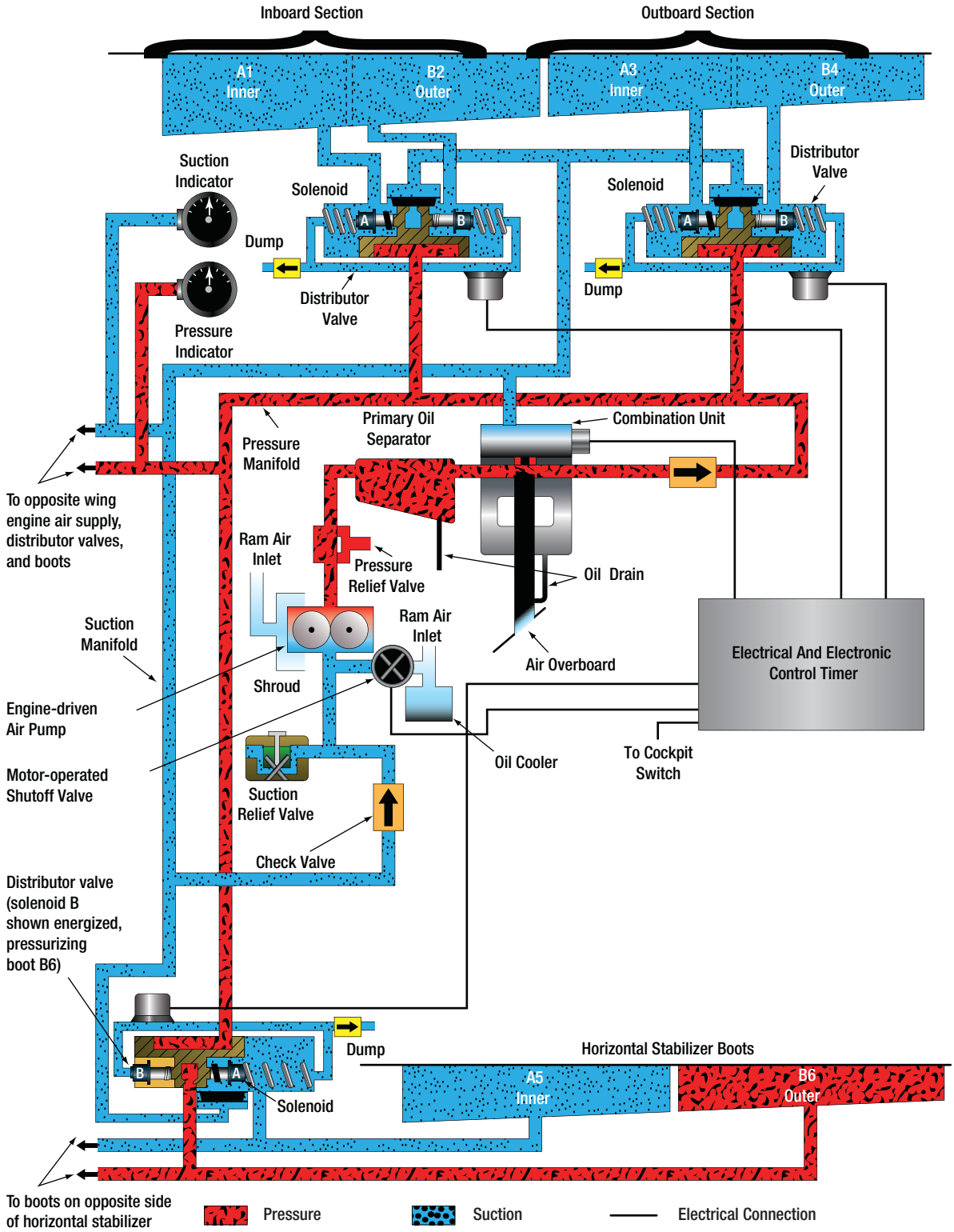


Figure 15-30. Right-side deice boot system on a large aircraft (left side similar).

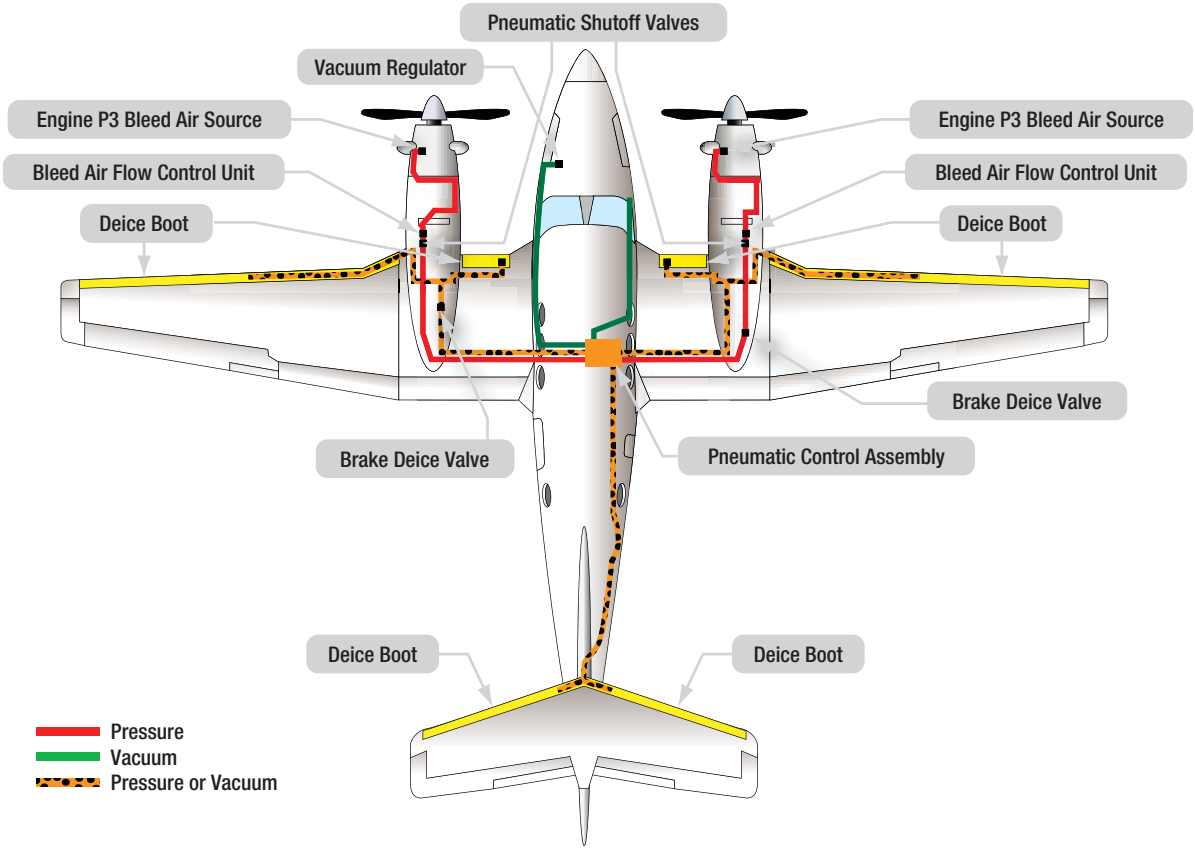


Figure 15-31. Wing de-ice system for turboprop aircraft.

Wet-Type Engine-Driven Air Pump

To provide pressure for the de-ice boots, older aircraft may use a wet-type engine driven air pump mounted on the accessory drive gear case of the engine. Some modern aircraft may also use a wet type air pump because of its durability. The pump is typically a four vane, positive displacement pump. Engine oil passes from the accessory case through the pump mounting base flange to lubricate the pump. Some of the oil is entrained in the output air and must be removed by an oil separator before it is sent through other components in the deice system. When installing a wet type pump, care should be taken to ensure that the oil passage in the gasket, pump, and mounting flange are aligned to ensure lubrication. (***Figure 15-33***)



Figure 15-32. Ice protection panel on a turboprop aircraft with deice boots.

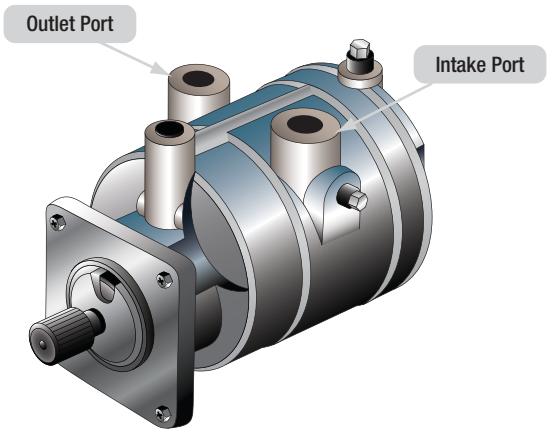


Figure 15-33. A wet type air pump with engine oil lubricating ports in the mounting flange.

Dry Type Engine Driven Air Pump

Most modern GA aircraft are equipped with a dry-type engine-driven air pump. It is also mounted on the engine accessory drive case; however, it is not lubricated with engine oil. The pump is constructed with carbon rotor vanes and bearings. The carbon material wears at a controlled rate to provide adequate lubrication without the need for oil. This keeps output air oil-free; thus, the use of an oil separator is not required. Caution should be used to prevent oil, grease, or degreasing fluids from entering the pump or the air system to ensure proper pump and system operation. (***Figure 15-34***)

Dry type and wet-type pumps are virtually maintenance free. Mounting bolts should be checked for security as should all hose connections. Wet-type pumps have a longer time before requiring overhaul, but dry-type pumps give the assurance that the deice system will not be contaminated with oil.

Oil Separator

An oil separator is required for each wet type air pump. The pump output air flows through the separator where most of the oil is removed and sent back to the engine though a drain line. Some systems may include a secondary separator to ensure oil free air is delivered to the deice system. There are no moving parts in an oil separator. A convoluted interior allows the air to pass, while the oil condenses and drains back to the engine. The only maintenance required on the separator is flushing the interior of the unit with a specified solvent. This should be done at intervals prescribed in the applicable maintenance manual. (***Figure 15-35***)



Figure 15-34. Dry-type engine-driven air pump.

Control Valve

A control valve is a solenoid operated valve that allows air from the pump to enter the deice system. When energized by the deice switch in the cockpit, the valve opens. The control valve dumps pump air overboard when the deice system is not in use. Many control valves are built in combination with pressure relief valves that keeps the deice system safe from over pressure. (***Figure 15-36***)

Deflate Valve

All deice boot systems require a means for connecting vacuum from the air pump to the boots when the boots are not in use. This ensures the boots are held tightly deflated against the aircraft structure to provide the significant change in size and shape needed to break off accumulated ice when the boots inflate. One single deflate valve is used on simple de-ice boot systems. The deflate valve is solenoid operated. It is located at a point in the system where when closed, air is delivered to the boots. When open, vacuum is applied. Often, the deflate function is built into another unit, such as a distributor valve discussed next.

Distributor Valve

A distributer valve is a type of control valve used in relatively complex de-ice boot systems. It is an electrically-operated solenoid valve controlled by the deice boot system timer or control unit. On some systems, a distributor valve is assigned to each set of deice boots



Figure 15-35. An oil separator used with a wet-type engine-driven air pump.

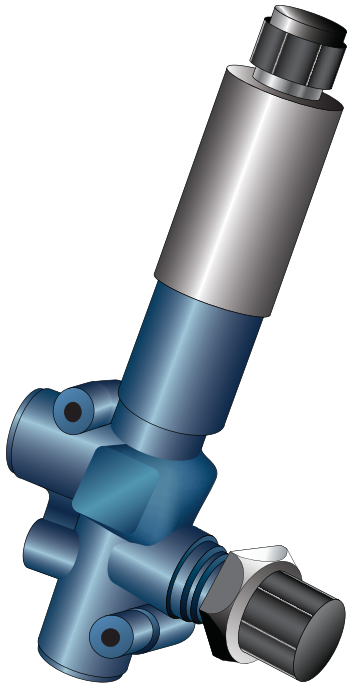


Figure 15-36. A solenoid operated deice control valve.

it controls. It differs from a control valve in that it has the deflate valve function built into it. Therefore, the distributor valve transfers connection of the boots from the pressure side of the air pump to the vacuum side of the pump once the proper inflation time has elapsed. The valve also dumps the unneeded air from the pump overboard. Another type of distributor valve exists that handles the inflation and deflation of numerous sets of deice boots in a single unit. It also connects the boots to vacuum and dumps pump air when deice is not needed. A servo motor is used to position the multi-position valve.

These centralized units are controlled by a timer or control unit. They inflate and deflate all of the boots on the aircraft. The timer may be built into the unit on some models.

Timer/Control Unit

All but the simplest of deice systems contain a timer or control unit. This device controls the action of the distributor valve(s) to ensure all boots are inflated in the proper sequence and for the correct duration. Six seconds of inflation is common to break off accumulated ice. The boot then must be immediately deflated so that ice does not adhere to the inflated geometry of the boot. This could cause it to fail to deflate or break off ice when the boot is re-inflated. The timer, or control unit, can also be made to cycle through the inflation and deflation of all boots periodically, thus relieving the flight crew of repetitive manual activation of the system. The function and capabilities of timers and control units vary. Consult the manufacturer's maintenance information for the performance characteristics of the timer/control unit on the aircraft in question.

The timer, or control unit, may be an independent device, or it may be built in as part of another deice system component, such as a central distribution valve.

NOTE: A modern system design may use a pressure switch to signal deflation of the deice boots. When pressure builds in the boots to a preset amount, the switch signals the control valve to close and connect the boots to vacuum. However, this system retains a control unit for automatic cycling of the system at a set time interval.

Regulators and Relief Valves

Both the pressure and vacuum developed by an air pump must be regulated for use in the deice boot system. Typical boot inflation air pressure is between 15 and 20 psi. Vacuum pressure is set for the requirements of the gyroscopic instruments operated by the vacuum side of the air pump. Measured in inches of mercury, normal vacuum pressure (suction) is 4.5 to 5.5 "Hg.

De-ice boot system air pressure is controlled by a pressure regulator valve located somewhere in the system downstream of the pump or oil separator, if installed. The regulator may be a stand-alone unit, or it may be combined into another deice system component. Regardless, the spring loaded valve relieves pressure overboard when it exceeds the limit for which the system is designed.

A vacuum regulator is installed in the vacuum manifold on the suction side of the air pump to maintain the vacuum at the designed level. Also known as a suction regulating valve or similar, the spring loaded valve contains a filter for the ambient air drawn through the valve during operation. This filter must be changed or kept clean per manufacturer's instructions. (***Figure 15-37***)

Manifold Assembly

In all pneumatic deice boot systems, it is necessary for check valves to be installed to prevent back flow of air in the system. The location(s) depend on system design. Sometimes, the check valve is built into another system



Figure 15-37. A vacuum regulator.



Figure 15-38. A manifold assembly used in multiengine aircraft deice systems.

component. On twin engine aircraft, it is common to unite the air supplied from each engine driven pump to provide redundancy.

Check valves are required to guard against backflow should one pump fail. A manifold assemble is commonly used to join both sides of the system. (***Figure 15-38***) It contains the required check valves in a single assembly.

Inlet Filter

The air used in a deice boot system is ambient air drawn in upstream of the gyroscopic instruments on the suction side of the engine driven air pump. This air must be free of contaminants for use spinning the gyros, as well as for inflation of the de-ice boots. To ensure clean air, an inlet filter is installed as the air intake point for the system. This filter must be regularly maintained as per manufacturer's instructions. ***Figure 15-39*** shows a typical inlet air filter. ***Figure 15-40*** shows the relationship of the vacuum regulator and inlet air filter to other system components.

INSPECTION, MAINTENANCE, AND TROUBLESHOOTING OF RUBBER DEICER BOOT SYSTEMS

Maintenance on pneumatic deicing systems varies with each aircraft model. The instructions of the airframe or system components manufacturer should be followed in all cases. Depending on the aircraft, maintenance usually consists of operational checks, adjustments, troubleshooting, and inspection.

Operational Checks

An operational check of the system can be made by operating the aircraft engines or by using an external source of air. Most systems are designed with a test plug to permit ground checking the system without operating the engines. When using an external air source, make certain that the air pressure does not exceed the test pressure established for the system. Before turning the deicing system on, observe the vacuum operated instruments.

If any of the gauges begin to operate, it is an indication that one or more check valves have failed to close and that reverse flow through the instruments is occurring. Correct the difficulty before continuing the test. If no movement of the instrument pointers occurs, turn on the deicing system.

With the deicer system controls in their proper positions, check the suction and pressure gauges for proper indications. The pressure gauge fluctuates as the deicer tubes inflate and deflate. A relatively steady reading should be maintained on the vacuum gauge. It should



Figure 15-39. Air filter for vacuum system.

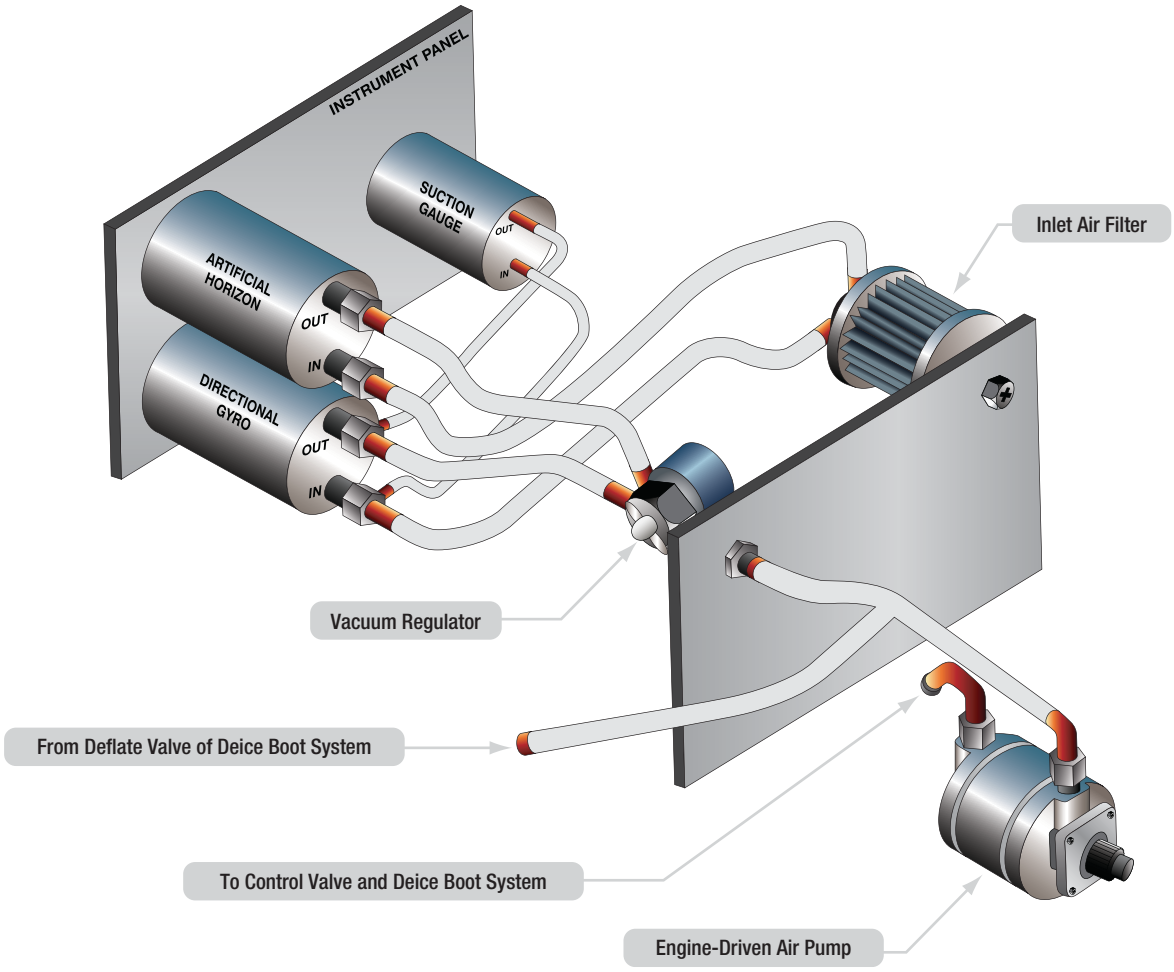


Figure 15-40. Location of the inlet air filter in relationship to other components in a gyro instrument/pneumatic deice boot system.

be noted that not all systems use a vacuum gauge. If the operating pressure and vacuum are satisfactory, observe the deicers for actuation. With an observer stationed outside the aircraft, check the inflation sequence to be certain that it agrees with the sequence indicated in the aircraft maintenance manual.

Check the timing of the system through several complete cycles. If the cycle time varies more than is allowable, determine the difficulty and correct it. Inflation of the deicers must be rapid to provide efficient deicing. Deflation of the boot being observed should be completed before the next inflation cycle. (***Figure 15-41***)

Adjustments

Examples of adjustments that may be required include adjusting the deicing system control cable linkages, adjusting system pressure relief valves, and deicing system vacuum(suction) relief valves. A pressure relief valve acts as a safety device to relieve excess pressure in the event of regulator valve failure.

To adjust this valve, operate the aircraft engines and adjust a screw on the valve until the deicing pressure gauge indicates the specified pressure at which the valve should relieve. Vacuum relief valves are installed in a system that uses a vacuum pump to maintain constant suction during varying vacuum pump speeds. To adjust a vacuum relief valve, operate the engines. While watching the vacuum (suction) gauge, an assistant should adjust the suction relief valve adjusting screw to obtain the correct suction specified for the system.

Troubleshooting

Not all troubles that occur in a deicer system can be corrected by adjusting system components. Some troubles must be corrected by repair or replacement of system components or by tightening loose connections. Several troubles common to pneumatic deicing systems are shown in the left-hand column of the chart in ***Figure 15-42***.

Note the probable causes and the remedy of each trouble listed in the chart. In addition to using troubleshooting



Figure 15-41. Test equipment used to test a wing deice system (left), and test equipment installed in the aircraft for testing (right).

charts, operational checks are sometimes necessary to determine the possible cause of trouble.

Inspection

During each preflight and scheduled inspection, check the deicer boots for cuts, tears, deterioration, punctures, and security; during periodic inspections, go a little further and check deicer components and lines for cracks. If weather cracking of rubber is noted, apply a coating of conductive cement. The cement, in addition to sealing the boots against weather, dissipates static electricity so that it does not puncture the boots by arcing to the metal surfaces.

Deice Boot Maintenance

The life of the deicers can be greatly extended by storing them when they are not needed and by observing these rules when they are in service:

1. Do not drag gasoline hoses over the deicers.
2. Keep deicers free of gasoline, oil, grease, dirt, and other deteriorating substances.
3. Do not lay tools on or lean maintenance equipment against the deicers.
4. Promptly repair or resurface the deicers when abrasion or deterioration is noted.
5. Wrap deice boots in paper or canvas when storing.

Thus far, preventive maintenance has been discussed. The actual work on the deicers consists of cleaning, resurfacing, and repairing. Cleaning should ordinarily be done at the same time the aircraft is washed, using a mild soap and water solution. Grease and oil can be removed with a cleaning agent, such as naptha, followed by soap and water scrubbing. Whenever the degree of wear is such that it indicates that the electrical conductivity of the deicer surface has been destroyed, it may be necessary to resurface the deicer. The resurfacing substance is a black, conductive neoprene cement. Prior to applying the resurfacing material, the deicer must be cleaned thoroughly and the surface roughened. Cold patch repairs can be made on a damaged deicer.

The deicer must be relieved of its installed tension before applying the patch. The area to be patched must be clean and buffed to roughen the surface slightly. Patches are glued in place. Follow manufacturer's instructions for all repairs.

ELECTRICAL DE-ICING SYSTEMS

De-icing is accomplishes with electricity on propellers and, occasionally on airfoils with electric de-ice boots.

ELECTRIC PROPELLER DE-ICE

The formation of ice on the propeller leading edges, cuffs, and spinner reduces the efficiency of the powerplant system. Deice systems using electrical heating elements are common.

Many propellers are deiced by an electrically heated boot on each blade. The boot, firmly cemented in place, receives current from a slip ring and brush assembly on the spinner bulkhead. The slip ring transmits current to the deice boot. The centrifugal force of the spinning propeller and air blast breaks the ice particles loose from the heated blades. (***Figure 15-43***)

On one aircraft model, the boots are heated in a preset sequence, which is an automatic function controlled

Problem	Causes (most of which can be identified with a 343 Test Kit)	Corrective Action(s)
Boots Do Not Inflate	• Open circuit breaker • Faulty deflate valve Solenoid inoperable: 1. Improper voltage at solenoid 2. Blocked air vent in solenoid 3. Inoperative plunger Diaphragm not seated 1. Blocked vent orifice located in rivet bottom at center of diaphragm 2. Dirty diaphragm seal area 3. Diaphragm ruptured • Two faulty deice control valves of faulty two-stage regulators • Faulty check valve • Relay not functioning • Leak in system boots	• Reset circuit breaker • Check deflate valves as follows: Solenoid inoperable: 1. Correct electrical system 2. Clean with alcohol or replace 3. Clean with alcohol or replace Diaphragm not seated 1. Clean with .010 diameter wire and alcohol 2. Clean with blunt instrument and alcohol 3. Replace valve • Clean or replace valve assembly as noted above • Replace check valve • Check wiring or replace relay • Repair as needed
Slow Boot Inflation	• Lines blocked or disconnected • Low air pump capacity • One or more deice control valves not functioning properly • Deflate valve not fully closed • Ball check in deflate valve inoperative • Leaks in system or boots	• Check and replace lines • Replace air pump • Clean or replace valve assembly as noted above • Clean or replace valve assembly as noted above • Clean check valve or replace deflate valve • Repair as needed
System Will Not Cycle	• Pressure in system not reaching specified psi to activate pressure switch • Leak in system or boots • Pressure switch on deflate valve inoperative	• Clean or replace deice control valve as noted above • Clean or replace deflate valve, as noted above • Repair as needed, tighten all hose connections • Replace switch
Slow Deflation	• Low vacuum • Faulty deflate valve (indicated by temporary reduction in suction gauge reading)	• Repair as needed • Clean or replace valve assembly as noted above
No Vacuum For Boot Hold Down	• Malfunctioning deflate valve or deice valve • Leak in system or boots	• Clean or replace valve assembly as noted above • Repair as needed
Boots Will Not Deflate During Cycle	• Faulty deflate valve	• Check and replace valve
Boots Appear To Inflate On Aircraft Climb	• Vacuum source for boot holddown inoperative • Lines running through pressurized cabin loose or disconnected	• Check operation of ball check in deflate valve • Check for loose or disconnected vacuum lines and repair

Figure 15-42. Troubleshooting guide for wing deice system.

by a timer. This sequence is as follows: 30 seconds for the right prop outer elements; 30 seconds for the right prop inner elements; 30 seconds for the left prop outer elements; and, 30 seconds for the left prop inner elements. Once the system is turned on for automatic is activated, it cycles continuously. A manual bypass of the timer is incorporated. (***Figure 15-44***)

ELECTRIC AIRFOIL DE-ICE BOOTS

A few modern aircraft are equipped with electric deice boots on wing sections or on the horizontal stabilizer. These boots contain electric heating elements which are bonded to the leading edges similarly to pneumatic deice boots. When activated, the boots heat up and melt the ice off of leading edge surfaces. The elements are controlled by a sequence timer in a deice controller. Ice detector and ram air temperature probe inputs initiate operation when other flight condition parameters exist. The boot elements turn ON and OFF in paired sections to avoid aerodynamic imbalance. The system is inoperative while the aircraft is on the ground.

HOT AIR DE-ICE/DE-FOG

Some laminated windshields on older aircraft have a space between the plies that allows the flow of hot air to be directed between the glass to keep it warm and fog free. The source of air is bleed air or conditioned air from the environmental control system. Small aircraft may utilize ducted warm air, which is release to flow over the windshield inner surface to defrost and defog. These

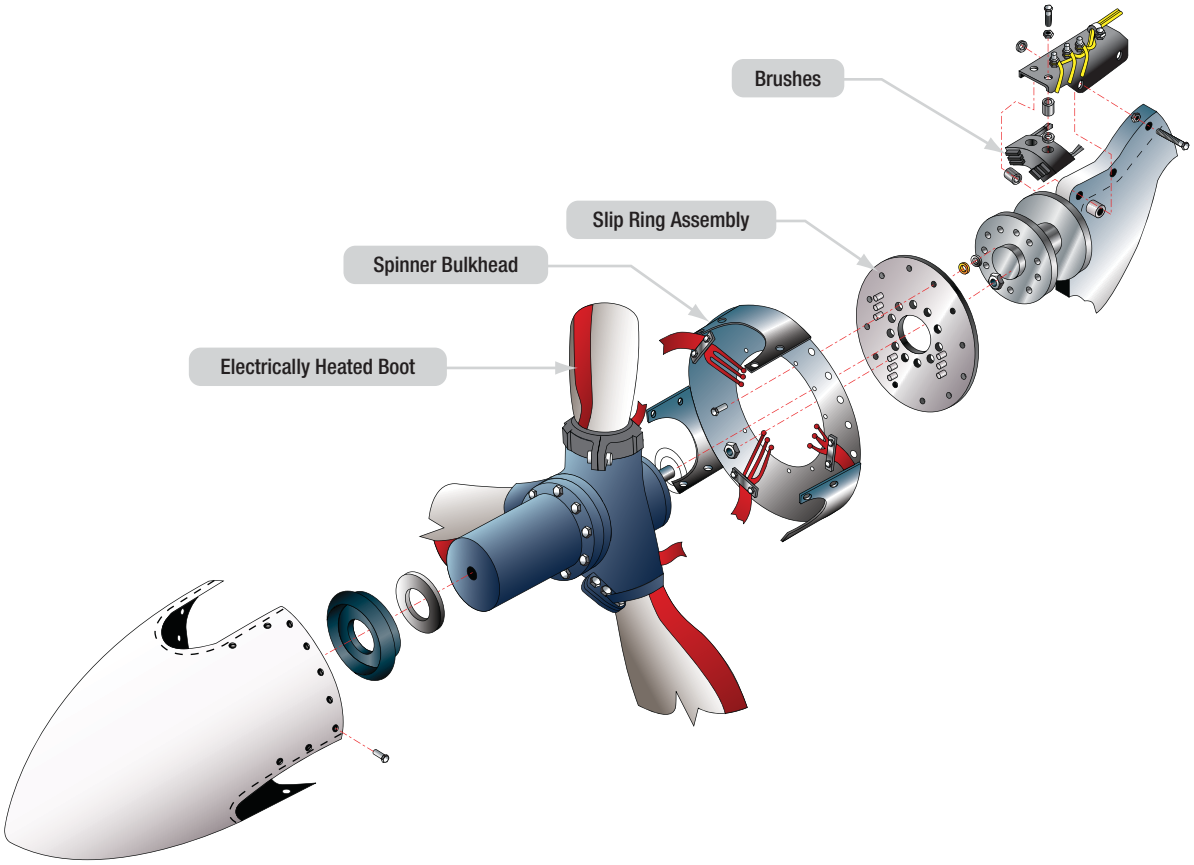


Figure 15-43. Electro thermal propeller deice system components.

systems are similar to those used in automobiles. The source of air could be ambient (defog only), the aircraft's heating system, or a combustion heater. While these pneumatic windshield heat systems are effective for de-icing/defogging the aircraft on which they are installed, they are not approved for flying into known icing conditions and, as such, are not effective for anti-ice.

Note that large aircraft equipped with pneumatic jet blast rain repellent systems achieve some hot air anti-icing effects from operating this system although electric windshield heat is usually used.

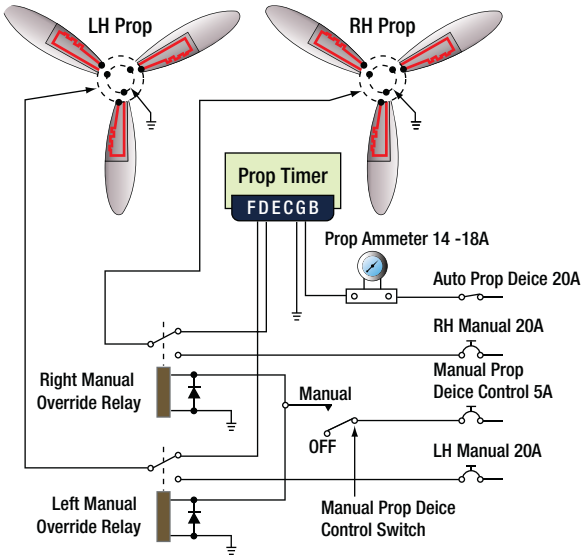


Figure 15-44. Propeller electrical deice system schematic.

CHEMICAL DE-ICE SYSTEMS

WINDSHIELD CHEMICAL DE-ICE

As previously mentioned in this chapter, chemical anti-ice systems exist generally for small aircraft. This type of anti-ice is also used on windshields. (***Figure 15-45***)

Whether alone or part of a TKS™ system or similar, the liquid chemical is sprayed through a nozzle onto the outside of the windshield which prevents ice from forming. The chemical can also de-ice the windshield of ice that may have already formed. Systems such as these have a fluid reservoir, pump, control valve, filter, and relief valve. Other components may exist. ***Figure 15-46*** shows a set of spray tubes for application of chemical anti-ice on an aircraft windshield.

CHEMICAL PROPELLER DE-ICE

Some aircraft models, especially single-engine GA aircraft, use a chemical deicing system for the propellers. Ice usually appears on the propeller before it forms on the wing. The glycol-based fluid is metered from a tank by a small electrically driven pump through a micro filter to the slinger rings on the prop hub. The propeller system can be a stand-alone system, or it can be part of a chemical wing and stabilizer deicing system such as the TKS™ weeping system.

CHEMICAL GROUND DEICING OF AIRCRAFT

The presence of ice on an aircraft may be the result of direct precipitation, formation of frost on integral fuel tanks after prolonged flight at high altitude, or accumulations on the landing gear following taxiing through snow or slush. In accordance with the Federal Aviation Administration (FAA) Advisory Circular (AC) 120-60, the aircraft must be free of all frozen contaminants adhering to the wings, control surfaces, propellers, engine inlets, or other critical surfaces before takeoff.

Any deposits of ice, snow, or frost on the external surfaces of an aircraft may drastically affect its performance. This may be due to reduced aerodynamic lift and increased aerodynamic drag resulting from the disturbed airflow over the airfoil surfaces, or it may be due to the weight of the deposit over the whole aircraft. The operation of an aircraft may also be seriously affected by the freezing of moisture in controls, hinges, valves, microswitches, or by the ingestion of ice into the engine. When aircraft are hangared to melt snow or frost, any melted snow or ice may freeze again if the aircraft is subsequently moved into subzero temperatures. Any measures taken to remove frozen deposits while the aircraft is on the ground must also prevent the possible refreezing of the liquid.

Frost Removal

Frost deposits can be removed by placing the aircraft in a warm hangar or by using a frost remover or deicing fluid. These fluids normally contain ethylene glycol and isopropyl alcohol and can be applied either by spray or by hand. It should be applied within 2 hours of flight. Deicing fluids may adversely affect windows or the

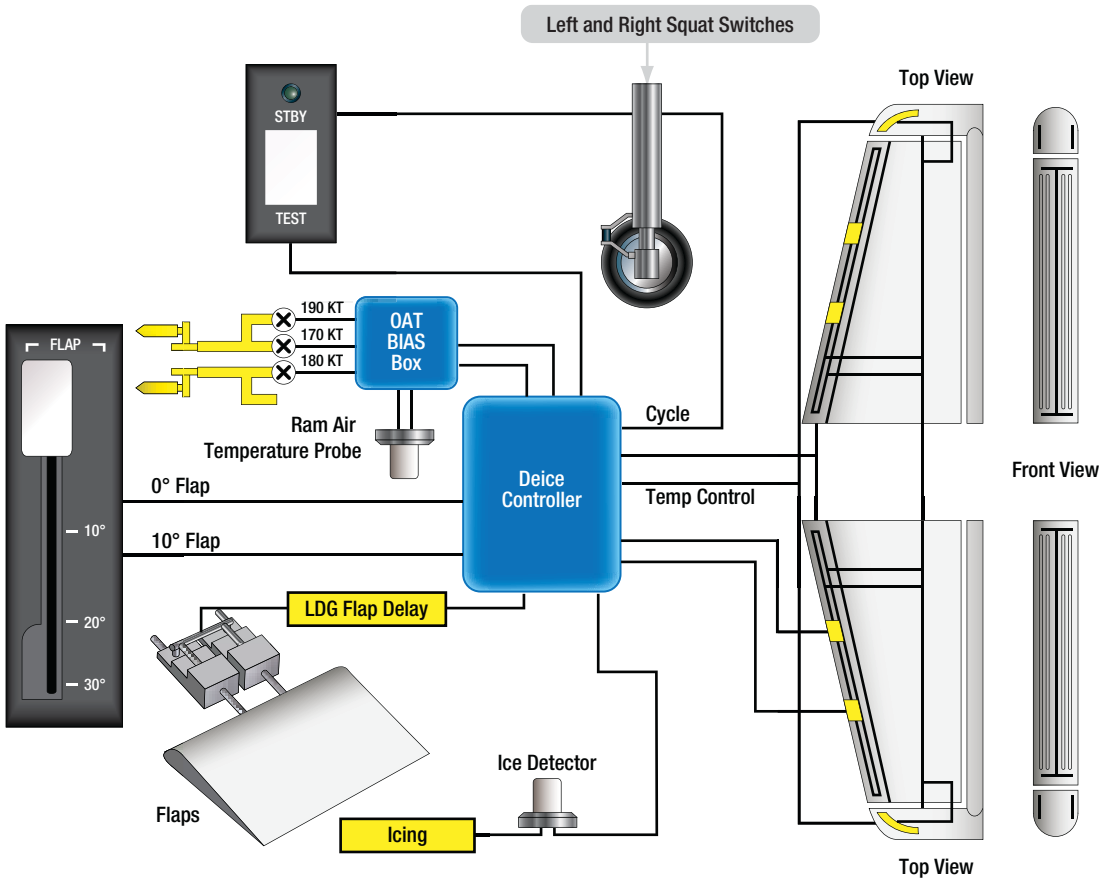


Figure 15-45. Electric stabilizer deice system.



Figure 15-46. Chemical deicing spray tubes.



Figure 15-47. Boom truck de-icing of an airliner.

exterior finish of the aircraft, only the type of fluid recommended by the aircraft manufacturer should be used. Transport category aircraft are often deiced on the ramp or a dedicated deicing location on the airport. Deicing trucks are used to spray the deicing and/or anti-icing fluid on aircraft surfaces. (***Figure 15-47***)

Deicing Fluid

As mentioned previously, various de-icing fluids exist for ground de-icing and anti—king of aircraft. A deicing fluid must be accepted according to its type for holdover times, aerodynamic performance, and material compatibility. The coloring of these fluids is standardized. In general, straight glycol is colorless, Type-I fluids are orange, Type-II fluids are white/pale yellow, and Type-IV fluids are green.

The color for Type-III fluid has not yet been determined. It is a fluid compound manufactured for use on slow aircraft that rotate at less than 100 knots. However, Type III de-ice fluid is approved for use on Boeing 757 and 767 aircraft.

As explained in the anti-ice section of this sub-module, Type I fluid is an ethylene or propylene glycol based de-ice fluid that contains wetting agents and corrosion inhibitors. This is a low viscosity fluid that leaves a thin film on aircraft surfaces. The film quickly flows off the de-iced surface resulting in minimal anti-icing properties. It is common for Type I fluid to be mixed with hot water in different ratios depending on weather/ icing conditions. Type II and VI de-icing fluids are thixotropic They share the property of high viscosity until acted upon by an outside force. So once applied, a thick layer of Type II and Type IV fluid results on the aircraft surface. This gives high anti-icing protection even when frozen precipitation is falling when de-icing is done.

If both deicing and anti-icing are required, the procedure may be performed in one or two steps. The selection of a one- or two-step process depends upon weather conditions, available equipment, available fluids, and the holdover time to be achieved. A one-step approach when anti-icing in desirable or required must be done with Type II or Type IV de-ice fluid since Type I fluid does not provide significant anti-icing. A two-step approach allows options. For example, The first step, de-icing, could be done with Type I fluid. With a high concentration of water to fluid, Type I fluid used to de-ice is less expensive than other types of fluid. Then, a second application can be made using Type II or Type IV fluid. Since the aircraft is already de-iced, a smaller quantity of these more expensive fluids can be used to give the thixotropic anti-ice properties needed.

Holdover Time (HOT)

Holdover Time (HOT) is the estimated time that deicing/anti-icing fluid prevents the formation of frost or ice and the accumulation of snow on the critical surfaces of an aircraft.

HOT begins when the final application of deicing/anti-icing fluid commences and expires when the deicing/anti-icing fluid loses its effectiveness. ***Figure 15-48*** shows a holdover timetable for Type IV fluid. HOT guidelines for other fluids are available and must be used when comparing options for different fluid use and options considering weather and traffic conditions.

FAA Type IV Holdover Time Guideline

Guidelines for holdover times anticipated for SAE type IV fluid mixtures as function of weather conditions and OAT.
CAUTION: This table is for use in departure planning only, and it should be used in conjunction with pretakeoff check procedures.

OAT		SAE type IV fluid concentration neat fluid water (vol. %/vol.%)	Approximate holdover times under various weather conditions (hours:minutes)						
°C	°F		Frost*	Freezing Fog	Snow◊	Freezing drizzle***	Light freezing rain	Rain on cold soaked wing	Other*
Above 0	Above 32	100/ 0	18:00	1:05–2:15	0:35–1:05	0:40–1:10	0:25–0:40	0:10–0:50	CAUTION: No Holdover Time Guidelines Exist
		72/25	6:00	1:05–1:45	0:30–1:05	0:35–0:50	0:15–0:30	0:05–0:35	
		50/50	4:00	0:15–0:35	0:05–0:20	0:10–0:20	0:05–0:10	CAUTION: Clear Ice May Require Louch For Confirmation	
0 through –3	32 through 27	100/ 0	12:00	1:05–2:15	0:30–0:55	0:40–1:10	0:15–0:40		
		75/25	5:00	1:05–2:15	0:25–0:50	0:35–0:50	0:15–0:30		
		50/50	3:00	1:15–0:35	0:05–0:15	0:10–0:20	0:05–0:15		
below –3 through –14	below 27 through 7	100/ 0	12:00	0:20–0:50	0:20–0:40	**0:20–0:45	**0:10–0:25		
		75/25	5:00	0:25–0:50	0:15–0:25	**0:15–0:30	**0:10–0:20		
below –14 through –25	below 7 through –13	100/ 0	12:00	0:15–0:40	0:15–0:30				
below –25	below –13	100/ 0	SAE type IV fluid may be used below –25 °C (–13 °F) if the freezing point of the fluid is at least 7 °C (13 °F) below the OAT and the aerodynamic acceptance criteria are met. Consider use of SAE type I when SAE type IV fluid cannot be used.						

°C = Degrees Celsius
°F = Degrees Fahrenheit
OAT= Outside Air Temperature
VOL = Volume

The responsiblity for the application of these data remains with the user.
* During conditions that apply to aircraft protection for ACTIVE FROST
** No holdover time guidelines exist for this condition below –10 °C (14 °F)
*** Use light freezing rain holdover times if positive identification of freezing drizzle is not possible
‡ Snow pellets, ice pellets, heavy snow, moderate and heavy freezing rain, hail.
◊ Snow includes snow grains

CAUTIONS:
- The time of protection will be shortened in heavy weather conditions: heavy precipitation rates or high moisture contents.
- High wind velocity or jet blast may reduce holdover time below the lowest time stated in the range.
- Holdover time may be reduced when aircraft skin temperature is lower than OAT.

Figure 15-48. FAA deice holdover time guidelines.

Critical Surfaces

Basically, all surfaces that have an aerodynamic, control, sensing, movement, or measuring function must be clean. These surfaces cannot necessarily be cleaned and protected in the same conventional deicing/anti-icing manner as the wings. Some areas require only a cleaning operation, while others need protection against freezing. The procedure of deicing may also vary according to aircraft limitations. The use of hot air may be required when deicing (e.g., landing gear or propellers). ***Figure 15-49*** shows critical areas on an aircraft that should not be sprayed directly with de-ice fluid. Some critical elements and procedures that are common for most aircraft are:

1. Deicing/anti-icing fluids must not be sprayed directly on wiring harnesses and electrical components (e.g., receptacles, junction boxes), onto brakes, wheels, exhausts, or thrust reversers.
2. Deicing/anti-icing fluid shall not be directed into the orifices of pitot heads, static ports, or directly onto airstream direction detectors probes/angle of attack airflow sensors.
3. All reasonable precautions shall be taken to minimize fluid entry into engines, other intakes/outlets, and control surface cavities.
4. Fluids shall not be directed onto flight deck or cabin windows as this can cause crazing of acrylics or penetration of the window seals.
5. Any forward area from which fluid can blow back onto windscreens during taxi or subsequent takeoff shall be free of residues prior to departure.
6. If Type II, III, or IV fluids are used, all traces of the fluid on flight deck windows should be removed prior to departure, particular attention being paid to windows fitted with wipers.
7. Landing gear and wheel bays shall be kept free from buildup of slush, ice, or accumulations of blown snow.
8. When removing ice, snow, slush, or frost from aircraft surfaces, care shall be taken to prevent it entering and accumulating in auxiliary intakes or control surface hinge areas (e.g., manually remove snow from wings and stabilizer surfaces forward toward the leading edge and remove from ailerons and elevators back towards the trailing edge)

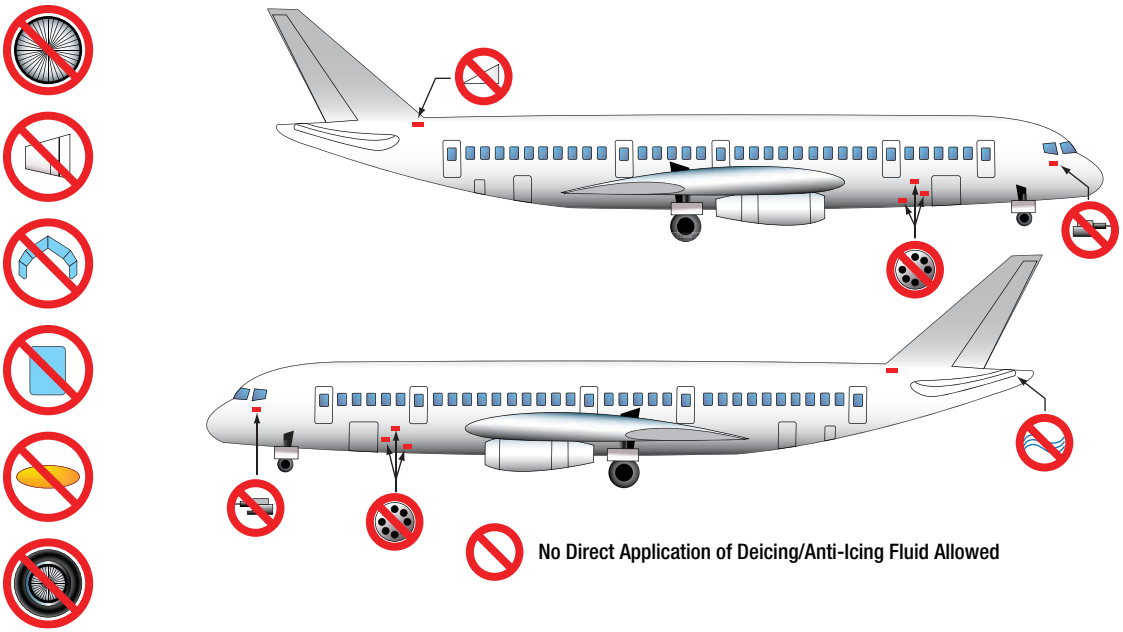


Figure 15-49. No direct application of deicing/anti-icing fluid allowed.

Ice and Snow Removal

Probably the most difficult deposit to deal with is deep, wet snow when ambient temperatures are slightly above the freezing point. This type of deposit should be removed with a soft brush or squeegee. Use care to avoid damage to antennas, vents, stall warning devices, vortex generators, etc., that may be concealed by the snow. Light, dry snow in subzero temperatures should be blown off whenever possible; the use of hot air is not recommended, since this would melt the snow, which would then freeze and require further treatment.

Moderate or heavy ice and residual snow deposits should be removed with a deicing fluid. No attempt should be made to remove ice deposits or break an ice bond by force. After completion of deicing operations, inspect the aircraft to ensure that its condition is satisfactory for flight. All external surfaces should be examined for signs of residual snow or ice, particularly in the vicinity of control gaps and hinges. Check the drain and pressure sensing ports for obstructions.

When it becomes necessary to physically remove a layer of snow, all protrusions and vents should be examined for signs of damage. Control surfaces should be moved to ascertain that they have full and free movement. The landing gear mechanism, doors and bay, and wheel brakes should be inspected for snow or ice deposits and the operation of uplocks and microswitches checked. Snow or ice can enter turbine engine intakes and freeze in the compressor. If the compressor cannot be turned by hand for this reason, hot air should be blown through the engine until the rotating parts are free.

RAIN CONTROL SYSTEMS

There are several different ways to remove the rain from the windshields. Most aircraft use one or a combination of the following systems: windshield wipers, chemical rain repellent, pneumatic rain removal (jet blast), or windshields treated with a hydrophobic surface seal coating.

WINDSHIELD WIPER SYSTEMS

In an electrical windshield wiper system, the wiper blades are driven by an electric motor(s) that receive (s) power from the aircraft's electrical system. On some aircraft, the pilot's and copilot's windshield wipers are operated by separate systems to ensure that clear vision is maintained through one of the windows should one system fail. Each windshield wiper assembly consists of a wiper, wiper arm and a wiper motor/converter. Almost all windshield wiper systems use electrical motors. Some older aircraft might be equipped with hydraulic wiper motors. (***Figure 15-50***)

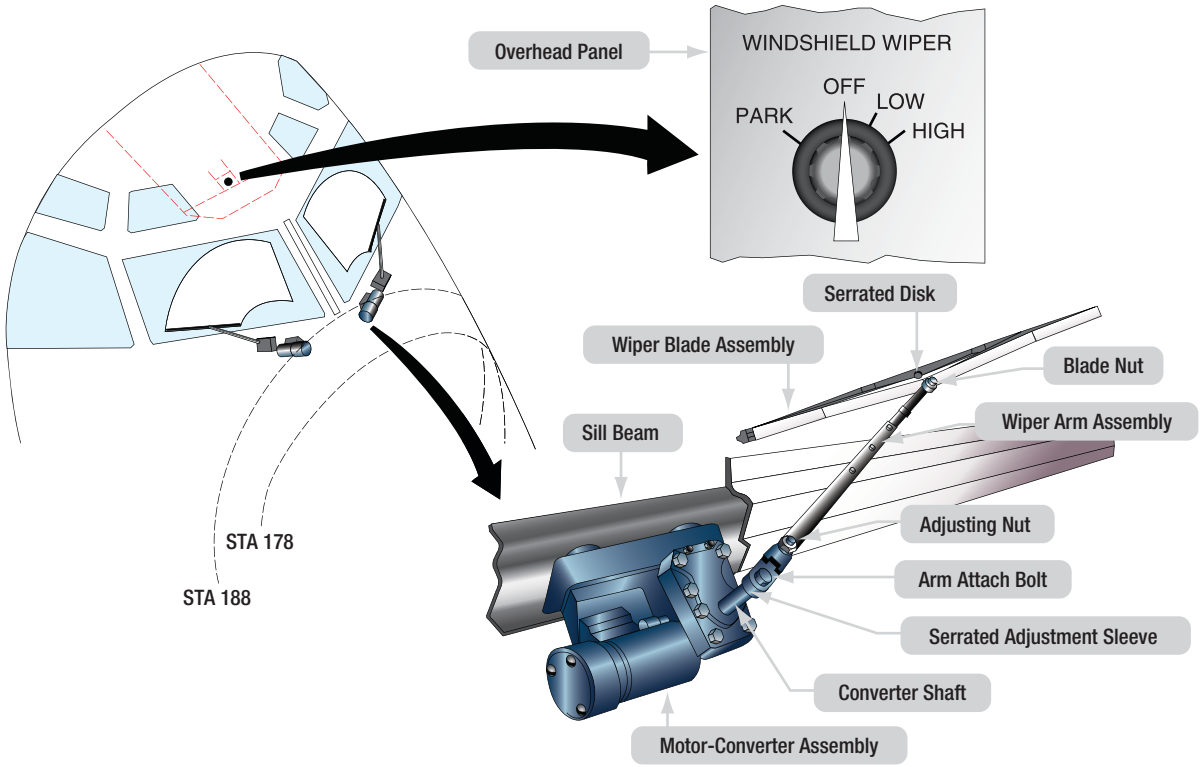


Figure 15-50. Windshield wiper assembly/installation on a transport category aircraft. The motor-converter is mounted under the aircraft skin.

Maintenance performed on windshield wiper systems consists of operational checks, adjustments, and troubleshooting. An operational check should be performed whenever a system component is replaced or whenever the system is suspected of not working properly. During the check, make sure that the windshield area covered by the wipers is free of foreign matter and is kept wet with water. Adjustment of a windshield wiper system consists of adjusting the wiper blade tension, the angle at which the blade sweeps across the windshield, and proper parking of the wiper blades.

CHEMICAL RAIN REPELLENT

Water poured onto clean glass spreads out evenly. Even when the glass is held at a steep angle or subjected to air velocity, the glass remains wetted by a thin film of water. However, when glass is treated with certain chemicals, a transparent film is formed that causes the water to behave very much like mercury on glass. The water draws up into beads that cover only a portion of the glass and the area between beads is dry. The water is readily removed from the glass. This principle lends itself quite naturally to removing rain from aircraft windshields. The high-velocity slipstream continually removes the water beads, leaving a large part of the window dry.

A rain repellent system permits application of the chemical repellent by a switch or push button in the cockpit. The proper amount of repellent is applied regardless of how long the switch is held. On some systems, a solenoid valve controlled by a time delay module meters the repellent to a nozzle which sprays it on the outside of the windshield. Two such units exist, one each for the forward glass of the pilot and copilot. (***Figure 15-51***)

This system should only be used in very wet conditions. The rain repellent system should not be operated on dry windows because heavy undiluted repellent restricts window visibility. Should the system be operated inadvertently, do not operate the windshield wipers or rain clearing system as this tends to increase smearing. Also, the rain repellent residues caused by application in dry weather or very light rain can cause staining or minor corrosion of the aircraft skin. To prevent this, any concentrated repellent or residue should be removed by a thorough fresh water rinse at the earliest opportunity. After application, the repellent film slowly deteriorates with continuing rain impingement. This makes periodic reapplication necessary. The length of time between applications depends upon rain intensity, the type of repellent used, and whether windshield wipers are used.

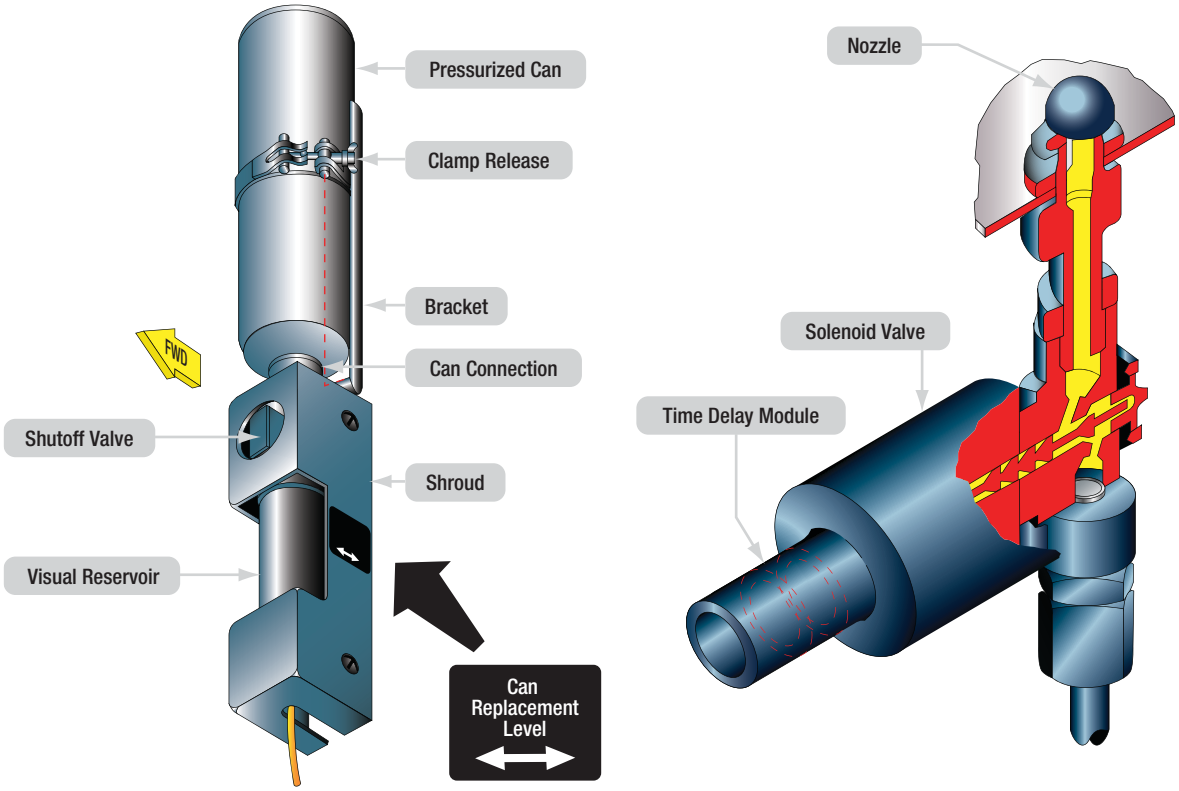


Figure 15-51. Cockpit rain repellant canister and reservoir.

WINDSHIELD SURFACE SEAL COATING

Some aircraft models use a surface seal coating, also called hydrophobic coating that is on the outside of the pilot's/copilot's windshield. (***Figure 15-52***)

The word hydrophobic means to repel or not absorb water. The windshield hydrophobic coating is on the external surface of the windows (windshields). The coatings cause raindrops to bead up and roll off, allowing the flight crew to see through the windshield with very little distortion. The hydrophobic windshield coating reduces the need for wipers and gives the flight crew better visibility during heavy rain. Most new aircraft windshields are treated with surface seal coating.

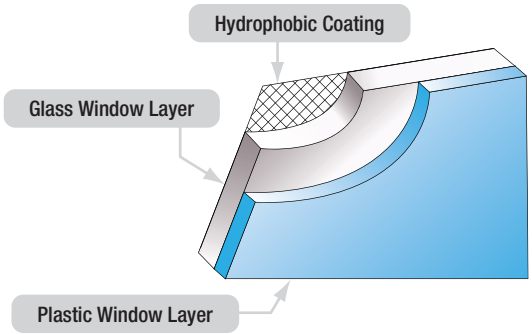


Figure 15-52. Hydrophobic coating on windshield.

The manufacturer's coating process deeply penetrates the windshield surface providing hydrophobic action for quite some time. When effectiveness declines, products made to be applied in the field are used. These liquid treatments rubbed onto the surface of the windshield maintain the beading action of rain water. They must be applied periodically or as needed.

PNEUMATIC RAIN REMOVAL SYSTEMS

Windshield wipers characteristically have two basic problem areas. One is the tendency of the slipstream aerodynamic forces to reduce the wiper blade loading pressure on the window, causing ineffective wiping or streaking. The other is in achieving fast enough wiper oscillation to keep up with high rain impingement rates during heavy rain falls. As a result, most aircraft wiper systems fail to provide satisfactory vision in heavy rain.

The rain removal system shown in ***Figure 15-53*** controls windshield icing and removes rain by directing a flow of heated air over the windshield. This heated air serves two purposes. First, the air breaks the rain drops into small particles that are then blown away. Secondly, the air heats the windshield to prevent the moisture from freezing. The air can be supplied by an electric blower or by bleed air.

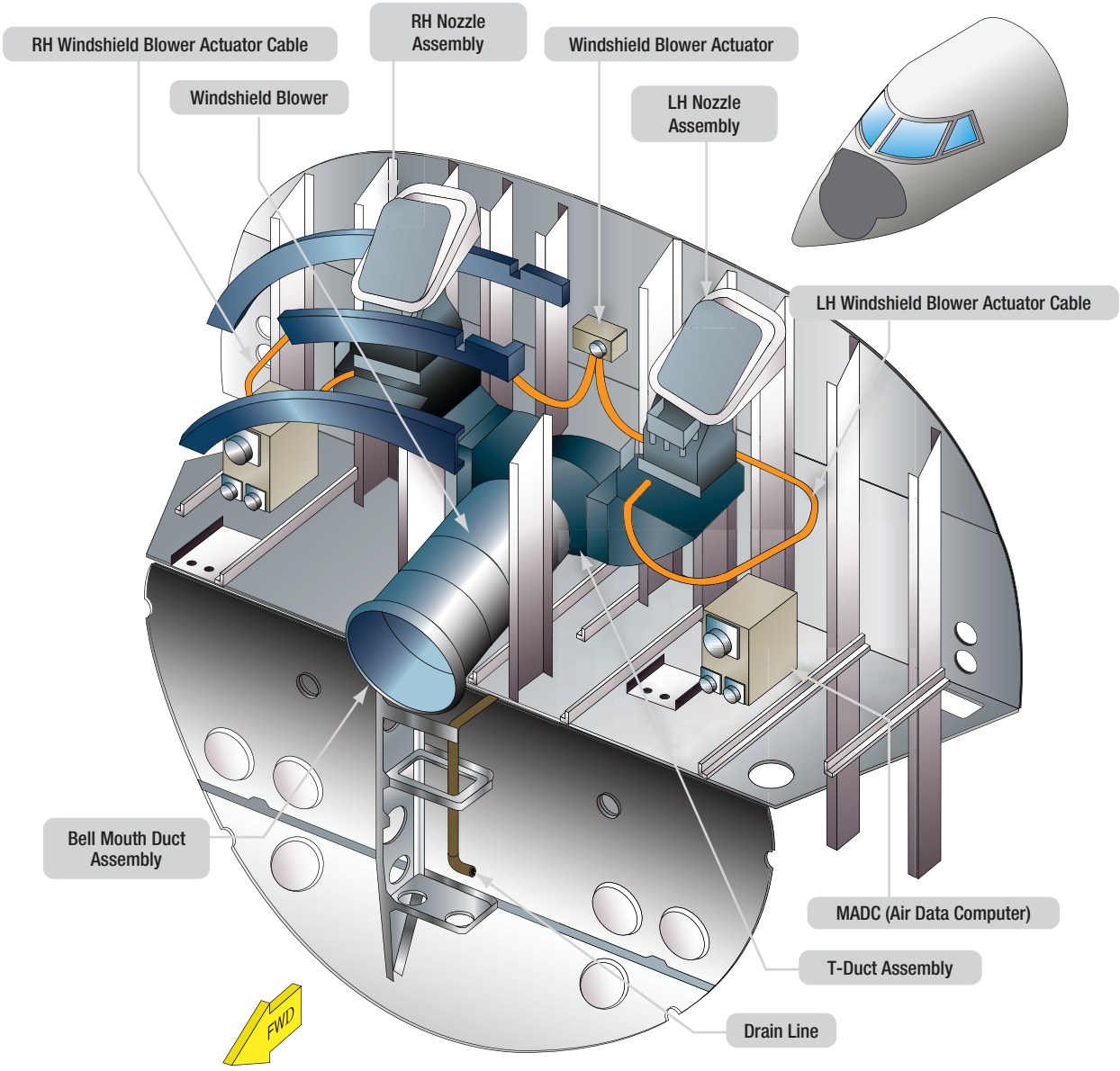


Figure 15-53. Windshield rain and frost removal system.

Question: 15-1
The two types of ice are encountered during flight are _____ ice and _____ ice.

Question: 15-2
When ice collects on an ice detector probe, the established _____ of vibration decreases.

Question: 15-3
The most common source of heated air for wing anti-icing is turbine engine _____.

Question: 15-4
The _____ controls the flow of bleed air from the pneumatic system to the WAI ducts.

Question: 15-5
An EAI _____ regulates air that operates the EAI anti-ice valve.

Question: 15-6
_____ that protrude into the ambient airstream are particularly susceptible to ice formation in flight.

Question: 15-7
The purpose of _____ is to create the strength and impact resistance of the windshield assembly.

Question: 15-8
Wing and stabilizer chemical anti-ice systems are often called _____ systems.

ANSWERS

Answer: 15-1
clear.
rime.

Answer: 15-2
frequency.

Answer: 15-3
compressor bleed air.

Answer: 15-4
WAI valve (wing anti-ice valve).

Answer: 15-5
controller.

Answer: 15-6
Data probes.

Answer: 15-7
laminations.

Answer: 15-8
weeping wing.

Question: 15-9
On modern aircraft, the deicer boots are bonded with _____ to the leading edge of wing and tail surfaces.

Question: 15-10
The source of air for pneumatic deice boots on turboprop aircraft is _____.

Question: 15-11
A _____ may be used to ensures deice boots are held tightly deflated against the aircraft structure to provide the significant change in size and shape needed to break off accumulated ice when the boots inflate.

Question: 15-12
An operational check of a deice boot system can be made by operating the engines or by using _____.

Question: 15-13
A propeller deice boot receives current from a _____ on the spinner bulkhead.

Question: 15-14
Deposits of ice, snow, or frost on the external surfaces of an aircraft may reduced aerodynamic lift and increased _____ resulting from the disturbed airflow over the airfoil surfaces.

Question: 15-15
Deep, wet snow when ambient temperatures are slightly above the freezing point should be removed from aircraft surfaces with _____.

Question: 15-16
Some aircraft models use a surface seal coating, also called _____ that is on the outside of the pilot's/ copilot's windshield.

ANSWERS

Answer: 15-9
an adhesive.

Answer: 15-10
engine bleed air.

Answer: 15-11
deflate valve.

Answer: 15-12
an external source of air.

Answer: 15-13
slip ring and brush assembly.

Answer: 15-14
aerodynamic drag.

Answer: 15-15
a soft brush or squeegee.

Answer: 15-16
hydrophobic coating.



PART-66 SYLLABUS **LEVELS**

CERTIFICATION CATEGORY → **B2**

Sub-Module 16

LANDING GEAR (ATA 32)

Knowledge Requirements

13.16 - Landing Gear (ATA 32)

Construction, shock absorbing;	1
Extension and retraction systems: normal and emergency;	3
Indications and warnings;	3
Wheels, brakes, antiskid and autobraking;	3
Tires;	1
Steering;	3
Air-ground sensing.	3

LANDING GEAR CONSTRUCTION

Aircraft landing gear supports the entire weight of an aircraft during landing and ground operations. They are attached to primary structural members of the aircraft. The type of gear depends on the aircraft design and its intended use. Most landing gear have wheels to facilitate operation to and from hard surfaces, such as airport runways. Other gear feature skids for this purpose, such as those found on helicopters, balloon gondolas, and in the tail area of some tail dragger aircraft.

Aircraft that operate to and from frozen lakes and snowy areas may be equipped with landing gear that have skis. Aircraft that operate to and from the surface of water have pontoon-type landing gear. Regardless of the type of landing gear utilized, shock absorbing equipment, brakes, retraction mechanisms, controls, warning devices, cowling, fairings, and structural members necessary to attach the gear to the aircraft are considered parts of the landing gear system. (***Figure 16-1***)

Numerous configurations of landing gear types can be found. Additionally, combinations of two types of gear are common. Amphibious aircraft are designed with gear that allow landings to be made on water or dry land.

NOTE: References to auxiliary landing gear refer to the nose gear, tail gear, or outrigger-type gear on any particular aircraft. Main landing gear are the two or more large gear located close to the aircraft's center of gravity.



Figure 16-1. Basic landing gear types include those with wheels (a), skids (b), skis (c), and floats or pontoons (d).

LANDING GEAR CONFIGURATIONS

Three basic arrangements of landing gear are used: tail wheel type landing gear (also known as conventional gear), tandem landing gear, and tricycle-type landing gear.

TAIL WHEEL-TYPE LANDING GEAR

Tail wheel-type landing gear is also known as conventional gear because many early aircraft use this type of arrangement. The main gear are located forward of the center of gravity, causing the tail to require support from a third wheel assembly. Typical configurations are shown in ***Figure 16-2***.

TANDEM LANDING GEAR

Few aircraft are designed with tandem landing gear. As the name implies, this type of landing gear has the main gear and tail gear aligned on the longitudinal axis of the aircraft. For example, sailplanes commonly use tandem gear, although many only have one actual gear forward on the fuselage with a skid under the tail. (***Figure 16-3***)

TRICYCLE-TYPE LANDING GEAR

The most commonly used landing gear arrangement is the tricycle-type landing gear. It is comprised of main gear and nose gear. (***Figure 16-4***) Tricycle-type landing gear is used on large and small aircraft. The nose gear of a few aircraft with tricycle-type landing gear is not controllable. It simply casters as steering is accomplished with differential braking during taxi. However, nearly all aircraft have steerable nose gear. On light aircraft, the nose gear is directed through mechanical linkage to the rudder pedals.

Heavy aircraft typically utilize hydraulic power to steer the nose gear. Control is achieved through an independent tiller in the flight deck. (***Figure 16-5***)

The main gear on a tricycle-type landing gear arrangement is attached to reinforced wing structure or fuselage structure. The number and location of wheels on the main gear vary. Many main gear have two or more wheels. (***Figure 16-6***)



Figure 16-2. Tail wheel configuration landing gear on a DC-3 (left) and a STOL Maule MX-7-235 Super Rocket.



Figure 16-3. Tandem landing gear along the longitudinal axis of the aircraft permits the use of flexible wings on sailplanes (left) and select military aircraft like the B-52 (center). The VTOL Harrier (right) has tandem gear with outrigger-type gear.



Figure 16-4. Tricycle-type landing gear with dual main wheels on a Learjet (left) and a Cessna 172, also with tricycle gear (right).

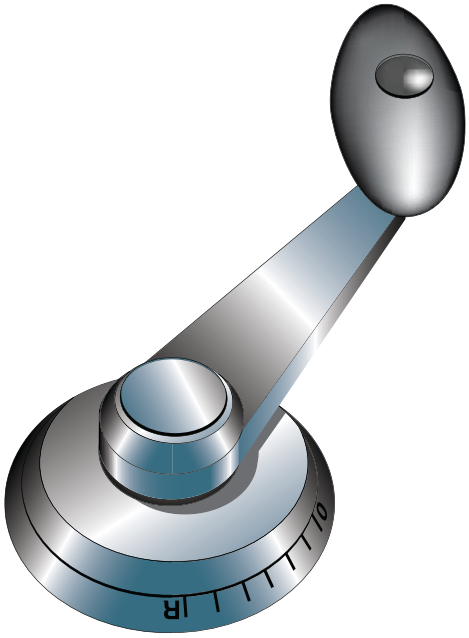


Figure 16-5. A nose wheel steering tiller located on the flight deck.

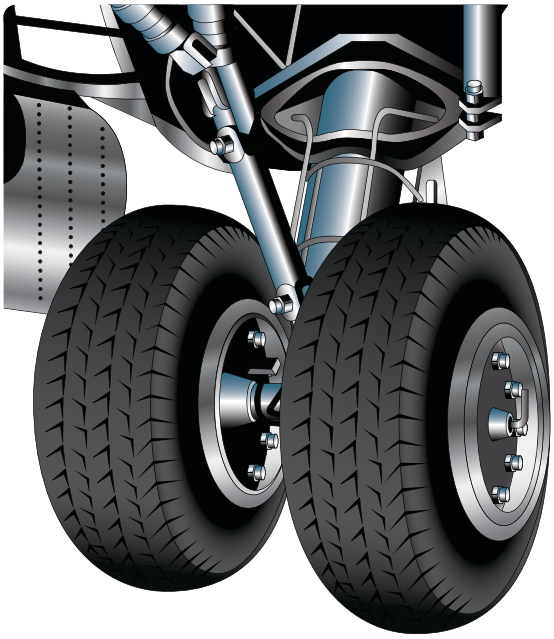


Figure 16-6. Dual main gear of a tricycle-type landing gear.

Multiple wheels spread the weight of the aircraft over a larger area. When more than two wheels are attached to a landing gear strut, the attaching mechanism is known as a bogie. (***Figure 16-7***)

The tricycle-type landing gear arrangement consists of many parts and assemblies. These include air/oil shock struts, gear alignment units, support units, retraction and safety devices, steering systems, wheel and brake assemblies, etc. A main landing gear of a transport category aircraft is illustrated in ***Figure 16-8*** with many of the parts identified as an introduction to landing gear nomenclature.



Figure 16-7. Triple bogie main landing gear assembly on a Boeing 777.

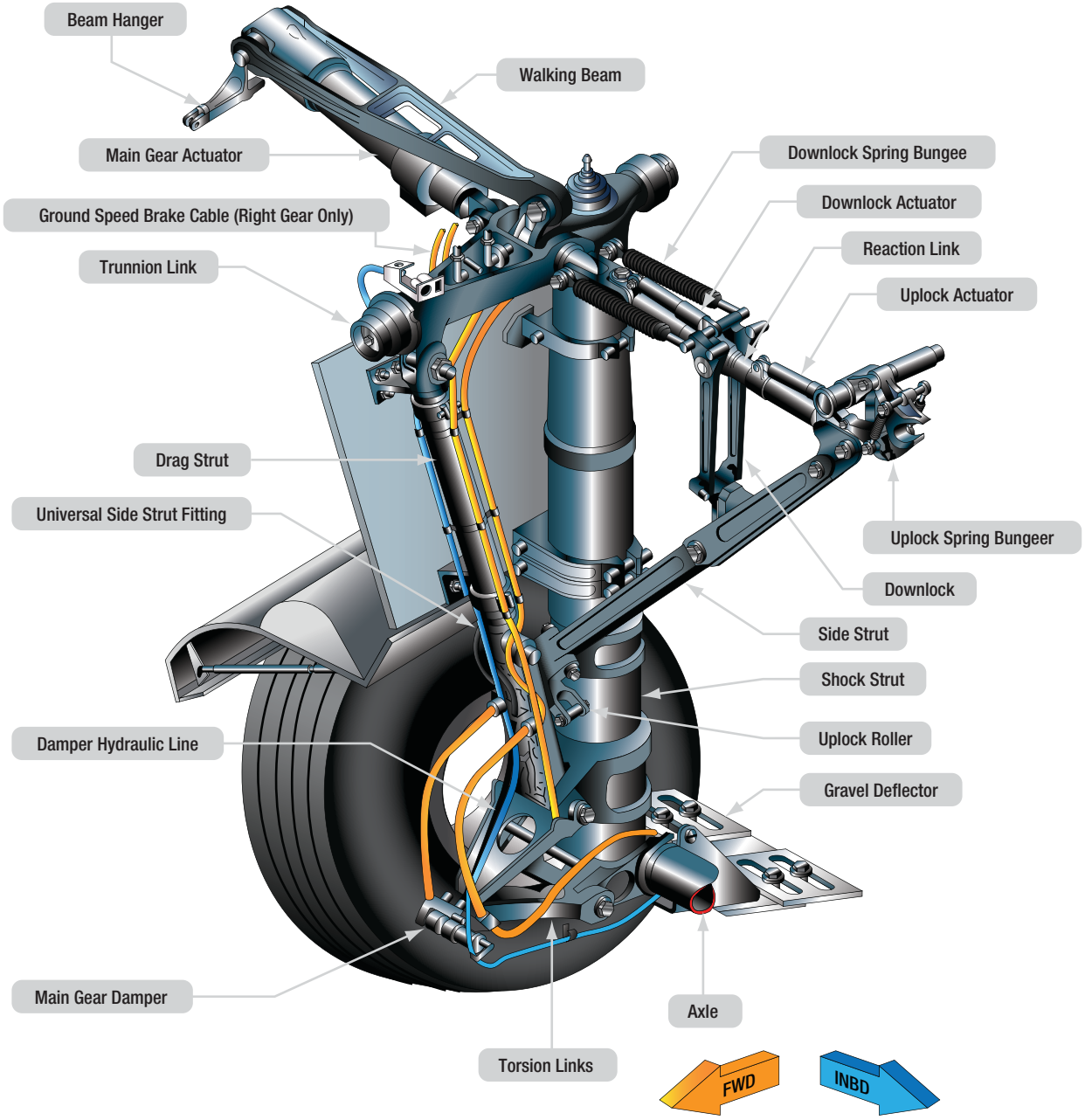


Figure 16-8. Nomenclature of a main landing gear bogie truck.

FIXED AND RETRACTABLE LANDING GEAR

Further classification of aircraft landing gear can be made into two categories: fixed and retractable. Many small, single engine light aircraft have fixed landing gear, as do a few light twins. Mechanisms to retract and stow the landing gear to eliminate drag add weight to the aircraft. On slow aircraft, the penalty of this added weight is not overcome by the reduction of drag, so fixed gear is used.

Retractable landing gear stow in fuselage or wing compartments while in flight. Most retractable gear have a close fitting panel attached to them that fairs with the aircraft skin when the gear is fully retracted. (***Figure 16-9***) Other aircraft have separate doors that open, allowing the gear to enter or leave, and then close again. The extension and retraction of most landing gear is usually accomplished with hydraulic power. Landing gear retraction systems are discussed below.



Figure 16-9. The retractable gear of a Boeing 737 recesses into the fuselage. Panels attached to the landing gear provide smooth airflow over the struts. The wheel assemblies mate with seals to provide aerodynamic flow without doors.

SHOCK ABSORBING

In addition to supporting the aircraft for taxi, the forces of impact on an aircraft during landing must be controlled by the landing gear. This is done in two ways: 1) the shock energy is altered and transferred throughout the airframe at a different rate and time than the single strong pulse of impact, and 2) the shock is absorbed by converting the energy into heat energy. A variety of non-shock absorbing land gear are used on general aviation aircraft such as leaf-type spring gear, gear with bungee cords and even rigid steel landing gear on early aircraft and aircraft with skids. Most turbine powered aircraft use shock strut landing gear.

SHOCK STRUTS

True shock absorption occurs when the shock energy of landing impact is converted into heat energy, as in a shock strut landing gear. This is the most common method of landing shock dissipation in aviation. It is used on aircraft of all sizes. Shock struts are self-contained hydraulic units that support an aircraft while on the ground and protect the structure during landing.

There are many different designs of shock struts, but most operate in a similar manner. A typical pneumatic/hydraulic shock strut uses compressed air or nitrogen combined with hydraulic fluid to absorb and dissipate shock loads. It is sometimes referred to as an air/oil or oleo strut. A shock strut is constructed of two telescoping cylinders or tubes that are closed on the external ends. The upper cylinder is fixed to the aircraft and does not move. The lower cylinder is called the piston and is free to slide in and out of the upper cylinder. Two chambers are formed. The lower chamber is always filled with hydraulic fluid and the upper chamber is filled with compressed air or nitrogen. Efficient operation of the shock struts requires that proper fluid and air pressure be maintained. An orifice located between the two cylinders provides a passage for the fluid from the bottom chamber to enter the top cylinder chamber when the strut is compressed. (***Figure 16-10***)

The upper cylinder of a shock strut typically contains a valve fitting assembly. It is located at or near the top of the cylinder. The valve provides a means of filling the strut with hydraulic fluid and inflating it with air or nitrogen as specified by the manufacturer.

To keep the piston and wheels aligned, most shock struts are equipped with torque links or torque arms. One end of the links is attached to the fixed upper cylinder. The other end is attached to the lower cylinder (piston) so it cannot rotate. This keeps the wheels aligned. The links also retain the piston in the end of the upper cylinder when the strut is extended, such as after takeoff. (***Figure 16-11***)

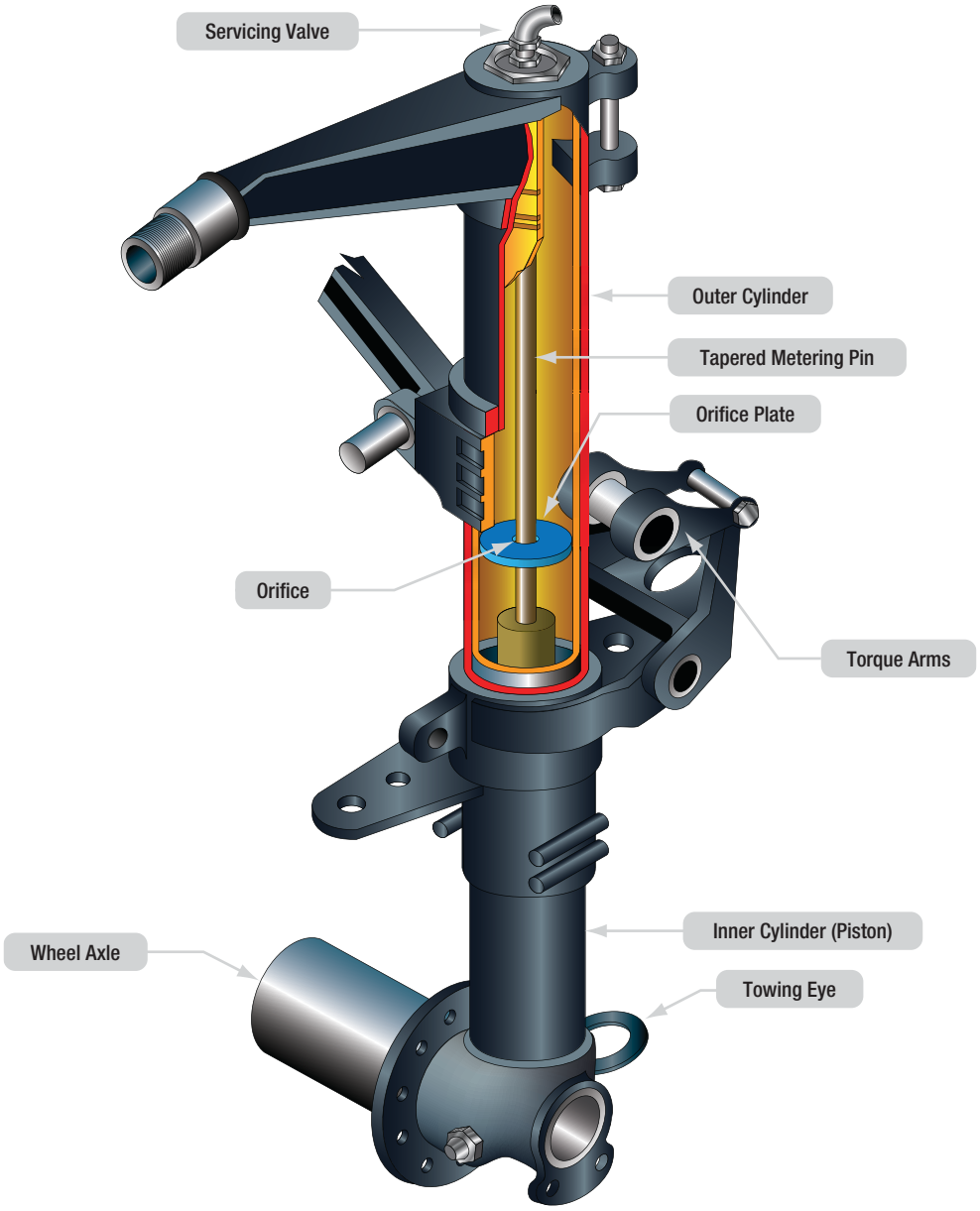


Figure 16-10. Some landing gear shock struts use an internal metering tube rather than a metering pin to control the flow of fluid from the bottom cylinder to the top cylinder.

EXTENSION AND RETRACTION SYSTEMS: NORMAL

Large aircraft extension and retraction systems are nearly always powered by hydraulics. Typically, the hydraulic pump is driven off of the engine accessory drive. Auxiliary electric hydraulic pumps are also common. Other devices used in a hydraulically-operated retraction system include actuating cylinders, selector valves, uplocks, downlocks, sequence valves, priority valves, tubing, and other conventional hydraulic system components. These units are interconnected so that they permit properly sequenced retraction and extension of the landing gear and the landing gear doors.

The correct operation of any aircraft landing gear retraction system is extremely important. ***Figure 16-12*** illustrates an example of a simple large aircraft hydraulic landing gear system. The system is on an aircraft that has doors that open before the gear is extended and close after the gear is retracted. The nose gear doors operate via mechanical linkage and do not require hydraulic power. There are many gear and gear door arrangements on various aircraft. Some aircraft have gear doors that close to fair the wheel well after the gear is extended.



Figure 16-11. Torque links align the landing gear and retain the piston in the upper cylinder when the strut is extended.

Others have doors mechanically attached to the outside of the gear so that when it stows inward, the door stows with the gear and fairs with the fuselage skin. In the system illustrated in ***Figure 16-12***, when the flight deck gear selector is moved to the gear-up position, it positions a selector valve to allow pump pressure from the hydraulic system manifold to access eight different components. The three downlocks are pressurized and unlocked so the gear can be retracted.

At the same time, the actuator cylinder on each gear also receives pressurized fluid to the gear-up side of the piston through an unrestricted orifice check valve. This drives the gear into the wheel well. Two sequence valves (C and D) also receive fluid pressure. Gear door operation must be controlled so that it occurs after the gear is stowed. The sequence valves are closed and delay flow to the door actuators. When the gear cylinders are fully retracted, they mechanically contact the sequence valve plungers that open the valves and allow fluid to flow into the close side of the door actuator cylinders. This closes the doors. Sequence valves A and B act as check valves during retraction. They allow fluid to flow one way from the gear-down side of the main gear cylinders back into the hydraulic system return manifold through the selector valve.

To lower the gear, the selector is put in the gear-down position. Pressurized hydraulic fluid flows from the hydraulic manifold to the nose gear uplock, which unlocks the nose gear. Fluid flows to the gear-down side of the nose gear actuator and extends it. Fluid also flows to the open side of the main gear door actuators. As the doors open, sequence valves A and B block fluid from unlocking the main gear uplocks and prevent fluid from reaching the down side of the main gear actuators. When the doors are fully open, the door actuator engages the plungers of both sequence valves to open the valves. The main gear uplocks, then receives fluid pressure and unlock. The main gear cylinder actuators receive fluid on the down side through the open sequence valves to extend the gear. Fluid from each main gear cylinder up-side flows to the hydraulic system return manifold through restrictors in the orifice check valves.

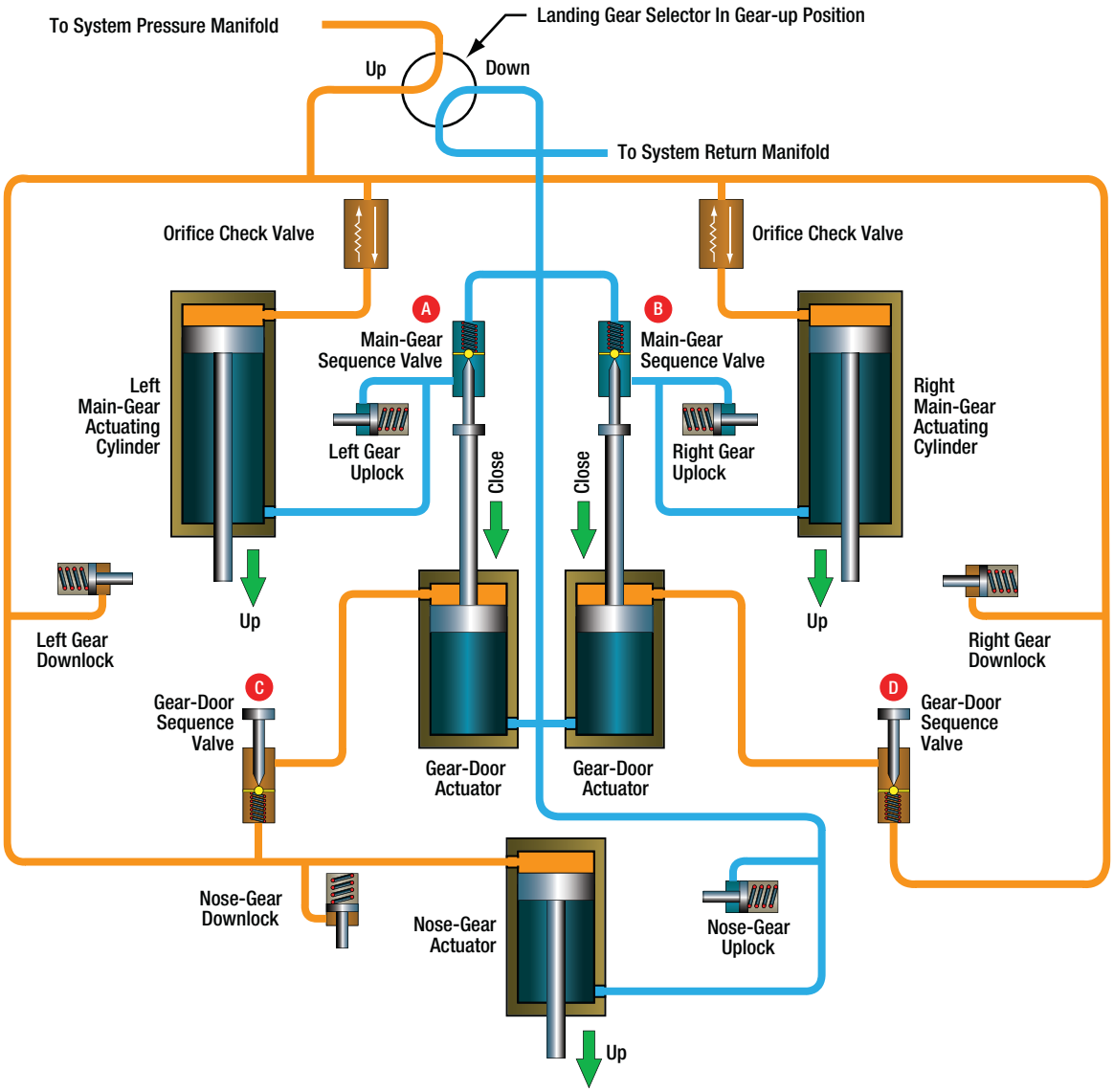


Figure 16-12. A simple large aircraft hydraulic gear retraction system.

The restrictors slow the extension of the gear to prevent impact damage. There are numerous hydraulic landing gear retraction system designs. Priority valves are sometimes used instead of mechanically operated sequence valves. This controls some gear component activation timing via hydraulic pressure. Particulars of any gear system are found in the aircraft maintenance manual. The aircraft technician must be thoroughly familiar with the operation and maintenance requirements of this crucial system.

EMERGENCY EXTENSION SYSTEMS

The emergency extension system lowers the landing gear if the main power system fails. There are numerous ways in which this is done depending on the size and complexity of the aircraft. Some aircraft have an emergency release handle in the flight deck that is connected through a mechanical linkage to the gear uplocks. When the handle is operated, it releases the uplocks and allows the gear to free-fall to the extended position under the force created by gravity acting upon the gear. Other aircraft use a non-mechanical back-up, such as pneumatic power, to unlatch the gear. Large and high performance aircraft are equipped with redundant hydraulic systems. This makes emergency extension less common since a different source of hydraulic power can be selected if the gear does not function normally. If the gear still fails to extend, some sort of unlatching device is used to release the uplocks and allow the gear to free fall. (***Figure 16-13***)

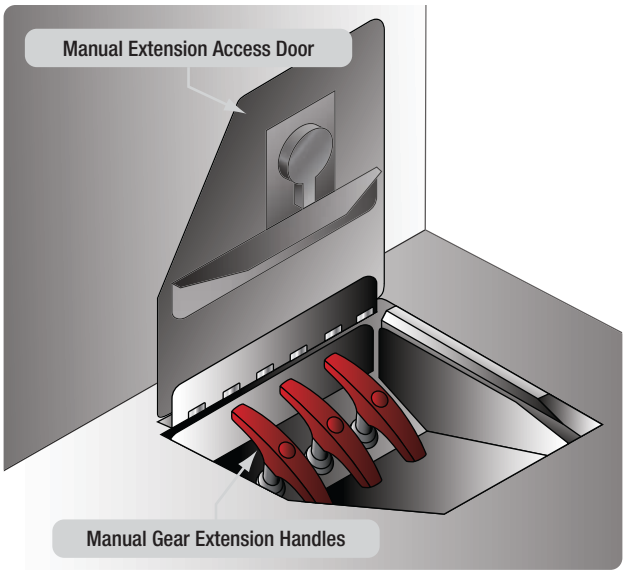


Figure 16-13. These emergency gear extension handles in a Boeing 737 are located under a floor panel on the flight deck. Each handle releases the gear uplock via a cable system so the gear can freefall into the extended position.

Consult the aircraft maintenance manual for all emergency landing gear extension system descriptions of operation, performance standards, and emergency extension tests as required.

LANDING GEAR RETRACTION TEST

The proper functioning of a landing gear system and components can be checked by performing a landing gear retraction test. This is also known as swinging the gear. The aircraft is properly supported on jacks for this check, and the landing gear should be cleaned and lubricated if needed. The gear is then raised and lowered as though the aircraft were in flight while a close visual inspection is performed. All parts of the system should be observed for security and proper operation. The emergency back-up extension system should be checked whenever swinging the gear.

Any time a landing gear component is replaced that could affect the correct functioning of the landing gear system, a retraction test should follow when adjustments to landing gear linkages or components that affect gear system performance are made. It may be necessary to swing the gear after a hard or overweight landing. It's also common to swing the gear while attempting to locate a malfunction within the system. For all required retraction tests and the specific inspection points to check, consult the manufacturer's maintenance manual for the aircraft in question as each landing gear system is unique.

The following is a list of general inspection items to be performed while swinging the gear:

1. Check the landing gear for proper extension and retraction.
2. Check all switches, lights, and warning devices for proper operation.
3. Check the landing gear doors for clearance and freedom from binding.
4. Check landing gear linkage for proper operation, adjustment, and general condition.
5. Check the alternate/emergency extension or retraction systems for proper operation.
6. Investigate any unusual sounds, such as those caused by rubbing, binding, chafing, or vibration.

Control of the landing gear and annunciating its position is done through a system of switches. Solid state circuits are controlled with solid state proximity switches located on the gear so that the position of the gear is known at all times. The condition of the gear is also known. For example, DOWN and LOCKED versus DOWN and NOT LOCKED.

Landing gear position indicators are located on the instrument panel adjacent to the gear selector handle. They are used to inform the pilot of gear position status. There are many arrangements for gear indication. Usually, there is a dedicated light for each gear. The most common display for the landing gear being down and locked is an illuminated green light. Three green lights means it is safe to land. All lights out typically indicates that the gear is up and locked, or there may be gear up indicator lights.

Gear in transit lights are used on some aircraft as are barber pole displays when a gear is not up or down and locked. Blinking indicator lights also indicate gear in transit. Some manufacturer's use a gear disagree annunciation when the landing gear is not in the same position as the selector. Many aircraft monitor gear door position in addition to the gear itself. Consult the aircraft manufacturer's maintenance and operating manuals for a complete description of the landing gear indication system. (***Figure 16-14***)



Figure 16-14. Landing gear selector panels with position indicator lights. The Boeing 737 panel illuminates red lights above the green lights when the gear is in transit.

LANDING GEAR SAFETY DEVICES

There are numerous landing gear safety devices. Most common are those that prevent the gear from retracting or collapsing while on the ground (ground locks). A nose wheel centering device prevents damage to the fuselage and gear by aligning the nose gear with the wheel well bay before retraction. Various safety and proximity switches and circuits that ensure sequential operation of the landing gear and other system components dependent on the air-ground status of the aircraft are also common. As mentioned above, gear position indicators are another safety device used to communicate to the pilot the position status of each individual landing gear at any time.

GROUND LOCKS

Ground locks are commonly used on aircraft landing gear as extra insurance that the landing gear will remain down and locked while the aircraft is on the ground. They are external devices that are placed in the retraction mechanism to prevent its movement. A ground lock can be as simple as a pin placed into the pre-drilled holes of gear components that keep the gear from collapsing. Another commonly used ground lock clamps onto the exposed piston of the gear retraction cylinder that prevents it from retracting. All ground locks should have a red streamers attached to them so they are visible and removed before flight. Ground locks are typically carried in the aircraft and put into place by the flight crew during the post landing walk-around. (***Figure 16-15***)

NOSE WHEEL CENTERING

Since most aircraft have steerable nose wheel gear assemblies for taxiing, a means for aligning the nose gear before retraction is needed. Centering cams built into the shock strut structure accomplish this.

SAFETY SWITCHES

At least one landing gear squat switch, or safety switch, is found on most aircraft. This is a switch positioned to open and close depending on the extension or compression of the main landing gear strut. (***Figure 16-16***) The squat switch is wired into any number of system operating circuits. One circuit prevents the gear from being retracted while the aircraft is on the ground. There are different ways to achieve this lockout. A solenoid that extends a shaft to physically disable the gear position selector is one such method found on many aircraft. When the landing gear is compressed, the squat safety switch is open, and the center shaft of the solenoid



Figure 16-15. Gear pin ground lock devices.

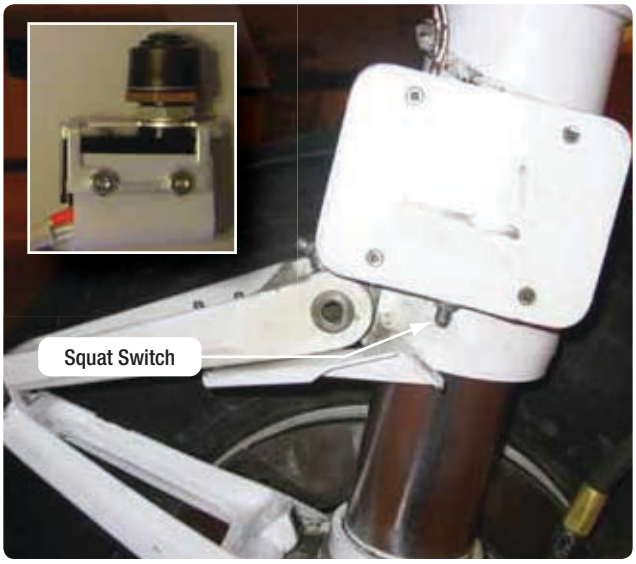


Figure 16-16. Typical landing gear squat switches.

protrudes a hardened lock-pin through the landing gear control handle so that it cannot be moved to the up position. At takeoff, the landing gear strut extends.

The safety switch closes and allows current to flow in the safety circuit. The solenoid energizes and retracts the lock-pin from the selector handle. This permits the gear to be raised. (***Figure 16-17***)

PROXIMITY SENSORS

The use of proximity sensors for gear position safety switches is common in high-performance aircraft. An electromagnetic sensor returns a different voltage to a gear logic unit depending on the proximity of a conductive target to the switch. No physical contact is made. When the gear is in the designed position, the metallic target is close to the inductor in the sensor which reduces the return voltage to an electronic logic unit located in the equipment bay. This type of sensing is especially useful in the landing gear environment where switches with moving parts can become contaminated with dirt and moisture from runways and taxi ways. The technician is required to ensure that sensor targets are installed the correct distance away from the sensor. Go-No-Go gauges are often used to set the distance. (***Figure 16-18***)

On the latest models of airline aircraft, use of proximity sensors has been expanded to monitor the position of cabin entry doors, cargo doors, access doors and thrust reversers. Two proximity sensor electronic units (PSEUs), each containing two integrated logic circuit cards, receive the proximity sensor information. It is then communicated to systems throughout the aircraft, typically through ARINC 629 or 429 data buses.

Condition warning annunciations originate in this manner as well as permissions for various systems to operate or not operate depending on the condition status of the proximity sensors. The following is a list of the systems that make use of the proximity sensor data from the PSEU's on the Boeing 777:

- Airplane information management system (AIMS)
- Electrical load management system (ELMS)
- Brake system control unit (BSCU)
- Cabin system management unit (CSMU)
- Cargo smoke detection system (CSDS)
- Audio management unit (AMU)

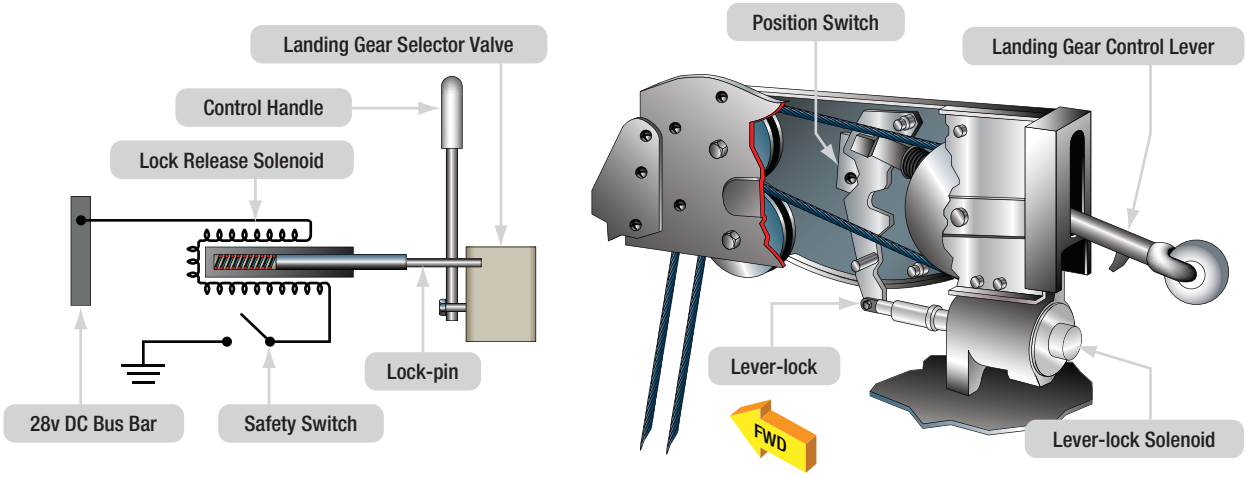


Figure 16-17. A landing gear safety circuit with solenoid that locks the control handle and selector valve from being able to move into the gear up position when the aircraft is on the ground. The safety switch, or squat switch, is located on the aircraft landing gear.

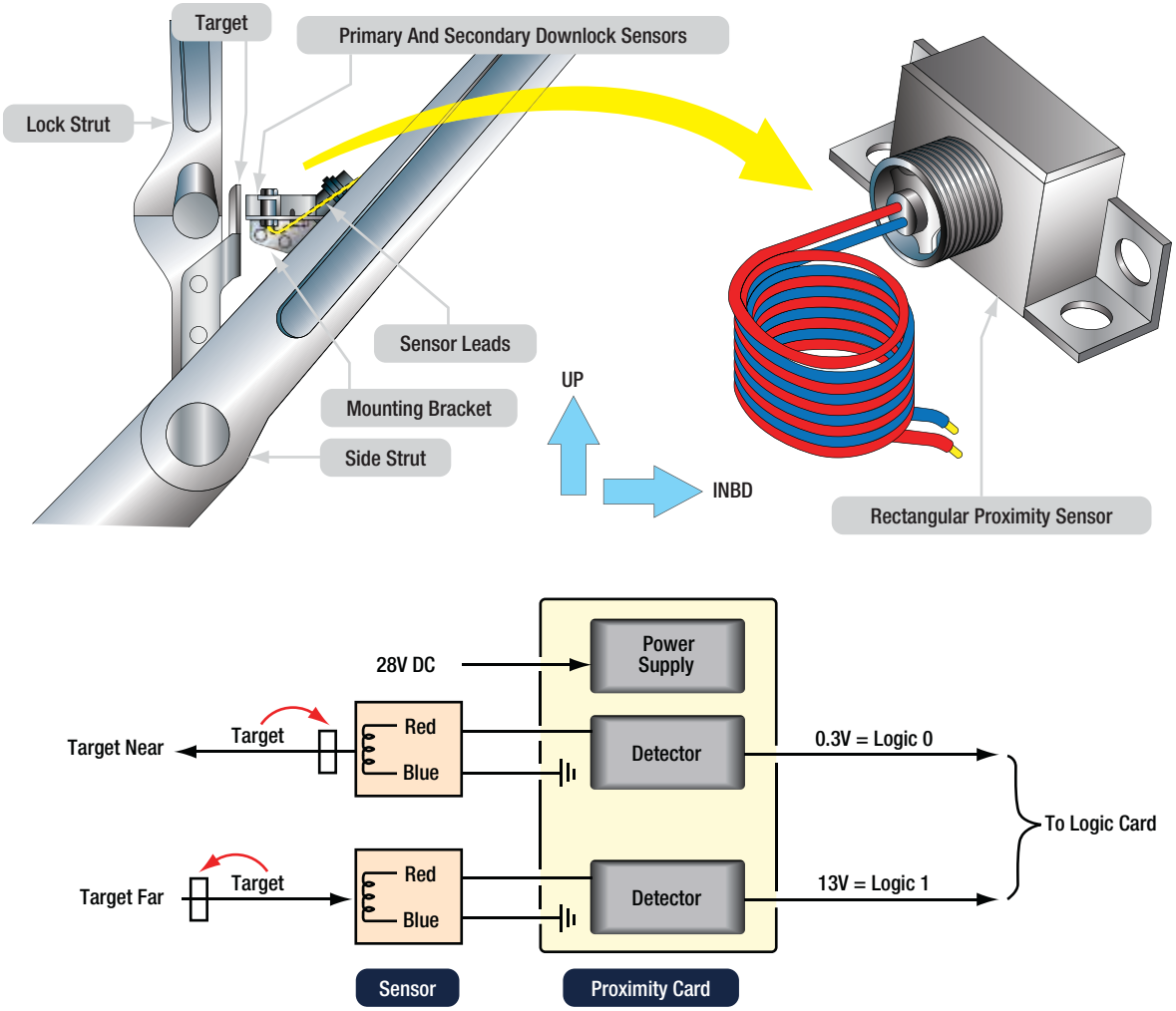


Figure 16-18. Proximity sensors are used instead of contact switches on many landing gear.

AIR-GROUND SENSING

Air-ground sensing in modern aircraft is part of an aircraft-wide control system. The operation (or the prevention of operation) of many on-board systems depends on whether or not the aircraft is in the air or on the ground. The following is a list of some of the systems on a large passenger aircraft that may require this status information from the air-ground sensing system:

- Airplane Information Management System (AIMS)
- Airborne Vibration Monitor Signal Conditioner Unit (AVMSCU)
- Autopilot Flight Director Computer (AFDC)
- Audio Management Unit (AMU)
- APU Controller (APUC)
- Air Supply Cabin Pressure Controller (ASCPC)
- Backup Generator Converter
- Bus Power Control Unit (BPCU)
- Cabin System Management Unit (CSMU)
- Cabin Temperature Controllers (CTC)
- Control Display Unit (CDU)
- Engine Data Interface Unit (EDIU)
- Electrical Load Management System (ELMS)
- Fuel Quantity Indication System (FQIS)
- Flap Slat Electronic Unit (FSEU)
- Generator Control Unit (GCU)
- Overhead Panel Bus Controller (OPBC)
- Passenger Address/Cabin Interphone (PA/CI)
- Proximity Sensor Electronics Unit (PSEU)
- Warning Electronic Unit (WEU)

As described above, squat switches have been used to initiate air ground status for aircraft. The newest airliners use strain gauge load sensors mounted on the landing gear beam structure to detect when the weight of the aircraft is on the main landing gear wheels (ground condition). As the gear beam structure bends under the weight of the aircraft, corresponding electrical signals

from the load sensors are sent to a unit or card file in the equipment bay for processing. There, weight on wheels (WOW) integrated circuit logic cards communicate the air-ground information with the other systems on the aircraft through a digital data bus. Responding to signal from the WOW cards, air-ground relays control the distribution of electrical power to the appropriate systems and devices on the aircraft.

Note that the WOW cards are electrostatic sensitive and all electrostatic sensitive material precautions must be observed by the technician. The WOW load sensors are two-piece, variable reluctance strain measurement devices. They are attached by mounting brackets to the landing gear beam. As the gear beam bends with the weight of the aircraft, the distance between the two pieces changes. The distance between sensor parts is what varies the signal sent to the WOW cards because one of the sensor parts is a target and the other contains two electric coils. Thus, the varied electric signal sent to the WOW cards is analog.

Communication with most other aircraft systems by the WOW cards is digital. The load sensors are covered for protection from the elements. Replacement of a sensor requires calibration of the sensor. Replacement of the WOW cards requires that all sensors be calibrated.

The use of load sensors instead of proximity switches for air-ground sensing adds the capability of capturing the weight of the aircraft when it is on the ground. Air-ground output from the sensors and cards are displayed on the flight deck on a landing gear maintenance page that is part of the aircraft information management system. Testing in air mode and/or ground mode of various air-ground related systems can be initiated here. Status of the air-ground system messages also appear on a landing gear maintenance display.

STEERING

NOSE WHEEL STEERING

The nose wheel on most aircraft is steerable from the flight deck via a nose wheel steering system. This allows the aircraft to be directed during ground operation. A few simple aircraft have nose wheel assemblies that caster. Such aircraft are steered during taxi by differential braking.

Most small aircraft have steering capabilities through the use of a simple system of mechanical linkages connected to the rudder pedals. Due to their mass and the need for positive control, large aircraft utilize a power source for nose wheel steering. Hydraulic power predominates. There are many different designs for large aircraft nose steering systems. Most share similar characteristics and components. Control of the steering is from the flight deck through the use of a small wheel, tiller, or joystick typically mounted on the left side wall. Switching the system on and off is possible on some aircraft.

Mechanical, electrical, or hydraulic connections transmit the controller input movement to a steering control unit. The control unit is a hydraulic metering or control valve. It directs hydraulic fluid under pressure to one or two actuators designed with various linkages to rotate the lower strut. An accumulator and relief valve, or similar pressurizing assembly keeps fluid in the actuators and system under pressure at all times. This permits the steering actuating cylinders to also act as shimmy dampers. A follow-up mechanism consists of various gears, cables, rods, drums, and/or bell-crank, etc. It returns the metering valve to a neutral position once the steering angle has been reached. Many systems incorporate an input subsystem from the rudder pedals for small degrees of turns made while directing the aircraft at high speed during takeoff and landing. Safety valves are typical in all systems to relieve pressure during hydraulic failure so the nose wheel can swivel.

The following explanation is accompanied by ***Figures 16-19, 16-20, and 16-21***, which illustrate a large aircraft nose wheel steering system and components. These figures and explanation are for instructional purposes only.

The nose wheel steering wheel connects through a shaft to a steering drum located inside the flight deck control pedestal. The rotation of this drum transmits the steering signal by means of cables and pulleys to the control drum of the differential

assembly. Movement of the differential assembly is transmitted by the differential link to the metering valve assembly where it moves the selector valve to the selected position. This provides the hydraulic power for turning the nose gear. (***Figure 16-19***)

As shown in ***Figure 16-20***, pressure from the aircraft hydraulic system is directed through the open safety shutoff valve into a line leading to the metering valve. The metering valve then routes the pressurized fluid out of port A, through the right turn alternating line, and into steering cylinder A. This is a one-port cylinder and pressure forces the piston to begin extension. Since the rod of this piston connects to the nose steering spindle on the nose gear shock strut which pivots at point X, the extension of the piston turns the steering spindle gradually toward the right. As the nose wheel turns, fluid is forced out of steering cylinder B through the left turn alternating line and into port B of the metering valve. The metering valve directs this return fluid into a compensator that routes the fluid into the aircraft hydraulic system return manifold.

As described, hydraulic pressure starts the nose gear turning. However, the gear should not be turned too far. The nose gear steering system contains devices to stop the gear at the selected angle of turn and hold it there. This is accomplished with follow-up linkage. As stated, the nose gear is turned by the steering spindle as the piston of cylinder A extends. The rear of the spindle contains gear teeth that mesh with a gear on the bottom of the orifice rod. (***Figure 16-19***)

As the nose gear and spindle turn, the orifice rod also turns but in the opposite direction. This rotation is transmitted by the two sections of the orifice rod to the scissor follow-up links located at the top of the nose gear strut. As the follow-up links return, they rotate the connected follow-up drum, which transmits the movement by cables and pulleys to the differential assembly.

Operation of the differential assembly causes the differential arm and links to move the metering valve back toward the neutral position. The metering valve and the compensator unit of the nose wheel steering system are illustrated in ***Figure 16-21***.

The compensator unit system keeps fluid in the steering cylinders pressurized at all times. This hydraulic unit consists of a three-port housing that encloses a spring-loaded piston and poppet. The left port is an air vent that prevents trapped air at the rear of the piston from interfering with the movement of the piston. The second port located at the top of the compensator connects through a line to the metering valve return port. The third port is located at the right side of the compensator. This port connects to the hydraulic system return manifold. It routes the steering system return fluid into the manifold when the poppet valve is open.

The compensator poppet opens when pressure acting on the piston becomes high enough to compress the spring. In this system, 100 psi is required. Therefore, fluid in the metering valve return line is contained under that pressure. The 100 psi pressure also exists throughout the metering valve and back through the cylinder return lines. This pressurizes the steering cylinders at all times and permits them to function as shimmy dampers.

STEERING DAMPER

As mentioned above, large aircraft with hydraulic steering hold pressure in the steering cylinders to provide the required damping. This is known as steering damping. Some older transport category aircraft have steering dampers that are vane-type. Nevertheless, they function to steer the nose wheel, as well as to dampen vibration.

SHIMMY DAMPERS

Torque links attached from the stationary upper cylinder of a nose wheel strut to the bottom moveable cylinder or piston of the strut are not sufficient to prevent most nose gear from the tendency to oscillate rapidly, or shimmy, at certain speeds. This vibration wheel shimmy must be controlled through hydraulic damping. The damper can be built integrally within the nose gear, but most often it is an external unit attached between the upper and lower shock struts. It is active during all phases of ground operation while permitting the nose gear steering system to function normally.

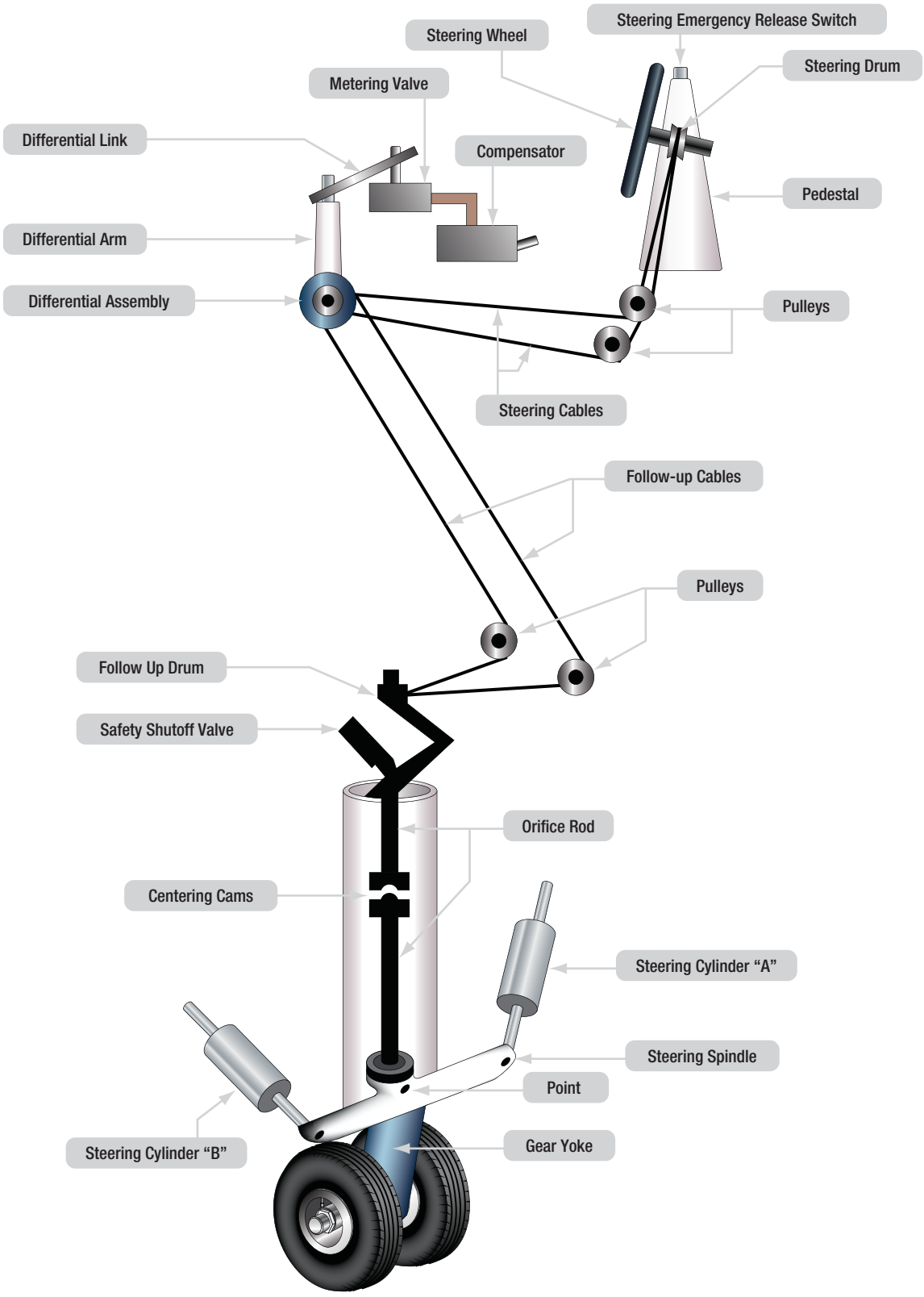


Figure 16-19. Example of a large aircraft hydraulic nose wheel steering system with hydraulic and mechanical units.

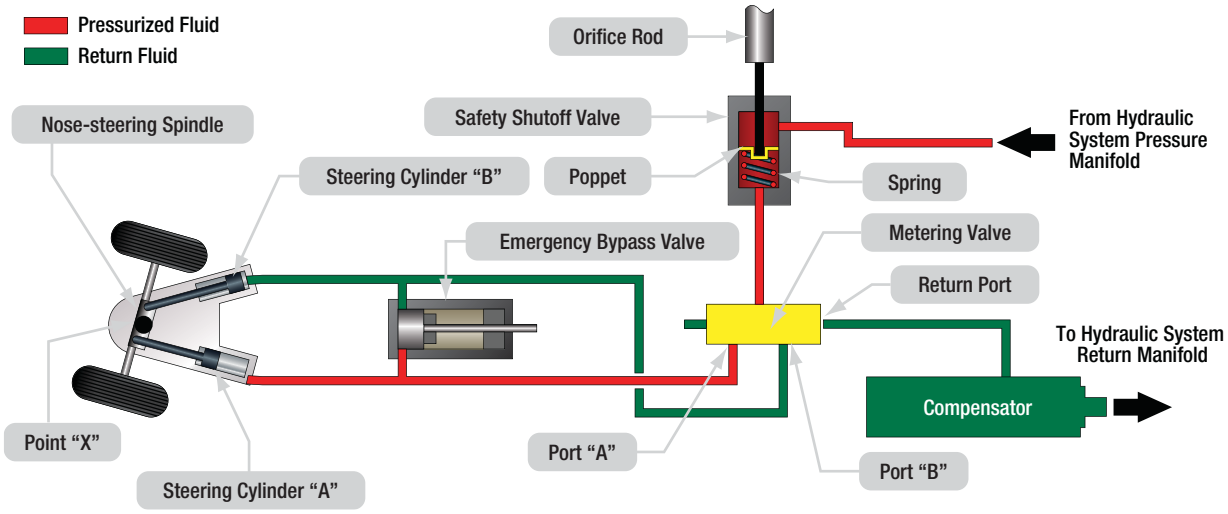


Figure 16-20. Hydraulic system flow diagram of large aircraft nose wheel steering system.

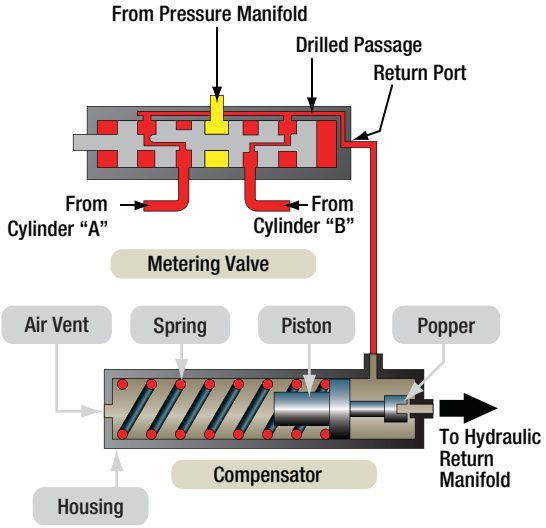


Figure 16-21. Hydraulic system flow diagram of large aircraft nose wheel steering system.

AIRCRAFT WHEELS

Aircraft wheels are an important component of a landing gear system. With tires mounted upon them, they support the entire weight of the aircraft during taxi, takeoff, and landing. The typical aircraft wheel is lightweight, strong, and made from aluminum alloy. Some magnesium alloy wheels also exist. Early aircraft wheels were of single piece construction, much the same as the modern automobile wheel. As aircraft tires were improved for the purpose they serve, they were made stiffer to better absorb the forces of landing without blowing out or separating from the rim. Stretching such a tire over a single piece wheel rim was not possible. A two-piece wheel was developed. Early two-piece aircraft wheels were essentially one-piece wheels with a removable rim to allow mounting access for the tire. These are still found on older aircraft. (***Figure 16-22***) Later, wheels with two nearly symmetrical halves were developed. Nearly all modern aircraft wheels are of this two piece construction. (***Figures 16-23 and 16-24***)

WHEEL CONSTRUCTION

The typical modern two-piece aircraft wheel is cast or forged from aluminum or magnesium alloy. The halves are bolted together and contain a groove at the mating surface for sealing the assembly for use with tubeless tires. The bead seat area of a wheel is where the tire actually contacts the wheel. It is the critical area that accepts the significant tensile loads from the tire during

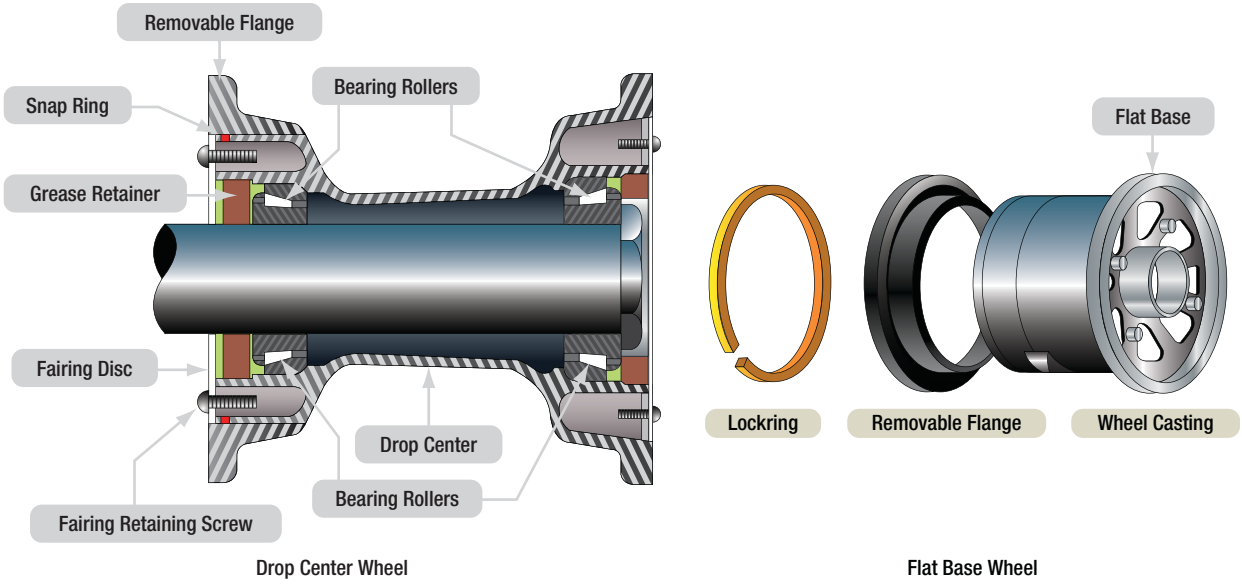


Figure 16-22. Removable flange wheels found on older aircraft are either drop center or flat base types.

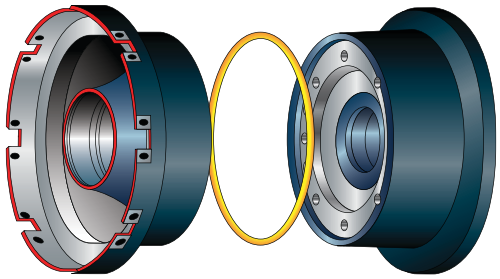


Figure 16-23. Two-piece split-wheel aircraft wheels found on modern light aircraft.

landing. To strengthen this area during manufacturing, the bead seat area is typically rolled to prestress it with a compressive stress load.

INBOARD WHEEL HALF

Wheel halves are not identical. The primary reason for this is that the inboard wheel half must have a means for accepting and driving the rotor(s) of the aircraft brakes that are mounted on both main wheels. Tangs on the rotor are fitted into steel reinforced keyways on many wheels. Other wheels have steel keys bolted to the inner wheel halves. These are made to fit slots in the perimeter of the brake rotor. Some small aircraft wheels have provisions for bolting the brake rotor to the inner wheel half. Regardless, the inner wheel half is distinguishable from the outer wheel half by its brake mounting feature. (***Figure 16-25***)

Both wheel halves contain a bearing cavity formed into the center that accepts the polished steel bearing cup, tapered roller bearing, and grease retainer of a typical wheel bearing set-up. A groove may also be machined to accept a retaining clip to hold the bearing assembly in place when the wheel assembly is removed. The wheel bearings are a very important part of the wheel assembly and are discussed in a later section of this chapter. The inner wheel half of a wheel used on a high performance aircraft is likely to have one or more thermal plugs. (***Figure 16-26***)

During heavy braking, temperatures can become so great that tire temperature and pressure rise to a level resulting in explosion of the wheel and tire assembly. The thermal plug core is filled with a low melting point alloy. Before tire and wheel temperatures reach the point of explosion, the core melts and deflates the tire. The tire must be removed from service, and the wheel must be inspected in accordance with the wheel manufacturer's instructions before return to service if a thermal plug melts.

Adjacent wheel assemblies should also be inspected for signs of damage. A heat shield is commonly installed under the inserts designed to engage the brake rotor to assist in protecting the wheel and tire assembly from overheating. An overinflation safety plug may also be installed in the inner wheel half. This is designed to rupture and release all of the air in the tire should it be over inflated. The fill valve is also often installed in the inner wheel half with the stem extending through holes in the outer wheel half to permit access for inflation and deflation.

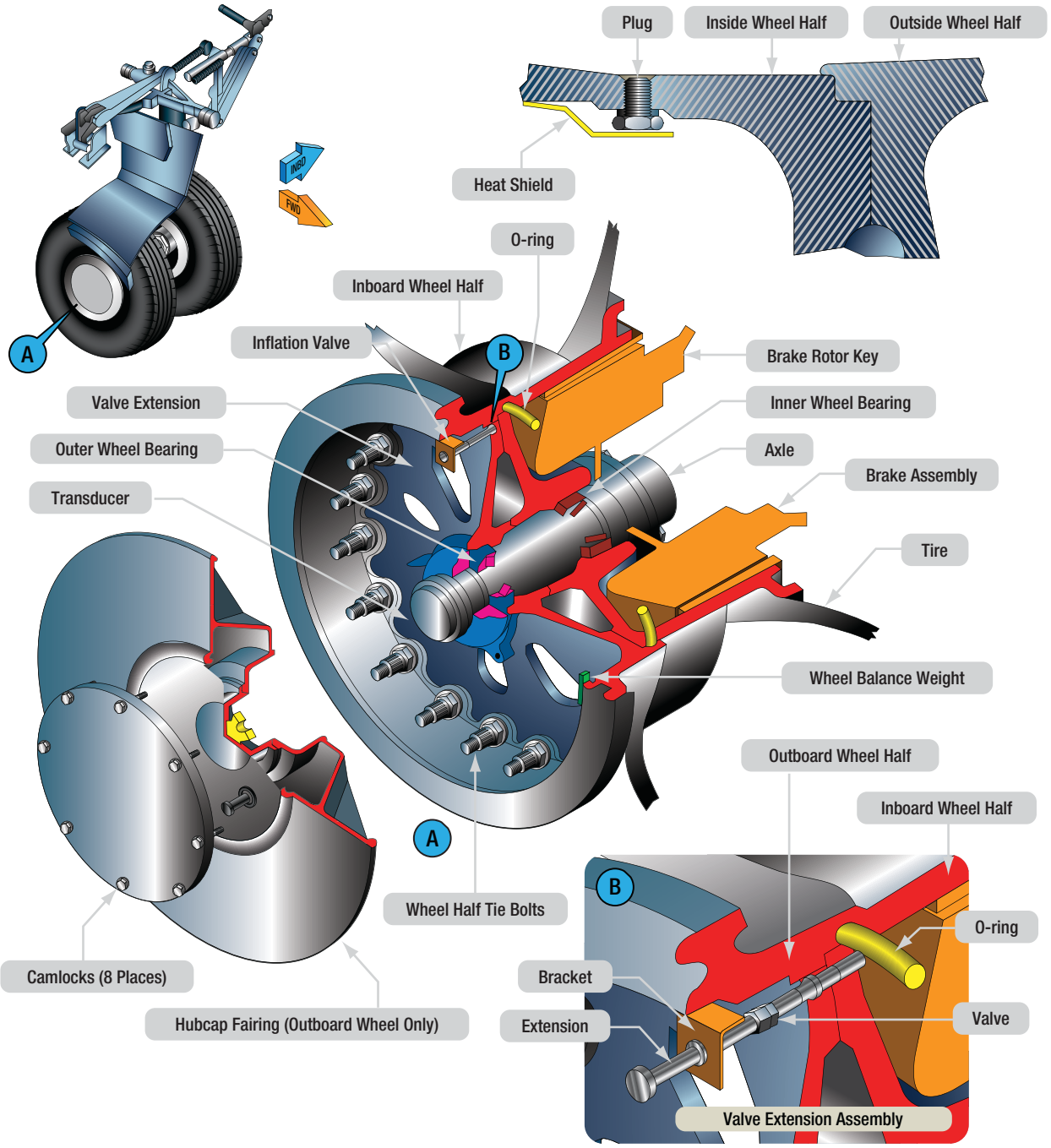


Figure 16-24. Features of a two piece aircraft wheel found on a modern airliner.

OUTBOARD WHEEL HALF

The outboard wheel half bolts to the inboard wheel half to make up the wheel assembly upon which the tire is mounted. The center boss is constructed to receive a bearing cup and bearing assembly as it does on the inboard wheel half. The outer bearing and end of the axle is capped to prevent contaminants from entering this area.

Aircraft with antiskid brake systems typically mount the wheel-spin transducer here. It is sealed and may also serve as a hub cap. The 737 outer wheel half illustrated in ***Figure 16-24*** also has a hub cap fairing over the entire wheel half. This is to fair it with the wind since the outer wheel half does not close behind a gear door on this aircraft. Hub caps may also be found on fixed gear aircraft. The outboard wheel half provides a convenient location of the valve stem used to inflate and deflate tubeless tires. Alternately, it may contain a hole through which a valve stem extension may pass from the inner wheel half or the valve stem itself may fit through such a hole if a tube-type tire is used.



Figure 16-25. Keys on the inner wheel half of an aircraft wheel used to engage and rotate the rotors of a disc brake.

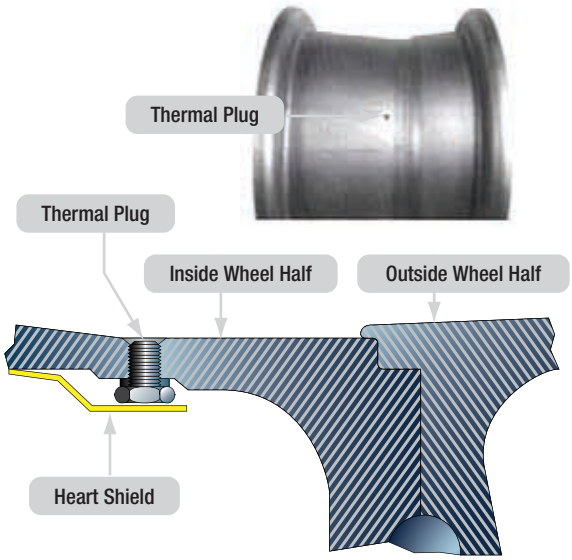


Figure 16-26. Heavy use of the aircraft brakes can cause tire air temperature and pressure to rise to a level resulting in explosion of the wheel assembly. To alleviate this, thermal plug(s) mounted in the inner wheel half of a high performance aircraft wheels are made with a fusible core that melts and releases the air from the tire before explosion.

WHEEL INSPECTION

An aircraft wheel assembly is inspected while on the aircraft as often as possible. A more detailed inspection and any testing or repairs may be accomplished with the wheel assembly removed from the aircraft.

ON AIRCRAFT INSPECTION

The general condition of the aircraft wheel assemblies can be inspected while on the aircraft. Any signs of suspected damage that may require removal of the wheel assembly from the aircraft should be investigated.

PROPER INSTALLATION

The landing gear area is such a hostile environment that the technician should inspect the landing gear including the wheels, tires, and brakes whenever possible. Proper installation of the wheels should not be taken for granted. All wheel tie bolts and nuts must be in place and secure A missing bolt is grounds for removal, and a thorough inspection of the wheel halves in accordance with the wheel manufacturer's procedures must be performed due to the stresses that may have occurred. The wheel hub dust cap and antiskid sensor should also be secure. The inboard wheel half should interface with the brake rotor with no signs of chafing or excessive movement. All brake keys on the wheel must be present and secure.

Examine the wheels for cracks, flaked paint, and any evidence of overheating. Inspect thermal plugs to ensure no sign of the fusible alloy having been melted. Thermal plugs that have permitted pressure loss in the tire require that the wheel assembly be removed for inspection. All other wheels with brakes and thermal plugs should be inspected closely while on the aircraft to determine if they too have overheated. Each wheel should be observed overall to ensure it is not abnormally tilted. Flanges should not be missing any pieces, and there should be no areas on the wheel that show significant impact damage.

AXLE NUT TORQUE

Axle nut torque is of extreme importance on an aircraft wheel installation. If the nut is too loose, the bearing and wheel assembly may have excessive movement. The bearing cup(s) could loosen and spin, which could damage the wheel. There could also be impact damage from the bearing rollers which leads to bearing failure. An over-torqued axle nut prevents the bearing from properly accepting the weight load of the aircraft. The bearing spins without sufficient lubrication to absorb the heat caused by the higher friction level. This too leads to bearing failure. All aircraft axle nuts must be installed and torqued in accordance with the airframe manufacturer's maintenance procedures. (***Figure 16-27***)

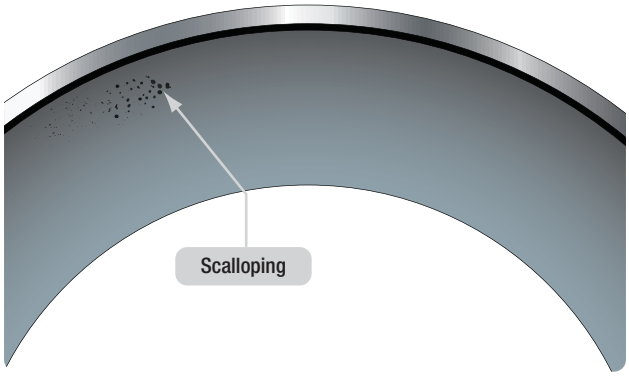


Figure 16-27. Improper loose torque on the axle nut can cause excessive end play leading to bearing race damage known as scalloping.

OFF AIRCRAFT WHEEL INSPECTION

Discrepancies found while inspecting a wheel mounted on the aircraft may require further inspection with the wheel removed from the aircraft. Other items such as bearing condition, can only be performed with the wheel assembly removed. A complete inspection of the wheel requires that the tire be removed from the wheel rim. Observe the following caution when removing a wheel assembly from an aircraft.

CAUTION: Deflate the tire before starting the procedure of removing the wheel assembly from the aircraft. Wheel assemblies have been known to explode while removing the axle nut, especially when dealing with high pressure, high performance tires. The torque of the nut can be the only force holding together a defective wheel or one with broken tie bolts. When loosened, the high internal pressure of the tire can create a catastrophic failure that could be lethal to the technician. It is also important to let aircraft tires cool before removal. Three hours or more is needed for cool down. Approach the wheel assembly from the front or rear, not broadside. Do not stand in the path of the released air and valve core trajectory when removing air from the tire as it could seriously injure the technician should it release from the valve stem.

NOTE: As a precautionary measure, remove only one tire and wheel assembly from a pair at a time. This leaves a tire and wheel assembly in place should the aircraft fall off its jack, resulting in less chance of damage to the aircraft and injury to personnel.

LOOSENING THE TIRE FROM THE WHEEL RIM

After inflation and usage, an aircraft tire has a tendency to adhere to the wheel, and the bead must be broken to remove the tire. There are mechanical and hydraulic presses designed for this purpose. In the absence of a device specifically made for the job, an arbor press can be used with patience working sequentially around the wheel as close as possible to the bead. (***Figure 16-28***)

As stated above, there should be no air pressure in the tire while it is being pressed off of the wheel. Never pry a tire off of the rim with a screwdriver or other device. The wheels are relatively soft. Any nick or deformation causes a stress concentration that can easily lead to wheel failure.

DISASSEMBLY OF THE WHEEL

Disassembly of the wheel should take place in a clean area on a flat surface, such as a table. Remove the wheel bearing first and set aside for cleaning and inspecting. The tie bolts can then be removed. Do not use an impact tool to disassemble the tie bolts. Aircraft wheels are made of relatively soft aluminum and magnesium alloys. They are not designed to receive the repeated hammering of an impact tool and will be damaged if used.

CLEANING THE WHEEL ASSEMBLY

Clean the wheel halves with the solvent recommended by the wheel manufacturer. Use of a soft brush helps this process. Avoid abrasive techniques, materials, and tools, such as scrapers, capable of removing the finish off of the wheel. Corrosion can quickly form and weaken the wheel if the finish is missing in an area. When the wheels are clean, they can be dried with compressed air.

INSPECTION OF THE WHEEL HALVES

A thorough visual inspection of each wheel half should be conducted for discrepancies specified in the wheel manufacturer's maintenance data. Use of a magnifying glass is recommended. Corrosion is one of the most common problems encountered while inspecting wheels. Locations where moisture is trapped should be checked closely. It is possible to dress out some corrosion according to the manufacturer's instructions. An approved protective surface treatment and finish must be applied before returning the wheel to service. Corrosion beyond stated limits is cause for rejection of the wheel.



Figure 16-28. Tire beads must be broken from the wheel to remove the tire. A mechanical removal tool designed for breaking the bead is shown in (A); a hydraulic press designed with the capacity for large aircraft wheels is shown in (B); and an arbor press is shown in (C). All are tools available to the technician for this purpose.

In addition to corrosion, cracks in certain areas of the wheel are particularly prevalent. One such area is the bead seat area. (***Figure 16-29***) The high stress of landing is transferred to the wheel by the tire in this contact area. Hard landings produce distortion or cracks that are very difficult to detect.

This is a concern on all wheels and is most problematic in high-pressure, forged wheels. Dye penetrant inspection is generally ineffective when checking for cracks in the bead area. There is a tendency for cracks to close up tightly once the tire is dismounted, and the stress is removed from the metal. Eddy current inspection of the bead seat area is required. Follow the wheel manufacturer's instruction when performing the eddy current check. The wheel brake disc drive key area is another area in which cracks are common. The forces experienced when the keys drive the disc against the stopping force of the brakes are high. Generally, a dye penetrant test is sufficient to reveal cracks in this area. All drive keys should be secure with no movement possible. No corrosion is permitted in this area. (***Figure 16-30***)

WHEEL TIE BOLT INSPECTION

Wheel half tie bolts are under great stress while in service and require inspection. The tie bolts stretch and change dimension usually at the threads and under the bolt head. These are areas where cracks are most common. Magnetic particle inspection can reveal these cracks. Follow the maintenance manual procedures for inspecting tie bolts.

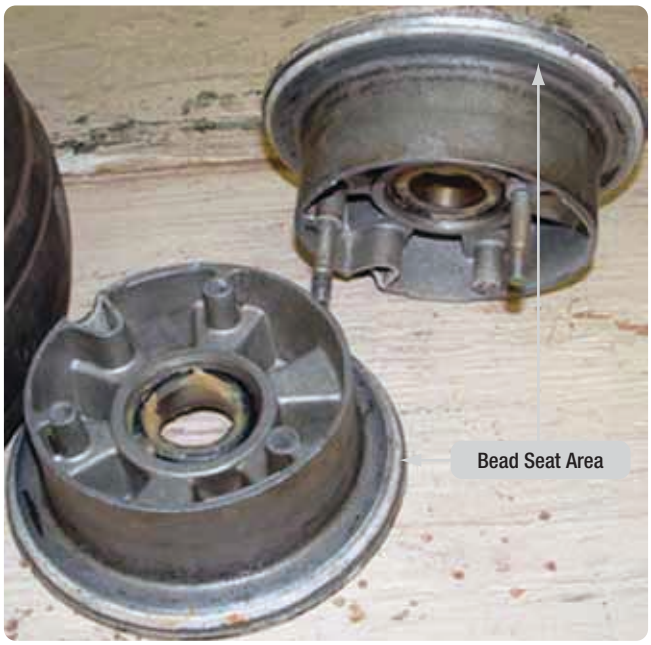


Figure 16-29. The bead seat areas of a light aircraft wheel set. Eddy current testing for cracks in the bead seat area is common.

KEY AND KEY SCREW INSPECTION

On most aircraft inner wheel halves, keys are screwed or bolted to the wheel to drive the brake disc(s). The drive keys are subject to extreme forces when the brakes are applied. As mentioned, there should be no movement between the wheel and the keys. The bolts should be checked for security, and the area around the keys should be inspected for cracks. There is also a limitation on how worn the keys can be since too much wear allows excessive movement. The wheel manufacturer's maintenance instructions should be used to perform a complete inspection of this critical area.

FUSIBLE PLUG INSPECTION

Fusible plugs or thermal plugs must be inspected visually. These threaded plugs have a core that melts at a lower temperature than the outer part of the plug. This is to release air from the tire should the temperature rise to a dangerous level. A close inspection should reveal whether any core has experienced deformation that might be due to high temperature. If detected, all thermal plugs in the wheel should be replaced with new plugs. (***Figure 16-31***)

BALANCE WEIGHTS

The balance of an aircraft wheel assembly is important. When manufactured, each wheel set is statically balanced. Weights are added to accomplish this if needed. They are a permanent part of the wheel assembly and must be installed to use the wheel. The balance weights are bolted to the wheel halves and can be removed when cleaning and inspecting the wheel. They must be re-fastened in their original position. When a tire is mounted to a wheel, balancing of the wheel and tire assembly may require that additional weights be added. These are usually installed around the circumference of the outside of the wheel and should not be taken as substitutes for the factory wheel set balance weights. (***Figure 16-32***)

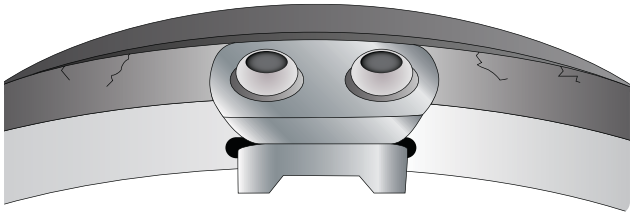


Figure 16-30. Inspection for cracks in the wheel disc drive key area is performed with dye penetrant on many wheels.

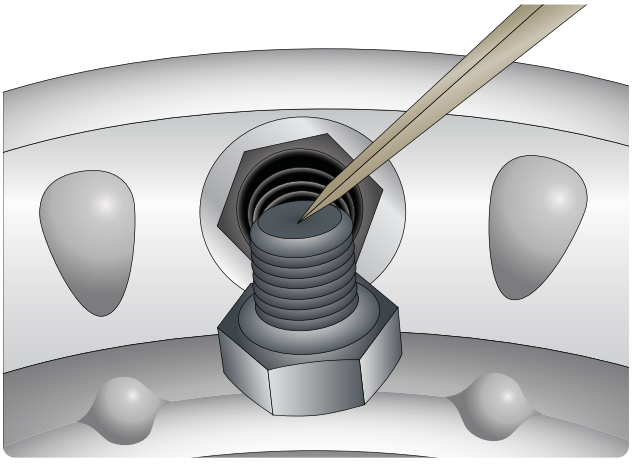


Figure 16-31. Visually inspect the core of a thermal or fusible plug for deformation associated with heat exposure. Replace all of the plugs if any appear to have begun to deform.

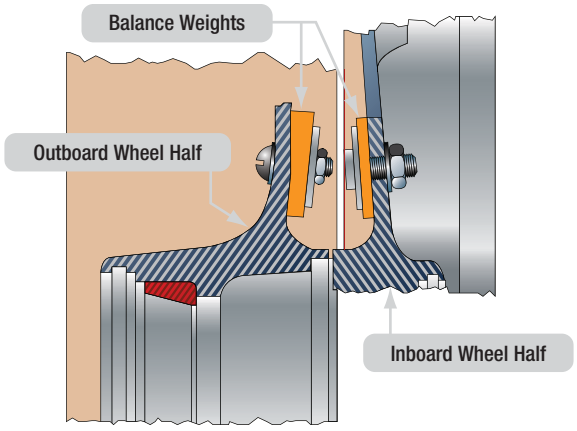


Figure 16-32. Two piece aircraft wheels are statically balanced when manufactured and may include weights attached to each wheel half that must stay with the wheel during its entire serviceable life.

CLEANING THE WHEEL BEARINGS

The bearings should be removed from the wheel to be cleaned with the recommended solvent, such as varsol, naptha, or Stoddard® solvent. Soaking the bearings in solvent is acceptable to loosen any dried-on grease. Bearings are brushed clean with a soft bristle brush and dried with compressed air. Never rotate the bearing while drying with compressed air. The high speed metal to metal contact of the bearing rollers with the race causes heat that damages the metal surfaces. The bearing parts could also cause injury should the bearing come apart. Always avoid steam cleaning of bearings. The surface finish of the metals will be compromised leading to early failure.

INSPECTION OF WHEEL BEARINGS

Once cleaned, the wheel bearing is inspected. There are many unacceptable conditions of the bearing and bearing cup, which are grounds for rejection. In fact, nearly any flaw detected in a bearing assembly is likely to be grounds for replacement. Common conditions of a bearing that are cause for rejection are as follows:

- Galling – caused by rubbing of mating surfaces. The metal gets so hot it welds, and the surface metal is destroyed as the motion continues and pulls the metal apart in the direction of motion. (***Figure 16-33***)
- Spalling – a chipped away portion of the hardened surface of a bearing roller or race. (***Figure 16-34***)
- Overheating – caused by lack of sufficient lubrication results in a bluish tint to the metal surface. The ends of the rollers shown were overheated causing the metal to flow and deform, as well as discolor. The bearing cup raceway is usually discolored as well. (***Figure 16-35***)
- Brinelling – caused by excessive impact. It appears as indentations in the bearing cup raceways. Any static overload or severe impact can cause true brinelling that leads to vibration and premature bearing failure. (***Figure 16-36***)
- False Brinelling – caused by vibration of the bearing while in a static state. Even with a static overload, lubricant can be forced from between the rollers and the raceway. Submicroscopic particles removed at the points of metal-to-metal contact oxidize. They work to remove more particles spreading the damage. This is also known as frictional corrosion. It can be identified by a rusty coloring of the lubricant. (***Figure 16-37***)
- Staining and surface marks – located on the bearing cup as grayish black streaks with the same spacing as the rollers and caused by water that has gotten into the bearing. It is the first stage of deeper corrosion that follows. (***Figure 16-38***)
- Etching and corrosion – caused when water and the damage caused by water penetrates the surface treatment of the bearing element. It appears as a reddish/brown discoloration. (***Figure 16-39***)
- Bruising – caused by fine particle contamination possibly from a bad seal or improper maintenance of bearing cleanliness. It leaves a less than smooth surface on the bearing cup. (***Figure 16-40***)



Figure 16-33. Galling is caused by rubbing of mating surfaces. The metal gets so hot it welds, and the surface metal is destroyed as the motion continues and pulls the metal apart in the direction of motion.



Figure 16-34. Spalling is a chipped away portion of the hardened surface of a bearing roller or race.

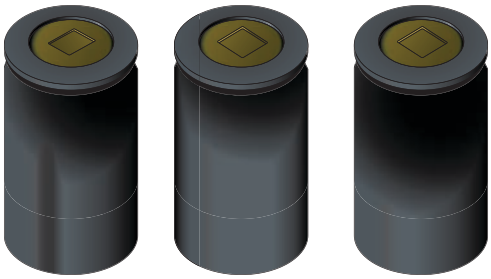


Figure 16-35. Overheating caused by lack of sufficient lubrication results in a bluish tint to the metal surface. The ends of the rollers shown were overheated causing the metal to flow and deform, as well as discolor. The bearing cup raceway is usually discolored as well.

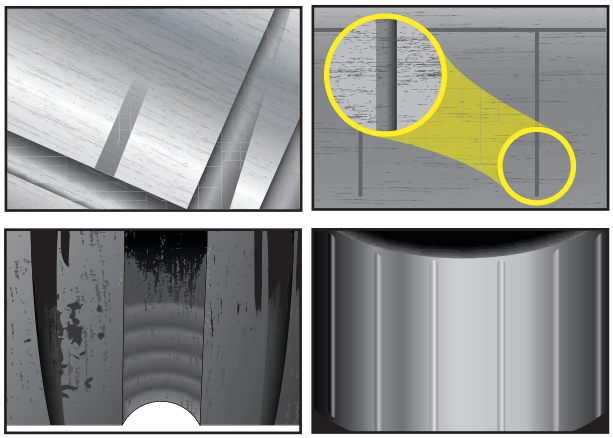


Figure 16-36. Brinelling is caused by excessive impact. It appears as indentations in the bearing cup raceways. Any static overload or severe impact can cause true brinelling, which leads to vibration and premature bearing failure.

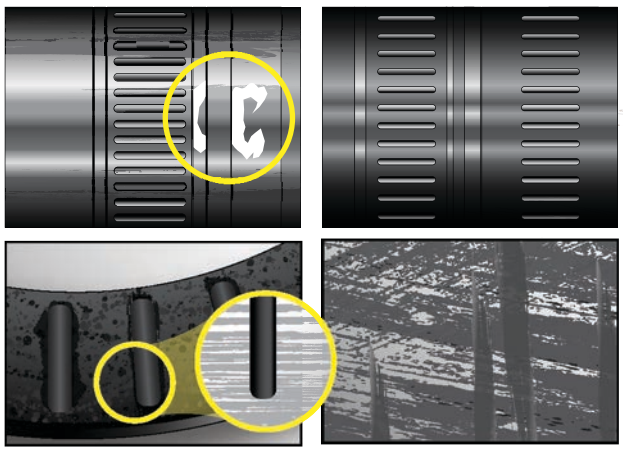


Figure 16-37. False brinelling is caused by vibration of the bearing while in a static state. Even with a static overload, it can force the lubricant from between the rollers and the raceway. Submicroscopic particles removed at the points of metal-to-metal contact oxidize. They work to remove more particles spreading the damage. This is also known as frictional corrosion. It can be identified by a rusty coloring of the lubricant.



Figure 16-38. Staining and surface marks on the bearing cup that are grayish black streaks with the same spacing as the rollers are caused by water that has gotten into the bearing. It is the first stage of deeper corrosion that will follow.



Figure 16-39. Etching and corrosion is caused when water, and the damage caused by water, penetrates the surface treatment of the bearing element. It appears as a reddish/brown discoloration.



Figure 16-40. Bruising is caused by fine particle contamination possibly from a bad seal or improper maintenance of bearing cleanliness. It leaves a less than smooth surface on the bearing cup.

The bearing cup does not require removal for inspection; however, it must be firmly seated in the wheel half boss. There should be no evidence that a cup is loose or able to spin. (***Figure 16-41***) The cup is usually removed by heating the wheel in a controlled oven and pressing it out or tapping it out with a non-metallic drift.

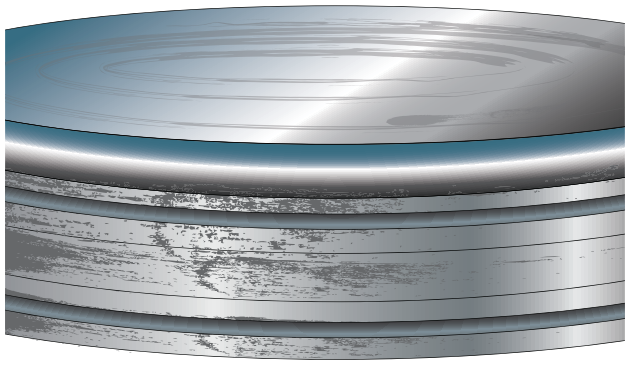


Figure 16-41. Bearing cups should be tight in the wheel boss and should never rotate. The outside of a bearing cup that was spinning while installed in the wheel is shown.

The installation procedure is similar. The wheel is heated and the cup is cooled with dry ice before it is tapped into place with a non-metallic hammer or drift. The outside of the race is often sprayed with primer before insertion. Consult the wheel manufacturer's maintenance manual for specific instructions.

BEARING HANDLING AND LUBRICATION

Periodically, wheel bearings must be removed, cleaned, inspected, and lubricated. When cleaning a wheel bearing, use the recommended cleaning solvent.

Do not use gasoline or jet fuel. Dry the bearing by directing a blast of dry air between the rollers. Do not direct the air so that it spins the bearing as without lubrication, this could cause the bearing to fly apart resulting in injury. When inspecting the bearing, check for defects that would render it unserviceable, such as cracks, flaking, broken bearing surfaces, roughness due to impact pressure or surface wear, corrosion or pitting, discoloration from excessive heat, cracked or broken bearing cages, and scored or loose bearing cups or cones that would affect proper seating on the axle or wheel. If any discrepancies are found, replace the bearing with a serviceable unit. Bearings should be lubricated immediately after cleaning and inspection to prevent corrosion.

Handling of bearings is of the utmost importance. Contamination, moisture, and vibration, even while the bearing is in a static state, can ruin a bearing. Avoid conditions where these may affect bearings and be sure to install and torque bearings into place according manufacturer's instructions. Proper lubrication is a partial deterrent to negative environmental impacts on a bearing. Use the lubricant recommended by the manufacturer. Use of a pressure bearing packing tool or adapter is also recommended as the best method to remove any contaminants from inside the bearing that may have remained after cleaning. (***Figure 16-42***)

To lubricate a tapered roller bearing without the use of a bearing lubrication tool, place a small amount of the approved grease on the palm of the hand. Grasp the bearing with the other hands and press the larger diameter side of the bearing into the grease to force it completely through the space between the bearing rollers and the cone. Gradually turn the bearing so that all of the rollers have been completely packed with grease. (***Figure 16-43***)



Figure 16-42. A pressure bearing lubricating tool.



Figure 16-43. Packing grease into a clean, dry bearing can be done by hand in the absence of a bearing grease tool. Press the bearing into the grease on the palm of the hand until it passes completely through the gap between the rollers and the inner race all the way around the bearing.

AIRCRAFT BRAKES

Very early aircraft have no brake system to slow and stop the aircraft while it is on the ground. Instead, they rely on slow speeds, soft airfield surfaces, and the friction developed by the tail skid to reduce speed during ground operation. Brake systems designed for aircraft became common after World War I as the speed and complexity of aircraft increased and the use of smooth, paved runway surfaces proliferated. All modern aircraft are equipped with brakes. Their proper functioning is relied upon for safe operation of the aircraft on the ground. The brakes slow the aircraft and stop it in a reasonable amount of time. They hold the aircraft stationary during taxi.

On most aircraft, each of the main wheels is equipped with a brake unit. The nose wheel or tail wheel does not have a brake. In the typical brake system, mechanical and/or hydraulic linkages to the rudder pedals allow the pilot to control the brakes. Pushing on the top of the right rudder pedal activates the brake on the right main wheel(s) and pushing on the top of the left rudder pedal operates the brake on the left main wheel(s). The basic operation of brakes involves converting the kinetic energy of motion into heat energy through the creation of friction. A great amount of heat is developed and forces on the brake system components are demanding. Proper adjustment, inspection, and maintenance of the brakes is essential for effective operation.

TYPES AND CONSTRUCTION OF AIRCRAFT BRAKES

Modern aircraft typically use disc brakes. The disc rotates with the turning wheel assembly while a stationary caliper resists the rotation by causing friction against the disc when the brakes are applied. The size, weight, and landing speed of the aircraft influence the design and complexity of the disc brake system. Single, dual, and multiple disc brakes are common types of brakes. Segmented rotor brakes are used on large aircraft. Expander tube brakes are found on older large aircraft. The use of carbon discs is increasing in the modern aviation fleet.

SINGLE DISC BRAKES

Small, light aircraft typically achieve effective braking using a single disc keyed or bolted to each wheel. As the wheel turns, so does the disc. Braking is accomplished by applying friction to both sides of the disc from a non-rotating caliper bolted to the landing gear axle flange. Pistons in the caliper housing under hydraulic pressure force wearable brake pads or linings against the disc when the brakes are applied. Hydraulic master cylinders connected to the rudder pedals supply the pressure when the upper halves of the rudder pedals are pressed.

FLOATING DISC BRAKES

A floating disk brake is illustrated in ***Figure 16-44***. A more detailed, exploded view of this type of brake is shown in ***Figure 16-45***. The caliper straddles the disc. It has three cylinders bored through the housing, but on other brakes this number may vary. Each cylinder



Figure 16-44. A single disc brake is a floating-disc, fixed caliper brake.

accepts an actuating piston assembly comprised mainly of a piston, a return spring, and an automatic adjusting pin. Each brake assembly has six brake linings or pucks.

Three are located on the ends of the pistons, which are in the outboard side of the caliper. They are designed to move in and out with the pistons and apply pressure to the outboard side of the disc. Three more linings are located opposite of these pucks on the inboard side of the caliper. These linings are stationary.

The brake disc is keyed to the wheel. It is free to move laterally in the key slots. This is known as a floating disk. When the brakes are applied, the pistons move out from the outboard cylinders and their pucks contact the disc. The disc slides slightly in the key slots until the inboard stationary pucks also contact the disc. The result is a fairly even amount of friction applied to each side of the disc and thus, the rotating motion is slowed. When brake pressure is released, the return spring in each piston assembly forces the piston back away from the disc. The spring provides a preset clearance between each puck and the disc. The self-adjusting feature of the brake maintains the same clearance, regardless of the amount of wear on the brake pucks. The adjusting pin on the back of each piston moves with the piston through a frictional pin grip.

When brake pressure is relieved, the force of the return spring is sufficient to move the piston back away from the brake disc, but not enough to move the adjusting pin held by the friction of the pin grip. The piston stops when it contacts the head of the adjusting pin. Thus, regardless of the amount of wear, the same travel of the piston is required to apply the brake. The stem of the pin protruding through the cylinder head serves as a wear indicator. The manufacturer's maintenance information states the minimum length of the pin that needs to be protruding for the brakes to be considered airworthy. (***Figure 16-46***)

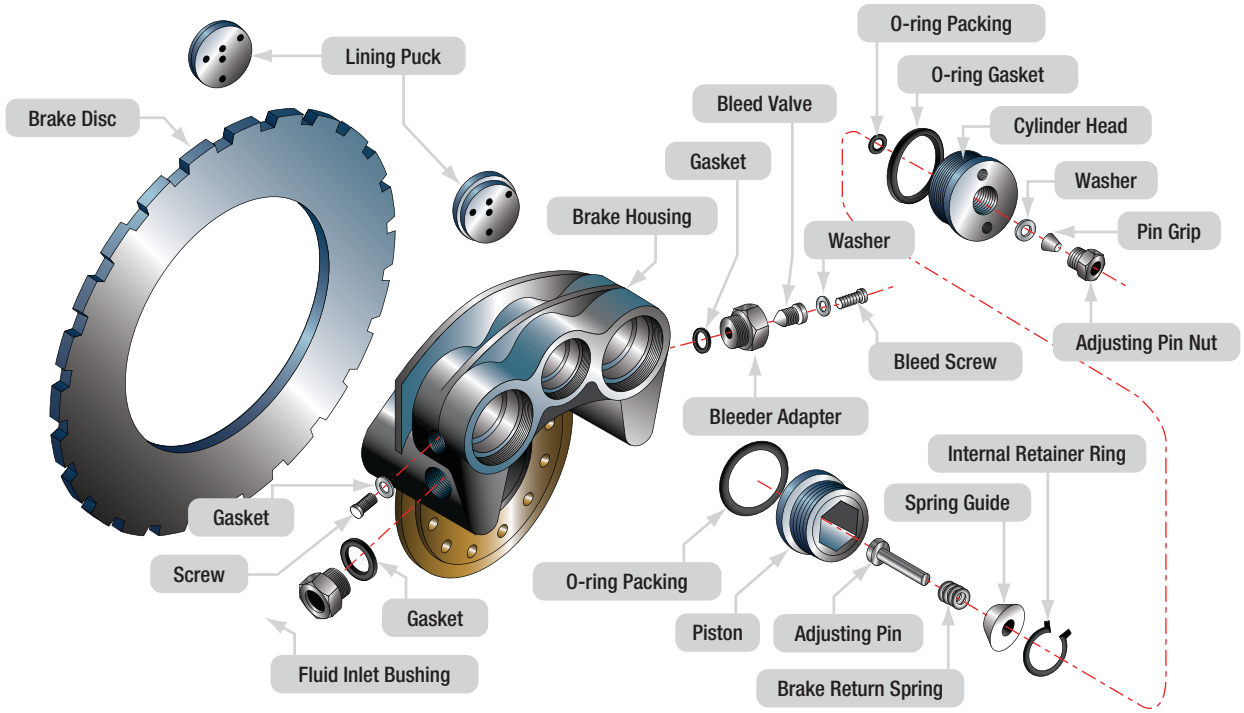


Figure 16-45. An exploded view of a single-disc brake assembly found on a light aircraft.

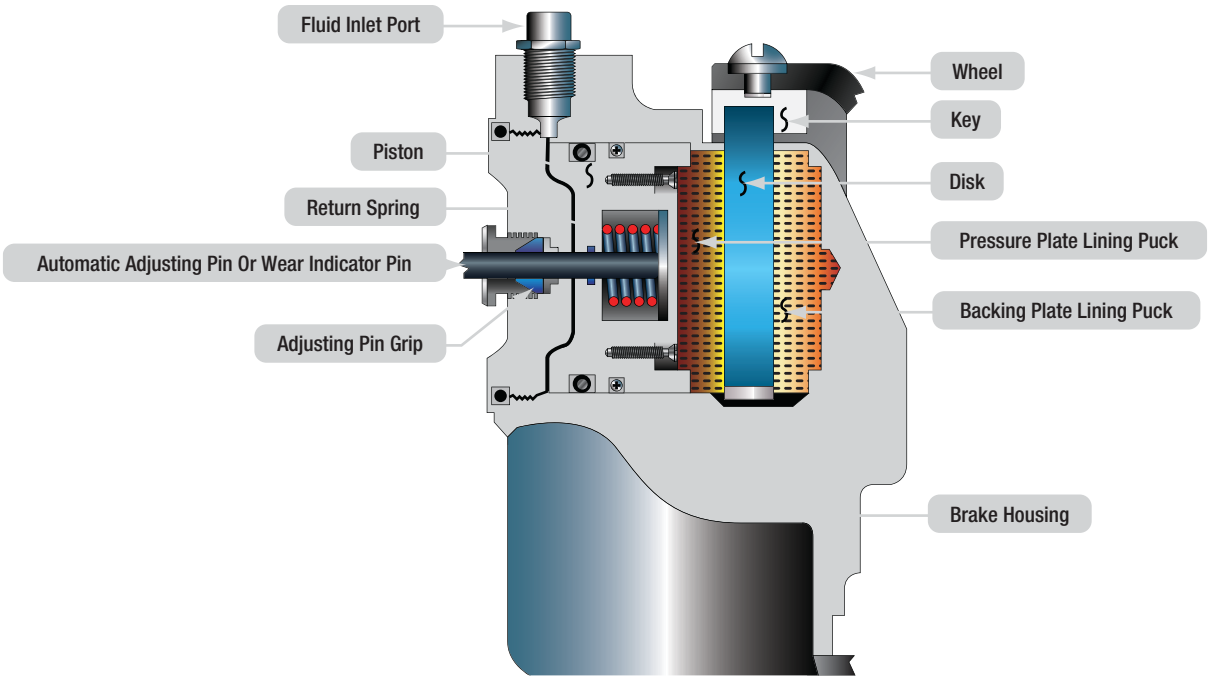


Figure 16-46. A cross-sectional view of a Goodyear single-disc brake caliper illustrates the adjusting pin assembly that doubles as a wear indicator.

The brake caliper has the necessary passages machined into it to facilitate hydraulic fluid movement and the application of pressure when the brakes are utilized. The caliper housing also contains a bleed port used by the technician to remove unwanted air from the system. Brake bleeding, as it is known, should be done in accordance with the manufacturer's maintenance instructions.

FIXED-DISC BRAKES

Even pressure must be applied to both sides of the brake disc to generate the required friction and obtain consistent wear properties from the brake linings. The floating disc accomplishes this as described above. It can also be accomplished by bolting the disc rigidly to the wheel and allowing the brake caliper and linings to float laterally when pressure is applied. This is the design of a common fixed-disc brake used on light aircraft. The brake is manufactured by the Cleveland Brake Company and is shown in ***Figure 16-47***. An exploded detail view of the same type of brake is shown in ***Figure 16-48***.

The fixed-disk, floating-caliper design allows the brake caliper and linings to adjust position in relationship to the disc. Linings are riveted to the pressure plate and backplate. Two anchor bolts that pass through the pressure plate are secured to the cylinder assembly.

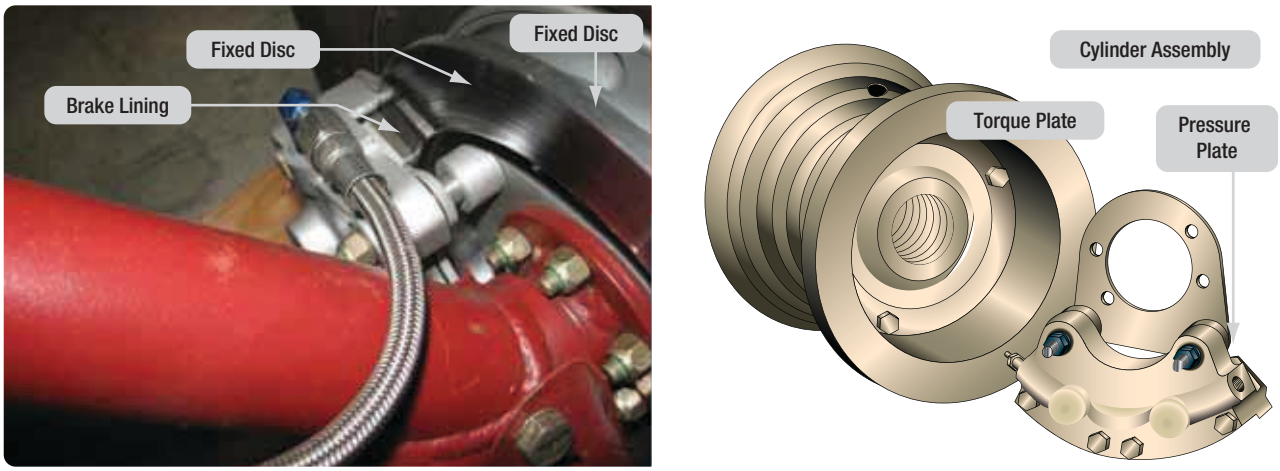


Figure 16-47. A Cleveland brake on a light aircraft is a fixed-disc brake. It allows the brake caliper to move laterally on anchor bolts to deliver even pressure to each side of the brake disc.

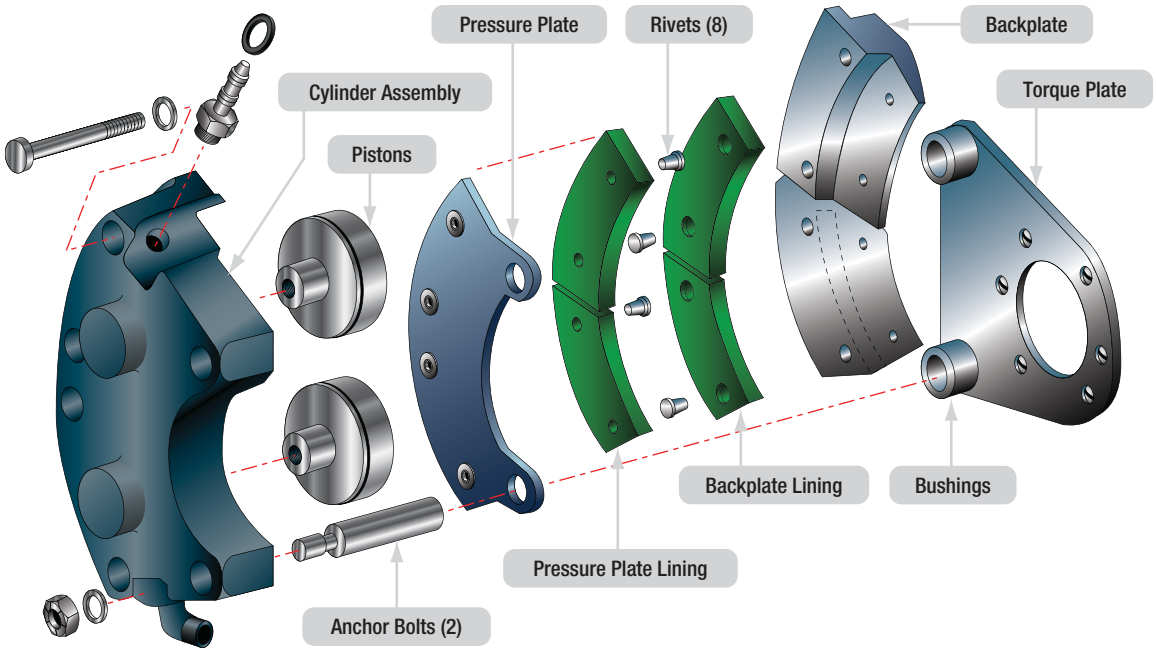


Figure 16-48. An exploded view of a dual-piston Cleveland brake assembly.

The other ends of the bolts are free to slide in and out of bushings in the torque plate, which is bolted to the axle flange. The cylinder assembly is bolted to the backplate to secure the assembly around the disc. When pressure is applied, the caliper and linings center on the disc via the sliding action of the anchor bolts in the torque plate bushings. This provides equal pressure to both sides of the disc to slow its rotation. A unique feature of the Cleveland brake is that the linings can be replaced without removing the wheel. Unbolting the cylinder assembly from the backplate allows the anchor bolts to slide out of the torque plate bushings. The entire caliper assembly is then free and provides access to all of the components.

Maintenance requirements on all single disc brake systems are similar to those on brake systems of any type. Regular inspection for any damage and for wear on the linings and discs is required. Replacement of parts worn beyond limits is always followed by an operational check. The check is performed while taxiing the aircraft. The braking action for each main wheel should be equal with equal application of pedal pressure. Pedals should be firm, not soft or spongy, when applied. When pedal pressure is released, the brakes should release without any evidence of drag.

DUAL-DISC BRAKES

Dual-disc brakes are used on aircraft where a single disc on each wheel does not supply sufficient braking friction. Two discs are keyed to the wheel instead of one. A center carrier is located between the two discs. It contains linings on each side that contact each of the discs when the brakes are applied. The caliper mounting bolts are long and mount through the center carrier, as well as the backplate which bolts to the housing assembly. (***Figure 16-49***)

MULTIPLE-DISC BRAKES

Large, heavy aircraft require the use of multiple-disc brakes. Multiple-disc brakes are heavy duty brakes designed for use with power brake control valves or power boost master cylinders, which is discussed later in this chapter. The brake assembly consists of an extended bearing carrier similar to a torque tube type unit that bolts to the axle flange. It supports the various brake parts, including an annular cylinder and piston, a series of steel discs alternating with copper or bronze-plated discs, a backplate, and a backplate retainer.

The steel stators are keyed to the bearing carrier, and the copper or bronze plated rotors are keyed to the rotating wheel. Hydraulic pressure applied to the piston causes the entire stack of stators and rotors to be compressed. This creates enormous friction and heat and slows the rotation of the wheel. (***Figure 16-50***)

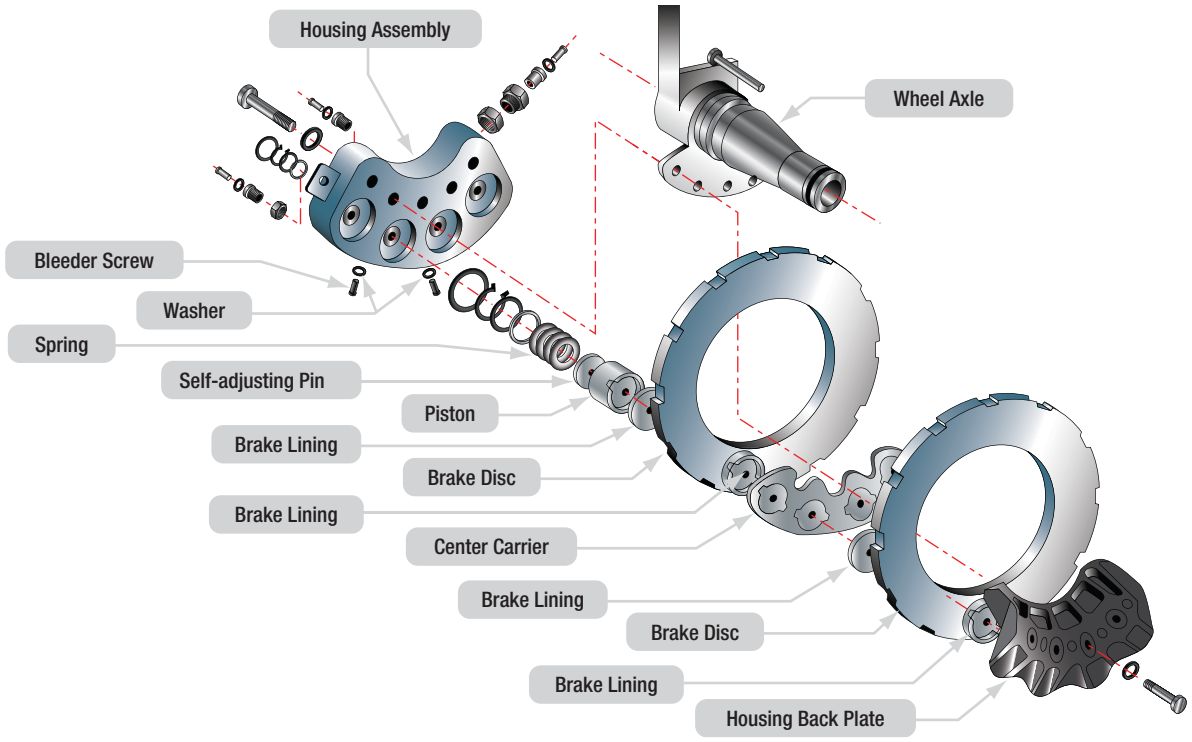


Figure 16-49. A dual-disc brake is similar to a single-disc brake. It uses a center carrier to hold brake linings against each of the discs.

As with the single and dual-disc brakes, retracting springs return the piston into the housing chamber of the bearing carrier when hydraulic pressure is relieved. The hydraulic fluid exits the brake to the return line through an automatic adjuster. The adjuster traps a predetermined amount of fluid in the brakes that is just sufficient to provide the correct clearances between the rotors and stators. (***Figure 16-51***)

Brake wear is typically measured with a wear gauge that is not part of the brake assembly. These types of brake are typically found on older transport category aircraft. The rotors and stators are relatively thin, only about 1/8-inch thick. They do not dissipate heat very well and have a tendency to warp.

SEGMENTED ROTOR-DISC BRAKES

The large amount of heat generated while slowing the rotation of the wheels on large and high performance aircraft is problematic. To better dissipate this heat, segmented rotor disc brakes have been developed. Segmented rotor-disc brakes are multiple-disc brakes but of more modern design than the type discussed earlier. There are many variations. Most feature numerous elements that aid in the control and dissipation of heat. Segmented rotor-disc brakes are heavy-duty brakes especially adapted for use with the high pressure hydraulic systems of power brake systems. Braking is accomplished by means of several sets of stationary, high friction type brake linings that make contact with rotating segments. The rotors are constructed with slots or in sections with space between them, which helps dissipate heat and give the brake its name. Segmented rotor multiple-disc brakes are the standard brake used on high performance and air carrier aircraft. An exploded view of one type of segmented rotor brake assembly is shown in ***Figure 16-52***.

The description of a segmented rotor brake is very similar to the multiple-disc type brake previously described. The brake assembly consists of a carrier, a piston and piston cup seal, a pressure plate, an auxiliary stator plate, rotor segments, stator plates, automatic adjusters, and a backing plate. The carrier assembly, or brake housing with torque tube, is the basic unit of the segmented rotor brake. It is the part that attaches to the landing gear shock strut flange upon which the other components of the brake are assembled. On some brakes, two grooves or cylinders are machined into the carrier to receive the piston cups and pistons. (***Figure 16-52***)

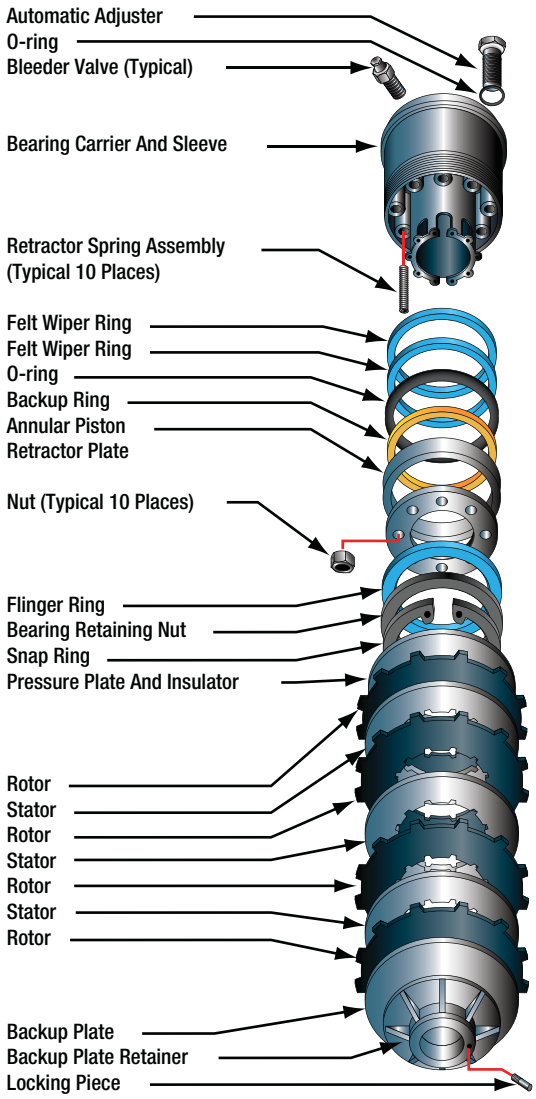


Figure 16-50. A multiple disc brake with bearing carrier upon which the parts of the brake are assembled including an annular cylinder and piston assembly that apply pressure evenly to a stack of rotors and stators.

Most segmented rotor-disc brakes have numerous individual cylinders machined into the brake housing into which fit the same number of actuating pistons. Often, these cylinders are supplied by two different hydraulic sources, alternating every other cylinder from a single source. If one source fails, the brake still operates sufficiently on the other. (***Figure 16-53***) External fittings in the carrier or brake housing admit the hydraulic fluid. A bleed port can also be found.

A pressure plate is a flat, circular, high-strength steel, non-rotating plate notched on the inside circumference to fit over the stator drive sleeves or torque tube spines. The brake actuating pistons contact the pressure plate. Typically, an insulator is used between the piston head and the pressure plate to impede heat conduction from the brake discs. The pressure plate transfers the motion of the pistons to the stack of rotors and stators that compress to slow the rotation of the wheels. On most designs, brake lining material attached directly to the pressure plate contacts the first rotor in the stack to transfer the motion of the piston(s). (***Figure 16-52***)

An auxiliary stator plate with brake lining material on the side opposite the pressure plate can also be used. Any number of alternating rotors and stators are sandwiched under hydraulic pressure against the backing plate of the brake assembly when the brakes are applied. The backing plate is a heavy steel plate bolted to the housing or torque tube at a fixed dimension from the carrier housing. In most cases, it has brake lining material attached to it and contacts the last rotor in the stack. (***Figure 16-52***)

Stators are flat plates notched on the internal circumference to be held stationary by the torque tube spines. They have wearable brake lining material riveted or adhered to each side to make contact with adjacent rotors. The liner is typically constructed of numerous isolated blocks. (***Figure 16-52***)

The space between the liner blocks aids in the dissipation of heat. The composition of the lining materials vary. Steel is often used. Rotors are slit or segmented discs that have notches or tangs in the external circumference that key to the rotating wheel. Slots or spaces between sections of the rotor create segments that allow heat to dissipate faster than it would if the rotor was solid. They also allow for expansion and prevent warping. (***Figure 16-52***)

Rotors are usually steel to which a frictional surface is bonded to both sides. Typically, sintered metal is used in creating the rotor contact surface. Segmented multiple-disc brakes use retraction spring assemblies with auto clearance adjusters to pull the backplate away from the rotor and stator stack when brake pressure is removed. This provides clearance so the wheel can turn unimpeded by contact friction between the brake parts, but keeps the units in close proximity for rapid contact and braking when the brakes are applied. The number of retraction devices varies with brake design.

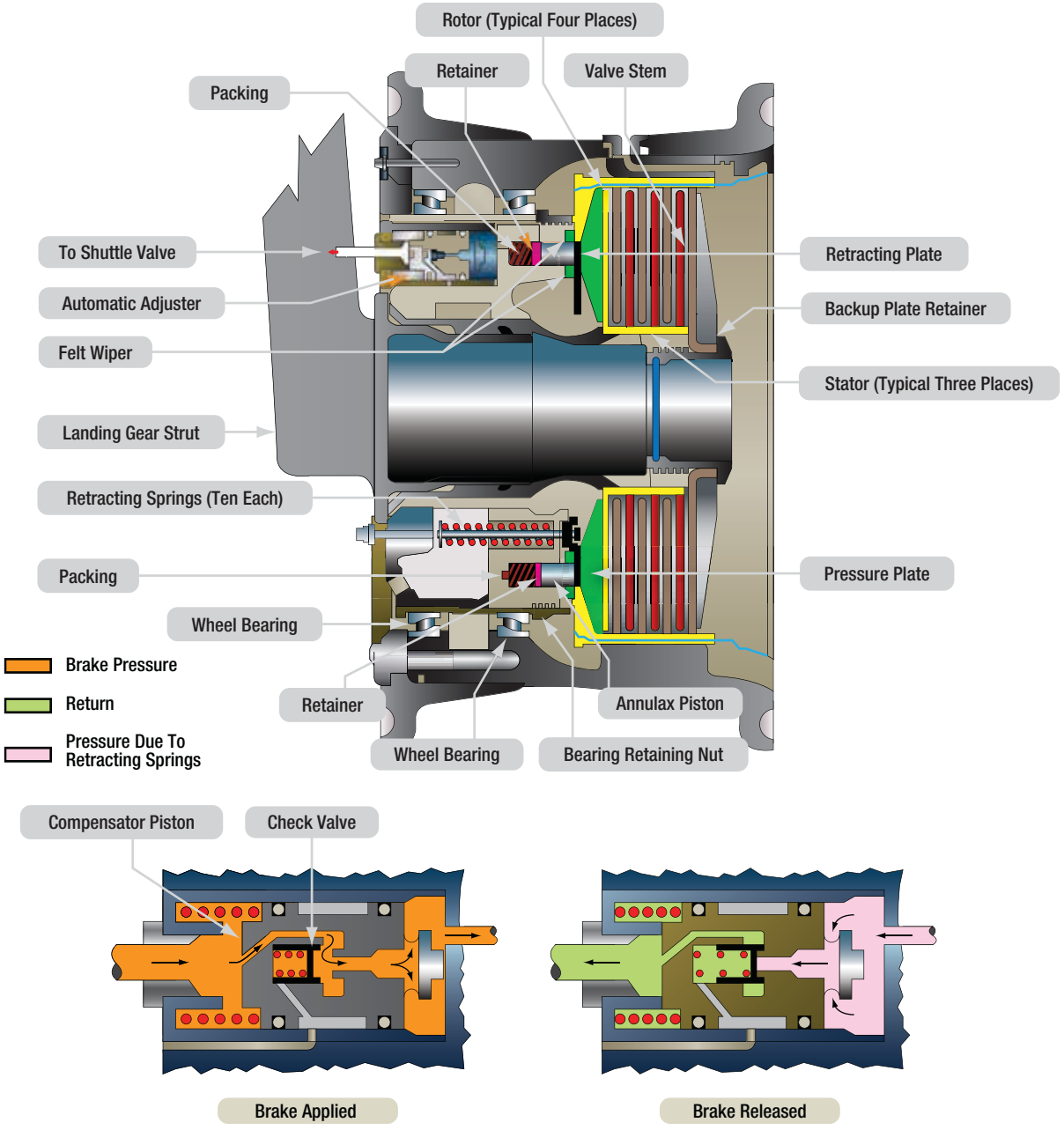


Figure 16-51. A multiple-disc brake with details of the automatic adjuster.

Figure 16-54 illustrates a brake assembly used on a Boeing 737 transport category aircraft. In the cutaway view, the number and locations of the auto adjustment retraction mechanisms can been seen. Details of the mechanisms are also shown. Instead of using a pin grip assembly for auto adjustment, an adjuster pin, ball, and tube operate in the same manner. They move out when brake pressure is applied, but the ball in the tube limits the amount of the return to that equal to the brake lining wear.

Two independent wear indicators are used on the brake illustrated. An indicator pin attached to the backplate protrudes through the carrier. The amount that it protrudes with the brakes applied is measured to ascertain if new linings are required.

NOTE: Other Segmented multiple-disc brakes may use slightly different techniques for pressure plate retraction and wear indication. Consult the manufacturer's maintenance information to ensure wear indicators are read correctly.

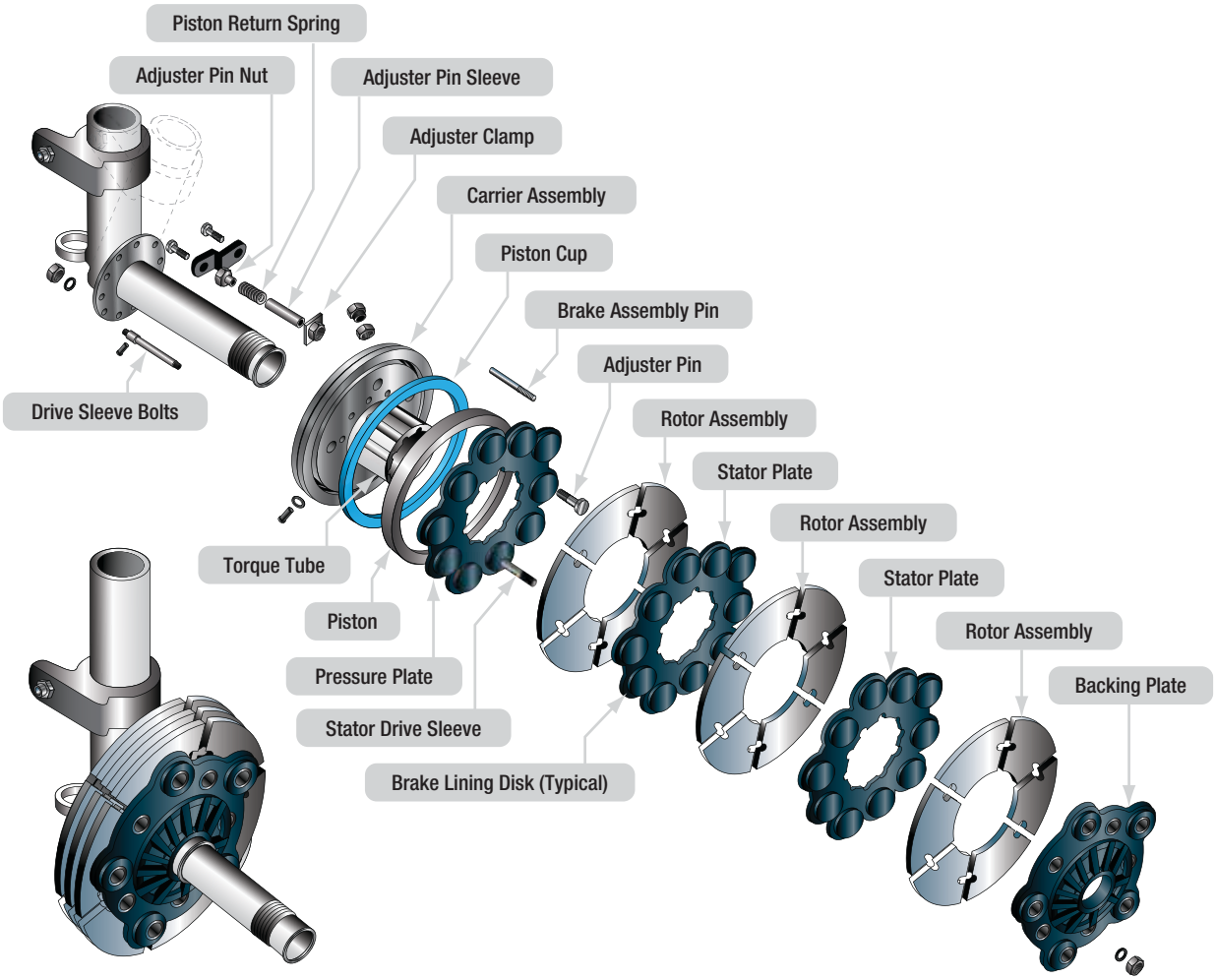


Figure 16-52. Exploded and detail views of segmented rotor brakes.

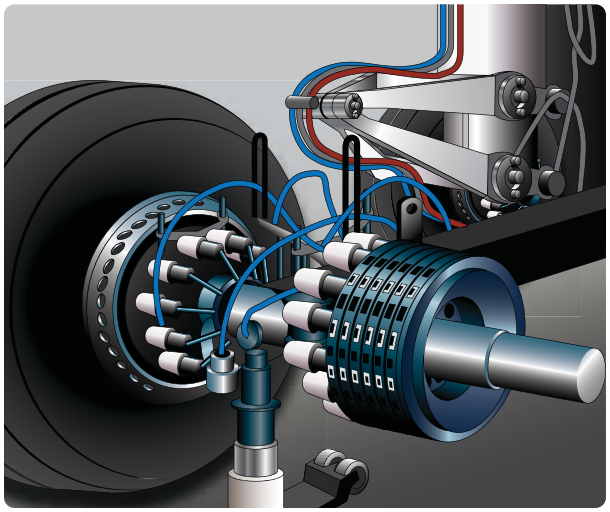


Figure 16-53. Many modern segmented rotor disc brakes use a housing machined to fit numerous individual actuating pistons.

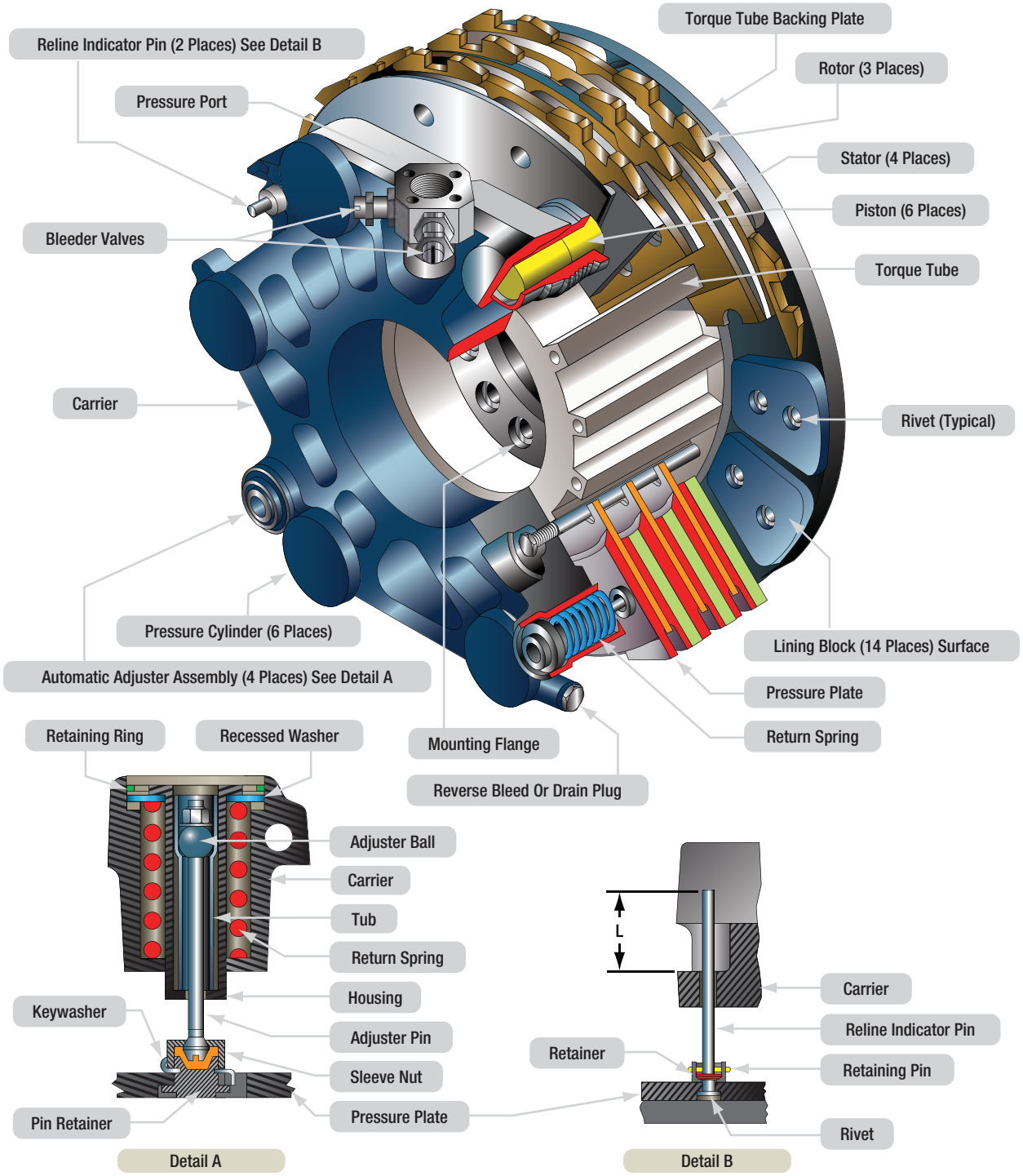


Figure 16-54. The multiple-disk brake assembly and details from a Boeing 737.

CARBON BRAKES

The segmented multiple-disc brake has given many years of reliable service to the aviation industry. It has evolved through time in an effort to make it lightweight and to dissipate the frictional heat of braking in a quick, safe manner. The latest iteration of the multiple-disc brake is the carbon-disc brake. It is currently found on high performance and air carrier aircraft. Carbon brakes are so named because carbon fiber materials are used to construct the brake rotors. (***Figure 16-55***)

Carbon brakes are approximately forty percent lighter than conventional brakes. On a large transport category aircraft, this alone can save several hundred pounds in aircraft weight. The carbon fiber discs are noticeably thicker than sintered steel rotors but are extremely light.

They are able to withstand temperatures fifty percent higher than steel component brakes. The maximum designed operating temperature is limited by the ability of adjacent components to withstand the high temperature. Carbon brakes have been shown to withstand two to three times the heat of a steel brake in non-aircraft applications. Carbon rotors also dissipate heat faster than steel rotors. A carbon rotor maintains its strength and dimensions at high temperatures. Moreover, carbon brakes last twenty to fifty percent longer than steel brakes, which results in reduced maintenance.

The only impediment to carbon brakes being used on all aircraft is the high cost of manufacturing. The price is expected to lower as technology improves and greater numbers of aircraft operators enter the market.

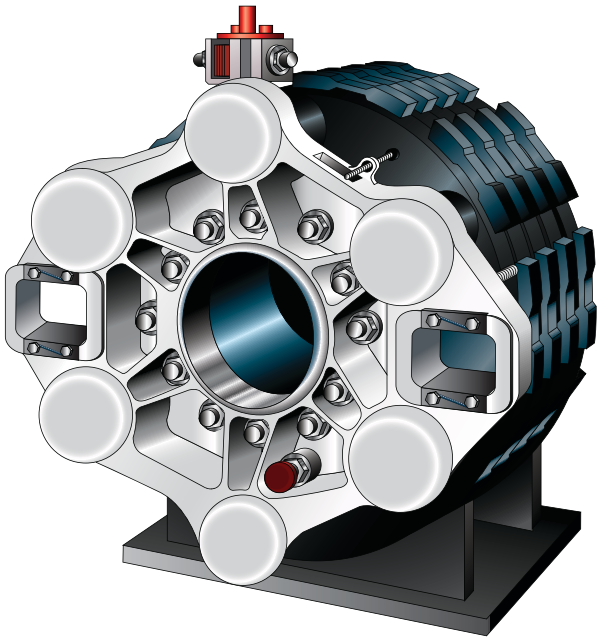


Figure 16-55. A carbon brake for a Boeing 737.

EXPANDER TUBE BRAKES

An expander tube brake is a different approach to braking that is used on aircraft of all sizes produced in the 1930s-1950s. It is a lightweight, low pressure brake bolted to the axle flange that fits inside an iron brake drum. A flat, fabric-reinforced neoprene tube is fitted around the circumference of a wheel-like torque flange. The exposed flat surface of the expander tube is lined with brake blocks similar to brake lining material. Two flat frames bolt to the sides of the torque flange. Tabs on the frames contain the tube and allow evenly spaced torque bars to be bolted in place across the tube between each brake block. These prevent circumferential movement of the tube on the flange. (***Figure 16-56***)

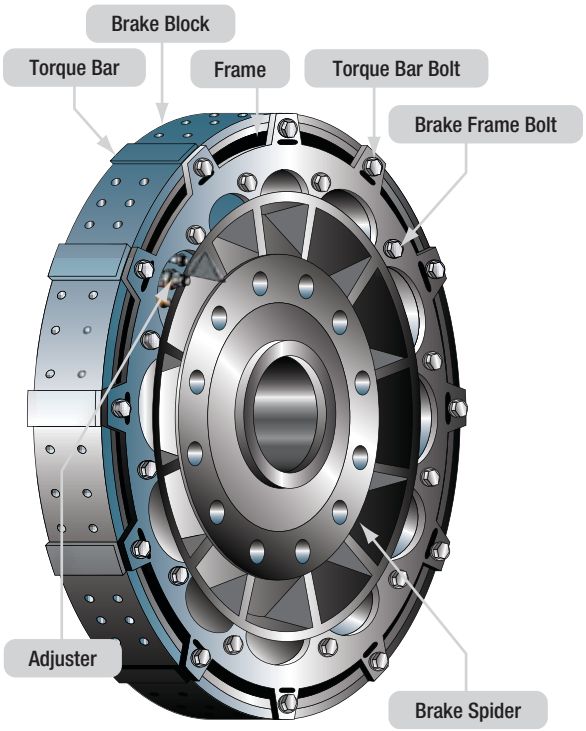


Figure 16-56. An expander tube brake assembly.

The expander tube is fitted with a metal nozzle on the inner surface. Hydraulic fluid under pressure is directed through this fitting into the inside of the tube when the brakes are applied. The tube expands outward, and the brake blocks make contact with the wheel drum causing friction that slows the wheel. As hydraulic pressure is increased, greater friction develops.

Semi-elliptical springs located under the torque bars return the expander tube to a flat position around the flange when hydraulic pressure is removed. The clearance between the expander tube and the brake drum is adjustable by rotating an adjuster on some expander tube brakes. Consult the manufacturer's maintenance manual for the correct clearance setting.

Figure 16-57 gives an exploded view of an expander tube brake, detailing its components. Expander tube brakes work well but have some drawbacks. They tend to take a setback when cold. They also have a tendency to swell with temperature and leak. They may drag inside the drum if this occurs. Eventually, expander brakes were abandoned in favor of disc brake systems.

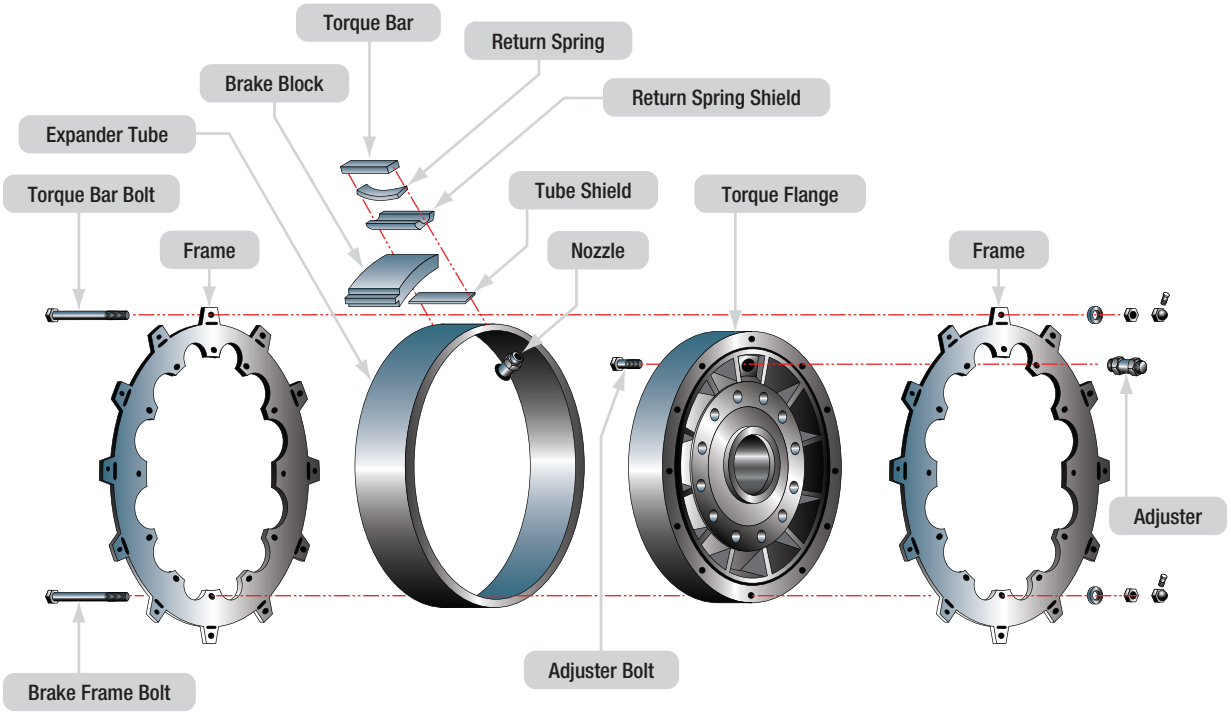


Figure 16-57. An exploded view of an expander tube brake.

BRAKE ACTUATING SYSTEMS

The various brake assemblies, described in the previous section, all use hydraulic power to operate. Different means of delivering the required hydraulic fluid pressure to brake assemblies are discussed in this section.

There are three basic actuating systems:

1. An independent system not part of the aircraft main hydraulic system;
2. A booster system that uses the aircraft hydraulic system intermittently when needed; and
3. A power brake system that only uses the aircraft main hydraulic system(s) as a source of pressure.

Systems on different aircraft vary, but the general operation is similar to those described.

INDEPENDENT MASTER CYLINDERS

In general, small, light aircraft and aircraft without hydraulic systems use independent braking systems. An independent brake system is not connected in any way to the aircraft hydraulic system. Master cylinders are used to develop the necessary hydraulic pressure to operate the brakes. This is similar to the brake system of an automobile. In most brake actuating systems, the pilot pushes on the tops of the rudder pedals to apply the brakes. A master cylinder for each brake is mechanically connected to the corresponding rudder pedal (i.e., right main brake to the right rudder pedal, left main brake to the left rudder pedal). (***Figure 16-58***)

When the pedal is depressed, a piston inside a sealed fluid-filled chamber in the master cylinder forces hydraulic fluid through a line to the piston(s) in the brake assembly. The brake piston(s) push the brake linings against the brake rotor to create the friction that slows the wheel rotation. Pressure is increased throughout the entire brake systems and against the rotor as the pedal is pushed harder. Many master cylinders have built-in reservoirs for the brake hydraulic fluid. Others have a single remote reservoir that services both of the aircraft's two master cylinders. (***Figure 16-59***)

A few light aircraft with nose wheel steering have only one master cylinder that actuates both main wheel brakes. This is possible because steering the aircraft during taxi does not require differential braking. Regardless of the set-up, it is the master cylinder that builds up the pressure required for braking.

A master cylinder used with a remote reservoir is illustrated in ***Figure 16-60***. This particular model is a Goodyear master cylinder. The cylinder is always filled with air-free, contaminant-free hydraulic fluid as is the

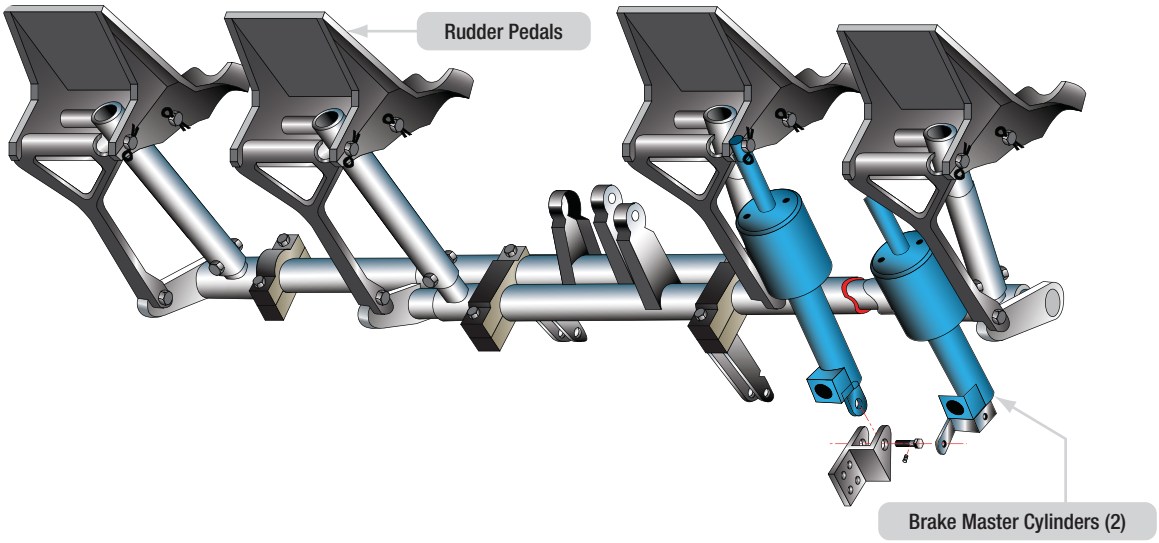


Figure 16-58. Master cylinders on an independent brake system are directly connected to the rudder pedals or are connected through mechanical linkage.

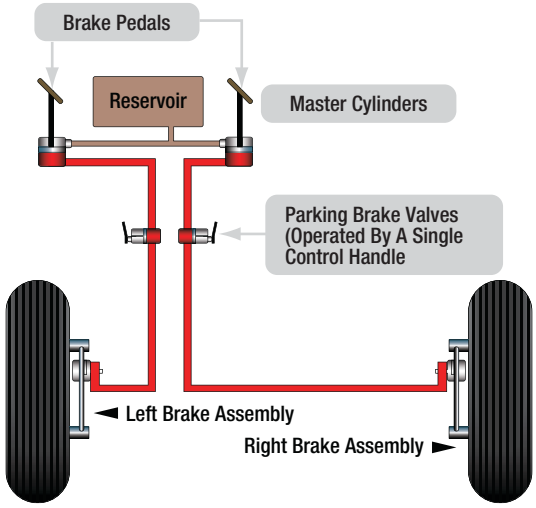


Figure 16-59. A remote reservoir services both master cylinders on some independent braking systems.

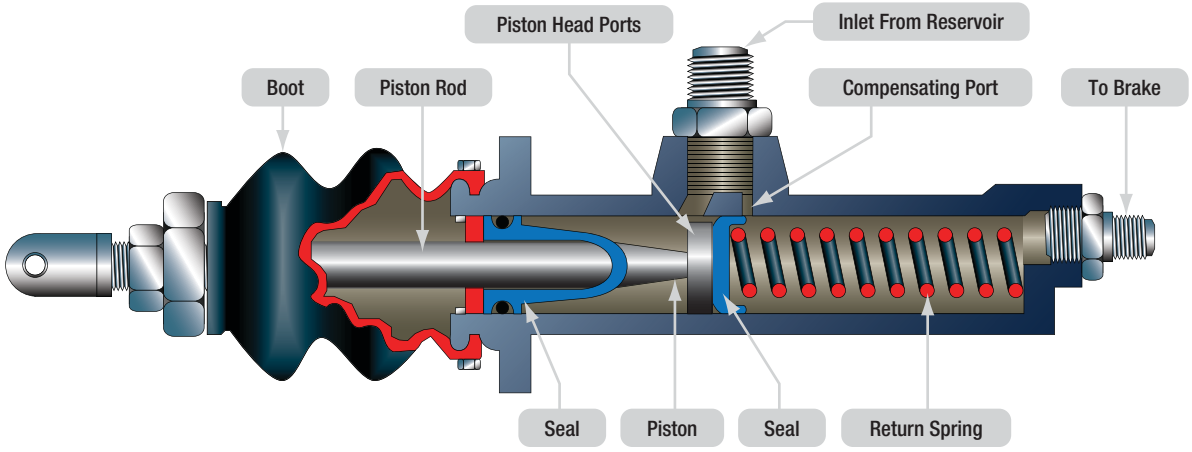


Figure 16-60. A Goodyear brake master cylinder from an independent braking system with a remote reservoir.

reservoir and the line that connects the two together. When the top of the rudder pedal is depressed, the piston arm is mechanically moved forward into the master cylinder. It pushes the piston against the fluid, which is forced through the line to the brake. When pedal pressure is released, the return springs in the brake assembly retract the brake pistons back into the brake housing. The hydraulic fluid behind the pistons is displaced and must return to the master cylinder. As it does, a return spring in the master cylinder move the piston, piston rod and rudder pedal back to the original position (brake off, pedal not depressed). The fluid behind the master cylinder piston flows back into the reservoir. The brake is ready to be applied again.

The forward side of the piston head contains a seal that closes off the compensating port when the brakes are applied so that pressure can build. The seal is only effective in the forward direction. When the piston is returning, or is fully retracted to the off position, fluid behind the piston is free to flow through piston head ports to replenish any fluid that may be lost downstream of the master cylinder. The aft end of the master cylinder contains a seal that prevents leakage at all times. A rubber boot fits over the piston rod and the aft end of the master cylinder to keep out dust. A parking brake for this remote reservoir master cylinder brake system is a ratcheting mechanical device between the master cylinder and the rudder pedals. With the brakes applied, the ratchet is engaged by pulling the parking brake handle. To release the brakes, the rudder pedals are depressed further allowing the ratchet to disengage. With the parking brake set, any expansion of hydraulic fluid due to temperature is relieved by a spring in the mechanical linkage.

A common requirement of all braking systems is for there to be no air mixed in with the hydraulic fluid. Since air is compressible and hydraulic fluid essentially is not, any air under pressure when the brakes are applied causes spongy brakes. The pedals do not feel firm when pushed down due to the air compressing. Brake systems must be bled to remove all air from the system. Instructions for bleeding the brakes are in the Manufacturer's maintenance information. Brake systems equipped with Goodyear master cylinders must be bled from the top down to ensure any air trapped behind the master cylinder piston is removed.

An alternative common arrangement of independent braking systems incorporates two master cylinders, each with its own integral fluid reservoir. Except for the reservoir location, the brake system is basically the same as just described. The master cylinders are mechanically linked to the rudder pedals as before. Depressing the top of a pedal causes the piston rod to push the piston into the cylinder forcing the fluid out to the brake assembly. The piston rod rides in a compensator sleeve and contains an O-ring that seals the rod to the piston when the rod is moved forward. This blocks the compensating ports.

When released, a spring returns the piston to its original position which refills the reservoir as it returns. The rod end seal retracts away from the piston head allowing a free flow of fluid from the cylinder through the compensating ports in the piston to the reservoir. (***Figure 16-61***)

Illustration A depicts the master cylinder when the brakes are off. The compensating port is open to allow fluid to expand into the reservoir should temperature increase. In B, the brakes are applied. The piston rod-end seal covers the compensating port as it contacts the piston head. The parking brake mechanism is a ratcheting type that operates as described.

A servicing port is supplied at the top of the master cylinder reservoir. Typically, a vented plug is installed in the port to provide positive pressure on the fluid.

BOOSTED BRAKES

In an independent braking system, the pressure applied to the brakes is only as great as the foot pressure applied to the top of the rudder pedal. Boosted brake actuating systems augment the force developed by the pilot with hydraulic system pressure when needed. The boost is only during heavy braking. It results in greater pressure applied to the brakes than the pilot alone can provide. Boosted brakes are used on medium and larger aircraft that do not require a full power brake actuating system. A boosted brake master cylinder for each brake is mechanically attached to the rudder pedals. However, the boosted brake master cylinder operates differently. (***Figure 16-62***)

When the brakes are applied, the pressure from the pilot's foot through the mechanical linkage moves the master cylinder piston in the direction to force fluid to the brakes. The initial movement closes the compensator poppet used to provide thermal

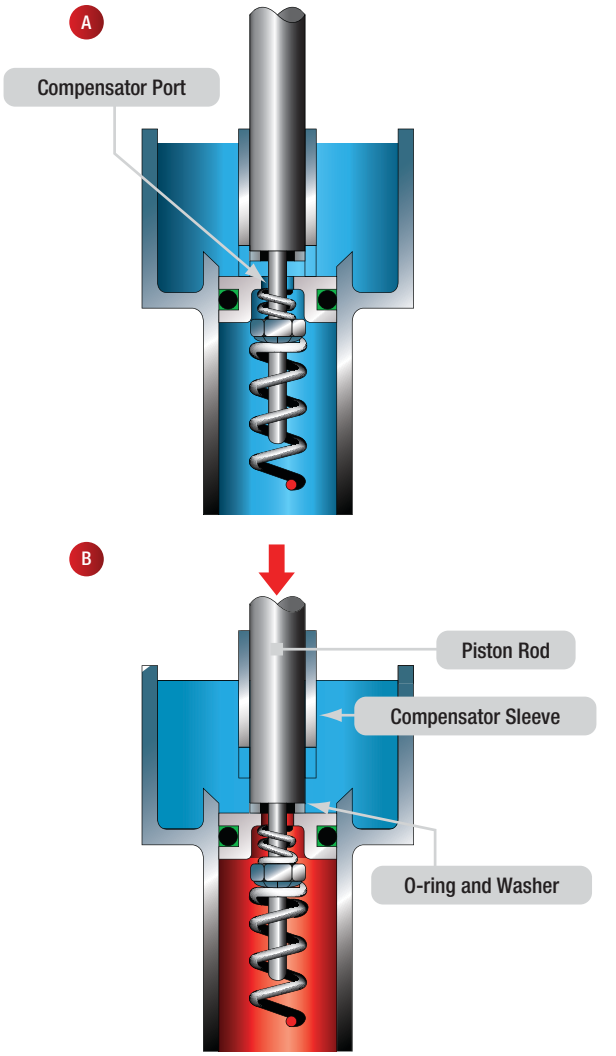


Figure 16-61. A common master cylinder with built-in reservoir is shown. Illustration A depicts the master cylinder when the brakes are off. The compensating port is open to allow fluid to expand into the reservoir should temperature increase. In B, the brakes are applied. The piston rod-end seal covers the compensating port as it contacts the piston head.

expansion relief when the brakes are not applied. As the pilot pushes harder on the pedal, a spring loaded toggle moves a spool valve in the cylinder. Aircraft hydraulic system pressure flows through the valve to the back side of the piston. Pressure's increased, as is the force developed to apply the brakes. When the pedal is released, the piston rod travels in the opposite direction, and the piston returns to the piston stop. The compensating poppet reopens. The toggle is withdrawn from the spool via linkages, and fluid pushes the spool back to expose the system return manifold port. System hydraulic fluid used to boost brake pressure returns through the port.

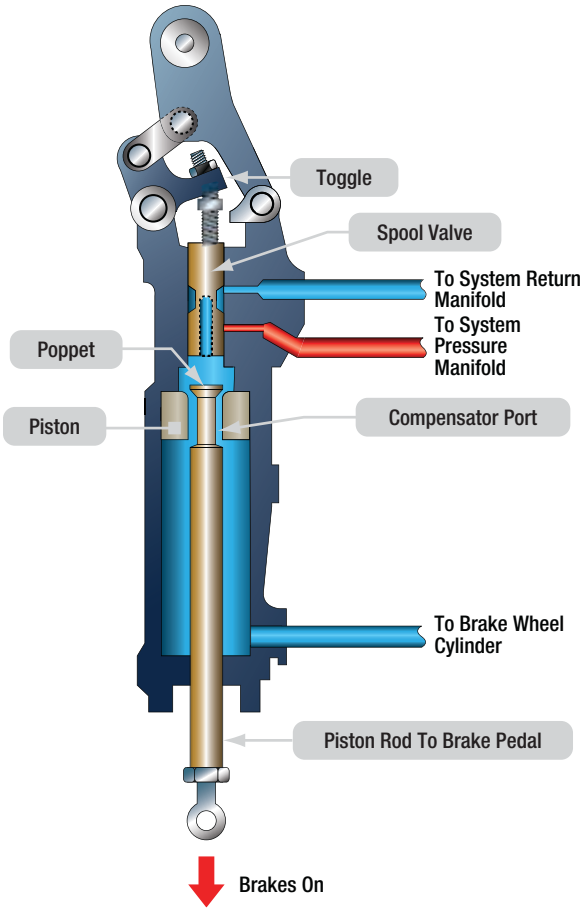


Figure 16-62. A master cylinder for a boosted brake system augments foot pedal pressure with aircraft system hydraulic pressure during heavy braking.

POWER BRAKES

Large and high performance aircraft are equipped with power brakes to slow, stop, and hold the aircraft. Power brake actuating systems use the aircraft hydraulic system as the source of power to apply the brakes. The pilot presses on the top of the rudder pedal for braking as with the other actuating systems. The volume and pressure of hydraulic fluid required cannot be produced by a master cylinder. Instead, a power brake control valve or brake metering valve receives the brake pedal input either directly or through linkages. The valve meters hydraulic fluid to the corresponding brake assembly in direct relation to the pressure applied to the pedal. Many power brake system designs are in use.

Most are similar to the simplified system illustrated in ***Figure 16-63A***. Power brake systems are constructed to facilitate graduated brake pressure control, brake pedal feel, and the necessary redundancy required in case of hydraulic system failure. Large aircraft brake systems integrate antiskid detection and correction devices.

These are necessary because wheel skid is difficult to detect on the flight deck without sensors. However, a skid can be quickly controlled automatically through pressure control of the hydraulic fluid to the brakes.

Hydraulic fuses are also commonly found in power brake systems. The hostile environment around the landing gear increases the potential for a line to break or sever, a fitting to fail, or other hydraulic system malfunctions to occur where hydraulic fluid is lost en route to the brake assemblies. A fuse stops any excessive flow of fluid when detected by closing to retain the remaining fluid in the hydraulic system. Shuttle valves are used to direct flow from optional sources of fluid,

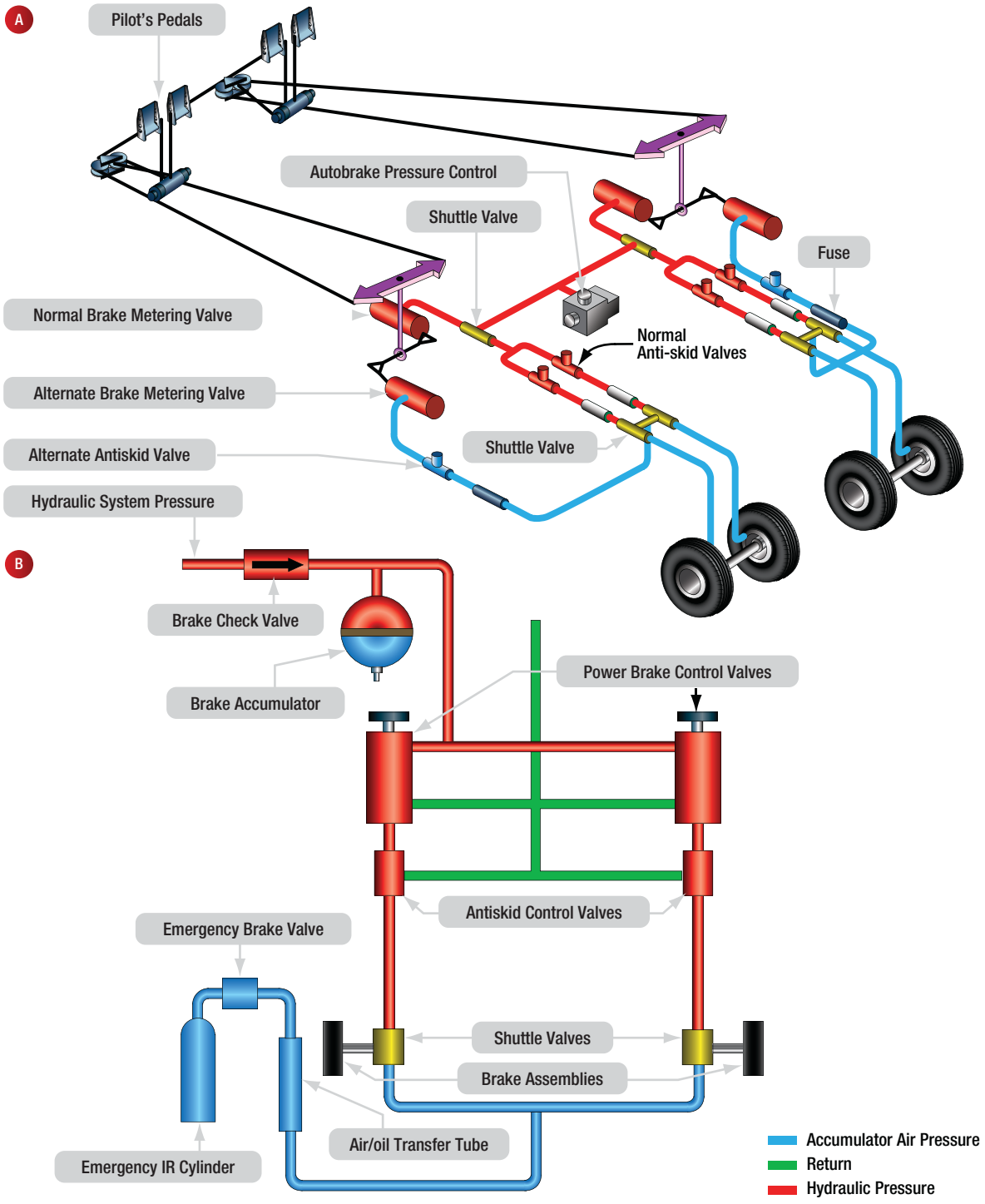


Figure 16-63. The orientation of components in a basic power brake system is shown in A. The general layout of an airliner power brake system is shown in B.

such as in redundant systems or during the use of an emergency brake power source. An airliner power brake system is illustrated in ***Figure 16-63B***.

BRAKE CONTROL VALVE/ BRAKE METERING VALVE

The key element in a power brake system is the brake control valve, sometimes called a brake metering valve. It responds to brake pedal input by directing aircraft system hydraulic fluid to the brakes. As pressure is increased on the brake pedal, more fluid is directed to the brake causing a higher pressure and greater braking action. A brake metering valve from a Boeing 737 is illustrated in ***Figure 16-64***. The system in which it is installed is diagrammed in ***Figure 16-65***.

Two sources of hydraulic pressure provide redundancy in this brake system. A brake input shaft, connected to the rudder/brake pedal through mechanical linkages, provides the position input to the metering valve. As in most brake control valves, the brake input shaft moves a tapered spool or slide in the valve so that it allows hydraulic system pressure to flow to the brakes. At the same time, the slide covers and uncovers access to the hydraulic system return port as required.

A machined slide or spool moves laterally to admit the correct amount of hydraulic system fluid to the brakes. The pressure developed is in proportion to the amount the rudder/brake pedal is depressed and the amount the slide is displaced. The slide/spool also simultaneously controls the return of fluid to the hydraulic system return manifold when brake pressure is released.

When the rudder/brake pedal is depressed, the slide in the metering valve moves to the left. (***Figure 16-64***) It covers the return port so pressure can build in the brake system. The hydraulic supply pressure chamber is connected to the brake system pressure chamber by the movement of the slide, which due to its taper, unblocks the passage between these two. As the pedal is depressed further, the valve slide moves farther to

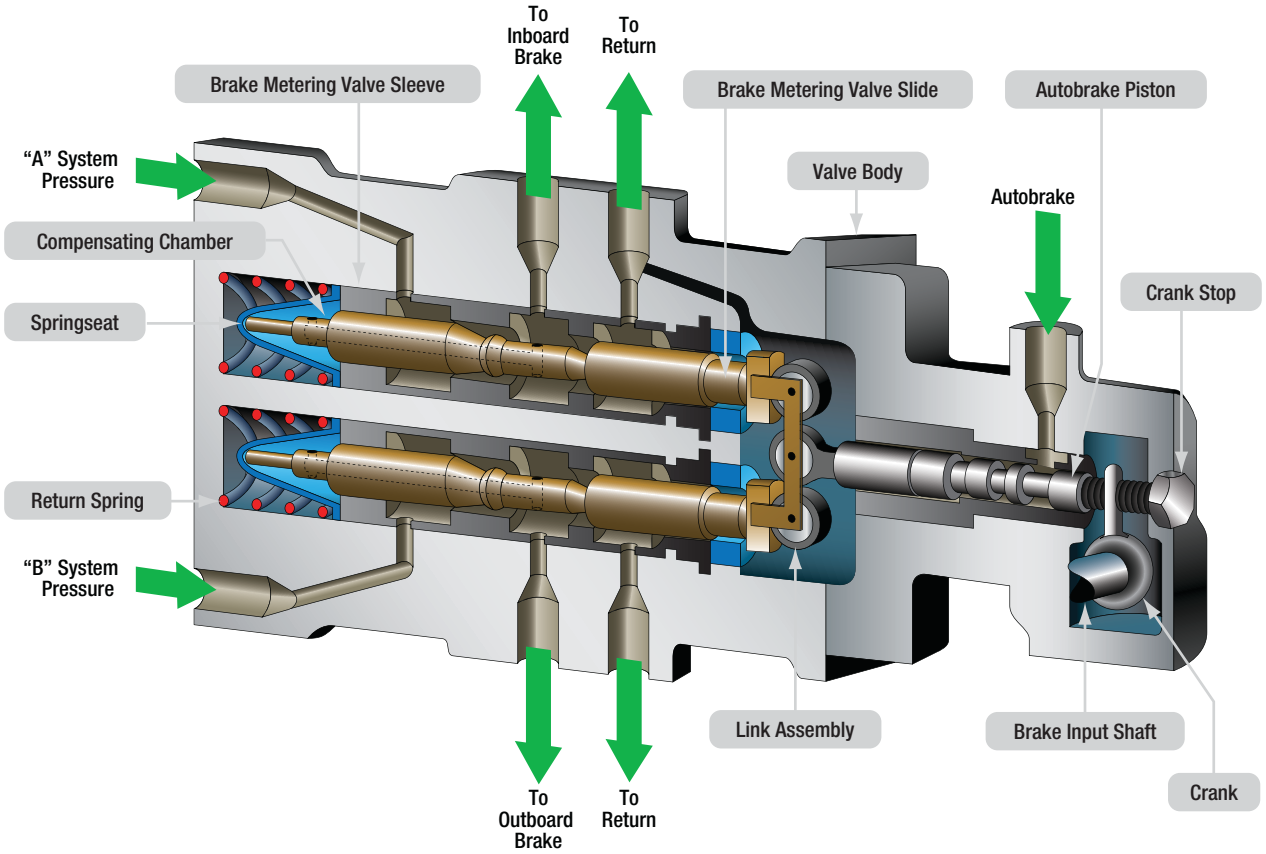


Figure 16-64. A brake metering valve from a Boeing 737. A machined slide or spool moves laterally to admit the correct amount of hydraulic system fluid to the brakes. The pressure developed is in proportion to the amount the rudder/brake pedal is depressed and the amount the slide is displaced. The slide/spool also simultaneously controls the return of fluid to the hydraulic system return manifold when brake pressure is released.

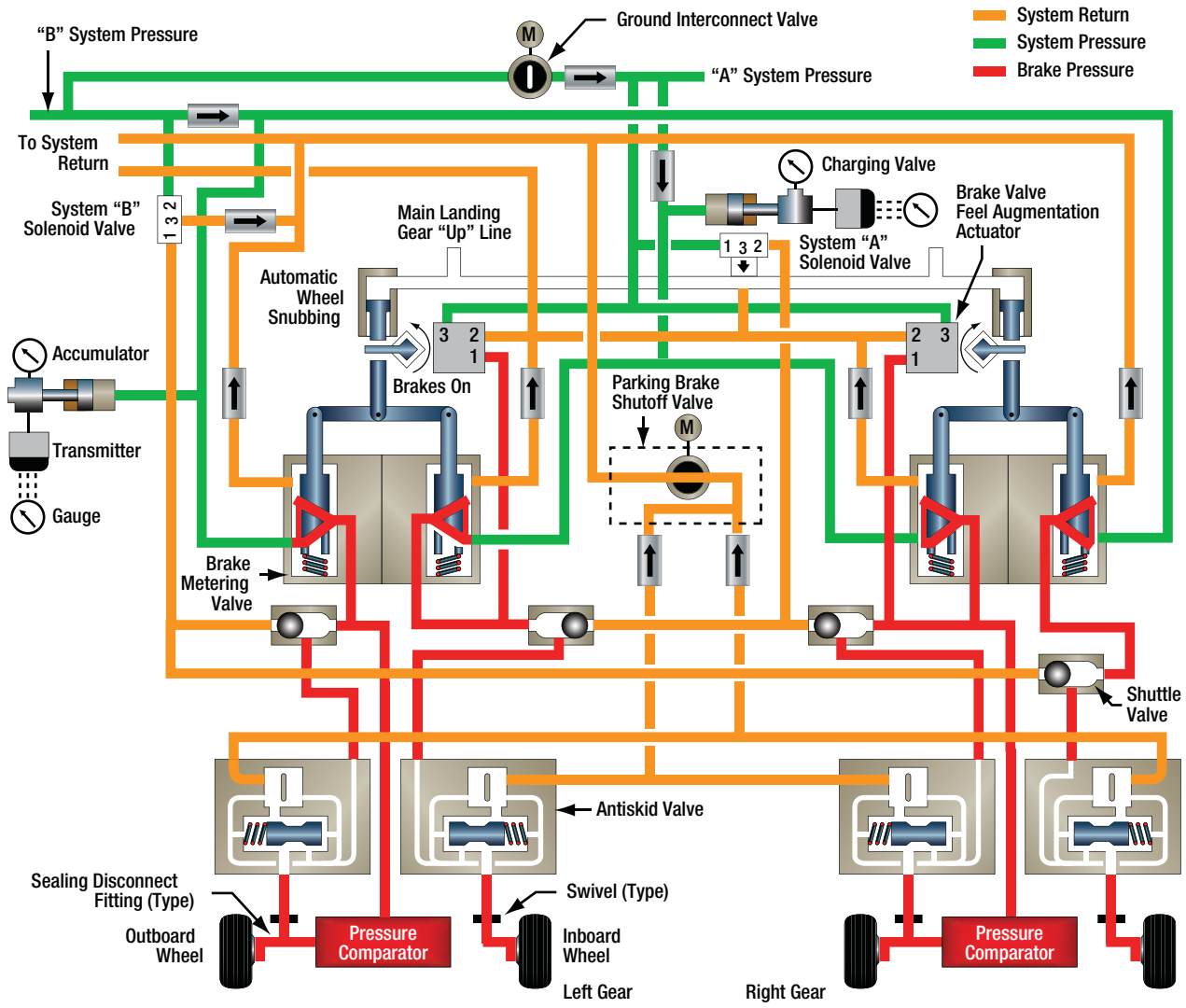


Figure 16-65. The power brake system on a Boeing 737.

the left. This enables more fluid to flow to the brakes due to the narrowing shape of the slide. Brake pressure increases with the additional fluid.

A passage in the slide directs brake pressure fluid into a compensating chamber at the end of the slide. This acts on the end of the slide creating a return force that counters the initial slide movement and gives feel to the brake pedal. As a result, the pressure and return ports are closed and pressure proportional to the foot pressure on the pedal is held on the brakes. When the pedal is released, a return spring and compensating chamber pressure drive the slide to the right into its original position (return port open, supply pressure chamber and brake pressure chambers blocked from each other). The metering valve operates as described simultaneously for the inboard and the outboard brakes. (***Figure 16-64***) The design of the link assembly is such that a single side of the metering valve can operate even if the other fails. Most brake control valves and metering valves function in a similar manner, although many are single units that supply only one brake assembly.

The auto brake, referenced in the metering valve diagram, is connected into the landing gear retraction hydraulic line. Pressurized fluid enters this port and drives the slide slightly to the left to apply the brakes automatically after takeoff. This stops the wheels from rotating when retracted into the wheel wells. Auto brake pressure is withheld from this port when the landing gear is fully stowed since the retraction system is depressurized. The majority of the rudder/brake pedal feel is supplied by the brake control or brake metering valve in a power brake system. Many aircraft refine the feel of the pedal with an additional feel unit. The brake valve feel augmentation unit, in the above system, uses a series of internal springs

and pistons of various sizes to create a force on the brake input shaft movement. This provides feel back through the mechanical linkages consistent with the amount of rudder/brake pedal applied. The request for light braking with slight pedal depression results in a light feel to the pedal and a harder resistance feel when the pedals are pushed harder during heavy braking. (***Figure 16-66***)

EMERGENCY BRAKE SYSTEMS

As can be seen in ***Figure 16-65***, the brake metering valves not only receive hydraulic pressure from two separate hydraulic systems, they also feed two separate brake assemblies. Each main wheel assembly has two wheels. The inboard wheel brake and the outboard wheel brake, located in their respective wheel rims, are independent from each other. In case of hydraulic system failure or brake failure, each is independently supplied to adequately slow and stop the aircraft without the other. More complicated aircraft may involve another hydraulic system for back-up or use a similar alternation of sources and brake assemblies to maintain braking in case of hydraulic system or brake failure.

NOTE: In the segmented rotor brake section above, a brake assembly was described that had alternating pistons supplied by independent hydraulic sources. This is another method of redundancy particularly suitable on, but not limited to, single main wheel aircraft.

In addition to supply system redundancy, the brake accumulator is also an emergency source of power for the brakes in many power brake systems. The accumulator is pre-charged with air or nitrogen on one side of its internal diaphragm. Enough hydraulic fluid is contained on the other side of the diaphragm to operate the brakes in case of an emergency. It is forced out of the accumulator into the brakes through the system lines under enough stored pressure to slow the aircraft. Typically, the accumulator is located upstream of the brake control/metering valve to capitalize on the control given by the valve. (***Figure 16-67***)

Some simpler power brake systems may use an emergency source of brake power that is delivered directly to the brake assemblies and bypasses the remainder of the brake system completely. A shuttle valve immediately upstream of the brake units shifts to accept this source when pressure is lost from the primary supply sources. Compressed air or nitrogen is sometimes used. A precharged fluid source can also be used as an alternate hydraulic source.

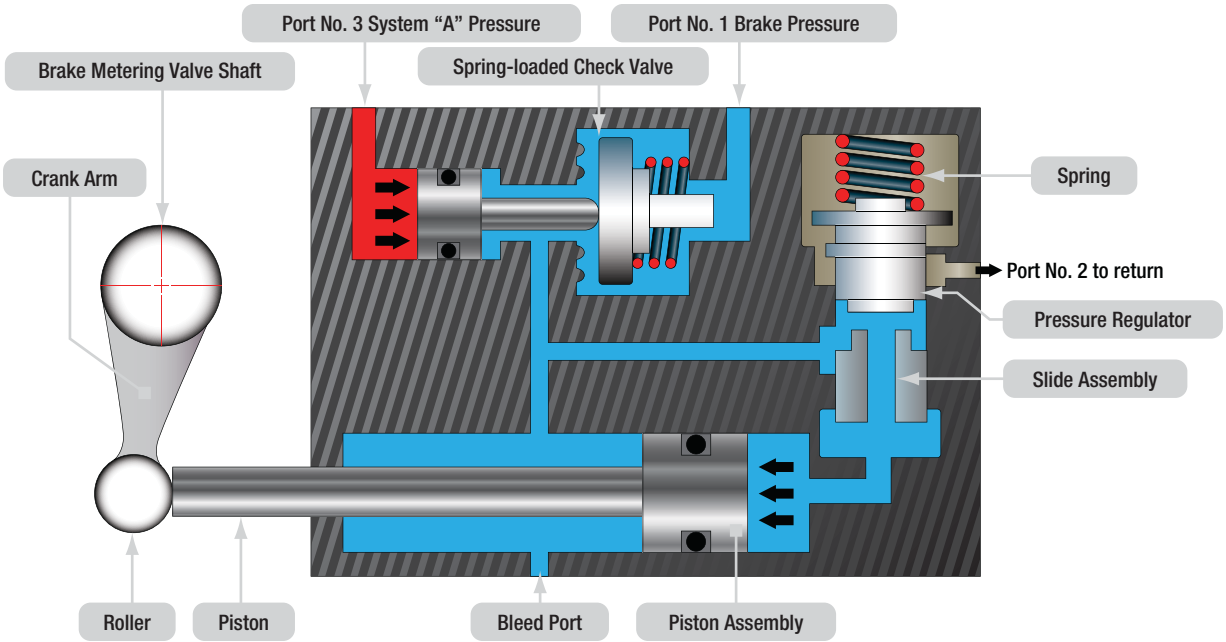


Figure 16-66. The power brake system on a Boeing 737.

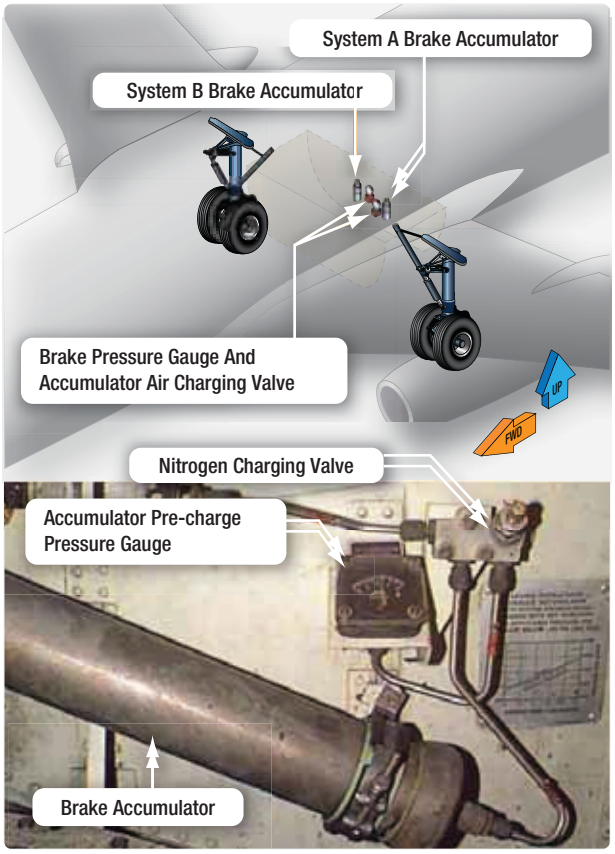


Figure 16-67. Emergency brake hydraulic fluid accumulators are precharged with nitrogen to deliver brake fluid to the brakes in the event normal and alternate hydraulic sources fail.

PARKING BRAKE

The parking brake system function is a combined operation. The brakes are applied with the rudder pedals and a ratcheting system holds them in place when the parking brake lever on the flight deck is pulled. (***Figure 16-68***) At the same time, a shut-off valve is closed in the common return line from the brakes to the hydraulic system. This traps the fluid in the brakes holding the rotors stationary. Depressing the pedals further releases the pedal ratchet and opens the return line valve.

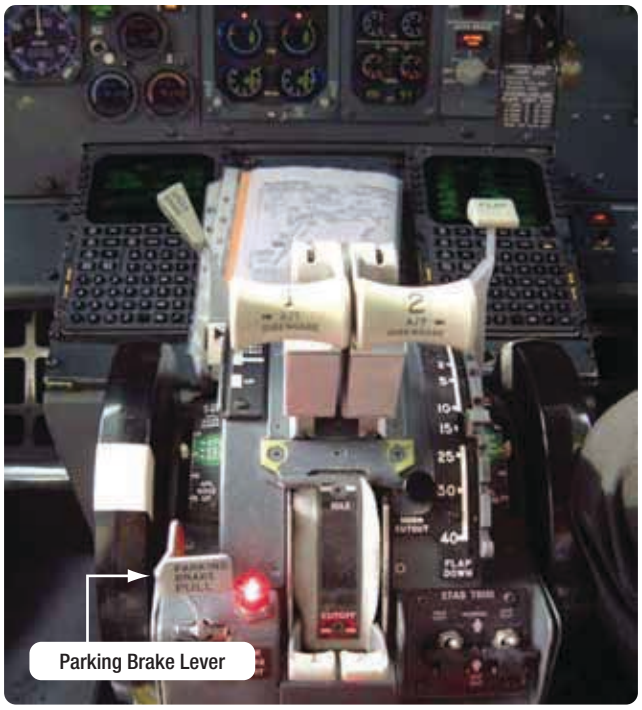


Figure 16-68. The parking brake lever on a Boeing 737 center pedestal throttle quadrant.

BRAKE DEBOOSTERS

Some aircraft brake assemblies that operate on aircraft hydraulic system pressure are not designed for such high pressure. They provide effective braking through a power brake system but require less than maximum hydraulic system pressure. To supply the lower pressure, a brake debooster cylinder is installed downstream of the control valve and antiskid valve. Brake deboosters are simple devices that use the application of force over different sized pistons to reduce pressure.

ANTI-SKID

Large aircraft with power brakes require antiskid systems. It is not possible to immediately ascertain in the flight deck when a wheel stops rotating and begins to skid, especially in aircraft with multiple-wheel main landing gear assemblies. A skid not corrected can quickly lead to a tire blowout, possible damage to the aircraft, and control of the aircraft may be lost.

SYSTEM OPERATION

The antiskid system not only detects wheel skid, it also detects when wheel skid is imminent. It automatically relieves pressure to the brake pistons of the wheel in question by momentarily connecting the pressurized brake fluid area to the hydraulic system return line. This allows the wheel to rotate and avoid a skid. Lower pressure is then maintained to the brake at a level that slows the wheel without causing it to skid.

Maximum braking efficiency exists when the wheels are decelerating at a maximum rate but are not skidding. If a wheel decelerates too fast, it is an indication that the brakes are about to lock and cause a skid. To ensure that this does not happen, each wheel is monitored for a deceleration rate faster than a preset rate. When excessive deceleration is detected, hydraulic pressure is reduced to the brake on that wheel. To operate the antiskid system, flight deck switches must be placed in the ON position. (***Figure 16-69***)

After the aircraft touches down, the pilot applies and holds full pressure to the rudder brake pedals. The antiskid system then functions automatically until the speed of the aircraft has dropped to approximately 20 mph. The system returns to manual braking mode for slow taxi and ground maneuvering. There are various designs of antiskid systems. Most contain three main types of components: wheel speed sensors, antiskid control valves, and a control unit. These units work together without human interference. Some antiskid systems provide complete automatic braking. The pilot needs only to turn on the auto brake system, and the antiskid components slow the aircraft without pedal input. (***Figure 16-69***)

Ground safety switches are wired into the circuitry for antiskid and auto brake systems. Wheel speed sensors are located on each wheel equipped with a brake assembly. Each brake also has its own antiskid control valve. Typically, a single control box contains the antiskid comparative circuitry for all of the brakes on the aircraft. (***Figure 16-70***)

WHEEL SPEED SENSORS

Wheel speed sensors are transducers. They may be alternating current (AC) or direct current (DC). The typical AC wheel speed sensor has a stator mounted in the wheel axle. A coil around it is connected to a controlled DC source so that when energized, the stator becomes an electromagnet. A rotor that turns inside the

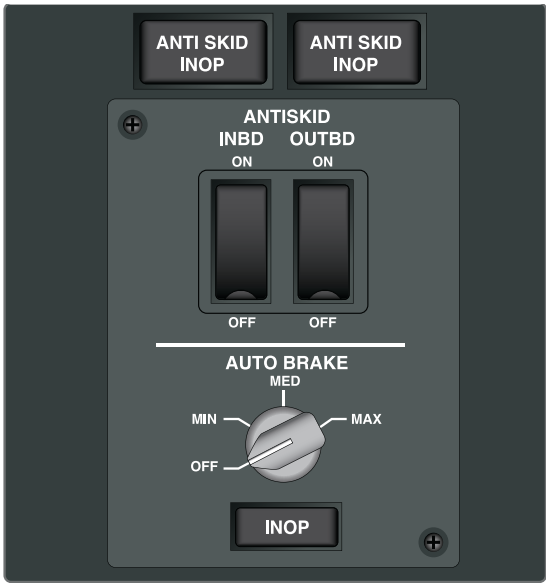


Figure 16-69. Antiskid switches in the cockpit.

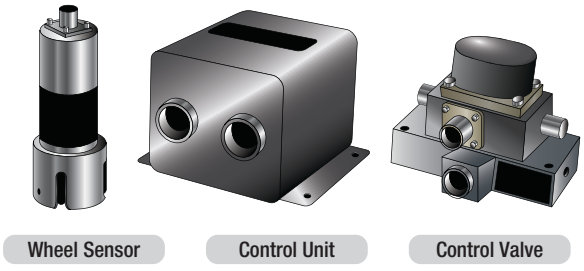


Figure 16-70. A wheel sensor (left), a control unit (center), and a control valve (right) are components of an antiskid system. A sensor is located on each wheel equipped with a brake assembly. An antiskid control valve for each brake assembly is controlled from a single central control unit.

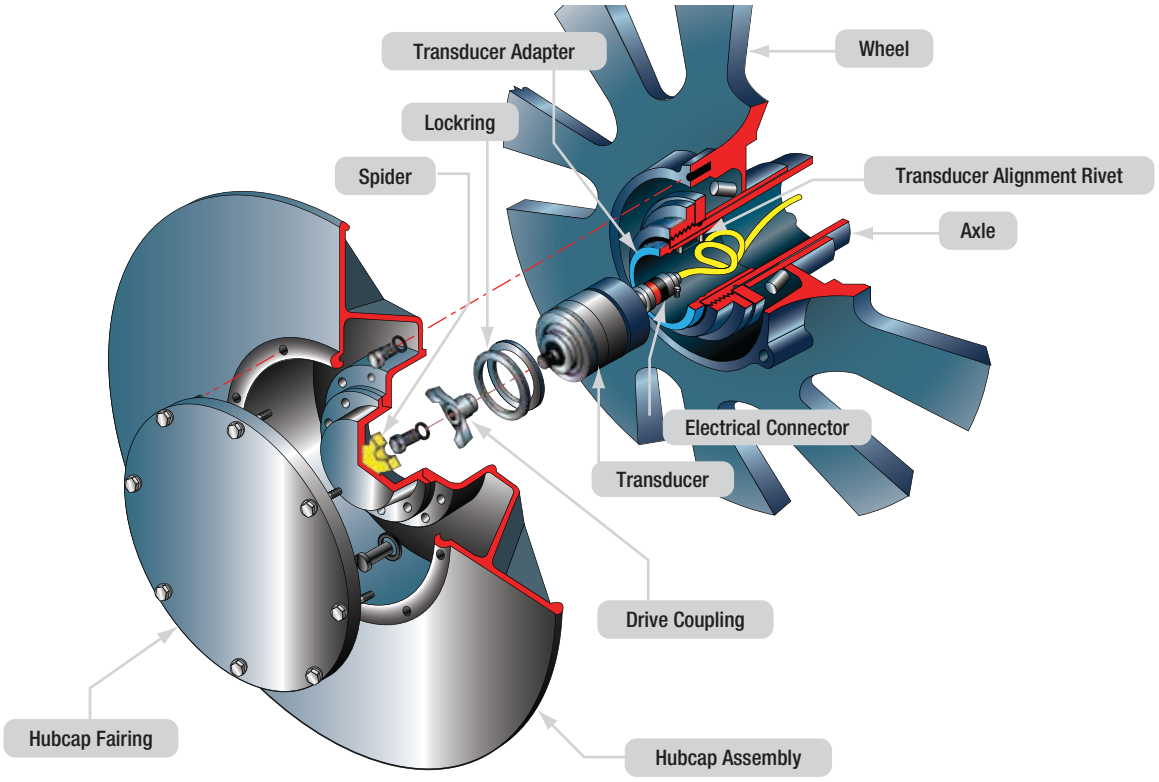


Figure 16-71. The stator of an antiskid wheel sensor is mounted in the axle, and the rotor is coupled to the wheel hub spider that rotates with the wheel.

stator is connected to the rotating wheel hub assembly through a drive coupling so that it rotates at the speed of the wheel. Lobes on the rotor and stator cause the distance between the two components to constantly change during rotation. This alters the magnetic coupling or reluctance between the rotor and stator. As the electromagnetic field changes, a variable frequency AC is induced in the stator coil. The frequency is directly proportional to the speed of rotation of the wheel. The AC signal is fed to the control unit for processing. A DC wheel speed sensor is similar, except that a DC is produced the magnitude of which is directly proportional to wheel speed. (***Figure 16-71***)

CONTROL UNITS

The control unit can be regarded as the brain of the antiskid system. It receives signals from each of the wheel sensors. Comparative circuits are used to determine if any of the signals indicate a skid is imminent or occurring on a particular wheel. If so, a signal is sent to the control valve of the wheel to relieve hydraulic pressure to that brake which prevents or relieves the skid. The control unit may or may not have external test switches and status indicating lights. It is common for it to be located in the avionics bay of the aircraft. (***Figure 16-72***)

The Boeing antiskid control valve block diagram in ***Figure 16-73*** gives further detail on the functions of an antiskid control unit. Other aircraft may have different logic to achieve similar end results. DC systems do not require an input converter since DC is received from the wheel sensors, and the control unit circuitry operates primarily with DC.

Only the functions on one circuit card for one wheel brake assembly are shown in ***Figure 16-73***. Each wheel has its own identical circuitry card to facilitate simultaneous operation. All cards are housed in a single control unit



Figure 16-72. A rack mounted antiskid control unit from an airliner.

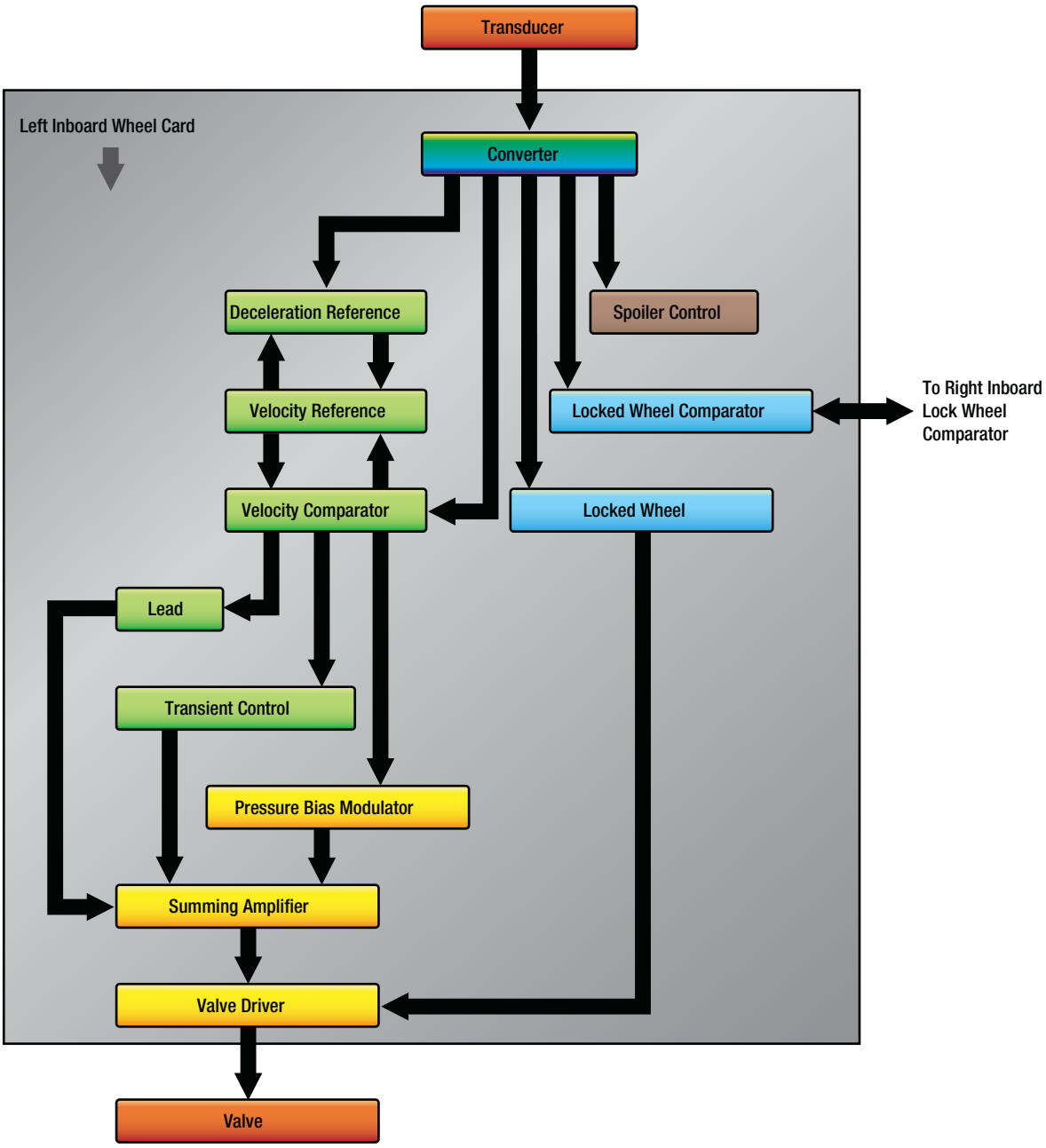


Figure 16-73. A Boeing 737 antiskid control unit internal block diagram.

that Boeing calls a control shield. The converter shown changes the AC frequency received from the wheel sensor into DC voltage that is proportional to wheel speed. The output is used in a velocity reference loop that contains deceleration and velocity reference circuits. The converter also supplies input for the spoiler system and the locked wheel system, which is discussed at the end of this section. A velocity reference loop output voltage is produced, which represents the instantaneous velocity of the aircraft. This is compared to converter output in the velocity comparator. This comparison of voltages is essentially the comparison of the aircraft speed to wheel speed. The output from the velocity comparator is a positive or negative error voltage corresponding to whether the wheel speed is too fast or too slow for optimum braking efficiency for a given aircraft speed.

The error output voltage from the comparator feeds the pressure bias modulator circuit. This is a memory circuit that establishes a threshold where the pressure to the brakes provides optimum braking. The error voltage causes the modulator to either increase or decrease the pressure to the brakes in attempt to hold the modulator threshold. It produces a voltage output that is sent to

the summing amplifier to do this. A lead output from the comparator anticipates when the tire is about to skid with a voltage that decreases the pressure to the brake. It sends this voltage to the summing amplifier as well. A transient control output from the comparator designed for rapid pressure dump when a sudden skid has occurred also sends voltage to the summing amp.

As the name suggests, the input voltages to the amplifier are summed, and a composite voltage is sent to the valve driver. The driver prepares the current required to be sent to the control valve to adjust the position of the valve. Brake pressure increases, decreases, or holds steady depending on this value.

ANTISKID CONTROL VALVES

Antiskid control valves are fast-acting, electrically controlled hydraulic valves that respond to the input from the antiskid control unit. There is one control valve for each brake assembly. A torque motor uses the input from the valve driver to adjust the position of a flapper between two nozzles. By moving the flapper closer to one nozzle or the other, pressures are developed in the second stage of the valve. These pressures act on a spool that is positioned to build or reduce pressure to the brake by opening and blocking fluid ports. (***Figure 16-74***)

As pressure is adjusted to the brakes, deceleration slows to within the range that provides the most effective braking without skidding. The wheel sensor signal adjusts to the wheel speed, and the control unit processes the change. Output is altered to the control valve. The control valve flapper position is adjusted and steady braking resumes without correction until needed. Antiskid control valves are typically located in the main wheel for close access to hydraulic pressure and return manifolds, as well as the brake assemblies. (***Figure 16-75***) Systematically, they are positioned downstream of the power brake control valves but upstream of debooster cylinders if the aircraft is so equipped.

TOUCHDOWN AND LOCK WHEEL PROTECTION

It is essential that the brakes are not applied when the aircraft contacts the runway upon landing. This could cause immediate tire blowout. A touchdown protection mode is built into most aircraft antiskid systems to prevent this. It typically functions in conjunction with the wheel speed sensor and the air/ground safety switch on the landing gear strut (squat switch). Until the aircraft has weight on wheels, the detector circuitry signals the antiskid control valve to open the passage between the brakes and the hydraulic system return, thus preventing pressure build-up and application of the brakes. Once the squat switch is open, the antiskid control unit sends a signal to the control valve to close and permit brake pressure build-up. As a back-up and when the aircraft is on the ground with the strut not compressed enough to open the squat switch, a minimum wheel speed sensor signal can override and allow braking. Wheels are often grouped with one relying on the squat switch and the other on wheel speed sensor output to ensure braking when the aircraft is on the ground, but not before then.

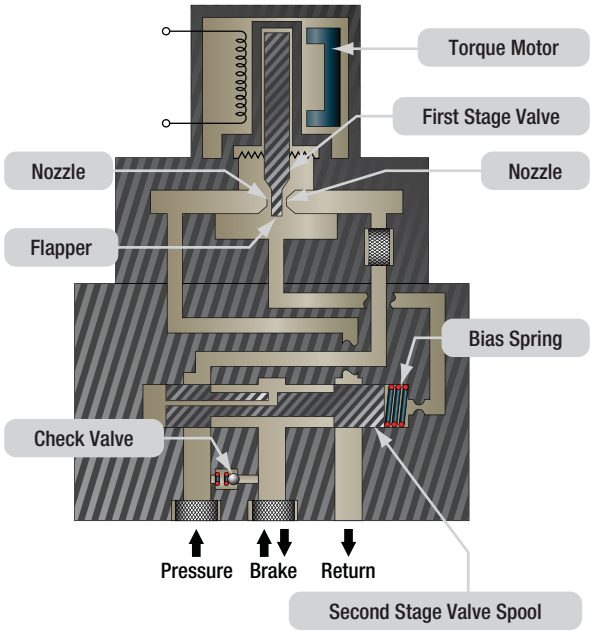


Figure 16-74. An antiskid control valve uses a torque motor controlled flapper in the first stage of the valve to adjust pressure on a spool in the second stage of the valve to build or relieve pressure to the brake.

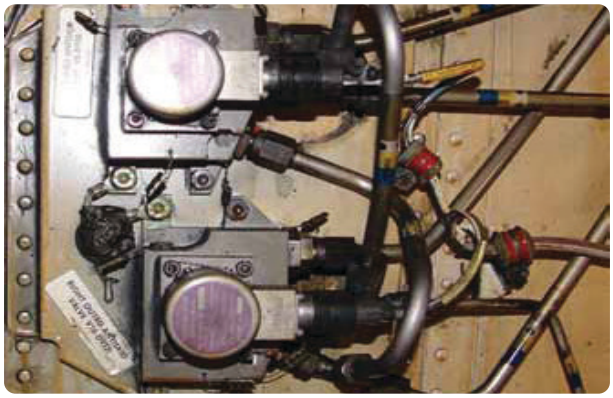


Figure 16-75. Two antiskid control valves with associated plumbing and wiring.

Locked wheel protection recognizes if a wheel is not rotating. When this occurs, the antiskid control valve is signaled to fully open. Some aircraft antiskid control logic, such as the Boeing 737 shown in ***Figure 16-74***, expands the locked wheel function. Comparator circuitry is used to relieve pressure when one wheel of a paired group of wheels rotates 25 percent slower than the other. Inboard and outboard pairs are used because if one of the pair is rotating at a certain speed, so should the other. If it is not, a skid is beginning or has occurred. On takeoff, the antiskid system receives input through a switch located on the gear selector that shuts off the antiskid system. This allows the brakes to be applied as retraction occurs so that no wheel rotation exists while the gear is stowed.

ANTISKID SYSTEM TESTS

It is important to know the status of the antiskid system prior to attempting to use it during a landing or aborted takeoff. Ground tests and in-flight tests are used. Built-in test circuits and control features allow testing of the system components and provide warnings should a particular component or part of the system become inoperative. An inoperative antiskid system can be shut off without affecting normal brake operation.

GROUND TEST

Ground tests vary slightly from aircraft to aircraft. Consult the manufacturer's maintenance manual for test procedures specific to the aircraft in question. Much of the antiskid system testing originates from testing circuits in the antiskid control unit. Built-in test circuits continuously monitor the antiskid system and provide warning if a failure occurs. An operational test can be performed before flight. The antiskid control switch and/or test switch is used in conjunction with system indicator light(s) to determine system integrity.

A test is first done with the aircraft at rest and then in an electrically simulated antiskid braking condition. Some antiskid control units contain system and component testing switches and lights for use by the technician. This accomplishes the same operational verification, but allows an additional degree of troubleshooting. Test sets are available for antiskid systems that produce electric signals that simulate speed outputs of the wheel transducer, deceleration rates, and flight/ground parameters.

IN-FLIGHT TEST

In-flight testing of the antiskid system is desirable and part of the pre-landing checklist so that the pilot is aware of system capability before landing. As with ground testing, a combination of switch positions and indicator lights are used according to information in the aircraft operations manual.

ANTISKID SYSTEM MAINTENANCE

Antiskid components require little maintenance. Troubleshooting antiskid system faults is either performed via test circuitry or can be accomplished through isolation of the fault to one of the three main operating components of the system. Antiskid components are normally not repaired in the field. They are sent to the manufacturer or a certified repair station when work is required. Reports of antiskid system malfunction are sometimes malfunctions of the brake system or brake assemblies. Ensure brake assemblies are bled and functioning normally without leaks before attempting to isolate problems in the antiskid system.

WHEEL SPEED SENSOR

Wheel speed sensors must be securely and correctly mounted in the axle. The means of keeping contamination out of the sensor, such as sealant or a hub cap, should be in place and in good condition. The wiring to the sensor is subject to harsh conditions and should be inspected for integrity and security. It should be repaired or replaced if damaged in accordance with the manufacturer's instructions. Accessing the wheel speed sensor and spinning it by hand or other recommended device to ensure brakes apply and release via the antiskid system is common practice.

CONTROL VALVE

Antiskid control valve and hydraulic system filters should be cleaned or replaced at the prescribed intervals. Follow all manufacturer's instructions when performing this maintenance. Wiring to the valve must be secure, and there should be no fluid leaks.

CONTROL UNIT

Control units should be securely mounted. Test switches and indicators, if any, should be in place and functioning. It is essential that wiring to the control unit is secure. A wide variety of control units are in use. Follow the manufacturer's instructions at all times when inspecting or attempting to perform maintenance on these units.

AUTO BRAKING

Aircraft equipped with auto brakes typically bypass the brake control valves or brake metering valves and use a separate auto brake control valve to provide this function. In addition to the redundancy provided, auto brakes rely on the antiskid system to adjust pressure to the brakes if required due to an impending skid.

Figure 16-76 shows a simplified diagram of the Boeing 757 brake system with the auto brake valve in relation to the main metering valve and antiskid valves in this eight-main wheel system. Note that in an auto brake system, the antiskid system must be functioning properly for the auto brake system to function.

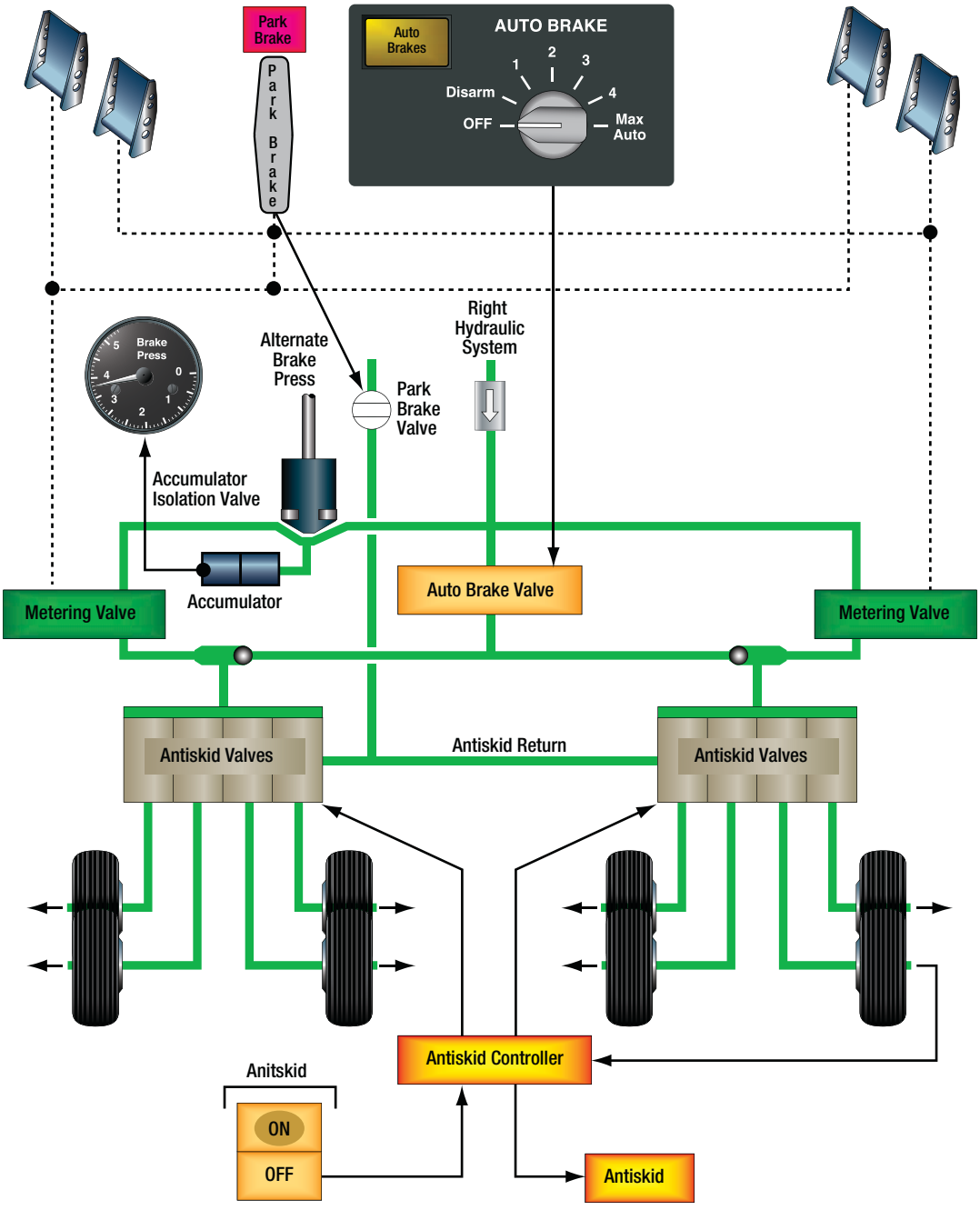


Figure 16-76. The Boeing 757 normal brake system with auto brake and antiskid.

Auto braking is activated by a switch on the flight deck. It can be set to various levels of deceleration. Typically, when turned ON, the auto brake system brakes automatically when weight is on the wheels and the throttles are in the idle position or thrust reversers are activated. Auto brake deactivates if the crew steps on the rudder/brake pedals, or the aircraft slows beyond a predetermined speed. An auto brake control box controlled by the selector switch regulates the function of the auto brake valve when auto braking in selected. It meters hydraulic fluid to the brakes instead of the regular brake metering valves Downstream of the auto brake valve, the antiskid functions normally.

Should a malfunction in the auto brake system occur, a warning light or flat screen annunciation will illuminate on the flight deck. The system will not function in this case and braking must be accomplished with the rudder brake pedals.

AIRCRAFT TIRES AND TUBES

Aircraft tires may be tube-type or tubeless. They support the weight of the aircraft while it is on the ground and provide the necessary traction for braking and stopping. The tires also help absorb the shock of landing and cushion the roughness of takeoff, rollout, and taxi operations. Aircraft tires must be carefully maintained to perform as required. They accept a variety of static and dynamic stresses and must do so dependably in a wide range of operating conditions. Aircraft tires are classified in various ways including by: type, ply rating, whether they are tube-type or tubeless, and whether they are bias ply tires or radials. Identifying a tire by its dimensions is also used.

PLY RATING

Tire plies are reinforcing layers of fabric encased in rubber that are laid into the tire to provide strength. In early tires, the number of plies used was directly related to the load the tire could carry. Nowadays, refinements to tire construction techniques and the use of modern materials to build up aircraft tires makes the exact number of plies somewhat irrelevant when determining the strength of a tire.

TUBE-TYPE OR TUBELESS

As stated, aircraft tires can be tube-type or tubeless. This is often used as a means of tire classification. Tires that are made to be used without a tube inserted inside have an inner liner specifically designed to hold air. Tube-type tires do not contain this inner liner since the tube holds the air from leaking out of the tire. Tires that are meant to be used without a tube have the word tubeless on the sidewall.

BIAS PLY OR RADIAL

Another means of classifying an aircraft tire is by the direction of the plies used in construction of the tire, either bias or radial. Traditional aircraft tires are bias ply tires. The plies are wrapped to form the tire and give it strength. The angle of the plies in relation to the direction of rotation of the tire varies between 30° and 60°. In this manner, the plies have the bias of the fabric from which they are constructed facing the direction of rotation and across the tire. Hence, they are called bias tires. The result is flexibility as the sidewall can flex with the fabric plies laid on the bias. (***Figure 16-77***)

Some modern aircraft tires are radial tires. The plies in radial tires are laid at a 90° angle to the direction of rotation of the tire. This configuration puts the non-stretchable fiber of the plies perpendicular to the sidewall and direction of rotation. This creates strength in the tire allowing it to carry high loads with less deformation. (***Figure 16-78***)

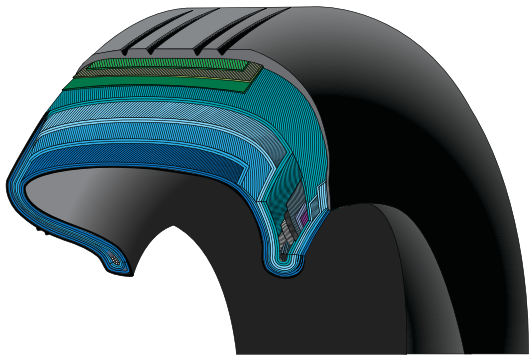


Figure 16-77. A bias ply tire has the fabric bias oriented with and across the direction of rotation and the sidewall. Since fabric can stretch on the bias, the tire is flexible, and can absorb loads. Strength is obtained by adding plies.

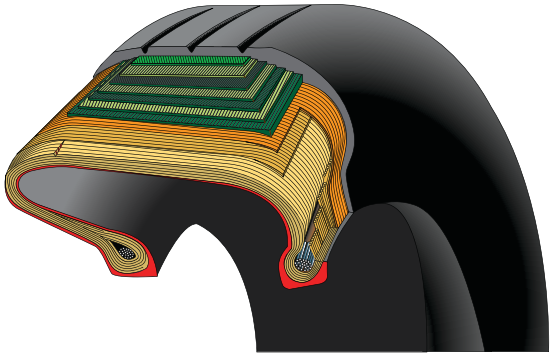


Figure 16-78. A radial tire has the fiber strands of the ply fabric oriented with and at 90 degrees to the direction of rotation and the tire sidewall. This restricts flexibility directionally and the flexibility of the sidewall while it strengthens the tire to carry heavy loads.

TIRE CONSTRUCTION

An aircraft tire is constructed for the purpose it serves. Unlike an automobile or truck tire, it does not have to carry a load for a long period of continuous operation. However, an aircraft tire must absorb the high impact loads of landing and be able to operate at high speeds even if only for a short time. The deflection built into an aircraft tire is more than twice that of an automobile tire. This enables it to handle the forces during landings without being damaged. Only tires designed for an aircraft as specified by the manufacturer should be used. It is useful to the understanding of tire construction to identify the various components of a tire and the functions contributed to the overall characteristics of a tire. Refer to ***Figure 16-79*** for tire nomenclature used in this discussion.

BEAD

The tire bead is an important part of an aircraft tire. It anchors the tire carcass and provides a dimensioned, firm mounting surface for the tire on the wheel rim. Tire beads are strong. They are typically made from high-strength carbon steel wire bundles encased in rubber.

CARCASS PLIES

Carcass plies, or casing plies as they are sometimes called, are used to form the tire. Each ply consists of fabric, usually nylon, sandwiched between two layers of rubber. The plies are applied in layers to give the tire strength and form the carcass body of the tire.

TREAD

The tread is the crown area of the tire designed to come in contact with the ground. It is a rubber compound formulated to resist wear, abrasion, cutting, and cracking. It also is made to resist heat build-up. Most modern aircraft tire tread is formed with circumferential grooves that create tire ribs. The grooves provide cooling and help channel water from under the tire in wet conditions to increase adhesion to the ground surface.

SIDEWALL

The sidewall of an aircraft tire is a layer of rubber designed to protect the carcass plies. It may contain compounds designed to resist the negative effects of ozone on the tire. It also is the area where information about the tire is contained. The tire sidewall imparts little strength to the cord body. Its main function is protection.

CHINE

Some tire sidewalls are mounded to form a chine. A chine is a special built-in deflector used on nose wheels of certain aircraft, usually those with fuselage mounted engines. The chine diverts runway water to the side and away from the intake of the engines.

TIRE INSPECTION ON THE AIRCRAFT

Tire condition is inspected while mounted on the aircraft on a regular basis. Inflation pressure, tread wear and condition, and sidewall condition are continuously monitored to ensure proper tire performance.

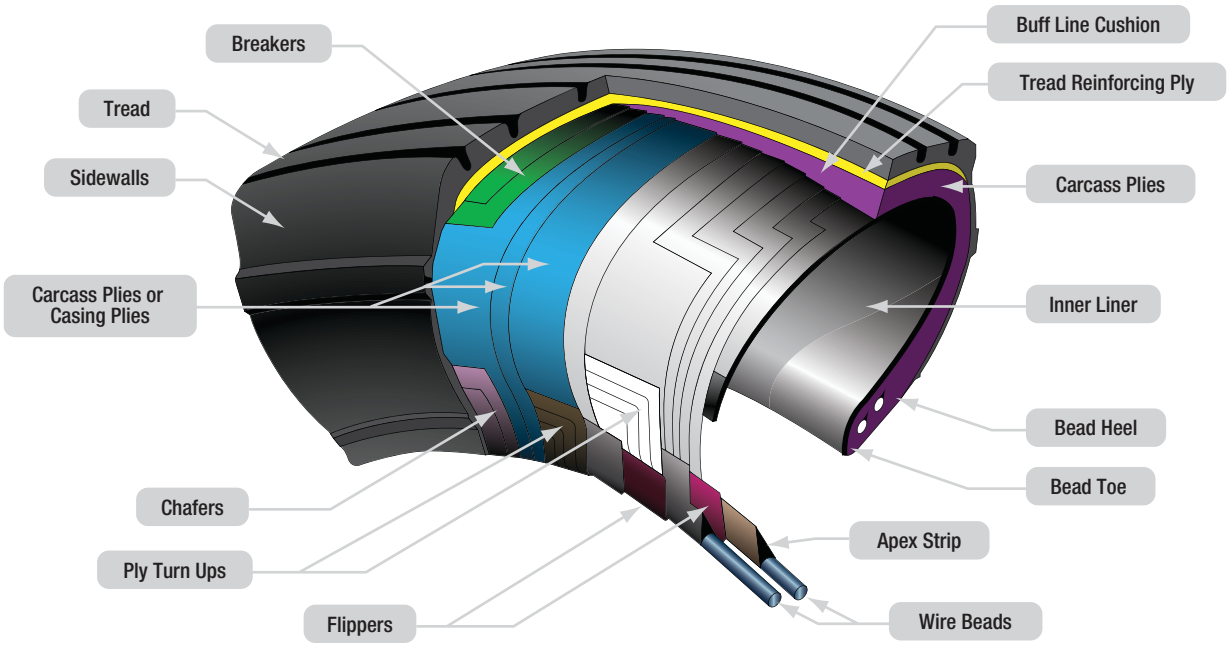


Figure 16-79. Construction nomenclature of an aircraft tire.

TIRE REMOVAL

Removal of any tire and wheel assembly should be accomplished following all aircraft manufacturer's instructions for the procedure. Safety procedures are designed for the protection of the technician and the maintenance of aircraft parts in serviceable condition. Follow all safety procedures to prevent personal injury and damage to aircraft parts and assemblies. An aircraft tire and wheel assembly, especially a high pressure assembly that has been damaged or overheated, should be treated as though it may explode. Never approach such a tire while its temperature is still elevated above ambient temperature. Once cooled, approach a damage tire and wheel assembly from an oblique angle advancing toward the shoulder of the tire. (***Figure 16-80***)

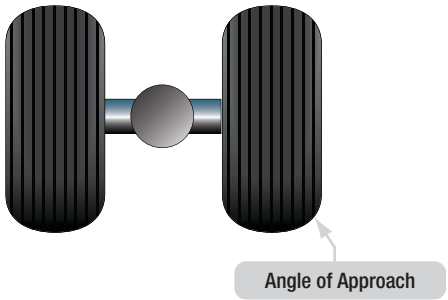


Figure 16-80. To avoid potential injury, approach a tire/wheel assembly that has damage or has been overheated at an angle toward the tire shoulder only after it has cooled to ambient temperature.

Question: 16-1
Three basic arrangements of landing gear are the tail wheel type landing gear (also known as conventional gear), tandem landing gear, and _____ landing gear.

Question: 16-2
Weight on wheels (WOW) integrated circuit logic cards communicate air-ground status information with the other systems on the aircraft through a _____.

Question: 16-3
Aircraft _____ support the entire weight of the aircraft during taxi, takeoff, and landing.

Question: 16-4
Large, heav y aircraft require the use of _____ brakes.

Question: 16-5
Power brake systems on large aircraft use the _____ as a source of power.

Question: 16-6
The majority of the rudder/brake pedal feel is supplied by the brake control or brake _____ in a power brake system.

Question: 16-7
Three main components of the brake antiskid systems are _____, _____ and a _____.

Question: 16-8
Anti-skid control valves are fast-acting, _____ controlled hydraulic valves that respond to the input from the anti-skid control unit.

ANSWERS

Answer: 16-1
tricycle-type.

Answer: 16-2
digital data bus.

Answer: 16-3
wheels.

Answer: 16-4
multiple-disc.

Answer: 16-5
aircraft main hydraulic system.

Answer: 16-6
metering valve.

Answer: 16-7
wheel speed sensors.
antiskid control valves.
control unit.

Answer: 16-8
Electrically.



PART-66 SYLLABUS **LEVELS**

CERTIFICATION CATEGORY → **B2**

Sub-Module 17

OXYGEN (ATA 35)

Knowledge Requirements

13.17 - Oxygen (ATA 35)

System lay-out: cockpit, cabin;	3
Sources, storage, charging and distribution;	3
Supply regulation;	3
Indications and warnings.	3

OXYGEN AND THE ATMOSPHERE

The mixture of gases that make up the earth's atmosphere is commonly called air. It is composed principally of 78 percent nitrogen and 21 percent oxygen. The remaining 1 percent is made up of various gases in smaller quantities. Some of these are important to human life, such as carbon dioxide, water vapor, and ozone. ***Figure 16-1*** indicates the respective percentage of the quantity of each gas in its relation to the total mixture.

As altitude increases, the total quantity of all the atmospheric gases reduces rapidly. However, the relative proportions of nitrogen and oxygen remain unchanged up to about 50 miles above the surface of the earth. The percentage of carbon dioxide is also fairly stable. The amounts of water vapor and ozone vary. Nitrogen is an inert gas that is not used directly by man for life processes; however, many compounds containing nitrogen are essential to all living matter.

The small quantity of carbon dioxide in the atmosphere is utilized by plants during photosynthesis. Thus, the food supply for all animals, including man, depends on it. Carbon dioxide also helps control breathing in man and other animals. The amount of water vapor in the atmosphere is variable but, even under humid conditions at sea level, it rarely exceeds 5 percent. Water also occurs in the atmosphere as ice crystals. All forms of water in the atmosphere absorb far more energy from the sun than do the other gases. Water plays an important role in the formation of weather.

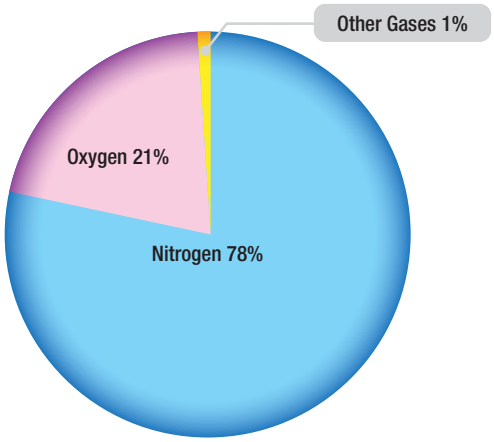


Figure 17-1. The percentage of the various gases that comprise the atmosphere.

Ozone is a form of oxygen. It contains three oxygen atoms per molecule, rather than the usual two. Most of the atmosphere's ozone is formed by the interaction of oxygen and the sun's rays near the top of the stratosphere in an area called the ozone layer. This is important to living organisms because ozone filters out most of the sun's harmful ultraviolet (UV) radiation. Ozone is also produced by electrical discharges, such as lightning strikes. It has a faint odor, somewhat like that of weak chlorine, that may be detected after a thunderstorm. Auroras and cosmic rays may also produce ozone. Ozone is of great consequence to living creatures on earth and to the circulation of the upper atmosphere.

HUMAN RESPIRATION AND CIRCULATION

The second most prevalent substance in the atmosphere, oxygen, is essential for most living processes. Without oxygen, humans and animals die very rapidly. A reduction in the normal oxygen supply alters the human condition. It causes important changes in body functions, thought processes, and the maintainable degree of consciousness. The resultant sluggish condition of mind and body produced by insufficient oxygen is called hypoxia.

There are several scenarios that can result in hypoxia. During aircraft operations, it is brought about by a decrease in the pressure of oxygen in the lungs at high altitudes. The air contains the typical 21 percent of oxygen, but the rate at which oxygen can be absorbed into the blood depends upon the oxygen pressure. Greater pressure pushes the oxygen from the lung alveoli into the bloodstream. As the pressure is reduced, less oxygen is forced into and absorbed by the blood.

At sea level, oxygen pressure in the lungs is approximately three pounds per square inch (psi). This is sufficient to saturate the blood with oxygen and permit the mind and body to function normally. As altitude is increased, this pressure decreases. Below 7 000 feet above sea level, the available oxygen quantity and pressure remain sufficient for saturation of the blood with oxygen. Above 7 000 feet, however, the oxygen pressure becomes increasingly insufficient to saturate the blood. At 10 000 feet mean

sea level (MSL), saturation of the blood with oxygen is only about 90 percent of normal. Long durations at this altitude can result in headache and fatigue, both symptoms of hypoxia.

At 15 000 feet MSL, oxygen transfer to the bloodstream drops to 81 percent of saturation. This typically results in sleepiness, headache, blue lips and fingernails, and increased pulse and respiration. Worse yet, vision and judgment become impaired and safe operation of an aircraft becomes compromised. Higher in the atmosphere, decreasing pressure causes even less oxygen to enter the bloodstream; only 68 percent saturation at 22 000 feet MSL. Remaining at 25 000 feet MSL for 5 minutes, where oxygen transfer to the blood is reduced to approximately 50 percent saturation, causes unconsciousness. (***Figure 17-2***)

Altitude MSL (feet)	Oxygen Pressure (psi)
0	3.08
5 000	2.57
10 000	2.12
15 000	1.74
20 000	1.42
25 000	1.15
30 000	0.92
35 000	0.76
40 000	0.57

Figure 17-2. Oxygen pressure in the atmosphere at various altitudes.

The negative effects of reduced atmospheric pressure at flight altitudes, forcing less oxygen into the blood, can be overcome.

There are two ways this is commonly done:
1. Increase the pressure of the oxygen or,
2. Increase the quantity of oxygen in the air mixture.

Large transport-category and high performance passenger aircraft pressurize the air in the cabin. This serves to push more of the normal 21 percent oxygen found in the air into the blood for saturation. When utilized, the percentage of oxygen available for breathing remains the same; only the pressure is increased.

By increasing the quantity of oxygen available in the lungs, less pressure is required to saturate the blood. This is the basic function of an aircraft oxygen system. Increasing the level of oxygen above the 21 percent found in the atmosphere can offset the reduced pressure encountered as altitude increases. Oxygen may be regulated into the air that is breathed so as to maintain a sufficient amount for blood saturation. Normal mental and physical activity can be maintained at indicated altitudes of up to about 40 000 feet with the sole use of supplemental oxygen.

Oxygen systems that increase the quantity of oxygen in breathing air are most commonly used as primary systems in small and medium size aircraft designed without cabin pressurization. Pressurized aircraft utilize oxygen systems as a means of redundancy should pressurization fail. Portable oxygen equipment may also be aboard for first aid purposes.

SOURCES OF OXYGEN

Oxygen is a colorless, odorless, and tasteless gas at normal atmospheric temperatures and pressures. It transforms into a liquid at -183 °C (its boiling point). Oxygen combines readily with most elements and numerous compounds. This combining is called oxidation. Typically, oxidation produces heat. When something burns, it is actually rapidly combining with oxygen. Oxygen itself does not burn because it does not combine with itself, except to form oxygen or ozone. But, pure oxygen combines violently with petroleum products creating a significant hazard when handling these materials in close proximity to each other. Nevertheless, oxygen and various petroleum fuels combine to create the energy produced in internal combustion engines.

Production of gaseous oxygen for commercial or aircraft cylinders is often through a process of liquefying air. By controlling temperature and pressure, the nitrogen in the air can be allowed to boil off leaving mostly pure oxygen. Oxygen may also be produced by the electrolysis of water. Passing electric current through water separates the oxygen from the hydrogen. One further method of

producing gaseous oxygen is by separating the nitrogen and oxygen in the air through the use of a molecular sieve. This membrane filters out nitrogen and some of the other gases in air, leaving nearly pure oxygen for use. Onboard oxygen sieeves, or oxygen concentrators as they are sometimes called, are used on some military aircraft. Their use in civil aviation is expected.

Another source of oxygen used frequently for emergency passenger oxygen on airline aircraft is chemical in nature. The chemical combination with oxygen when burning some materials causes a release of excess oxygen that can be captured and used. This unique production of oxygen is explained further below.

OXYGEN STORAGE

GASEOUS OXYGEN TANKS

Pure gaseous oxygen, or nearly pure gaseous oxygen, is stored and transported in high-pressure cylinders that are typically painted green. Technicians should be cautious to keep pure oxygen away from fuel, oil, and grease to prevent unwanted combustion. Not all oxygen in containers is the same. Aviator's breathing oxygen is tested for the presence of water. This is done to avoid the possibility of it freezing in the small passage ways of valves and regulators. Ice could prevent delivery of the oxygen when needed. Aircraft often operate in subzero temperatures, increasing the possibility of icing. The water level should be a maximum of .02ml per liter of oxygen. The words "Aviator's Breathing Oxygen" should be marked clearly on any cylinders containing oxygen for this purpose. (***Figure 17-3***)



Figure 17-3. "Aviator's breathing oxygen" is marked on all oxygen cylinders designed for this purpose.

Traditionally, the cylinders used to store high pressure oxygen have been heavy steel tanks rated for 1 800 - 1 850 psi of pressure and capable of maintaining pressure up to 2 400 psi. While these performed adequately, lighter weight tanks have been developed. Some newer cylinders are comprised of a lightweight aluminum shell wrapped by Kevlar™. These cylinders are capable of carrying the same amount of oxygen at the same pressure as steel tanks, but weigh much less. Also available are heavy-walled all-aluminum cylinders. These units are common as carry-on portable oxygen used in light aircraft. Most oxygen storage cylinders are painted green, but yellow and other colors may be used as well. The tanks are typically certified to department of transportation (DOT) specifications in the country of manufacture. To ensure continued serviceability, cylinders must be hydrostatically tested periodically. In general, a hydrostatic test consists of filling the container with water and pressurizing it to ⅝ of its certified rating. It should not leak, rupture, or deform beyond an established limit. ***Figure 17-4*** shows a hydrostatic cylinder testing apparatus.

Most cylinders also have a limited service life after which they can no longer be used. After a specified number of filling cycles or calendar age, the cylinders must be removed from service. The most common high-pressure steel oxygen cylinders used in aviation are the 3AA and the 3HT. They come in various sizes but are certified to the same specifications. Cylinders certified under U.S. DOT-E-8162 are popular for their extremely light weight. These cylinders typically have an aluminum core around which Kevlar™ is wrapped. The DOT-E- 8162 approved cylinders are also approved under DOT-SP-8162 specifications. The SP certification has extended the required time between hydrostatic testing to 5 years (previously 3 years). (***Figure 17-5***)

The manufactured date and certification number is stamped on each cylinder near the neck. Subsequent hydrostatic test dates are also stamped there as well.

Composite cylinders use placards rather than stamping. The placard must be covered with a coat of clear epoxy when additional information is added, such as a new hydrostatic test date.

Oxygen cylinders are considered empty when the pressure inside drops below 50 psi. This ensures that air containing water vapor has not entered the cylinder.



Figure 17-4. This test stand is used for hydrostatic testing of oxygen cylinders. The water-filled cylinder is lowered into the barrel on the left where it is pressurized to the proper level as monitored via gauges mounted on the control panel. A displacement container on the top left of the control board collects water from the barrel to measure the expansion of the cylinder when pressurized to ensure it is within limits.

Water vapor could cause corrosion inside the tank, as well as presenting the possibility of ice forming and clogging a narrow passageway in the cylinder valve or oxygen system. Any installed tank allowed to fall below this pressure should be removed from service.

CHEMICAL OR SOLID OXYGEN

Sodium chlorate has a unique characteristic. When ignited, it produces oxygen as it burns. This can be filtered and delivered through a hose to a mask that can be worn and breathed directly by the user. Solid oxygen candles, as they are called, are formed chunks of sodium chlorate wrapped inside insulated stainless steel housings to control the heat produced when activated. The chemical oxygen supply is often ignited by a spring-loaded firing pin that when pulled, releases a hammer that smashes a cap creating a spark to light the candle. Electric ignition via a current-induced hot wire also exists. Once lit, a sodium chlorate oxygen generator cannot be extinguished. It produces a steady flow of breathable oxygen until it burns out, typically generating 10-20 minutes of oxygen. (***Figure 17-6***)

Solid oxygen generators are primarily used as backup oxygen devices on pressurized aircraft. They are one-third as heavy as gaseous oxygen systems that use heavy storage tanks for the same quantity of oxygen available. Sodium chlorate chemical oxygen generators also have a long shelf life, making them perfect as a standby form of oxygen. They are inert below 400 °F and can remain stored with little maintenance or inspection until needed, or until their expiration date is reached.

The feature of not extinguishing once lit limits the use of solid oxygen since it becomes an all-or-nothing source. The generators must be replaced if used, which can greatly increase the cost of using them as a source of oxygen for short periods of time. Moreover, chemical

Certification Type	Material	Rated Pressure (psi)	Required Hydrostatic Test	Service Life (years)	Service Life (fillings)
DOT 3AA	Steel	1 800	5	Unlimited	N/A
DOT 3HT	Steel	1 850	3	24	4 380
DOT-E-8162	Composite	1 850	3	15	N/A
DOT-SP-8162	Composite	1 850	5	15	N/A
DOT 3AL	Aluminum	2 216	5	Unlimited	N/A

Figure 17-5. Common cylinders used in aviation with some certification and testing specifications.

oxygen candles must be transported with extreme caution and as hazardous materials. They must be properly packed and their ignition devices deactivated.

ONBOARD OXYGEN GENERATING SYSTEMS (OBOGS)

The molecular sieve method of separating oxygen from the other gases in air has application in flight, as well as on the ground. The sieves are relatively light in weight and relieve the aviator of a need for ground support for the oxygen supply. Onboard oxygen generating systems on military aircraft pass bleed air from turbine engines through a sieve that separates the oxygen for breathing use. Some of the separated oxygen is also used to purge the sieve of the nitrogen and other gases that keep it fresh for use. Use of this type of oxygen production in civilian aircraft is anticipated. (***Figure 17-7***)

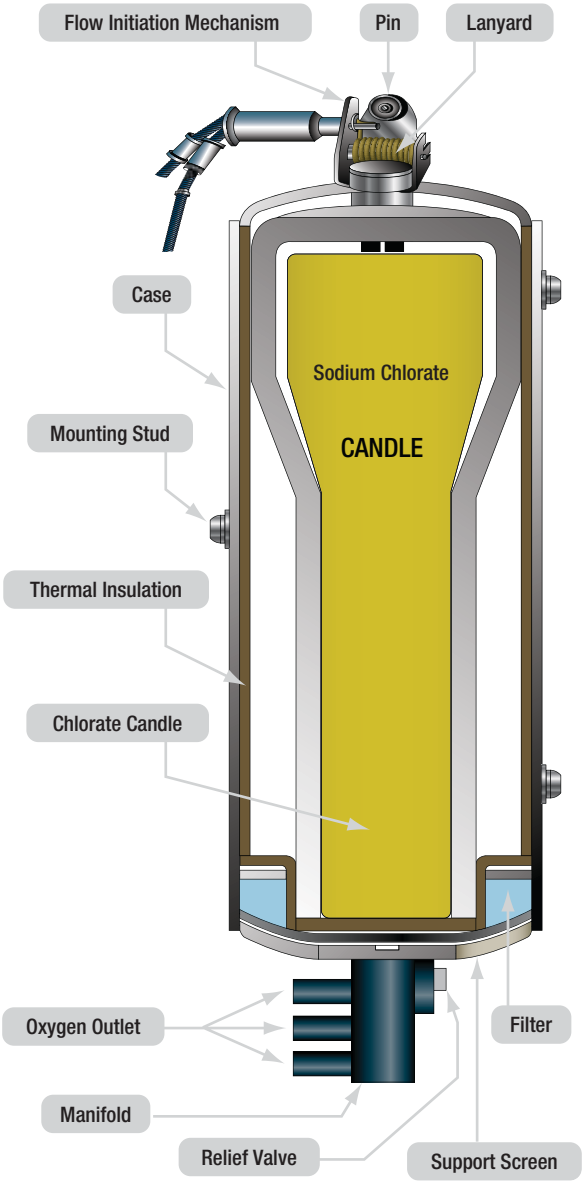


Figure 17-6. A sodium chlorate solid oxygen candle is at the core of a chemical oxygen generator.

LIQUID OXYGEN

Liquid oxygen (LOX) is a pale blue, transparent liquid. Oxygen can be made liquid by lowering the temperature to below -183 °C or by placing gaseous oxygen under pressure. A combination of these is accomplished with a Dewar bottle. This special container is used to store and transport liquid oxygen. It uses an evacuated, double-walled insulation design to keep the liquid oxygen under pressure at a very low temperature. (***Figure 17-8***) A controlled amount of oxygen is allowed to vaporize and is plumbed into a gaseous oxygen delivery system downstream of a converter that is part of the container assembly.



Figure 17-7. This on-board oxygen generating system uses molecular sieve technology.



Figure 17-8. A spherical liquid oxygen on-board container used by the military.

A small quantity of LOX can be converted to an enormous amount of gaseous oxygen, resulting in the use of very little storage space compared to that needed for high-pressure gaseous oxygen cylinders. However, the difficulty of handling LOX, and the expense of doing so, has resulted in the container system used for gaseous oxygen to proliferate throughout civilian aviation. LOX is used nearly exclusively in military aviation.

OXYGEN CHARGING

Charging procedures for oxygen systems vary. Many general aviation aircraft are set up to simply replace an empty cylinder with one that is fully charged. This is also the case with a portable oxygen system. High performance and air transport category aircraft often have built-in oxygen systems that contain plumbing designed to refill gaseous oxygen cylinders while they are in place. A general discussion of the procedure to fill this type of installation follows. Before charging any oxygen system, consult the aircraft manufacturer's maintenance manual. The type of oxygen to be used, safety precautions, equipment to be used, and the procedures for filling and testing the system must be observed. Several general precautions should also be observed when servicing a gaseous oxygen system. Oxygen valves should be opened slowly and filling should proceed slowly to avoid overheating. The hose from the refill source to the oxygen fill valve on the aircraft should be purged of air before it is used to transfer oxygen into the system. Pressures should also be checked frequently while refilling.

Airline and fixed-base operator maintenance shops often use oxygen filler carts to service oxygen systems. These contain several large oxygen supply cylinders connected to the fill cart manifold. This manifold supplies a fill hose that attaches to the aircraft. Valves and pressure gauges allow awareness and control of the oxygen dispensing process. (***Figure 17-9***)

Be sure all cylinders on the cart are aviator's breathing oxygen and that all cylinders contain at least 50 psi of oxygen pressure. Each cylinder should also be within its hydrostatic test date interval. After a cart cylinder has dispensed oxygen, the remaining pressure should be recorded. This is usually written on the outside of the cylinder with chalk or in a cylinder pressure log kept with the cart. As such, the technician can tell at a glance the status of each oxygen bottle. No pump or mechanical device is used to transfer oxygen from the fill cart manifold to the aircraft system. Objects under pressure flow from high pressure to low pressure. Thus, by connecting the cart to the aircraft and systematically opening oxygen cylinders with increasingly higher pressure, a slow increase in oxygen flow to the aircraft can be managed. The following is a list of steps to safely fill an aircraft oxygen system from a typical oxygen refill cart.

1. Check hydrostatic dates on all cylinders, especially those that are to be filled on the aircraft. If a cylinder is out of date, remove and replace it with a specified unit that is serviceable.
2. Check pressures on all cylinders on the cart and in the aircraft. If pressure is below 50 psi, replace the cylinder(s). On the aircraft, this may require purging the system with oxygen when completed. Best practices dictate that any low-pressure or empty cylinder(s) on the cart should also be removed and replaced when discovered.
3. Take all oxygen handling precautions to ensure a safe environment around the aircraft.
4. Ground the refill cart to the aircraft.
5. Connect the cart hose from the cart manifold to the aircraft fill port. Purge the air from the refill hose with oxygen before opening the refill valve on the aircraft. Some hoses are equipped with purge valves to do this while the hose is securely attached to the aircraft. Others hoses need to be purged while attached to the refill fitting but not fully tightened.
6. Observe the pressure on the aircraft bottle to be filled. Open it. On the refill cart, open the cylinder with the closest pressure to the aircraft cylinder that exceeds it.
7. Open the aircraft oxygen system refill valve. Oxygen will flow from cart cylinder (manifold) into the aircraft cylinder.
8. When the cylinder pressures equalize, close the cylinder on the cart, and open the cart cylinder with the next highest pressure. Allow it to flow into the aircraft cylinder until the pressures equalize and flow ceases. Close the cart cylinder, and proceed to the cart cylinder with the next highest pressure.
9. Continue the procedure in step 8 until the desired pressure in the aircraft cylinder is achieved.



Figure 17-9. Typical oxygen servicing cart used to fill an aircraft system.

10. Close the aircraft refill valve, and close all cylinders on the cart.
11. The aircraft oxygen cylinder valve(s) should be left in the proper position for normal operations. Remotely mounted cylinders are usually left open.
12. Disconnect the refill line from the refill port on the aircraft. Cap or cover both.
13. Remove the grounding strap.

Temperature has a significant effect on the pressure of gaseous oxygen. Manufacturers typically supply a fill chart or a placard at the aircraft oxygen refill station to guide technicians in compensating for temperature/pressure variations. Technicians should consult the chart and fill cylinders to the maximum pressure listed for the prevailing ambient temperature. (***Figure 17-10***)

When it is hot, oxygen cylinders are filled to a higher pressure than 1 800 psi or 1 850 psi, the standard maximum pressure ratings of most high-pressure aircraft oxygen cylinders. This is allowable because at altitude the temperature and pressure of the oxygen can decrease significantly. Filling cylinders to temperature-compensated pressure values ensures a full supply of oxygen is available when needed.

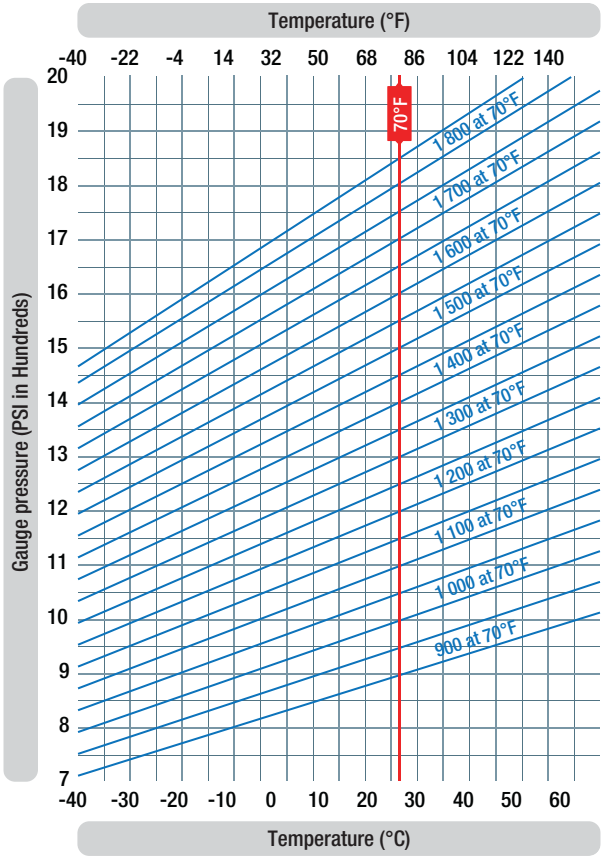


Figure 17-10. A temperature-compensating pressure refill chart is used by the technician to ensure proper oxygen cylinder pressure in the aircraft system.

When filling cylinders on a cold day, compensation for temperature and pressure changes dictates that cylinders be filled to less than the maximum rated capacity to allow for pressure increases as temperature rises. Strict adherence to the temperature/pressure compensation chart values is mandatory to ensure safe storage of aircraft oxygen.

Note that some aircraft have temperature compensation features built into the refill valve. After setting the ambient temperature on the valve dial, the valve closes when the correct amount of oxygen pressure has been established in the aircraft cylinder. A chart can be used to ensure proper servicing.

OXYGEN SYSTEMS AND SUPPLY REGULATION

The use of gaseous oxygen in aviation is common; however, applications vary. On a light aircraft, it may consist of a small carry-on portable cylinder with a single mask attached via a hose to a regulator on the bottle. Larger portable cylinders may be fitted with a regulator that divides the outlet flow for 2-4 people. Built-in oxygen systems on high performance and light twin-engine aircraft typically have a location where oxygen

cylinders are installed to feed a distribution system via tubing and a regulator. The passenger compartment may have multiple breathing stations plumbed so that each passenger can individually plug in a hose and mask if oxygen is needed. A central regulator is normally controlled by the flight crew who may have their own separate regulator and oxygen cylinder. Transport category aircraft may use a more elaborate built-in gaseous oxygen system as a backup system to cabin pressurization. In all of these cases, oxygen is stored as a gas at atmospheric temperature in high-pressure cylinders. It is distributed through a system with various components that are described in this section.

The design of the various oxygen systems used depends largely on the type of aircraft, its operational requirements, and whether the aircraft has a pressurization system. Systems are often characterized by the type of regulator used to dispense the oxygen: continuous-flow and demand flow. In some aircraft, a continuous-flow oxygen system is installed for both passengers and crew. The pressure demand system is widely used as a crew system, especially on the larger transport aircraft. Many aircraft have a combination of both systems that may be augmented by portable equipment.

CONTINUOUS-FLOW SYSTEMS

In its simplest form, a continuous-flow oxygen system allows oxygen to exit the storage tank through a valve and passes it through a regulator/reducer attached to the top of the tank. The flow of high-pressure oxygen passes through a section of the regulator that reduces the pressure of the oxygen, which is then fed into a hose attached to a mask worn by the user. Once the valve is opened, the flow of oxygen is continuous. Even when the user is exhaling, or when the mask is not in use, a preset flow of oxygen continues until the tank valve is closed.

On some systems, fine adjustment to the flow can be made with an adjustable flow indicator that is installed in the hose in line to the mask. A portable oxygen setup for a light aircraft exemplifies this type of continuous-flow system and is shown in ***Figure 17-11***.

A more sophisticated continuous-flow oxygen system uses a regulator that is adjustable to provide varying amounts of oxygen flow to match increasing need as altitude increases.



Figure 17-11. A typical portable gaseous oxygen cylinder complete with valve, pressure gauge, regulator/reducer, hose, adjustable flow indicator, and rebreather cannula. A padded carrying case/bag can be strapped to the back of a seat in the cabin to meet certification and testing specifications.

These regulators can be manual or automatic in design. Manual continuous-flow regulators are adjusted by the crew as altitude changes. Automatic continuous-flow regulators have a built in aneroid. As the aneroid expands with altitude, a mechanism allows more oxygen to flow though the regulator to the users. (***Figure 17-12***)

SYSTEM LAYOUT: CABIN (CONTINUOUS FLOW)

Many continuous-flow systems include a fixed location for the oxygen cylinders with permanent delivery plumbing installed to all passenger and crew stations in the cabin. Fully integrated oxygen systems usually have separate, remotely mounted components to reduce pressure and regulate flow. A pressure relief valve is also typically installed in the system, as is some sort of filter and a gauge to indicate the amount of oxygen pressure remaining in the storage cylinder(s). ***Figure 17-13*** diagrams the type of continuous-flow system that is found on small to medium sized aircraft.



Figure 17-12. A manual continuous flow oxygen system may have a regulator that is adjusted by the pilot as altitude varies. By turning the knob, the left gauge can be made to match the flight altitude thus increasing and decreasing flow as altitude changes.

Built-in continuous-flow gaseous oxygen systems accomplish a final flow rate to individual user stations through the use of a calibrated orifice in each mask. Larger diameter orifices are usually used in crew masks to provide greater flow than that for passengers. Special oxygen masks provide even greater flow via larger orifices for passengers traveling with medical conditions requiring full saturation of the blood with oxygen. Allowing oxygen to continuously flow from the storage cylinder can be wasteful. Lowest sufficient flow rates can be accomplished through the use of rebreather apparatus. Oxygen and air that is exhaled still contains usable oxygen. By capturing this oxygen in a bag, or in a cannula with oxygen absorbing reservoirs, it can be inhaled with the next breath, reducing waste. (***Figure 17-14***)

The passenger section of a continuous-flow oxygen system may consist of a series of plug-in supply sockets fitted to the cabin walls adjacent to the passenger seats to which oxygen masks can be connected. Flow is inhibited until a passenger manually plugs in.

When used as an emergency system in pressurized aircraft, such as an airliner, depressurization automatically triggers the deployment of oxygen ready continuous-flow masks at each passenger station. A lanyard attached to the mask turns on the flow to each mask when it is pulled toward the passenger for use.

The masks are normally stowed overhead in the passenger service unit (PSU). (***Figure 17-15***) Deployment of the emergency continuous-flow passenger oxygen masks may also be controlled by the crew. (***Figure 17-16***) Continuous-flow oxygen masks are simple devices made to direct flow to the nose and mouth of the wearer. They fit snugly but are not air tight. Vent holes allow

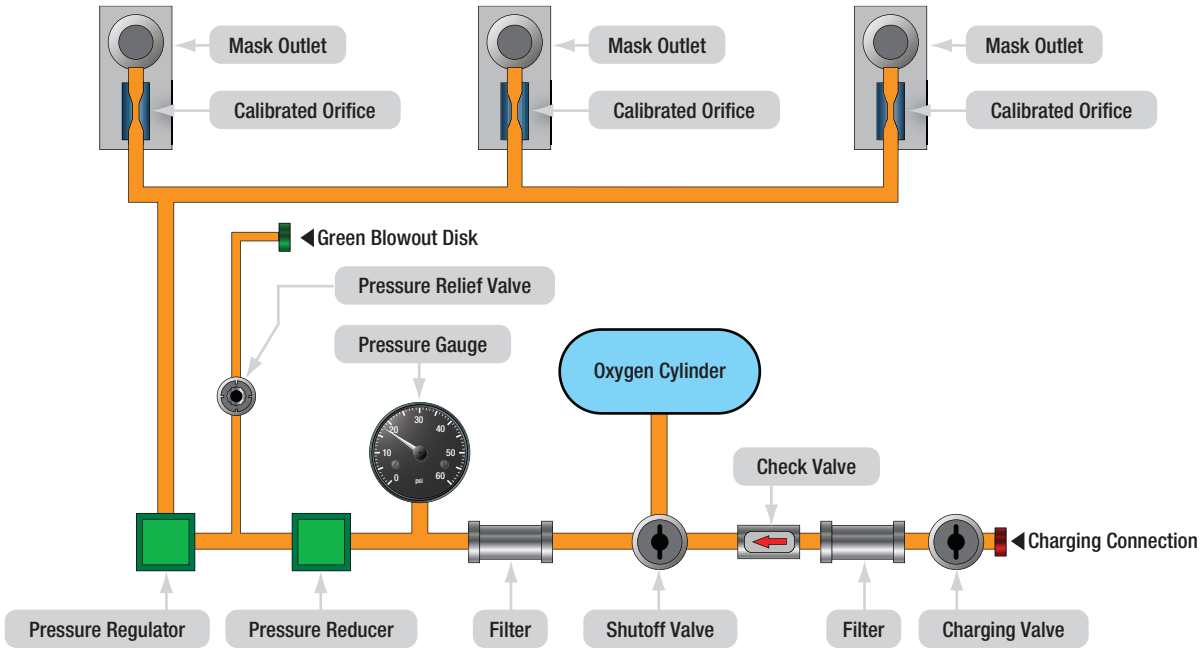


Figure 17-13. Continuous flow oxygen system found on small to medium size aircraft.

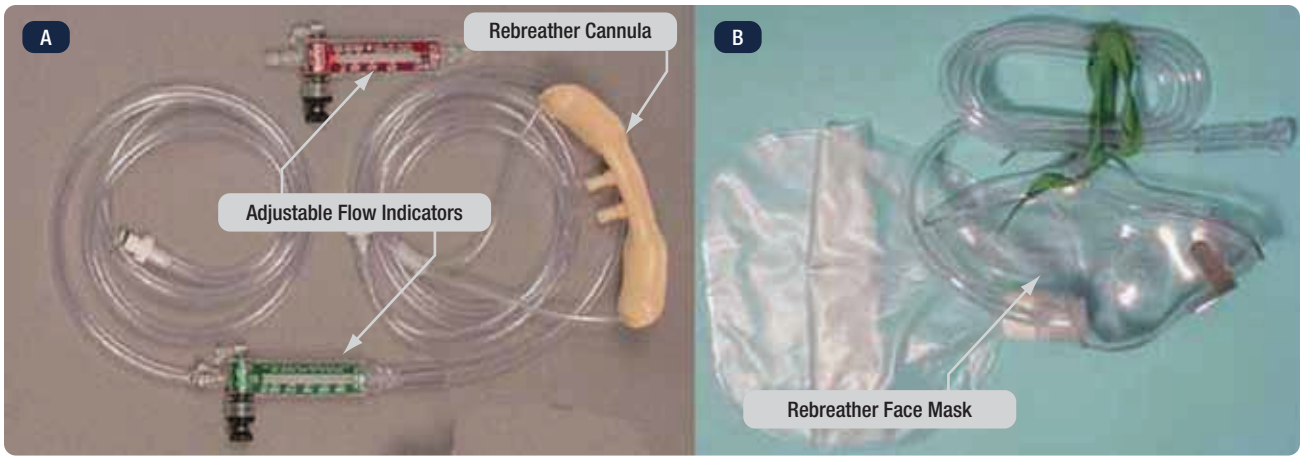


Figure 17-14. A rebreather cannula (A) and rebreather bag (B) capture exhaled oxygen to be inhaled on the next breath. This conserves oxygen by permitting lower flow rates in continuous flow systems. The red and green devices are optional flow indicators that allow the user to monitor oxygen flow rate. The type shown also contains needle valves for final regulation of the flow rate to each user.

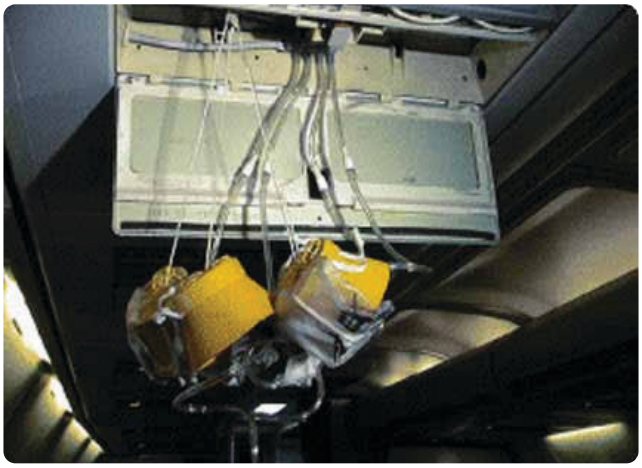


Figure 17-15. A passenger service unit (psu) is hinged over each row of seats in an airliner. Four yellow continuous flow oxygen masks are shown deployed. They are normally stored behind a separate hinged panel that opens to allow the masks to fall from the PSU for use.

cabin air to mix with the oxygen and provide escape for exhalation. In a rebreather mask, the vents allow the exhaled mixture that is not trapped in the rebreather bag to escape. This is appropriate, because this is the air-oxygen mixture that has been in the lungs the longest, so it has less recoverable oxygen to be breathed again. (***Figure 17-17***)

DEMAND-FLOW SYSTEMS

When oxygen is delivered only as the user inhales, or on demand, it is known as a demand-flow system. During the hold and exhalation periods of breathing, the oxygen supply is stopped. Thus, the duration of the oxygen supply is prolonged as none is wasted.



Figure 17-16. The crew can deploy passenger emergency continuous-flow oxygen masks and supply with a switch in the cockpit.

Demand-flow systems are similar to continuous-flow systems in that a cylinder delivers oxygen through a valve when opened. The tank pressure gauge, filter(s),



Figure 17-17. Examples of different continuous-flow oxygen masks.

pressure relief valve, and any plumbing installed to refill the cylinder while installed on the aircraft are all similar to those in a continuous flow system. The high-pressure oxygen also passes through a pressure reducer and a regulator to adjust the pressure and flow to the user. But, demand-flow oxygen regulators differ significantly from continuous-flow oxygen regulators. They work in conjunction with close-fitting demand-type masks to control the flow of oxygen. (***Figure 17-18***)

A pressure reduction occurs at the inlet of each individual demand regulator by limiting the size of the inlet orifice at the pressure reducing inlet valve. When the user inhales (demands) oxygen through the mask, this valve unseats and allows oxygen to flow through the regulator. When the user exhales, the valve reseats to conserve oxygen, There are two types of individual regulators: the diluter-demand type and the pressure-demand type. (***Figure 17-19***)

DILUTER-DEMAND

The diluter-demand type regulator holds back the flow of oxygen until the user inhales with a demand-type oxygen mask. The regulator dilutes the pure oxygen supply with cabin air each time a breath is drawn. With its control toggle switch set to normal, the amount of dilution depends on the cabin altitude. As altitude increases, an aneroid allows more oxygen and less cabin air to be delivered to the user by adjusting flows through a metering valve. At approximately 34 000 feet, the diluter-demand regulator meters 100 percent oxygen. This should not be needed unless cabin pressurization fails.

Additionally, the user may select 100 percent oxygen delivery at any time by positioning the oxygen selection lever on the regulator. A built-in emergency switch also delivers 100 percent oxygen, but in a continuous flow as the demand function is bypassed. (***Figure 17-20***)

PRESSURE DEMAND

Pressure-demand oxygen systems operate similarly to diluter demand systems, except that oxygen is delivered through the individual pressure regulator(s) under higher pressure. When the demand valve is unseated, oxygen under pressure forces its way into the lungs of the user. The demand function still operates, extending the overall supply of oxygen beyond that of a continuous-flow system. Dilution with cabin air also occurs if cabin altitude is less than 34 000 feet.

Pressure-demand regulators are used on aircraft that regularly fly at 40 000 feet and above. They are also found on many airliners and high-performance aircraft that may not typically fly that high. Forcing oxygen into the lungs under pressure ensures saturation of the blood, regardless of altitude or cabin altitude. Both diluter-demand and pressure-demand regulators also come in mask-mounted versions. The operation is essentially the same as that of panel-mounted regulators. (***Figure 17-21***)

SYSTEM LAYOUT: COCKPIT

Diluter-demand and pressure-demand flow systems are used most frequently by the crew on high performance and air transport category aircraft. (***Figure 17-22***)

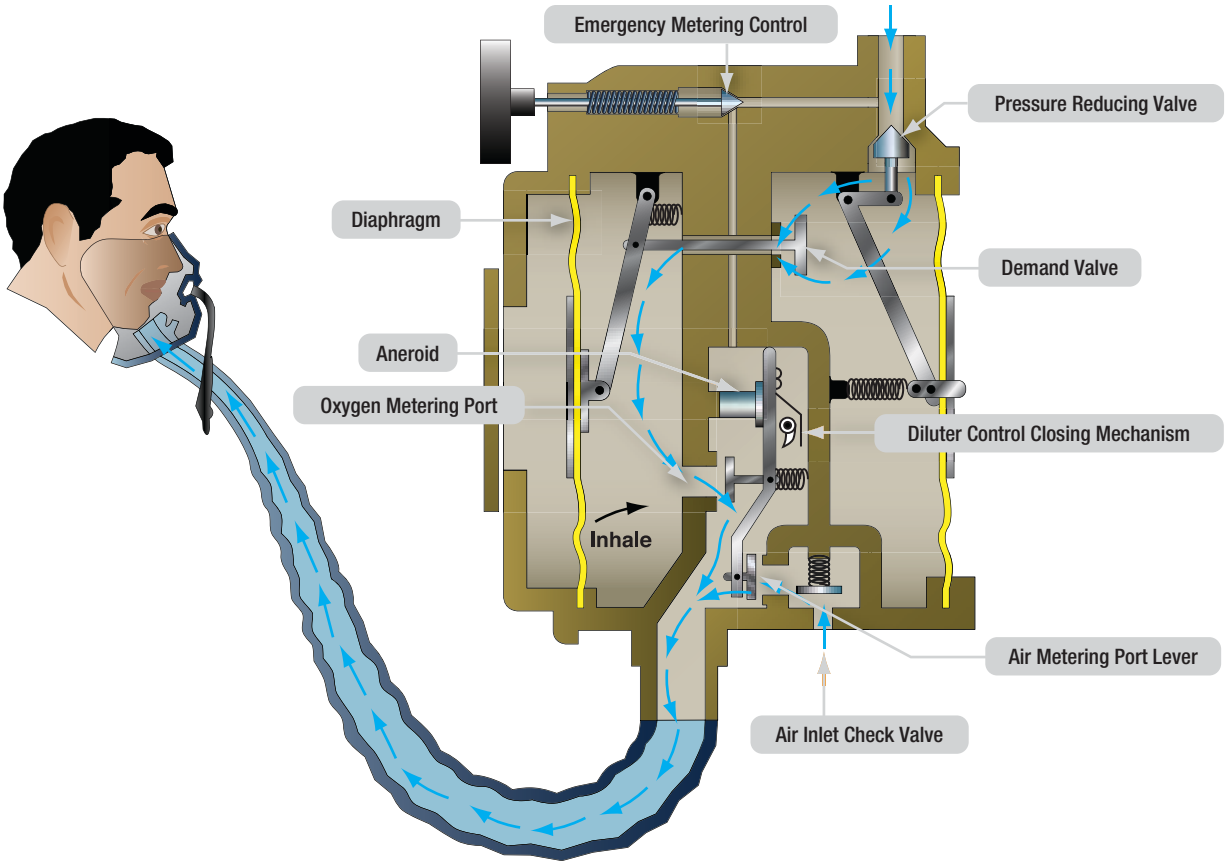


Figure 17-18. A demand regulator and demand-type mask work together to control flow and conserve oxygen. Demand-flow masks are close fitting so that when the user inhales, low pressure is created in the regulator, which allows oxygen to flow. Exhaled air escapes through ports in the mask, and the regulator ceases the flow of oxygen until the next inhalation.

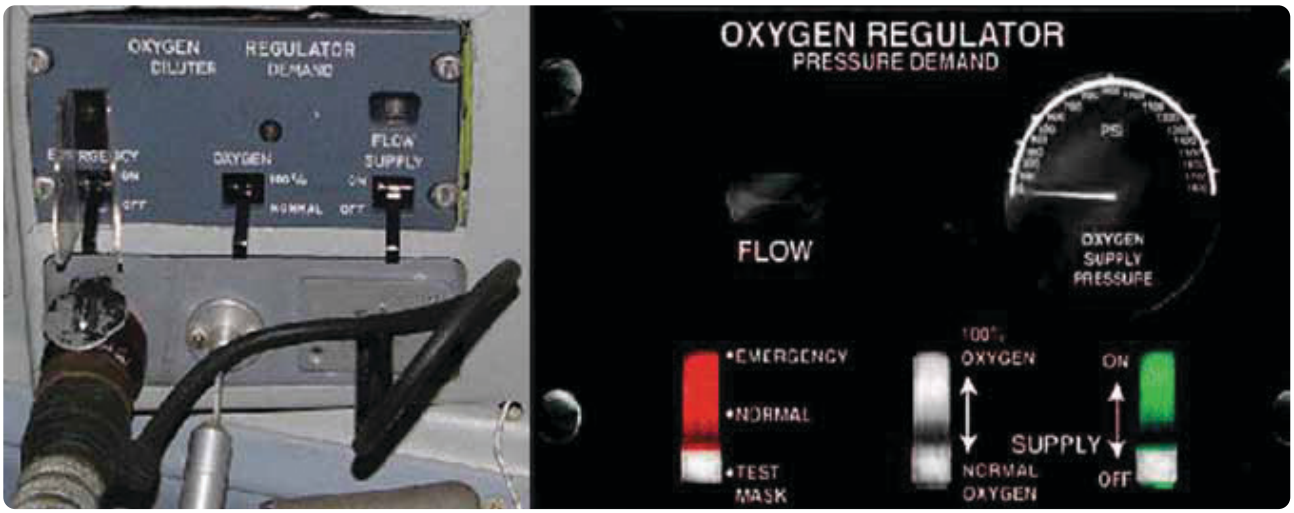


Figure 17-19. The two basic types of regulators used in demand flow oxygen systems. The panel below the diluter demand regulator on the left is available for mask hose plug in (left), lanyard mask hanger (center), and microphone plug in (right). Most high performance demand type masks have a microphone built-in.

The tanks and servicing panel are remotely mounted, often near the forward baggage compartment. In large aircraft, separate storage cylinders for crew and passengers are typical.

Also located near the tank is the system pressure reducing valve, sometimes called a pressure regulator. This device lowers the oxygen pressure from the storage cylinder(s) to roughly 60-85 psi. This oxygen is then

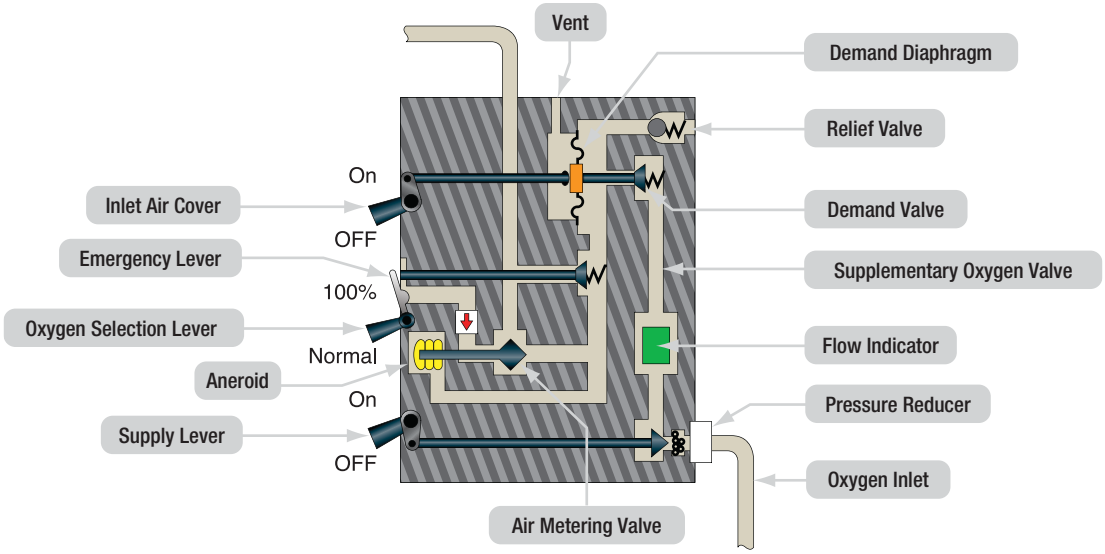


Figure 17-20. A diluter-demand regulator operates when low pressure caused by inhalation moves the demand diaphragm. A demand valve connected to the diaphragm opens, letting oxygen flow through the metering valve. The metering valve adjusts the mixture of cabin air and pure oxygen via a connecting link to an aneroid that responds to cabin altitude.



Figure 17-21. A mask-mounted version of a miniature diluter-demand regulator designed for use in general aviation (left), a mechanical quick-donning diluter-demand mask with the regulator on the mask (center), and an inflatable quick-donning mask (right). Squeezing the red grips directs oxygen into the hollow straps.

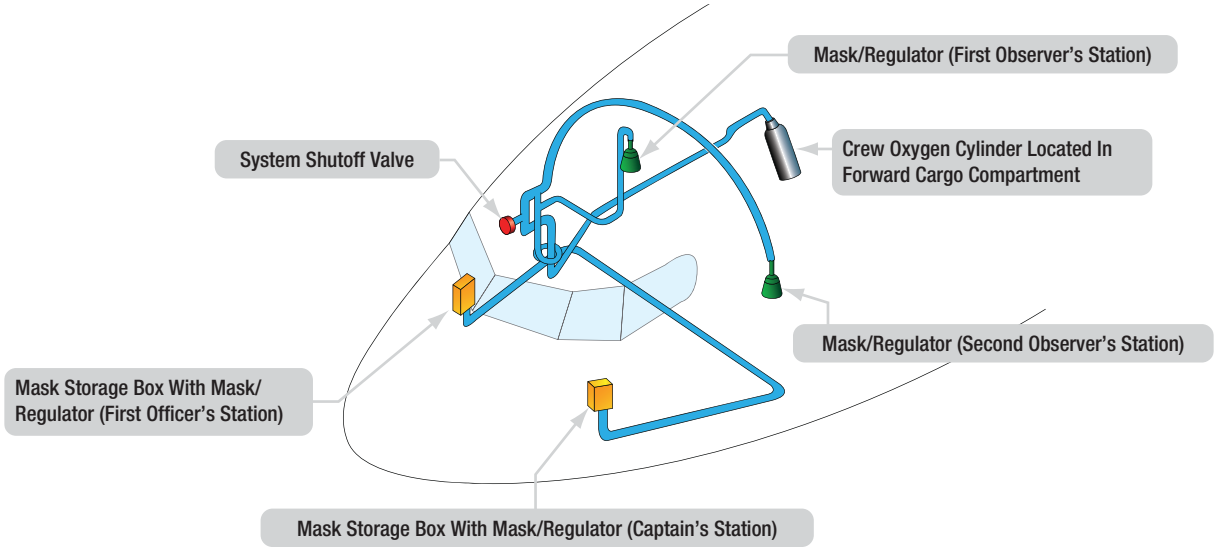


Figure 17-22. Location of demand-flow oxygen components on a transport category aircraft.

distributed to the flight deck though a system of tubing and valves and into the individual demand regulators located at each crew station.

Each demand regulator is controlled by the user. A system shutoff valve is located upstream of the station regulators and masks. An electric switch, often on the flight deck overhead panel, controls the shutoff valve. Tank pressure indicators may be located near the switch. Manual/cable activation can be achieved through use of a handle below a panel in the flight deck flooring on Boeing aircraft.

CHEMICAL OXYGEN SYSTEMS

The two primary types of chemical oxygen systems are the portable type, much like a portable carry-on gaseous oxygen cylinder, and the fully integrated supplementary oxygen system used as backup on pressurized aircraft in case of pressurization failure. (***Figure 17-23***)

This latter use of solid chemical oxygen generators is most common on airliners. The generators are stored in the overhead PSU attached to hoses and masks for every passenger on board the aircraft. When a depressurization occurs, or the flight crew activates a switch, a compartment door opens and the masks and hoses fall out in front of the passengers. The action of pulling the mask down to a usable position actuates an electric current, or ignition hammer, that ignites the oxygen candle and initiates the flow of oxygen. Typically, 10 to 20 minutes of oxygen is available for each user. This is calculated to be enough time for the aircraft to descend to a safe altitude for unassisted breathing. Chemical oxygen systems are unique in that they do not produce the oxygen until it is time to be used. This allows safer transportation of the oxygen supply with less maintenance.

Chemical oxygen-generating systems also require less space and weigh less than gaseous oxygen systems supplying the same number of people. Long runs of tubing, fittings, regulators, and other components are avoided, as are heavy gaseous oxygen storage cylinders. Each passenger row grouping has its own fully independent chemical oxygen generator. The generators, which often weigh less than a pound, are insulated and can burn completely without getting hot. The size of the orifice opening in the hose-attach nipples regulates the continuous flow of oxygen to the users.

ELECTRONIC PULSE DEMAND SYSTEMS

A recent development in general aviation oxygen systems is the electronic pulse demand oxygen delivery system (EDS). A small, portable EDS unit is made to connect between the oxygen source and the mask in a continuous-flow oxygen system. It delivers timed pulses of oxygen to the wearer on demand, saving oxygen normally lost during the hold and exhale segments of the breathing cycle. Advanced pressure sensing and processing allows the unit to deliver oxygen only when an inhalation starts. It can also sense differences in users' breathing cycles and physiologies and adjust the flow of oxygen accordingly. A built-in pressure-sensing device adjusts the amount of oxygen released as altitude changes. (***Figure 17-24***)

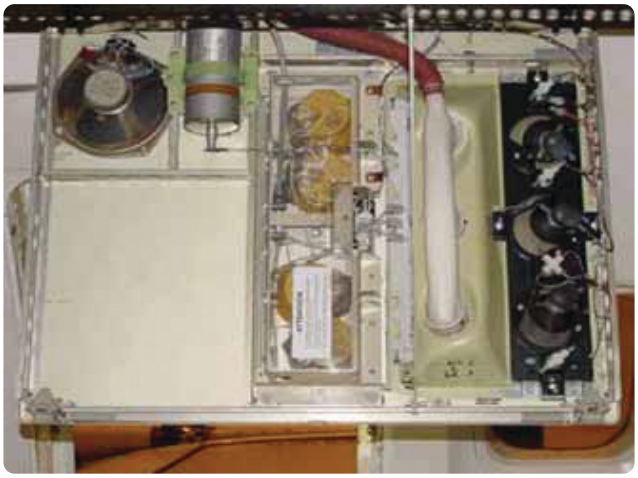


Figure 17-23. An oxygen generator mounted in place in an overhead passenger service unit of an air transport category aircraft.



Figure 17-24. A portable two-person electronic pulse-demand (EPD) oxygen regulating unit.

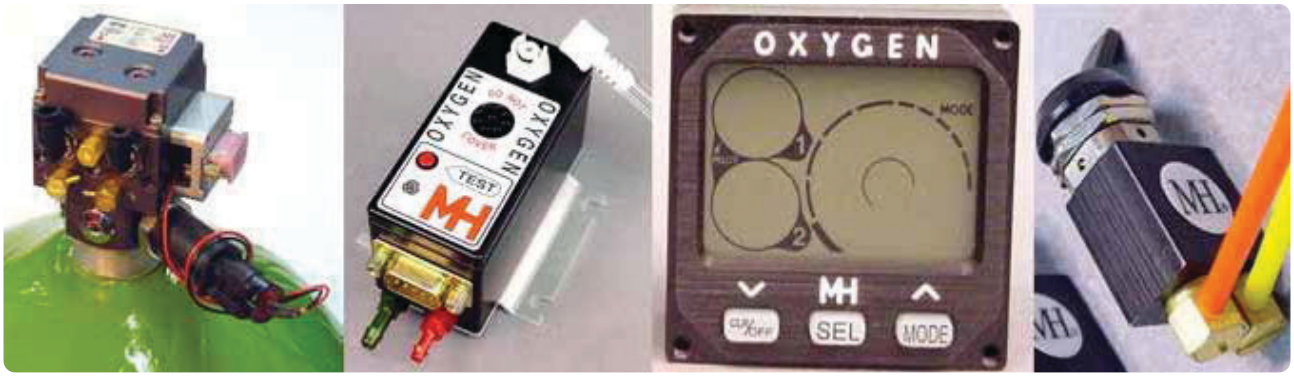


Figure 17-25. The key components of a built-in electronic pulse demand oxygen metering system: (A) electronic regulator, (B) oxygen station distributer unit, (C) command/display unit, (D) emergency bypass switch.

Permanently mounted EPD systems are also available. They typically integrate with an electronic valve/regulator on the oxygen cylinder and come with an emergency bypass switch to provide continuous-flow oxygen should the system malfunction. A liquid crystal display (LCD) monitor/control panel displays numerous system operating parameters and allows adjustments to the automatic settings. This type of electronic metering of oxygen has also been developed for passenger emergency oxygen use in airliners. (***Figure 17-25***)

LOX SYSTEMS

LOX systems are rarely used in civilian aviation. They may be encountered on former military aircraft now in the civilian fleet. As mentioned, the storage of LOX requires a special container system. The plumbing arrangement to convert the liquid to a usable gas is also unique. It basically consists of a controlled heat exchange assembly of tubing and valves. Overboard pressure relief is provided for excessive temperature situations. Once gaseous, the LOX system is the same as it is in any comparable gaseous oxygen delivery system. Use of pressure-demand regulators and masks is common. Consult the manufacturer's maintenance manual for further information if a LOX system is encountered.

OXYGEN SYSTEM DISTRIBUTION

Tubing and fittings make up most of the oxygen system distribution plumbing and connect the various components of the oxygen system. Most oxygen lines are metal in permanent installations. High-pressure lines are usually stainless steel. Tubing lines in the low-pressure parts of the oxygen system are typically aluminum.

Flexible plastic hosing is used to deliver oxygen to the masks; its use is increasing in permanent installations to save weight. Installed oxygen tubing is usually identified with color coded tape applied to each end of the tubing, and at specified intervals along its length. The tape coding consists of a green band overprinted with the words "BREATHING OXYGEN" and a black rectangular symbol overprinted on a white background border strip. (***Figure 17-26***)

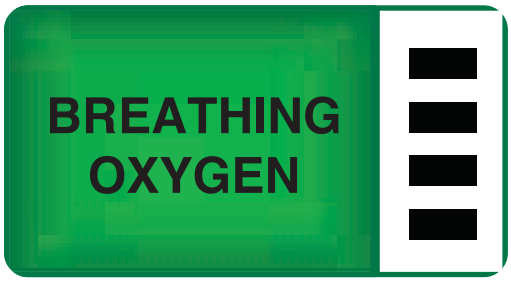


Figure 17-26. Color-coded tape used to identify oxygen tubing.

Tubing-to-tubing fittings in oxygen systems are often designed with straight threads to receive flared tube connections. Tubing-to-component fittings usually have straight threads on the tubing end and external pipe threads (tapered) on the other end for attachment to the component. The fittings are typically made of the same material as the tubing (i.e., aluminum or steel). Flared and flareless fittings are both used, depending on the system. Five types of valves are commonly found in high-pressure gaseous oxygen systems: filler, check, shutoff, pressure reducer, and pressure relief. They

function as they would in any other system with one exception: oxygen system shutoff valves are specifically designed to open slowly.

The ignition point for any substances is lower in pure oxygen than it is in air. When high-pressure oxygen is allowed to rush into a low-pressure area, its velocity could reach the speed of sound. If it encounters an obstruction (a valve seat, an elbow, a piece of contaminant, etc.), the oxygen compresses. With this compression, known as adiabatic compression (since it builds so quickly no heat is lost to its surroundings), comes high temperature. Under pressure, this high temperature exceeds the ignition point of the material the oxygen encounters and a fire or explosion results.

A stainless steel line, for example, would not normally burn and is used for carrying numerous fluids under high pressure. But under high pressure and temperature in the presence of 100 percent oxygen, even stainless steel can ignite. To combat this issue, all oxygen shutoff valves are slow, opening valves designed to decrease velocity. (***Figure 17-27***)

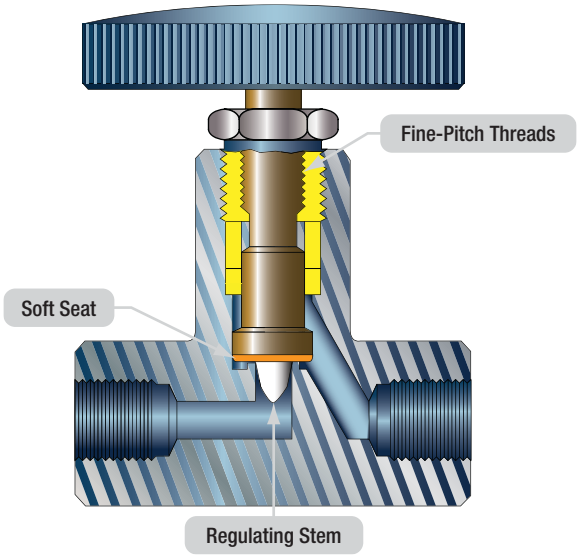


Figure 17-27. This high-pressure oxygen system shutoff valve has fine- pitch threads and a regulating stem to slow the flow of oxygen through the valve. A soft valve seat is also included to assure the valve closes completely.

Additionally, technicians should always open all oxygen valves slowly. Keeping oxygen from rushing into a low pressure area should be a major concern when working with high-pressure gaseous oxygen systems.

OXYGEN INDICATION AND WARNINGS

Oxygen cylinder valves and high-pressure systems are often provided with a relief valve should the desired pressure be exceeded. Often, the valve is ported to an indicating or blowout disk. This is located in a conspicuous place, such as the fuselage skin, where it can be seen during walk-around inspection. Most blowout disks are green. The absence of the green disk indicates the relief valve has opened, and the cause should be investigated before flight. (***Figure 17-28***) In addition to the blowout disc, various indications and warning are included in oxygen systems.

Flow indicators, or flow meters, are common in all oxygen systems. They usually consist of a lightweight object, or apparatus, that is moved by the oxygen stream. When flow exists, this movement signals the user in some way. (***Figure 17-29***)

Many flow meters in continuous-flow oxygen systems also double as flow rate adjusters. Needle valves fitted into the flow indicator housing can fine-adjust the oxygen delivery rate. Demand-flow oxygen systems usually have flow indicators built into the individual regulators at each user station. Some contain a blinking device that activates when the user inhales and oxygen is delivered. Others move a colored pith object into a window. Regardless, flow indicators provide a quick verification that an oxygen system is functioning. (***Figure 17-29***)



Figure 17-28. An oxygen blowout plug on the side of the fuselage indicates when pressure relief has occurred and should be investigated.



Figure 17-29. Different flow indicators are used to provide verification that the oxygen system is functioning: continuous-flow, in-line (left); continuous-flow, in-line with valve adjuster (center); and old style demand flow (right).

Different types of independent flow indicators are used to provide verification that the oxygen system is functioning. Many demand-flow system flow indicators are built into the individual oxygen regulators as was seen ***Figure 17-19***.

The amount of pressure in the supply tanks of an airliner using gaseous oxygen is an indication of the quantity of oxygen on board. Most tanks contain a direct reading gauge on the tank outlet. Maintenance personnel use this gauge. The flight crew relies on a transducer that sends an electric signal to a gauge on the flight deck as was seen in ***Figure 17-16***. On modern aircraft, transducer output is sent to the aircraft information management system for display of pressure on a multifunctional display screen. The system status page must be selected on the display select panel. ***(Figure 17-30***)

Note that some aviators flying in smaller aircraft may use a portable pulse oximeters. This device measures the oxygen saturation level of the blood. With this information, adjustments to the oxygen flow rates of onboard oxygen equipment can be made to prevent hypoxia. ***Figure 17-31*** shows an oximeter into which a finger is inserted to measure oxygen saturation of the blood in percentage. Heart rate is also displayed.

OXYGEN SYSTEM MAINTENANCE

All oxygen systems require servicing and maintenance. Various procedures and requirements are used to perform common maintenance functions are covered in this section.

LEAK TESTING GASEOUS OXYGEN SYSTEMS

Leaks in a continuous-flow oxygen system may be difficult to detect because the system is open at the user end. Blocking the flow of oxygen allows pressure to build and leak check procedures can be followed that are similar to those used in the high pressure sections of the systems. Detection of leaks should be performed with oxygen-safe leak check fluid. This is a soapy liquid free from elements that might react with pure oxygen or contaminate the system. As with leak detection on an inflated tire or tube assembly, the oxygen leak detection solution is applied to the outside of fittings and mating surfaces. The formation of bubbles indicates a leak. (***Figure 17-32***)

Careful assembly of oxygen components and fittings without over-tightening or under-tightening is required. If a leak is found at a fitting, it should be checked for the proper torque. Tightening may not always stop the leak. If the fitting is torqued properly and a leak still exists, pressure must be released from the system and the fitting must be examined for flaws or contamination. If necessary, the fitting must be replaced. All system components, lines, and fittings must be replaced with the proper parts, which should be cleaned and inspected thoroughly before installation. Follow the manufacturer's instructions and repeat the leak check when completed.



Status Display

Figure 17-30. The display select panel and oxygen system status display on a Boeing 777 multifunctional display.



Figure 17-31. A portable pulse-type oximeter displays percentage of oxygen saturation of the blood and heart rate. Pilots can adjust oxygen supply levels to maintain saturation and avoid hypoxia.

Use caution when maintaining the high pressure portion of a gaseous oxygen system. An open tank valve pressurizes the lines and components with up to 1 850 pounds per square inch (psi) of oxygen. Identify the high-pressure section of the system as that portion upstream of the reducer or regulator that has stainless steel tubing. Note that no attempt should be made to tighten a leaky oxygen fitting while the system is charging. The oxygen supply should be isolated in the cylinder and the system depressurized to reduce the consequences of a spark or to minimize spillage and injury should a complete fitting failure occur.

DRAINING AN OXYGEN SYSTEM

The biggest factor in draining an oxygen system is safety. The oxygen must be released into the atmosphere without causing a fire, explosion, or hazard. Draining outside is highly recommended. The exact method of draining can vary. The basic procedure involves establishing a continuous flow in a safe area until the system is empty.

If the cylinder valve is operative, close the valve to isolate the oxygen supply in the cylinder. All that remains is to empty the lines and components. This can be done without disassembling the system by letting oxygen flow from the delivery point(s). If the environment is safe to receive the oxygen, positioning a demand-flow regulator



Figure 17-32. Oxygen system leak check solution.

to emergency delivers a continuous flow of oxygen to the mask when plugged in. Hang the mask(s) out of a window while the system drains. Plug in all mask(s) to allow oxygen to drain from a continuous-flow oxygen system. Systems without check valves can be drained by opening the refill valve.

PURGING AN OXYGEN SYSTEM

The inside of an oxygen system becomes completely saturated with oxygen during use. This is desirable to deliver clean, odor-free oxygen to the users and to prevent corrosion caused by contamination. An oxygen system needs to be purged if it has been opened or depleted for more than 2 hours, or if it is suspected that the system has been contaminated. Purging is accomplished to evacuate contaminants and to restore oxygen saturation to the inside of the system.

The main cause of contamination in an oxygen system is moisture. In very cold weather, the small amount of moisture contained in the breathing oxygen can condense. With repeated charging, a significant amount of moisture may collect. Additionally, systems that are opened contain the moisture from the air that has entered. Damp charging equipment, or poor refill procedures, can also introduce water into the system. Always follow manufacturer's instructions when performing maintenance, refilling, or purging an oxygen system.

Cumulative condensation in an oxygen system cannot be entirely avoided. Purging is needed periodically. The procedure for purging may vary somewhat with each aircraft model. Generally speaking, oxygen is run through a sound oxygen system for a number of minutes at a given pressure to perform the purging. This can be as little as 10 minutes at normal delivery pressure. Other systems may require up to 30 minutes of flow at an elevated pressure. Regardless, the removal of contaminants and the re-saturation of the inside of the system with oxygen is the basis for purging. It is acceptable to use nitrogen, or dry air, to blow through lines and components when performing maintenance. However, a final purging with pure oxygen is required before the system is serviceable for use.

It is important to ensure storage cylinders are refilled if they are used during the purging process. Be certain that there are no open lines and all safety caps are installed before returning the aircraft to service.

INSPECTION OF MASKS AND HOSES

The wide varieties of oxygen masks used in aviation require periodic inspection. Mask and hose integrity ensure effective delivery of oxygen to the user when it is needed. Sometimes this is in an emergency situation. Leaks, holes, and tears are not acceptable. Most discrepancies of this type are remedied by replacement of the damaged unit.

Some continuous-flow masks are designed for disposal after use. Be sure there is a mask for each potential user on board the aircraft. Masks designed to be reused should be clean, as well as functional. This reduces the danger of infection and prolongs the life of the mask. Various mild cleaners and antiseptics that are free of petroleum products can be used. A supply of individually wrapped alcohol swabs are often kept in the cockpit.

Built-in microphones should be operational. Donning straps and fittings should be in good condition and function so that the mask is held firm to the user's face. Note that the diameter of mask hoses in a continuous-flow system is quite a bit smaller than those used in a demand-flow system. This is because the inside diameter of the hose aids in controlling flow rate. Masks for each kind of system are made to only connect to the proper hose.

Smoke masks are required on transport aircraft and are used on some other aircraft as well. These cover the eyes, as well as the user's nose and mouth. Smoke masks are usually available within easy grasp of the crew members. They are used when the situation in the cockpit demands the increased level of protection offered. Smoke mask hoses plug into demand regulators in the same port used for regular demand type masks and operate in the same manner. Most include a built-in microphone. (***Figure 17-33***)

Some portable oxygen systems are also fitted with smoke masks.

REPLACEMENT OF TUBING, VALVES, AND FITTINGS

The replacement of aircraft oxygen system tubing, valves, and fittings is similar to the replacement of the same components in other aircraft systems. There is, however, an added emphasis on cleanliness and compatible sealant use. Any oxygen system component should be cleaned thoroughly before installation. Often tubing comes with leftover residue from the bending or flaring processes. Cleaning should be accomplished with non-petroleum-based cleansers. Trichlorethylene, acetone, and similar cleaners can be used to flush new

tubing. Tubing should be blown or baked dry before installation. Follow the manufacturer's procedures for cleaning oxygen system components.

Some oxygen components make use of tapered pipe fittings. This type of connection is usually sealed with the application of thread lubricant/sealant. Typical thread sealers are petroleum based and should not be used; only oxygen compatible thread lubricant/sealers should be used. Alternatively, Teflon™ tape is also used on oxygen pipe fitting connections. Be sure to begin wrapping the Teflon™ tape at least two threads from the end of the fitting. This prevents any tape from coming loose and entering the oxygen system.

OXYGEN SYSTEM SAFETY

Precautions must be observed when working with or around pure oxygen. It readily combines with other substances, some in a violent and explosive manner. As mentioned, it is extremely important to keep distance between pure oxygen and petroleum products. When allowed to combine, an explosion can result. Additionally, there are a variety of inspection and maintenance practices that should be followed to ensure safety when working with oxygen and oxygen systems. Care should be used and, as much as possible, maintenance should be done outside.

When working on an oxygen system, it is essential that the warnings and precautions given in the aircraft maintenance manual be carefully observed. Before any work is attempted, an adequate fire extinguisher should be on hand. Cordon off the area and post NO SMOKING placards. Ensure that all tools and servicing equipment are clean and avoid power on checks and use of the aircraft electrical system.

When working around oxygen and oxygen systems, cleanliness enhances safety. Clean, grease-free hands, clothes, and tools are essential. A good practice is to use only tools dedicated for work on oxygen systems. There should be absolutely no smoking or open flames within a minimum of 50 feet of the work area. Always use protective caps and plugs when working with oxygen cylinders, system components, or plumbing. Do not use any kind of adhesive tape. Oxygen cylinders should be stored in a designated, cool, ventilated area in the hanger away from petroleum products or heat sources.

Oxygen system maintenance should not be accomplished until the valve on the oxygen supply cylinder is closed and pressure is released from the system. Fittings should be unscrewed slowly to allow any residual pressure to dissipate. All oxygen lines should be marked and should have at least 2 inches of clearance from moving parts, electrical wiring, and all fluid lines. Adequate clearance must also be provided from hot ducts and other sources that might heat the oxygen. A pressure and leak check must be performed each time the system is opened for maintenance. Do not use any lubricants, sealers, cleaners, etc., unless specifically approved for oxygen system use.



Figure 17-33. Smoke masks cover the eyes as well as the nose and mouth of the user.

Question: 17-1
The atmosphere of the earth contains about _____ % oxygen.

Question: 17-2
What are two ways the negative effects of reduced atmospheric pressure at flight altitudes forcing less oxygen into the blood can be overcome?

Question: 17-3
Pure oxygen combines violently with _____, which creates a significant hazard when handling these materials in close proximity to each other.

Question: 17-4
To ensure continued serviceability, oxygen cylinders must be _____ periodically.

Question: 17-5
When _____ is ignited, it produces oxygen as it burns.

Question: 17-6
Built-in oxygen systems contain _____ designed to refill gaseous oxygen cylinders while they are in place.

Question: 17-7
Question text. _____ has a significant effect on the pressure of gaseous oxygen.

Question: 17-8
Transport category aircraft use a more elaborate, built-in _____ oxygen system as a backup system to cabin pressurization.

ANSWERS

Answer: 17-1
21.

Answer: 17-2
Increase the pressure of the oxygen or,
Increase the quantity of oxygen in the air mixture.

Answer: 17-3
petroleum products.

Answer: 17-4
hydrostatically tested.

Answer: 17-5
sodium chlorate.

Answer: 17-6
plumbing.

Answer: 17-7
Temperature.

Answer: 17-8
gaseous.

Question: 17-9
Built-in _____ gaseous oxygen systems accomplish the final f low rate to an individual user station through the use of a calibrated orifice in each mask.

Question: 17-10
When oxygen is delivered only as the user inhales, it is known as a _____ system.

Question: 17-11
Two types of oxygen regulators for flight crew use on jet transport aircraft are _____ and _____.

Question: 17-12
High-pressure oxygen distribution lines are usually made from _____.

Question: 17-13
An oxygen _____ usually consists of a lightweight object, or apparatus, that is moved by the oxygen stream.

Question: 17-14
To purge an oxygen system, pure _____ is used.

Question: 17-15
Oxygen fittings should be unthreaded _____ to allow any residual pressure to dissipate.

Question: 17-16
Detection of leaks should be performed with _____.

ANSWERS

Answer: 17-9
continuous-flow.

Answer: 17-10
demand-flow.

Answer: 17-11
demand flow.
diluter demand.
pressure demand.

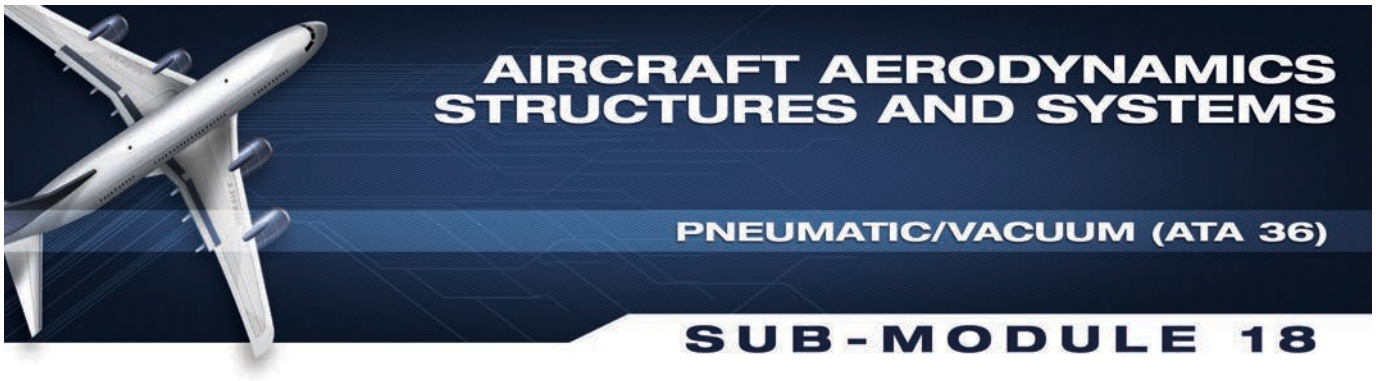
Answer: 17-12
stainless steel.

Answer: 17-13
flowmeter (flow indicator).

Answer: 17-14
oxygen.

Answer: 17-15
slowly.

Answer: 17-16
oxygen-safe leak check fluid.



PART-66 SYLLABUS **LEVELS**

CERTIFICATION CATEGORY → **B2**

Sub-Module 18

PNEUMATIC/VACUUM (ATA 36)

Knowledge Requirements

13.18 - Pneumatic/Vacuum (ATA 36)

System lay-out;	2
Sources: engine/APU, compressors, reservoirs, ground supply;	2
Pressure control;	3
Distribution;	1
Indications and warnings;	3
Interfaces with other systems.	3

PNEUMATIC / VACUUM (ATA 36)

VACUUM SYSTEMS

Large transport aircraft powered by reciprocating engines may contain original instrument technology that includes gyroscopic instruments powered by a vacuum system.

SYSTEM LAYOUT

Twin-engine aircraft vacuum systems contain an engine driven vacuum pump on each engine. The associated lines and components for each pump are isolated from each other and act as two independent vacuum systems. The vacuum lines are routed from each vacuum pump through a vacuum relief valve and through a check valve to the vacuum four-way selector valve. The four-way valve permits either pump to supply a vacuum manifold.

From the manifold, flexible hoses connect the vacuum operated instruments into the system. To reduce the vacuum for the turn and bank indicators, needle valves are included in both lines to these units. Lines to the artificial horizons and the directional gyro receive full vacuum. From the instruments, lines are routed to the vacuum gauge through a turn and bank selector valve. This valve has three positions: main, left turn and bank (T&B), and right T&B. In the main position, the vacuum gauge indicates the vacuum in the lines of the artificial horizons and directional gyro. In the other positions, the lower value of vacuum for the turn and bank indicators are displayed.

A schematic of this type of twin-engine aircraft vacuum system is shown in ***Figure 18-1***. Note the following components: two engine-driven pumps, two vacuum relief valves, two flapper type check valves, a vacuum manifold, a vacuum restrictor for each turn and bank indicator, an engine four-way selector valve, one vacuum gauge, and a turn-and-bank selector valve. Not shown are system and individual instrument filters. A drain line may also be installed at the low point in the system.

COCKPIT

Much of the vacuum system is located behind the instrument panel. Mounted in the panel are the 4-way selector valve, turn and bank selector valve and a system vacuum gauge. Tubing and connections for the gyroscopic instruments are behind the instrument panel.

CABIN

There are no vacuum system components in the passenger cabin.

PNEUMATIC SYSTEMS

Pneumatic and hydraulic systems are similar in that they use confined fluids. Since liquids and gases flow, they are both considered fluids; however, there is a great difference in the characteristics of the two. Liquids are practically incompressible; a quart of water still occupies about a quart of space regardless of how hard it is compressed. But gases are highly compressible; a quart of air can be compressed into a thimbleful of space. In spite of this difference, gases and liquids are both fluids which are confined and made to transmit power. The type of unit used to provide pressurized air for pneumatic systems is determined by the system's air pressure requirements.

HIGH PRESSURE SYSTEMS

In the past, some aircraft manufacturers equipped their aircraft with a high pressure pneumatic system (3 000 psi). An aircraft that utilizes this type of system is the Fokker F27. Such systems operate a great deal like hydraulic systems, except they employ air instead of a liquid for transmitting power.

High pressure pneumatic systems are sometimes used for:

- Brakes
- Opening and closing dooirs
- Driving hydraulic pumps, alternators, starters, water injection pumps, etc.
- Operating emergency devices such as brakes and landing gear.

SYSTEM LAYOUT

Many high pressure pneumatic systems are installed for one-time emergency or back-up use and are completely discharged when used. They use pressurized air or nitrogen pressurized containers with no on-board means provided to re-pressurize the system once deployed. Other high pressure pneumatic systems use pressurized containers that are re-charged during flight through the action of compressors installed in the system. This type of installation allows the pneumatic system to operate components repeatedly rather than just once in a manner similar to a hydraulic system.

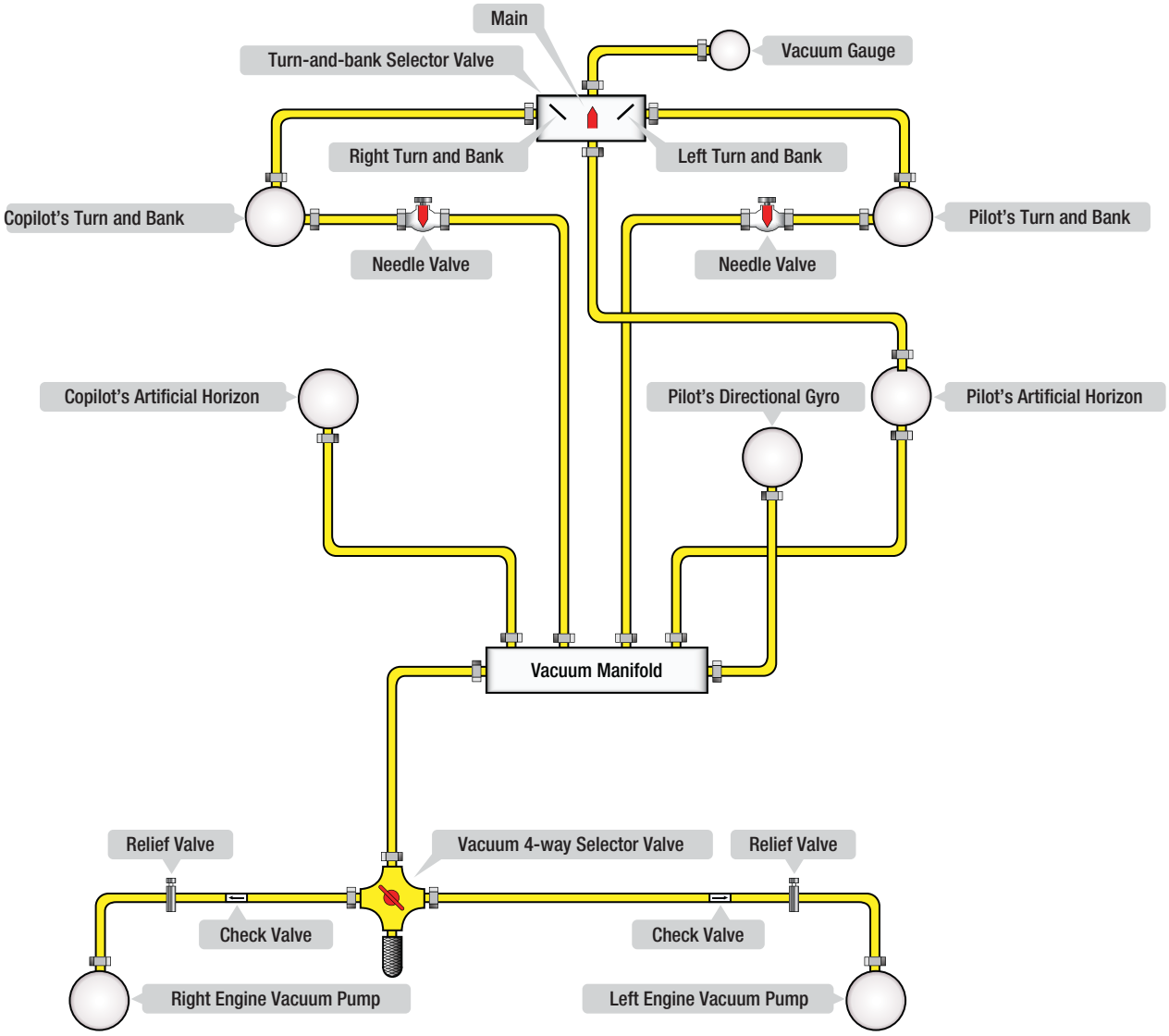


Figure 18-1. An example of a twin-engine instrument vacuum system.

Figure 18-2 shows a typical layout of a high pressure pneumatic system on a twin reciprocating engine aircraft equipped with on-board.

SOURCES

Sources for high pressure pneumatic systems include engine-driven and other on-board compressors, ground air and ground nitrogen sources. As stated, some aircraft employ permanently installed air compressors which recharge air bottles whenever pressure is used for operation of a unit. Several types of compressors are used for this purpose. Some have two stages of compression, while others have three, depending on the maximum desired operating pressure. Details on compressor operation are found in the aircraft maintenance manual.

Alternately, air and nitrogen storage containers for pneumatic systems are filled on the ground with either a ground-based compressor or a high pressure bottle transfer for nitrogen.

STORAGE

For high-pressure systems, air is usually stored in metal bottles (pressures ranging from 1 000 to 3 000 psi, depending on the particular system). The high pressure storage cylinder is typically a light weight steel-constructed reservoir. The date of manufacture and safe working pressure should be on the reservoir as well as a date stamped for the performance of the last hydrostatic test. It is common practice for these high pressure containers to be inspected often and removed periodically for hydrostatic checks. A standpipe is commonly used at the discharge port to prevent any water that has collected

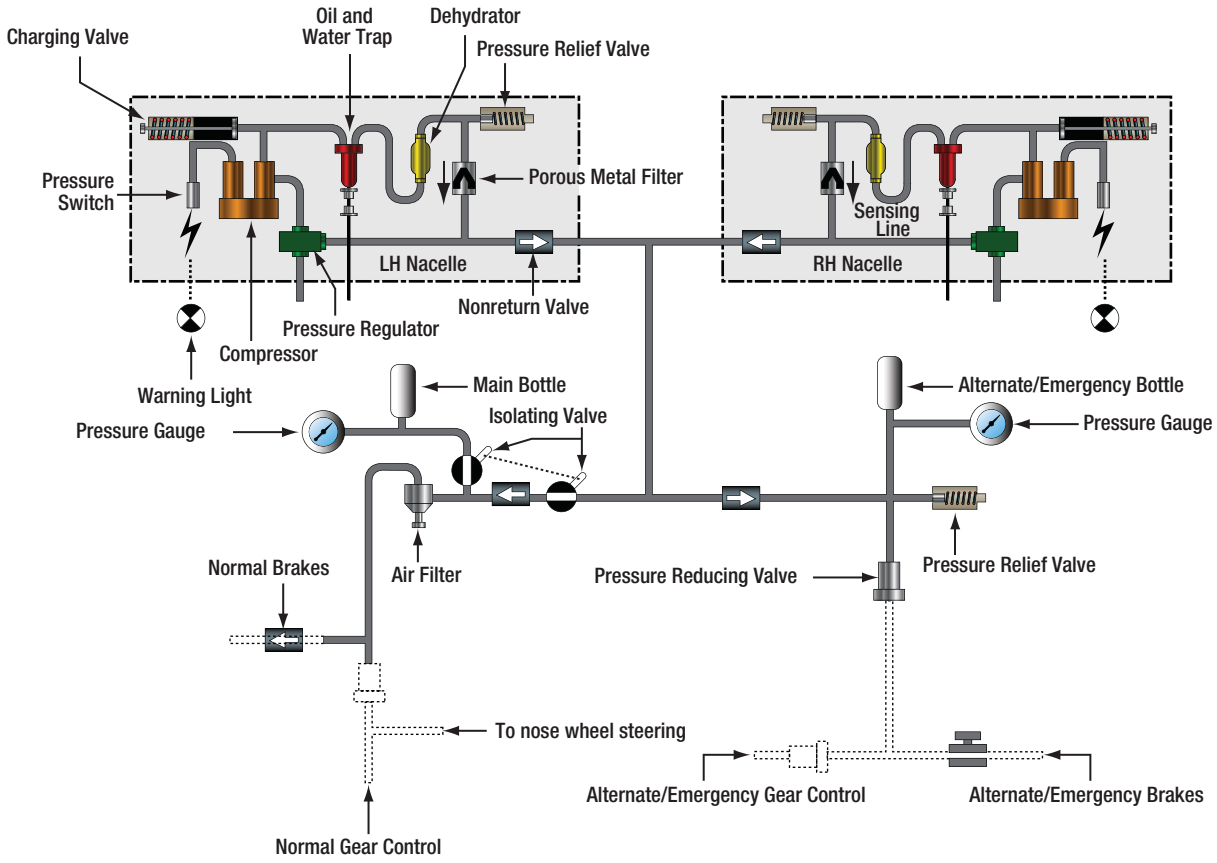


Figure 18-2. High-pressure pneumatic system.

inside the container from entering the system. Air flowing out of the container must go through the end of the standpipe which is elevated above any conceivable water level. This type of container is used in both one-time and multi-deployment systems.

CHARGING

Charging of high pressure bottles is done with either an on-board compressor or a ground source. The typical high-pressure storage bottle has two ports, one of which is equipped with a charging valve. A ground-operated compressor or air bottle can be connected to this valve to add air. Nitrogen may also be introduce through this valve.

For on-board charging, the charging valve is plumbed to the compressor outlet. The other valve on a typical high pressure pneumatic reservoir is a control valve. It acts as a shutoff valve, keeping air trapped inside the bottle until the system is operated. This valve must be opened if fitted when installed in a chargeable system. Reservoir contents stays held in the bottle with system pressure. A pressure switch is used for flight deck warnings.

DISTRIBUTION

Pneumatic power is distributed through high pressure steel or stainless steel lines. The use of check valves is common to prevent back flow. The lines are routed in the same manner as hydraulic lines to reach the components. In systems that operate one time and emergency systems, a shuttle valve is often used to close off the normal system flow and allow flow of high pressure pneumatic air to operate the component.

SUPPLY REGULATION

A pressure regulator maintains system pressure with a relief valve to limit pressure in case of regulator failure. Check valves are used to prevent back flow to the compressor. In addition to the use of a selector valve or control valve to direct the air to the portion of the system thro ugh which it must be distributed, isolation valves are often installed in the distribution system to isolate working components from those that are inoperative or to isolate part of the system that has a leak. Note that all components in a high pressure pneumatic system do not necessarily operate at full system pressure. Pressure reducing valves are used to lower the system pressure to that require by a

particular component or sub system. Restrictors and variable restrictors are used to control the speed of the component(s) operated by pneumatic. (***Figure 18-3***)

The few high pressure pneumatic systems on aircraft that the technician encounters are either one-time use or multi-use or both. ***Figure 18-4*** illustrates part of a pneumatic system that uses a rechargeable system for normal operation of gear extension and retraction as well as brake operation.

For emergency brake application, a completely redundant distribution system supplies high pressure air from a reservoir (not shown) independent of the normal system.

EMERGENCY BACK-UP SYSTEMS / PNEUDRAULICS

Many aircraft use a high-pressure pneumatic back-up source of power to extend the landing gear or actuate the brakes if the main hydraulic braking system fails. High pressure nitrogen is not directly used to actuate the landing gear actuators or brake units but, instead, it applies the pressurized nitrogen to move hydraulic fluid to the actuator. This process is called pneudraulics. The following paragraph discusses the components and operation of an emergency pneumatic landing gear extension system used on a business jet. (***Figure 18-5***)

NITROGEN BOTTLES

Nitrogen used for emergency landing gear extension is stored in two bottles, one bottle located on each side of the nose wheel well. Nitrogen from the bottles is released by actuation of an outlet valve. Once depleted, the bottles must be recharged by maintenance personnel.

GEAR EMERGENCY EXTENSION CABLE AND HANDLE

The outlet valve is connected to a cable and handle assembly. The handle is located on the side of the copilot's console and is labeled EMER LDG GEAR. Pulling the handle fully upward opens the outlet valve, releasing compressed nitrogen into the landing gear extension system. Pushing the handle fully downward closes the outlet valve and allows any nitrogen present in the emergency landing gear extension system to be vented overboard. The venting process takes approximately 30 seconds.

DUMP VALVE

As compressed nitrogen is released to the landing gear selector/dump valve during emergency extension, the pneudraulic pressure actuates the dump valve portion of the landing gear selector/dump valve to isolate the landing gear system from the remainder of hydraulic system.

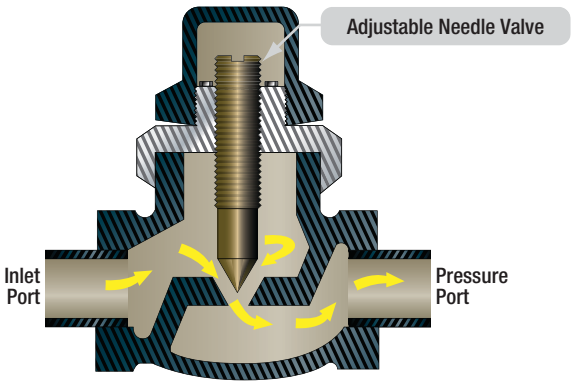


Figure 18-3. Variable pneumatic restrictor.

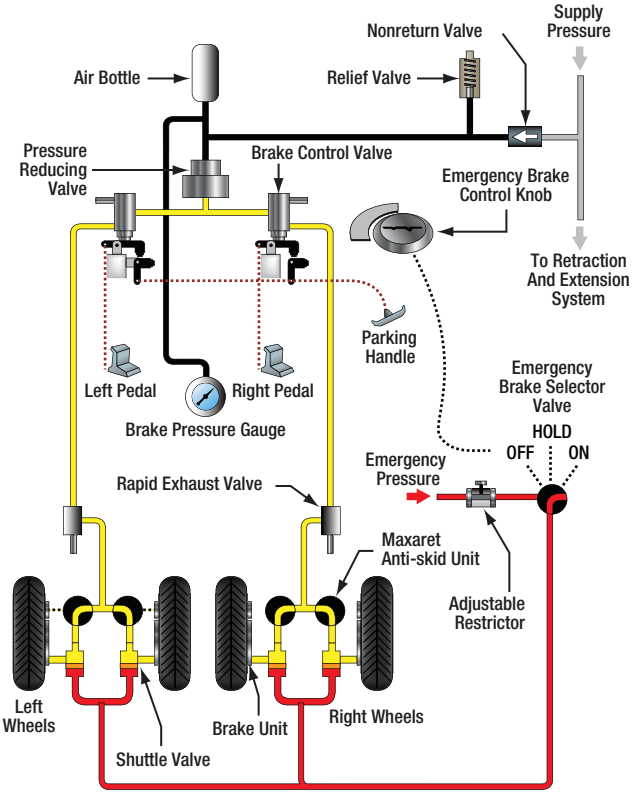


Figure 18-4. Normal rechargeable and emergency non-rechargeable pneumatic brake systems on the same aircraft.

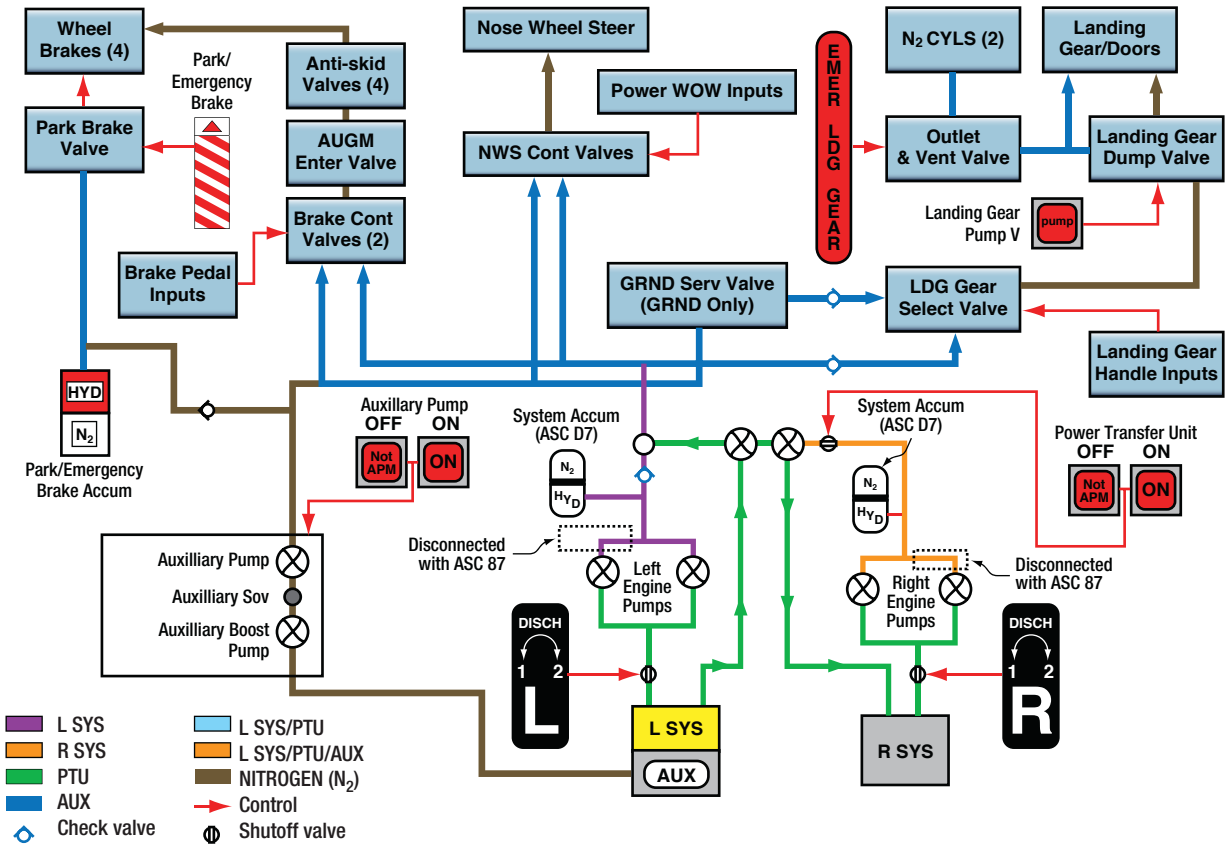


Figure 18-5. Pneumatic emergency landing gear extension system.

When activated, a blue DUMP legend is illuminated on the LDG GR DUMP V switch, located on the flight deck overhead panel. A dump valve reset switch is used to reset the dump valve after the system has been used and serviced. Emergency Extension Sequence:

1. Landing gear handle is placed in the DOWN position.
2. Red light in the landing gear control handle is illuminated.
3. EMER LDG GEAR handle is pulled fully outward.
4. Compressed nitrogen is released to the landing gear selector/dump valve.
5. Pneudraulic pressure actuates the dump valve portion of the landing gear selector/dump valve.
6. Blue DUMP legend is illuminated on the LDG GR DUMP switch.
7. Landing gear system is isolated from the remainder of hydraulic system.
8. Pneudraulic pressure is routed to the OPEN side of the landing gear door actuators, the UNLOCK side of the landing gear uplock actuators, and the EXTEND side of the main landing gear sidebrace actuators and nose landing gear extend/retract actuator.
9. Landing gear doors open.
10. Uplock actuators unlock.
11. Landing gear extends down and locks.
12. Three green DOWN AND LOCKED lights on the landing gear control panel are illuminated.
13. Landing gear doors remain open.

HIGH PRESSURE PNEUMATIC POWER SYSTEM MAINTENANCE

Maintenance of high pressure pneumatic power system consists of servicing, troubleshooting, removal, and installation of components, and operational testing.

The air compressor's lubricating oil level should be checked daily in accordance with the applicable manufacturer's instructions. The oil level is indicated by means of a sight gauge or dipstick. The pneumatic system should be purged periodically to remove the contamination, moisture, or oil from the components and lines. Purging the system is accomplished by pressurizing it and removing the plumbing from various components throughout the system.

MEDIUM-PRESSURE PNEUMATIC SYSTEMS

SYSTEM LAYOUT

Medium pressure pneumatic systems on large passenger aircraft are typically designed around the sources for pneumatic air that feed a common manifold. Each engine contains an independent bleed air subsystem that is designed to extract and regulate pneumatic bleed air from the engine. It is then forwarded to the pneumatic manifold for use. The pneumatic manifold contains the control valves that are operated to supply the systems that require pneumatic power. An isolation valve separates the pneumatic manifold from each engine bleed air supply and regulation subsystem so as to be able to turn the supply ON and OFF from that engine. The APU is similarly designed although the APU may turn a dedicated load compressor to supply the air rather than tapping bleed air off the compressor section of the engine.

A pneumatic power supply cart provides already regulated air pressure. When it is used to supply the manifold, the aircraft engines are not operated. A ground pneumatic air supply adapter with check valve is located directly in the pneumatic manifold. Closing the engine and APU isolation valves isolates the ground air supply. The supply cart must be powered down to de-energize the pneumatic manifold and remove the hose.

SOURCES

A medium-pressure pneumatic system (35-150 psi) does not include an air bottle/ storage reservoir. Instead, it draws air from the compressor section of a turbine engine. This is known as bleed air and is used to provide pneumatic power for engine starts, engine de-icing, wing de-icing, air conditioning and more. In some cases, it provides hydraulic power to the aircraft systems (if the hydraulic system is equipped with an air-driven hydraulic pump). Engine bleed air is also used to pressurize the aircraft's hydraulic reservoirs, anti-ice the TAT probe and other applications specific to particular aircraft.

Ground sources of pneumatic air also are used. Fixed and portable cart-type units containing engine-driven air supply compressors are connected into the pneumatic manifold to power the pneumatic system without running the engines. A ground air supply connector and check valve is provided in the manifold for the duct diameter sized hose from the ground source.

STORAGE

Bleed air pneumatic systems normally do not store pneumatic air in any particular container like the reservoir bottles of a high pressure pneumatic systems. Each turbine engine and the APU supply the bleed air. A shutoff or regulating and shutoff valve is typically located between the engine bleed air tap-offs and the pneumatic ducting that makes up the pneumatic manifold. A shutoff type valve is also used to control the flow of pneumatic air from the APU.

Thus, the pneumatic manifold may be considered a storage location. It is located downstream of the pneumatic shutoff valves from the engines and APU. Control valves allow pneumatic air to be routed from the manifold into pneumatically powered components such as engine starters, pneumatically driven hydraulic pumps, and into the wing anti-ice ducts and air conditioning packages.

PRESSURE CONTROL

Airliner pneumatic system pressure control begins with control of engine compressor bleed air. Intermediate stage compressor bleed air normally supplies the bulk of the pneumatic system demand. However, in times of high demand or reduced engine throttle, a second, and sometimes a third tap off of high-stage compressor bleed air is combined with intermediate-stage air to main sufficient volume for operating pneumatic system component demands. Various pressure regulating and sourcing valves are used to deliver the optimum volume of air into the pneumatic manifold at any given time.

On the most modern aircraft, regulation is maintained electronically. Digital data buses supply inputs to central pneumatic system control computers. The computers set the position of the various valves in the system to meet demand. The dominant use of pneumatic air is cabin air conditioning.

On a Boeing 777, therefore, the computer that regulates the pneumatic system is called the air supply cabin pressure controller (ASCPC). There are two ASCPCs located in the main equipment center. They use data about the air sources and air end user components and sub-systems to select the regulating valve positions. Environmental conditions, engine, APU and airframe

status conditions as well as flight status conditions are all factors considered for control. The data for automatic operation comes from these:

- Airfoil and cowl ice protection system (ACIPS)
- Autopilot flight director system (AFDS)
- Airplane information management system (AIMS)
- Air supply cabin pressure controllers (ASCPC)
- Auxiliary power unit controller (APUC)
- Cabin temperature controller (CTC)
- Duct leak and overheat detection (DLODS)
- ECS miscellaneous card (ECSMC)
- Electronic engine control (EEC)
- Electrical engine control (EEC)
- Electrical load management system (ELMS)
- Flap slat electronics unit (FSEU)
- Hydraulic interface module (HYDIM) cards
- Overhead panel ARINC 629 system (OPAS)
- Warning electronic unit (WEU)
- Weight on wheels (WOW) cards.

Figure 18-6 illustrates the pneumatic system on a Boeing 777. It shows how the data buses provide a flow of information to and from the pneumatic system, its components and the myriad of other related systems on the aircraft listed above. Older airliners have fewer inputs to regulating and shutoff valves than modern aircraft. Most valves modulate with a handful or fewer inputs from sensors, sensing lines, related valves, components and pneumatic manifold pressure. Systems are designed so that pneumatic manifold pressure remains relatively constant despite the demands of pneumatic components. Cabin air conditioning, in particular requires a relatively large continuous flow of pneumatic air. Reduced pressure in the pneumatic manifold is fed back to the mechanism that controls the valve positions of regulating and shutoff valves for various compressor bleed air stages. Typically these are positioned to maintain volume requirements by manipulating valve positions to maintain a set pressure in the manifold. On this aircraft, the 13th stage modulating and shutoff valve supplies any addition air required to the pneumatic manifold.

Pressure relief valves are provided to protect pneumatic ducts from excessive pressure. Typically, a relief valve is located in each section of the pneumatic manifold separated by an isolation or shutoff valve. The APU supply duct may also have a relief valve. Relief settings on some Boeing 737's are in the 80-110 psi range.

Without the benefit of computer control found on the most modern aircraft, many aircraft control and operate bleed air pneumatic system pressure regulating valves solely with pneumatic pressure. No electricity is needed. Pneumatic pressure in the pneumatic manifold is routed to a pressure regulator that also receives pneumatic/bleed air input from other locations. Pneumatic temperature inputs are also used. The regulator's internal mechanism balances the air inputs as required by system demands. This is largely done with springs and chambers with diaphragms for comparing pressures. Pneumatic lines then run from the regulator output to pressure regulating and shutoff valve mechanism(s) which modulate valve position accordingly using the pressurized air signal from the regulator.

DISTRIBUTION

The engine bleed air distribution system interconnects the engine bleeds of the engines and APU and contains the necessary valves to shut off bleed air at each engine and isolate various ducts. The medium pressure pneumatic system is generally characterized by the use of high volume ducting. The pneumatic manifold, which is itself ducting, distributes the air through the use of control valves leading to various pneumatic systems components and sub-systems.

The ducts into which the control valves direct the air are of various sizes. High volume ducting is used for engine starting and wing anti-ice and air conditioning. Smaller diameter (low volume) ducting is used for many other components such as windscreen anti fogging and total air temp gauge anti-ice. The temperature of pneumatic air is controlled within an acceptable range.

Air conditioning systems accepted air from the pneumatic manifold that is too hot to be released directly into the cabin. Air conditioning packages use heat exchangers and well as an air cycle machine to adjust the temperature of the pneumatic air so that it is comfortable in the cabin. Wing anti-ice and engine starter air temperatures are not as critical and make use of pneumatic manifold air without further temperature adjustment.

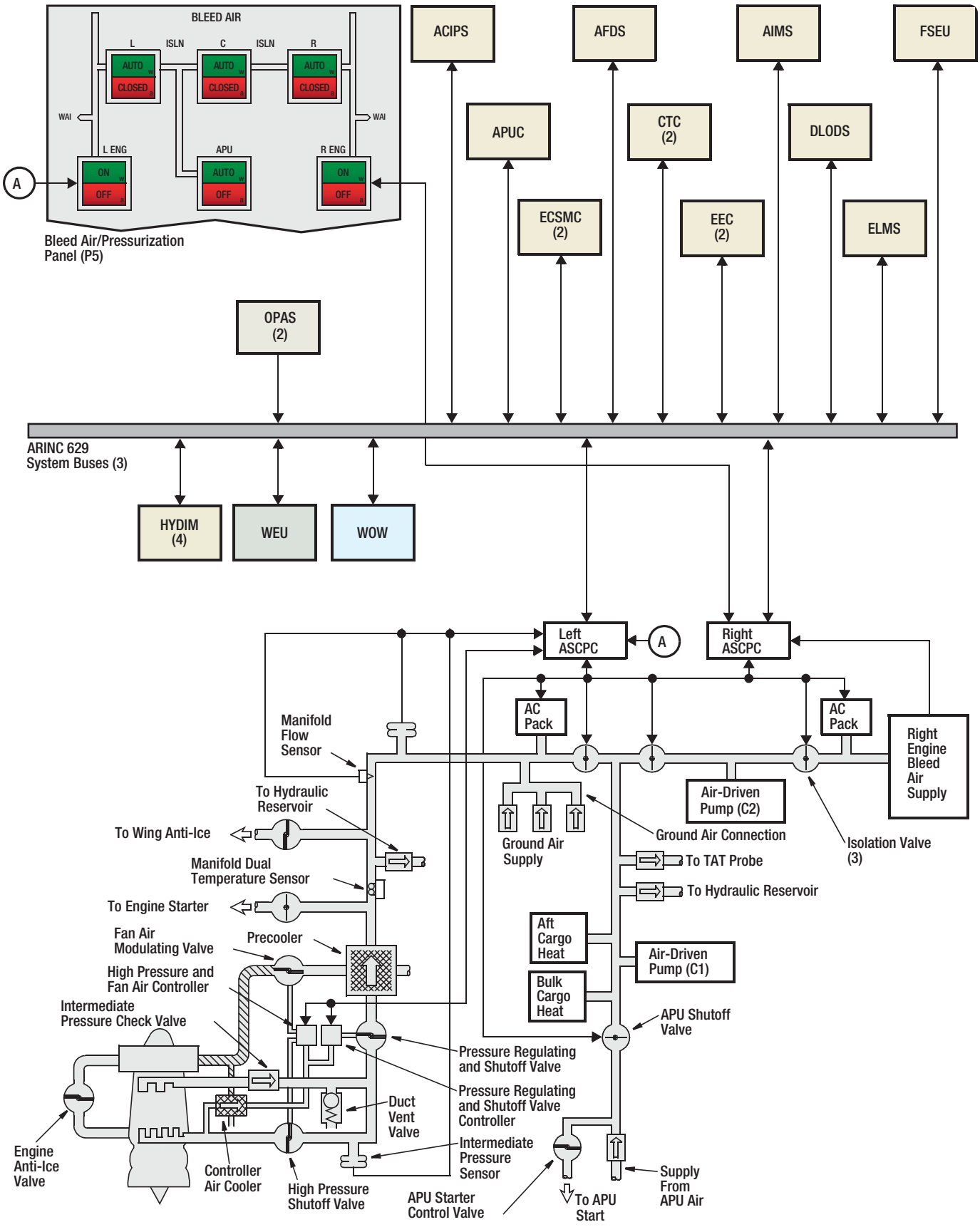


Figure 18-6. Pneumatic system components and integration on a Boeing 777.

INDICATIONS AND WARNINGS

There are few indications and warnings associated with the pneumatic system. Pneumatic manifold pressure is a key parameter monitored on the flight deck. Twin engine airliners typically have a pressure transmitter mounted in each section of the pneumatic manifold associated with an engine. These transducers send an electric signal to a dual gauge on the pneumatic control panel. Isolation valve control switches are located nearby. A low or no pressure situation can be handled by closing an isolation valve and using the remaining pneumatic source engine to supply all pneumatic requirements for each side of the pneumatic system. One transmitter is used for each engine. Both pressure transmitters are connected to a dual pressure indicator on the overhead panel. Engine bleed pneumatic system temperature is also monitored. On a Boeing 737, indication consists of over temperature switches located in the ducting system.

The switches are wired to trip lights on the flight deck overhead panel. The temperature switches illuminate the trip lights and the corresponding engine bleed valve closes automatically when the bleed air temperature exceeds approximately 490° F.

On digital aircraft, pneumatic system operation is completely automatic. Redundant computer controlled systems factor all supply and demand parameters including pneumatic pressure and temperature as in a non-digital aircraft. Valve positions and flow data are also factored. Over-pressure and over-temperature bleed air and duct leak conditions cause protective shutdown of the affected part of the system. BITE equipped components provide self-monitoring information to the central maintenance computer system. Synoptic display of pneumatic air user system parameters and those of the pneumatic system itself is given via a multifunctional display panel. (***Figure 18-7***)

For the pneumatic system, the air synoptic display shows the following information:

- Ground air in use
- Duct pressures
- Engine bleed air pressure regulating and shutoff valve position
- Isolation valve position
- APU shutoff valve position

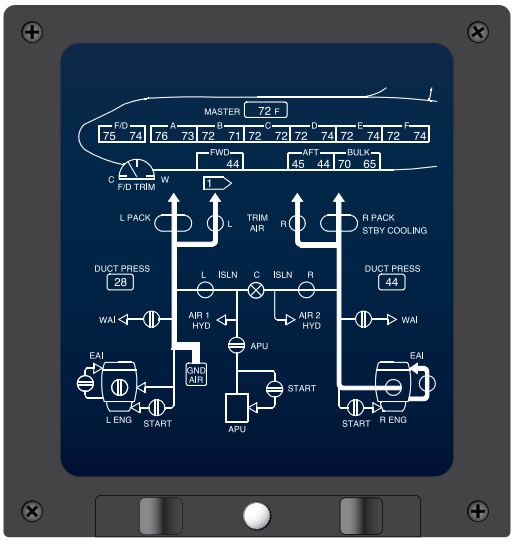
An X on an isolation valve symbol or the APU shutoff valve symbol shows the valve has failed or the switch on the bleed air/pressurization panel for the valve is in the non-normal position. An air supply maintenance page viewable on the multifunctional display is shown in ***Figure 18-8***.

INTERFACE WITH OTHER SYSTEMS

Pneumatic systems interface with other types of systems on many aircraft. The most common interface is with portions of the aircraft hydraulic system. As mentioned in the discussion on high pressure pneumatic systems, interface of emergency high pressure pneumatic system air with a normally hydraulic braking system is common. Use of a shuttle valve prioritizes the flow of air and directs it into the brake actuating mechanisms. The seals installed for hydraulic use are sufficient for use in the one-type deployment of emergency brakes by pressurized air.

Hydraulic system actuators are designed primarily for use with hydraulic fluid. Use with air is limited to one-time emergency operations. However, pneumatic power may be used to supplement and back-up hydraulic system components without loss of performance. This is done by turning a hydraulic pump with pneumatic power. The pneumatically driven hydraulic pump then supplies the hydraulic system components with fluid in the usual engineered manner. Cross utilizing hydraulic components with pneumatic air is eliminated.

Traditional benefits of hydraulic power are retained such as those from the incompressibility of the fluid. Control of a pneumatically driven hydraulic pump is through the use of a control valve in the pneumatic manifold. Selection of the pump via a switch on the flight deck causes the control valve to open and supply pneumatic air to drive to the pump. All hydraulic system controls are then operated normally either by the flight crew or automatically by computer. The Boeing 777 is an aircraft that incorporates pneumatically driven hydraulic pumps.



Synoptic Display

Figure 18-7. Synoptic display of a modern airliner pneumatic system.



Pneumatic Air Supply Display

Figure 18-8. Pneumatic air supply maintenance page on a digital airliner.

Question: 18-1
Twin-engine aircraft vacuum systems contain an _____ on each engine.

Question: 18-2
The pressure developed in a high pressure pneumatic system is approximately ____ psi.

Question: 18-3
When high pressure pneumatic air or nitrogen is used to move hydraulic fluid to an actuator, it is called _____.

Question: 18-4
The _____ contains the control valves that are operated to supply the systems that require pneumatic power.

Question: 18-5
_____-stage compressor bleed air normally supplies the bulk of the pneumatic system demand.

Question: 18-6
A _____ is located in each section of the pneumatic manifold separated by an isolation or shutoff valve.

Question: 18-7
Pneumatic manifold _____ is a key parameter monitored on the flight deck.

Question: 18-8
Pneumatic power may be used to back-up hydraulic system components without loss of performance by turning a _____ pump.

ANSWERS

Answer: 18-1
engine-driven vacuum pump.

Answer: 18-2
3 000.

Answer: 18-3
pneudraulics.

Answer: 18-4
pneumatic manifold.

Answer: 18-5
Intermediate.

Answer: 18-6
Relief valve.

Answer: 18-7
pressure.

Answer: 18-8
hydraulic.



PART-66 SYLLABUS **LEVELS**

CERTIFICATION CATEGORY → **B2**

Sub-Module 19

WATER/WASTE (ATA 38)

Knowledge Requirements

13.19 - Water/Waste (ATA 38)

Water system lay-out, supply, distribution, servicing and draining; — 2

Toilet system lay-out, flushing and servicing.

WATER/WASTE SYSTEMS

Large passenger aircraft are fitted with food preparation galleys and lavatories for passenger comfort. To support these installations, potable water, toilet and waste water/drainage systems are also installed.

LAYOUT

The water and waste system is composed of related systems as mentioned. The potable water system stores, delivers, monitors and controls potable (drinkable) water for galley and lavatory components. The toilet system provides sanitary toilets in the lavatory compartments and a means to dispose of toilet waste. The waste water system disposes of all waste water from the lavatory compartments. A water tank pressurization system is used to pressurize the potable water system. ***Figure 19-1*** illustrates these systems.

SUPPLY

The potable water system is the water supply system for the aircraft. Potable water is stored below the cabin floor in a single or multiple tanks. Typically, the aircraft pneumatic system pressurizes the tank(s). Pressure inside the tanks pushes the water through distribution lines to the lavatories and galleys.

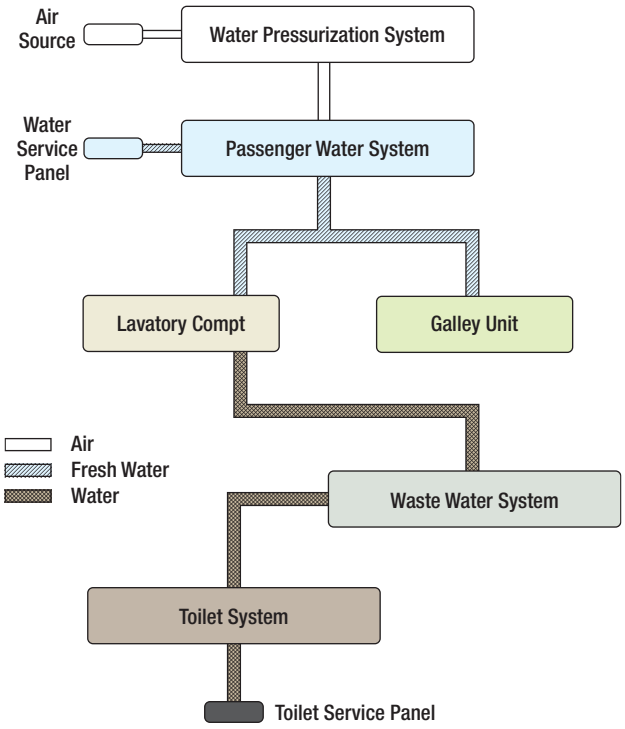


Figure 19-1. Related water/waste systems.

Potable water also goes to the toilets in lavatories equipped with vacuum flush toilet bowls. A fitting for pressurizing the tank on the ground with an external air source is available on the ground service panel. This is also where the tank can be filled and flushed. ***Figure 19-2*** illustrates a typical water servicing panel located behind a hinged door on the exterior of the aircraft. The water servicing panels are usually heated to prevent icing. Note that on some aircraft, a separate electrically powered compressor is used to pressurize the potable water tanks.

The water tank pressurization system passes air through an air filter and pressure regulator on way to the water tank. The air filter prevents contamination. A replaceable 10-micron filtration cellulose element is common. The pressure regulator reduces the air pressure and maintains approximately 20-50 psi in the water tank depending on the system. Check valves in the system prevent reverse pressurization. A system pressure relief valve prevents damage to the tank from over pressurization. ***Figure 19-3*** illustrates a typical potable water system. Most potable water supply systems are fitted with a pressure transmitter for tank pressure indication. Tank water quantity indication is also common.

DISTRIBUTION

Plumbing lines connect the water supply tank(s) to the lavatory and galley sink faucets. They are typically reinforced plastic hoses sometimes enclosed in protective conduit. In-line electric element heaters are installed just upstream of all hot water faucets. The units have a small reservoir and high wattage electric heating elements with overheat protection. Distribution lines are typically run behind the cabin ceiling panels. Shutoff valves are use to isolate sections of the water system for maintenance.

Since galleys and lavatories are modular units, often the technician must connect or disconnect the water distribution lines to the modular units. Plumbing inside the units includes lines, valves, heaters, etc. as well as components of the water drain and waste systems. Potable water distribution plumbing spans a great portion of the fuselage. Water lines pass near or through cold sections of the fuselage. In these areas, the plumbing must be insulated or heated. Heat tapes, inline heaters and heated hoses are all used to prevent ice in the water system.



Figure 19-2. A typical water servicing panel on an airliner.

SERVICING

It is necessary to completely drain the water system before disinfecting it or when parking the airplane in freezing weather. The water should be drained at least every 3 days and fresh water added to the system to prevent the growth of bacteria. Use of a disinfectant regularly in the potable water system is standard. Disinfection is also accomplished after completion of certain maintenance procedures or if the water system is contaminated.

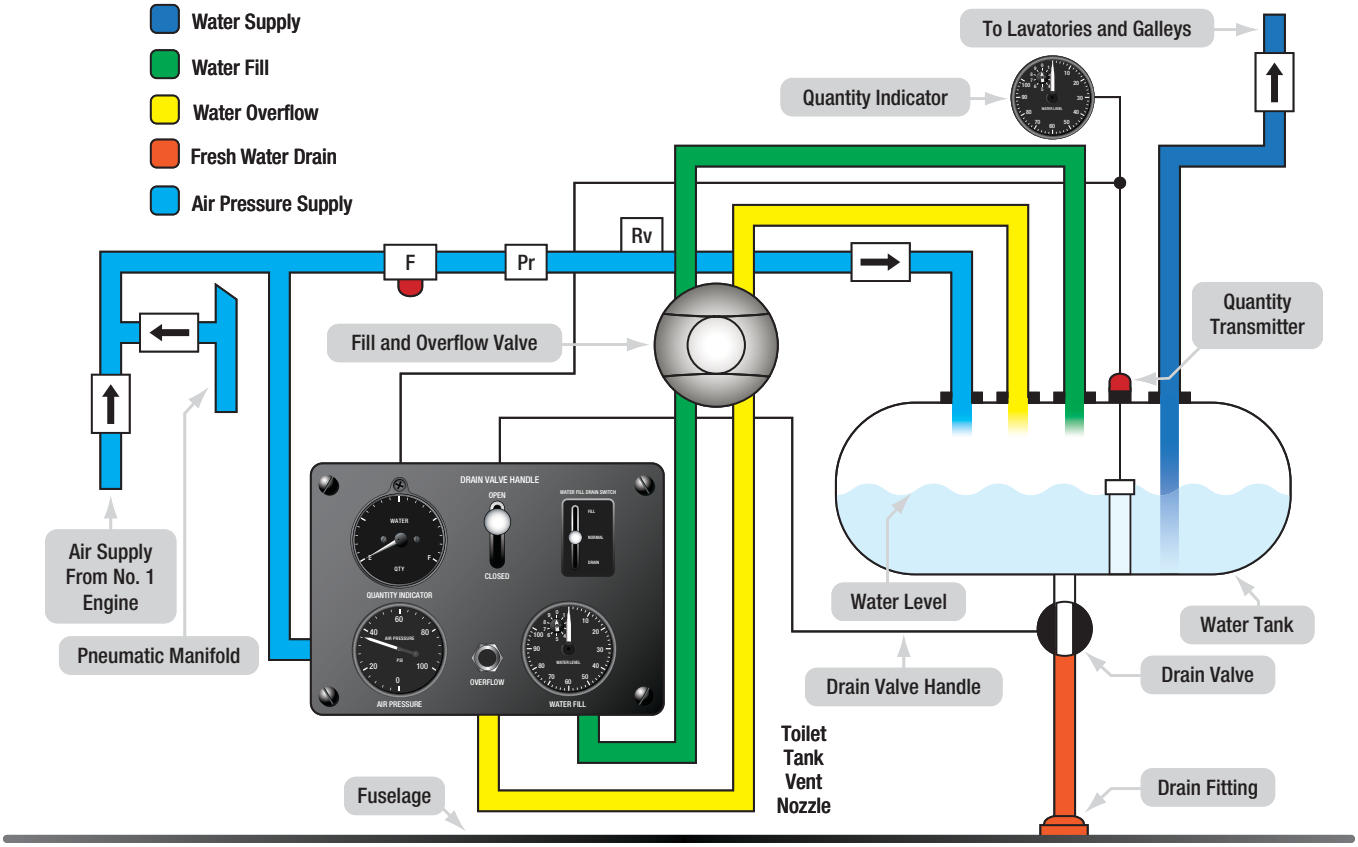


Figure 19-3. A typical potable water supply system.

The water servicing panel contains a fitting for the attachment of a fill hose. Once the hose is attached, a fill valve is positioned to allow the water to flow into the tank. The quantity indicator on the panel is used to fill the tank. An overfill drain line is plumbed from the top of the tank to the aircraft exterior. When the water reaches the level of the overfill fitting it spills into the overfill line and is drained overboard. Often, tanks are filled until water comes out of the overflow. Note that a maximum water fill pressure must be observed to avoid damage to the tank. The fill valve handle located on the service panel must be returned to the closed position for flight. Often the service panel door cannot be closed until the fill valve handle is in the proper position.

The fill/overflow valve allows filling of the passenger water system. On many aircraft, the valve is a four-port machined body with a rotary core. The rotary core is spring-loaded to ensure positive seating and prevent leakage. The valve is operated by a handle attached to the rotary core by a short stem. It is a 2-way valve with an OPEN and a CLOSED position. As stated, during flight the valve must be in the CLOSED position. The OPEN position permits filling of the water tank. There is typically only one water service panel installed, which allows filling, monitoring and draining of the system. The panel is located in the lower side of the fuselage, usually positioned in an area that does not interfere with other servicing and loading activities during turn-around checks.

DRAINING

A potable water tank drain valve is a two-port valve designed to permit drainage of the potable water tank. It is typically located below the tank and is controlled from the water service panel.

Each lavatory is also provided with a drain valve which controls potable water drainage of lavatory compartment plumbing. This valve typically has 3 positions: OFF, DRAIN and ON. In the OFF position all lines are completely closed. In the DRAIN position all lines are completely open. In the ON (normal) position, the water inlet and outlet lines are open and the drain line is closed. The valve is accessible in the lower part of the lavatory sink cabinet below the water heater. Galley and lavatory units are usually built in such a way as to allow any spills and leaks to drain through a port in the floor of the unit. This water is typically drained overboard through a heated drain mast.

Waste water is either drained directly overboard or to a holding tank. The tank is usually the same tank used for collecting toilet waste. The waste water drainage system disposes of all waste water from the lavatory compartments sinks. The system consists of plumbing necessary to drain the waste water into the toilet tank and moisture condensation and seepage to the toilet drain tube. Sink stoppers are spring loaded to the CLOSED position to maintain the integrity of cabin pressurization. Check valves are commonly used to prevent odors from backing up into the lavatory. Cabin pressure is used to close the drain system until needed and to force waste water overboard through a drain mast(s) mounted on the lower outside of the fuselage. (***Figure 19-4***)

On some aircraft, waste water is known as gray water. This does not include toilet waste which is collected in a tank and emptied from the ground lavatory service panel when the aircraft is on the ground. The gray water is used to assist toilet flushing or is drained overboard. (***Figure 19-5***)

Waste water drain masts are heated either electrically or with warm pneumatic system air. As with the water distribution lines, waster water lines are also heated in cold areas to prevent freezing. Heaters in some the gray water drain masts give high heat in flight and low heat on the ground. Heated gaskets are used in some installations to protect the waste drains. Heater blankets are sometimes used to heat the waste tank drain lines.

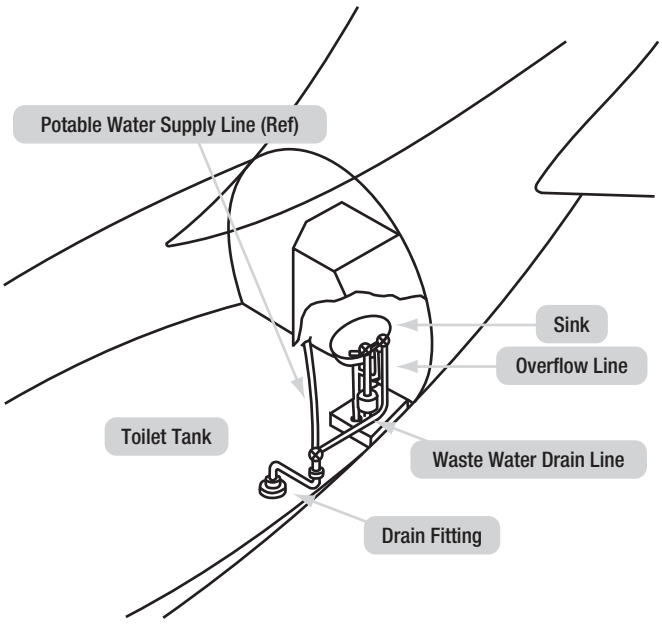


Figure 19-4. A typical waste waster installation on an airliner.

TOILET SYSTEMS

Large passenger aircraft and freight aircraft are fitted with lavatories. These lavatories include a flushable toilet system. For many years, toilet flushing systems used an electric motor-driven flushing mechanism to remove toilet waste into a holding tank positioned directly under the toilet. The most modern aircraft incorporate a vacuum flush system with remote waste holding tank(s).

As mentioned, modular lavatory installations are common. Most have all of the components immediately needed for flushing installed inside the unit including the waste holding tank. A waterproof floor pan or catch basin with drain makes up the floor of the lavatory unit which is installed on top of the passenger cabin floor. Tiedowns attach the unit to aircraft structure. Drain and vent lines extend through the floor to the outside of the aircraft. (***Figure 19-6***)

FLUSHING

All flushing system components are installed in the lavatory assembly. Typically, the toilet flush handle activates a timer which controls a toilet flush motor mounted on top of the toilet waste holding tank. When flushed, the motor turns a pump that sprays flushing agent from the tank through a filter and into the toilet bowl. Nozzles direct the spray in a swirling motion. After the set amount of flush time (usually 8-12 seconds) the waste and agent drain into the tank below the toilet. A separator between the tank and the bowl prevents splash and vision of tank contents.

Modern vacuum toilets are of different construction than the electric flush unit described and are not interchangeable. (***Figure 19-7***) The vacuum flush toilet is electronically controlled by a flush control unit mounted close to the flush valve at the outlet of the toilet. In addition to opening and closing the valve, the flush control module opens a rinse valve that releases potable water to rinse the bowl as the waste is removed and is connected into the aircraft's central monitoring system and the cabin services system.

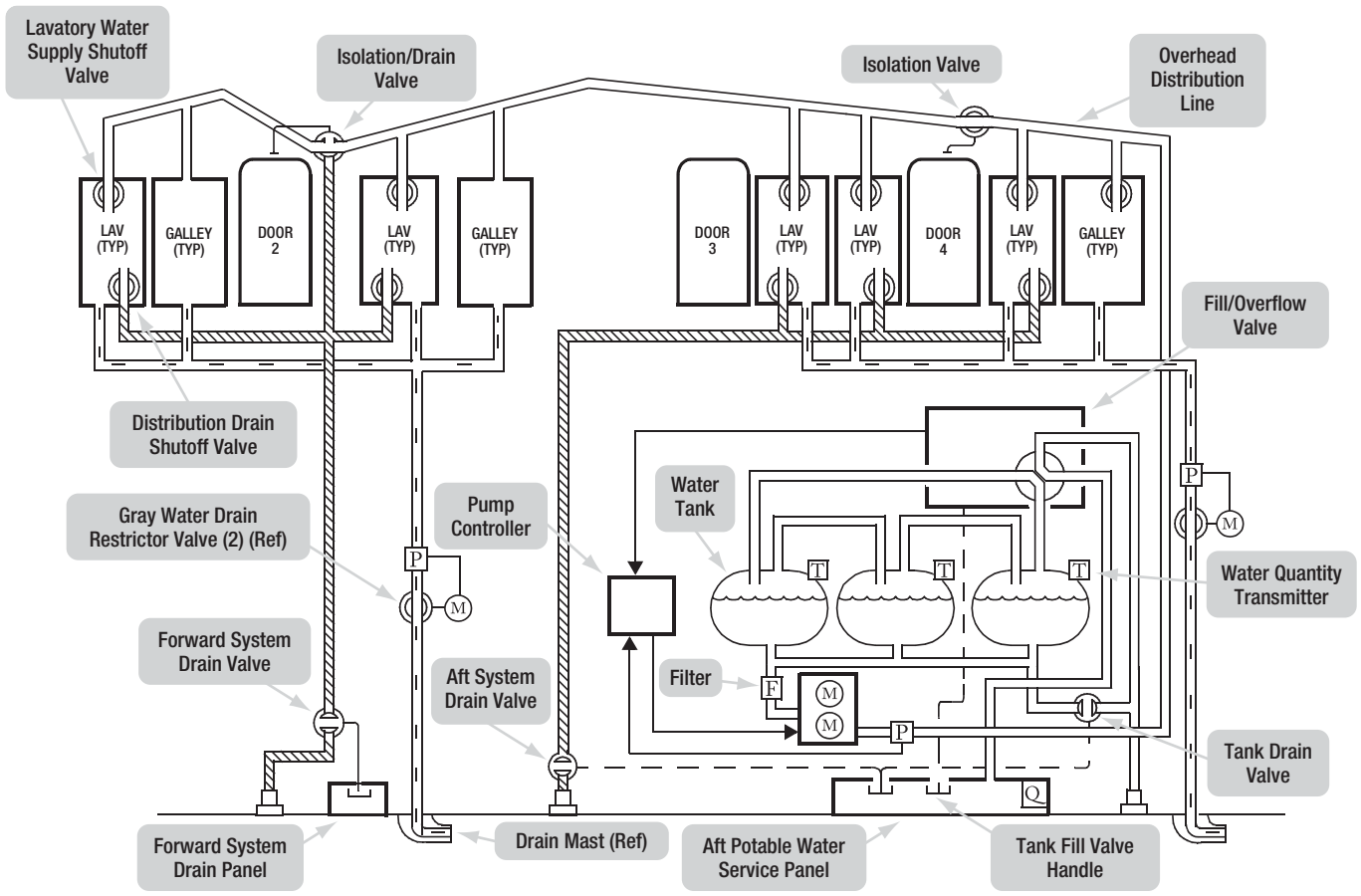


Figure 19-5. Gray water and potable water layout on a modern airliner.



Figure 19-6. Self contained modular lavatory.

Vacuum waste systems remove waste from the toilet with suction and deposits it in a waste holding tank. Potable water is injected for approximately one second to rinses the toilet when flushed. The waste tanks are remotely located below the cabin floor. The suction for flushing is made by an electric vacuum blower below 16 000 feet and by the differential pressure between the cabin pressure and the outside air pressure above 16 000 feet. A barometric switch controls the blower. Vacuum check valves are used to prevent reverse airflow through the vacuum blower bypass lines.

The electronic flush control unit controls a normally closed flush valve to open when the flush handle is activated. The waste is sucked from the toilet when flushed and sent to the remote waste storage tank. The tank is an integral part of the vacuum waste system. It contains a quantity sensing system and a separate sensor to indicate the tank is full. An electronic control module shuts the system down when the tank is full. It also provides indications that are sent to the flight attendants' cabin control panel as well as the EICAS maintenance page displayed on the flight deck. The tank contains a liquid separator at the inlet (top) of the tank. The liquid separator removes both moisture and waste particles from the waste tank air before venting the air overboard. The waste remains in the tank(s).

LAYOUT

There are a variety of lavatory system layouts owing to the number and configuration of lavatories on any particular aircraft. The basic systems remains as described. ***Figure 19-8*** illustrates a simple lavatory system of a Boeing 737.



Figure 19-7. A vacuum flush system toilet.

Each lavatory is positioned near an external service panels to which the waste tank drain and flush lines as well as the waste tank vent line are routed. Waste is removed from the waste tanks through the waste service panel. A "T" handle in the service panel opens the waste tank drain ball valve which permits the waste tank to drain. The flush line is used to rinse the waste holding tank.

Figure 19-9 illustrates a vacuum flush waste system. A vacuum blower is associated with each waste tank regardless of the number of toilets that drain to the tank. Drain and flush lines run from the tank to the exterior servicing panel as usual. Also plumbed into the system are the potable water lines used to delivery a small quantity of water with each flush.

SERVICING

Servicing components in a toilet system allow ground draining and cleansing of the toilet units. The waste tank is drained by pulling the waste drain valve handle on the toilet servicing panel after attaching the ground service cart to the 4 inch drain outlet and removing the drain plug. The tank is cleaned by attaching water pressure to the ground flush connection. The water enters the tank through spray nozzles that rinse the insides of the filter and tanks. The latest toilet waste holding tanks have built in separators to permit particle free vacuum air to be expelled overboard as waste is drawn into the tank from the toilet. A waste tank rinse nozzle is fitted to the top of the tank to rinse it when servicing. (***Figure 19-10***)

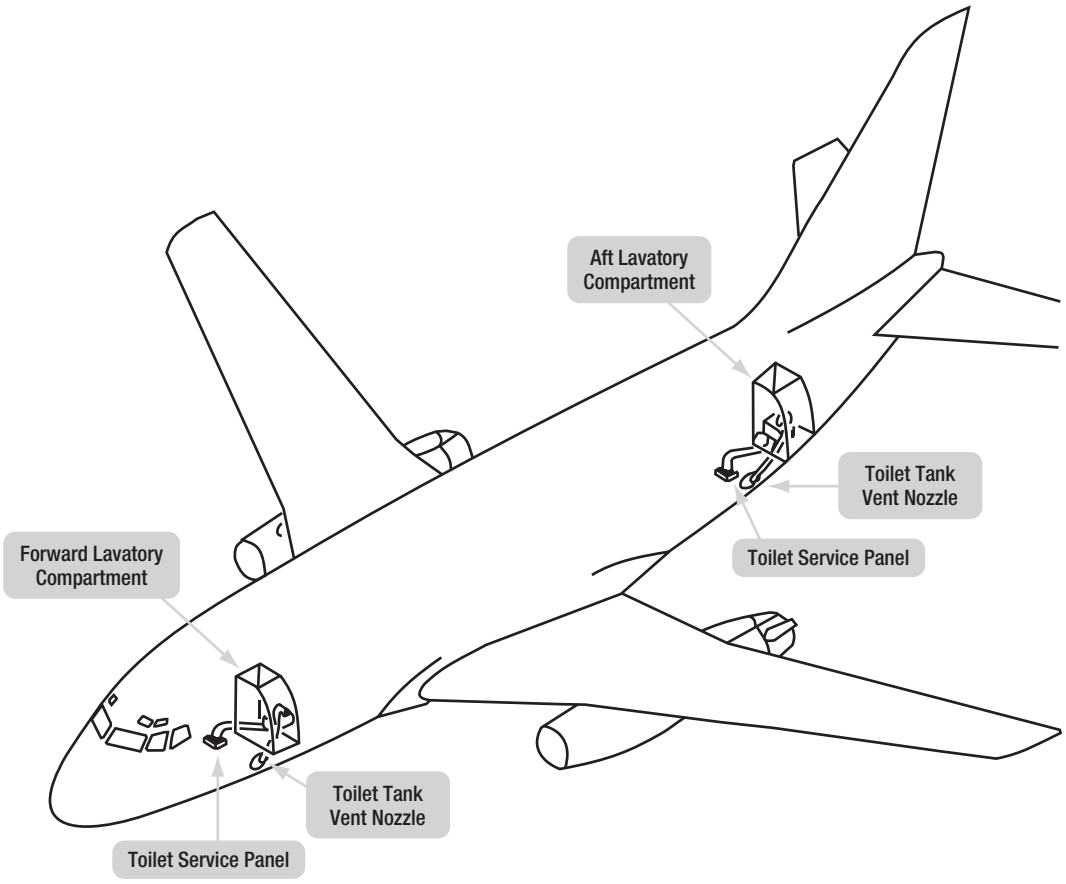


Figure 19-8. A typical toilet waste system on an airliner.

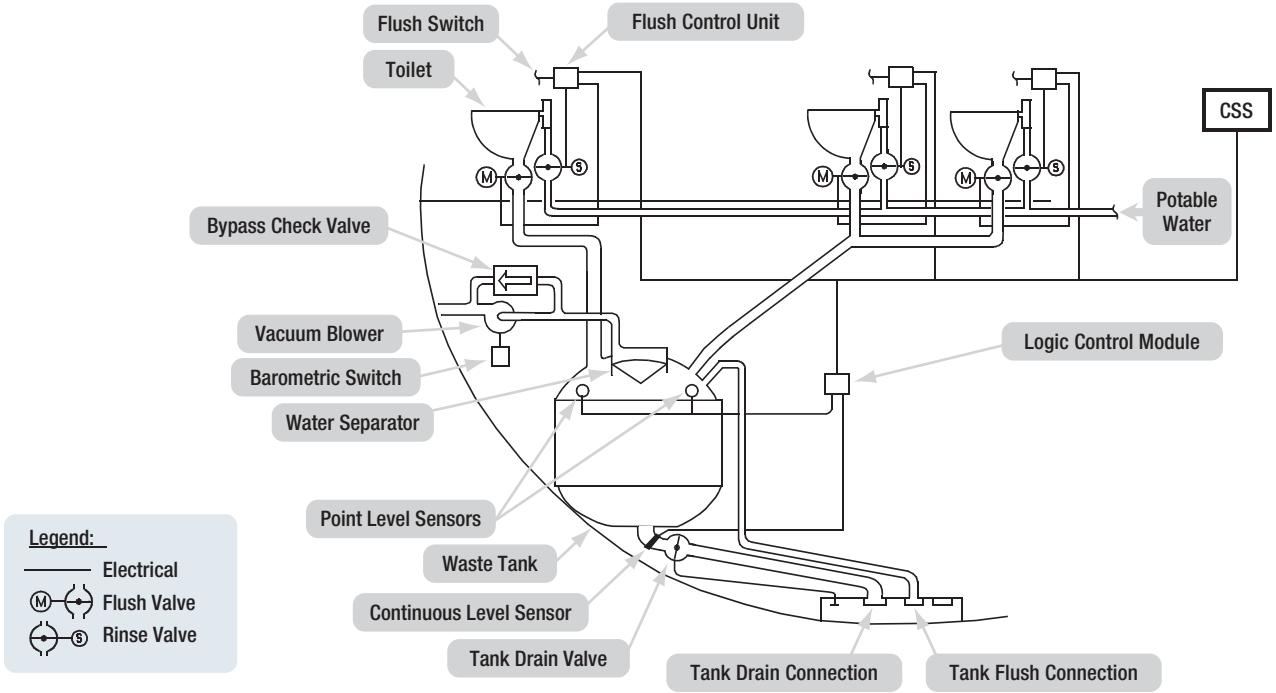


Figure 19-9. Toilet waste system on a vacuum flush aircraft.

CORROSION ASPECTS

The presence of water in lavatories and galleys makes the structure around and below these areas highly susceptible to corrosion. Most structure is aluminum although some manufacturer's include titanium structure because it is much more resistant to corrosive effects. Waterproof flooring, built in catch basins and drains help reduce corrosion by controlling where water can flow. But operation with thousands of passengers over time typically results in the loss of water and corrosion of the entire area around a lav or galley installation. Anti-corrosion measures are increased around and under lavatories and galleys. Some common actions are as follows:

- Corrosion Preventative Fluids (CPFs) applied to the structure.
- Floor panel clips taped to prevent scratching.
- Floor panels cushioned to prevent scratching and damage.
- Floor panels sealed to prevent fluid getting into the structure.
- Moisture barriers applied over the entire area.
- Leaks in the toilet system traced and cured at an early stage.

CPF is a water displacing fluid that adheres to the surface of clean (or primed) aluminum and it's alloys. It is a slightly waxy substance that can either be sprayed by aerosol can or by a 5 gallon (221) dispenser onto the structure. Care needs to be taken during this operation, as the fluid spray will damage the lungs if breathed in. Use only in well ventilated areas and wear face masks.

A thickened version of the fluid is available for brushing. Consult the AMM prior to application. The biggest draw-back with CPF is that it acts as a debris collector. It will collect dust and debris as it is very sticky. During routine maintenance and certainly prior to any structural inspection this needs to be removed. Always record any work done (re-application, etc.) and clear all work with a CRS signature. In addition to CPF, sealing of the floor panels in the area of lavatories and galleys is common. This is done in a variety of ways including taping and two-part sealants. Consult the airframe manufacturer's maintenance manual for the correct materials and procedures. In addition to sealing the floor panels, some operations utilize a Mylar moisture barrier over the floor before lavatories and galleys are installed. A caution with

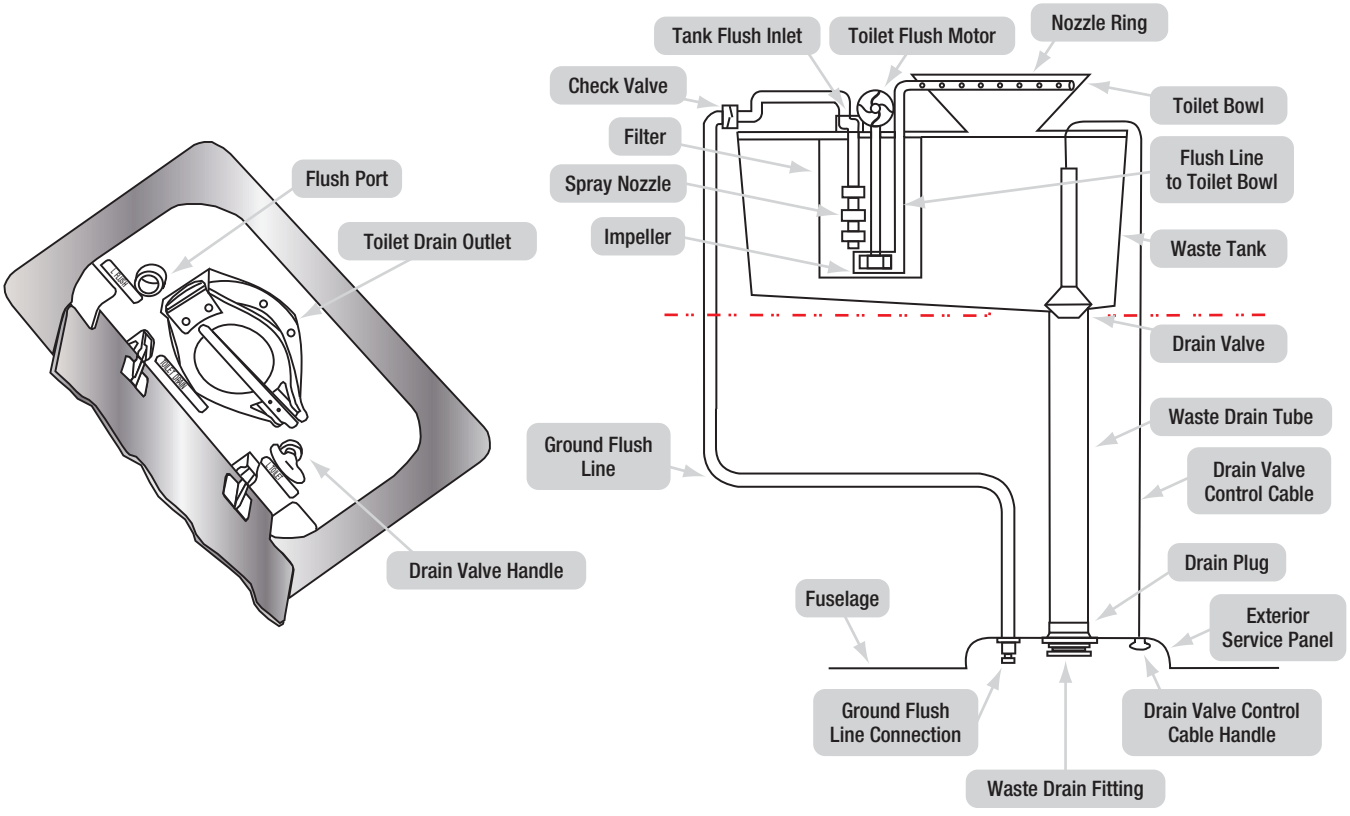


Figure 19-10. A typical toilet waste system on an airliner.

this method is that protection may be provided by the impermeable Mylar however water may migrate to the edge of the protected area and cause structural damage there. The most effective means of corrosion control is to prevent the liquid getting to the floor in the first place. Vacuum flush systems are a great improvement over the re-usable water toilet systems. The under-floor structure on these aircraft is usually in much better condition.

On all toilet systems it is important that leaks are detected and rectified early. Often the first sign that there is trouble under the toilets is water coming from the lower bilge drains on the underside of the fuselage. Any liquid from these needs immediate investigation. Liquid coming from the bilge drain could be (a) clean water (b) liquid from some cargo on board, or (c) toilet water. The clean water could be rain water, condensation water or potable water. In all cases, the source must be found and the fault rectified.

Question: 19-1
_____ inside the potable water tank(s) pushes the water through distribution lines to the lavatories and galleys.

Question: 19-2
What is used to heated potable water on passenger aircraft?

Question: 19-3
The fill valve handle located on the water service panel must be _____ for flight.

Question: 19-4
Galley and lavatory units are usually built in such a way as to allow any spills and leaks to drain through _____.

Question: 19-5
_____ are commonly used to prevent odors from backing up into the lavatory.

Question: 19-6
Waste water drain masts are heated either _____ or with _____.

Question: 19-7
Vacuum waste systems remove waste from the toilet with _____ and deposits it in a waste holding tank.

Question: 19-8
Waterproof flooring, built in catch basins and drains help reduce _____ by controlling where water can flow.

ANSWERS

Answer: 19-1
Air pressure.

Answer: 19-2
In-line electric element heaters.

Answer: 19-3
closed.

Answer: 19-4
a port in the floor of the unit.

Answer: 19-5
Check valves.

Answer: 19-6
electrically.
warm pneumatic system air.

Answer: 19-7
suction.

Answer: 19-8
corrosion.



PART-66 SYLLABUS **LEVELS**

CERTIFICATION CATEGORY → **B2**

Sub-Module 20

INTEGRATED MODULAR AVIONICS (ATA 42)

Knowledge Requirements

13.20 - Integrated Modular Avionics (ATA 42)

Functions that may be typically integrated in the Integrated Modular Avionic (IMA) modules are, among others: 3

Bleed Management, Air Pressure Control, Air Ventilation and Control, Avionics and Cockpit Ventilation Control, Temperature Control, Air Traffic Communication, Avionics Communication Router, Electrical Load Management, Circuit Breaker Monitoring, Electrical System BITE, Fuel Management, Braking Control, Steering Control, Landing Gear Extension and Retraction, Tyre Pressure Indication, Oleo Pressure Indication, Brake Temperature Monitoring, etc.;

Core System; Network Components.

INTRODUCTION

This section describes integrated modular avionics (IMA). As the name indicates, an aircraft with IMA has avionics systems that are integrated and modular. Integrated means that multiple functions are combined into a single piece of equipment. Modular refers to a design method that allows the system to be divided into separate, replaceable modules. The term Avionics itself derives from aviation electronics, and it refers to electronic systems used in aircraft. Avionics encompasses a wide range of systems. Avionics is used for navigation, for communication, for control of the aircraft, and for other purposes. On a modern aircraft, there are dozens of systems that can be considered avionics systems.

Integrated modular avionics is a design methodology, not an avionics system per se. In other words, it describes the way the avionics systems are put together, not the specific functions of the individual avionics systems. IMA represents and advance in avionics technology. Aircraft with IMA can realize reductions in the bulk, space, and weight of their avionics systems. Also, the overall reliability of the avionics can be improved, as less components are required.

Ease of fault isolation, and additional facilities such as BITE (built–in-test-equipment), is able to be achieved with the inherent architecture design. With aircraft systems having more software based functions, and computers becoming more powerful, computer requirements for equipment can be shared.

The following is just a partial list of functions that may be Integrated into an IMA System:

- Bleed Management
- Air Pressure Control
- Air Ventilation and Control
- Avionics and Cockpit Ventilation Control
- Air Traffic Communication
- Avionics Communication Router
- Electrical Load Management
- Circuit Breaker Monitoring
- Electrical System Built In Test Equipment (BITE)
- Fuel Management
- Braking Control
- Steering Control
- Landing Gear Extension and Retraction
- Tire Pressure Indication
- Brake Temperature Monitoring

Referring to the above list, IMA can be considered divided into three functions:

- Cockpit (fly-by-wire electrical flight controls, communication and warning systems)
- Utilities (incorporating the numerous systems above)
- Cabin (air-conditioning and other systems)

INTEGRATION OF AVIONICS

Design methods for avionics systems have evolved over time. From the late 1980's Business Jets have incorporated concepts of IMA, but on a smaller scale found in modern airliners. Military design developments for the F-22 and F-35 have also migrated into the commercial aviation field. Initially, avionics systems were discrete, stand-alone systems. This means that each system was separate. (***Figure 20-1***)

For example, an aircraft's compass system might have consisted of a flux detector in the wing, a gyroscope in the avionics compartment, and a heading indicator on the instrument panel. These components were used only by the compass system. They were not shared by other systems on the aircraft. Although the compass system's components were connected to each other with wiring, the compass system itself was not normally connected to any of the other systems on the aircraft. (An exception



Figure 20-1. An older instrument panel with many discrete indicators.

to this might be where an aircraft's auto-pilot has a Heading mode facility where compass heading data is fed as an outer loop source.)

Traditionally, on non-IMA aircraft, each avionics system had its own separate indicator and its own separate controls. As more and more avionics systems were developed and installed in aircraft, more indicators and controls had to be installed. Instrument panels became more complex and crowded. Figure 19 - 1 shows such an instrument panel. In addition, as more avionics systems were developed and installed, more LRUs (line replaceable units) or "black boxes" were installed in avionics compartments. More wiring was needed to interconnect these LRUs with their associated cockpit controls and indicators. More electrical power was needed to operate the systems.

Each additional indicator, LRU, and wire that is installed on an aircraft takes up space and adds weight. Because both space and weight-carrying capability are at a premium, it is desirable to keep the number of indicators, LRUs, and wires to a minimum. In the case of indicators, engineers began to develop designs that used the same indicator to display information from more than one system. For example, older designs had separate indicators for the compass system, the radio navigation system, and the weather radar system.

In newer designs, these systems are all connected to single, "integrated" indicator such as a navigation display (ND). The use of integrated indicators saves space and weight, and it streamlines pilot workload by reducing the number of indicators that must be scanned during flight. Much more information can be displayed on one instrument, or in a smaller area, which ergonomically creates an easier workload for the pilot.

Figure 20-2 shows a modern, integrated instrument panel. The design concept that was first used to combine indicator functions in the cockpit has been carried further with integrated modular avionics. In an aircraft with IMA, the same concept – integration – is applied to many of the LRUs as well. Instead of individual, independent LRUs, an aircraft with IMA uses modules which are integrated into a single system. The modules perform the functions formerly performed by the independent LRUs, but they are not completely independent. They share circuitry.



Figure 20-2. An instrument panel with fewer, integrated instruments.

Traditional (non-IMA) avionics suites have many separate LRUs ("black boxes") located in an avionics compartment. In such systems, there is a considerable amount of duplication among the black boxes. For example, each black box in the avionics compartment typically contains its own power supply. These power supplies receive aircraft power and use it to provide the various voltages needed by the circuits within that box. Also, each power supply is connected to the aircraft's electrical power system by a separate wire.

These power supplies might be functionally identical to each other. In a non-IMA aircraft, there could be twenty identical power supplies in twenty separate boxes in the avionics compartment. However, if the power supply in one of the black boxes fails, that system fails because that box cannot use power from another black box. In an aircraft with integrated modular avionics, some of the self-contained black boxes are replaced by modules. The modules form part of an integrated system because they are plugged into a mainframe or rack which is a single piece of equipment. This results in a bulk and weight reduction because some circuitry is now shared among the various modules.

For example, instead of having twenty duplicate, non-redundant power supplies, an IMA aircraft might have three redundant power supplies that are each capable of supplying all twenty modules. If one of the IMA power

supplies fails, there are still two redundant power supplies that can power the modules. A failure of one or even two of the power supplies does not result in the failure of any of the avionics modules. Thus, the IMA aircraft is more fault-tolerant, resulting in better reliability.

The integration concept reduces the total number of LRUs needed. Often, the same kind of data processing circuitry is required for different avionics functions. In previous (non-IMA) designs, this data processing circuitry had to be duplicated in separate LRUs for each separate system. With IMA, the data processing circuitry can be contained in fewer LRUs (modules), and it is shared among the systems that require it. This combining of functions results in an overall reduction in the number of separate LRUs. Less wiring is needed in an IMA aircraft. This is partially due to the integration of functions that eliminates unnecessary duplication of circuitry.

Using the example of the power supplies, with IMA there is no need to provide twenty separate wires from the aircraft electrical system for power supplies because there are only three power supplies for the IMA modules. In addition to this, the fact that IMA aircraft make extensive use of digital data buses also reduces the amount of wiring needed. (***Figure 20-3***)

DIGITAL DATA BUS USE REDUCES WIRING

The use of digital data buses can result in a tremendous reduction in the amount of wiring used in an aircraft. Digital data buses are used to transfer information from one piece of equipment to another, using far fewer wires than were previously required. A number of different digital data bus systems are used in various aircraft. Some of the more important ones are ARINC 429, ARINC 629, and AFDX (Avionics Full Duplex). An example that illustrates this reduction in wiring is radio tuning. When a radio is tuned, frequency information must be transferred from a radio tuning unit to the radio receiver or transceiver that is being tuned. This frequency information might consist of four or more digits.

Say a pilot wishes to tune a VHF communications transceiver to the frequency 128.35 MHz. The pilot enters this frequency into the tuning unit in the cockpit. From there, it must be carried to the VHF transceiver located in the avionics compartment. (***Figure 20-4***)

With traditional avionics systems, those not using digital data buses, each piece of information to be transferred from one location to another requires at least one separate wire. Often, a far larger number of wires is needed. Using our example of a pilot tuning the VHF communications transceiver to 128.35 MHz, each digit of the selected frequency must be transferred from the

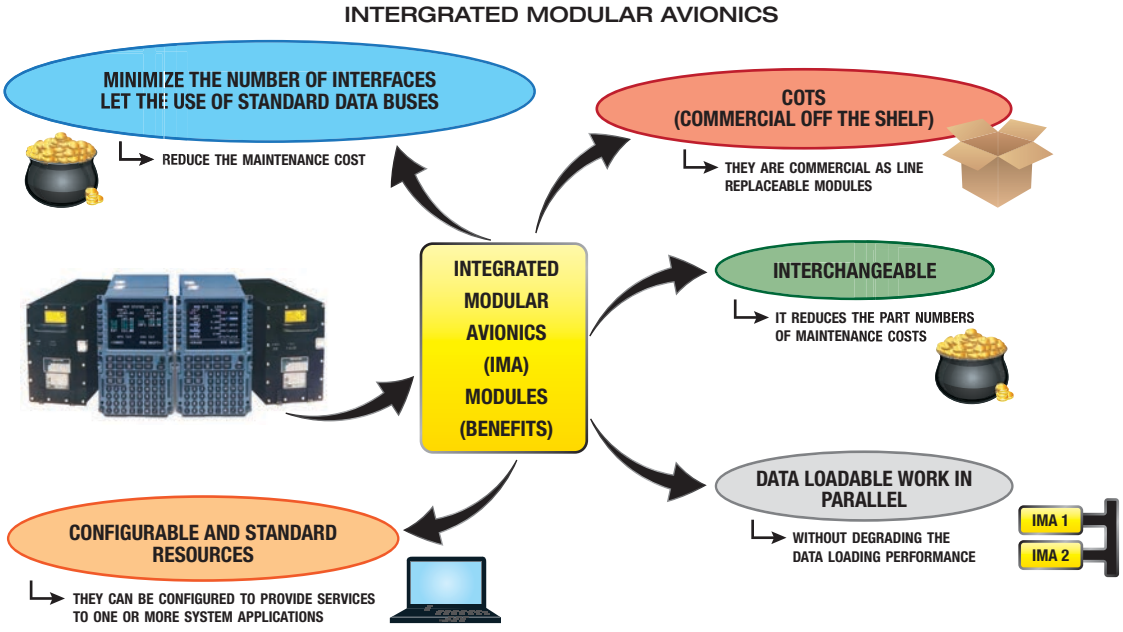


Figure 20-3. Integrated modular avionics concept diagram.

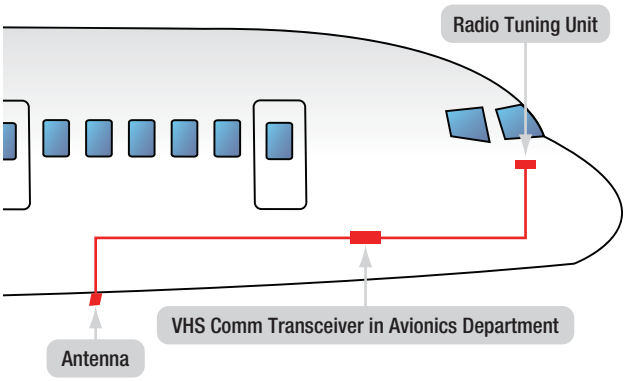


Figure 20-4. VHF Comm system interconnection.

2 x 5 Tuning Scheme

	0	1	2	3	4	5	6	7	8	9
A		✚	✚						✚	✚
B	✚	✚		✚	✚					
C			✚	✚		✚	✚			
D					✚	✚		✚	✚	
E	✚						✚	✚		✚

*Wires marked with ✚ are grounded.

Figure 20-5. Two out of five tuning scheme.

tuning unit to the transceiver. Because there are five digits in this frequency, it might seem that five wires would be needed to transfer this data. However, because the information being transferred is complex (each digit might be anything from a zero through a nine), even more wires are needed. A typical pre-data-bus method of accomplishing this transfer was the ARINC "two out of five" or 2×5 tuning scheme.

Figure 20-5 shows the 2×5 tuning scheme. 2×5 tuning requires the use of five wires for each digit of information being transferred. Of these five wires, two are be connected to ground and three are not. The particular wires which are grounded determine whether the digit transferred is a zero, a one, a two, etc. Since all VHF communication frequencies begin with the number "1," it is not necessary to transfer that digit from the tuning unit to the transceiver. However, each of the other four digits in the selected frequency must be transferred. Four digits at five wires per digit results in 20 wires. A ground wire is also needed, bringing the total number of tuning wires to 21. ***Figure 20-6*** illustrates this.

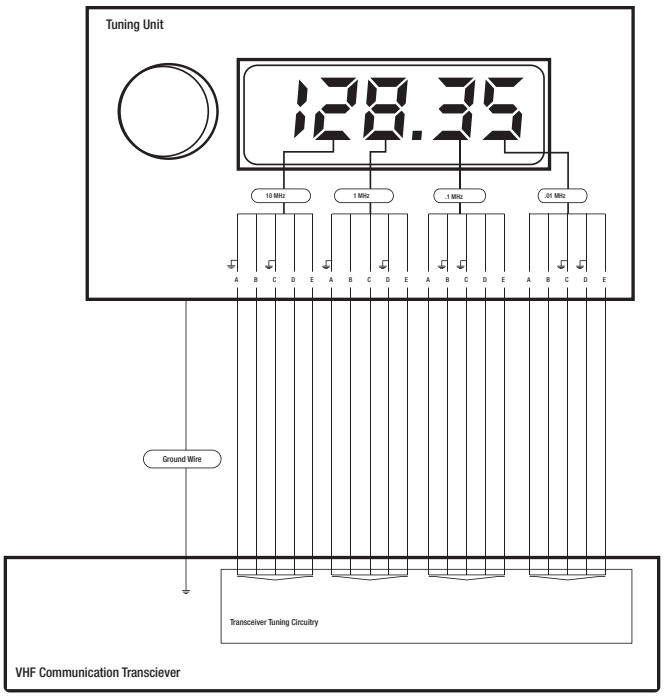


Figure 20-6. VHF Comm system tuning wires using two out of five tuning.

Using a digital data bus like ARINC 429, this same tuning information can be transferred using only two wires. With ARINC 429, the two information-carrying wires are twisted together, and they are covered with a braided shield. The shield is usually kept grounded, and protects the two inner wires from electromagnetic interference. The two inner wires are referred to as a "twisted pair". (***Figure 20-7***)

The reason it is possible to transfer so much information on only two wires is that the information is sent serially. This means that the same two wires carry one bit of information at a time. One bit of information is sent, then another, and then another. Soon, all the required information has been transferred from the tuning unit to the radio. The information transfer is done at a rapid rate: ARINC 429 can transfer up to 100 000 bits of information each second. ARINC 629 can transfer up to two million bits per second. AFDX is a newer digital data bus system used that can transfer data at rates up to 100 Mb/sec. Fewer wires are needed for each successive developed data bus system and so a substantial weight savings is realized.

AFDX is the latest development for avionics data buses. It has been successfully developed for the A380 and B787 and will be used for the A350 and A400M aircraft. AFDX uses a star topology switching network

and it's frame format is fully compliant with IEEE STD 802.3 (Ethernet). Relative to other formats, AFDX has characteristics of: good integration and ease of interfacing; reduction in cable interconnects and wiring; high bandwidth; high reliability and fault tolerances.

COMPUTER ARCHITECTURE

Examples of practical computer assemblies utilized by leading IMA design companies are listed below:

- Cabinet Modules, with each module connected to a BACKPLANE for intermodule communication, and gateways. (Honeywell AIMS)
- Cabinet of Cards; using a propriety DEOS operating system – commonly found in Business and Regional jets; (Honeywell Primus EPRIC)
- Independent Modules as LRU's; an AIMS derivative, (Honeywell VIA)

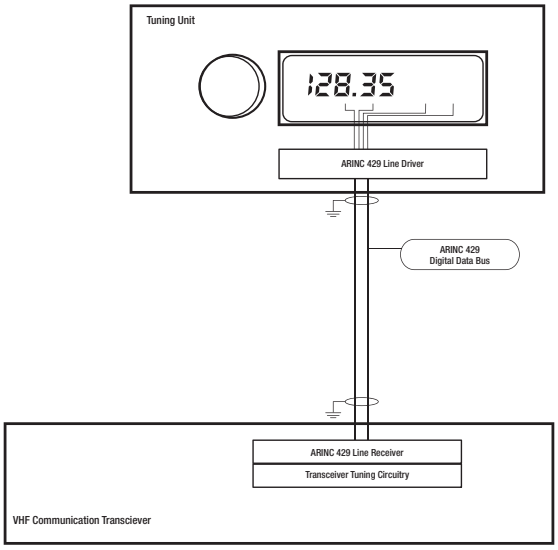


Figure 20-7. VHF Comm system tuning wires using ARINC 429 digital data bus.

DESIGN CONCEPTS

Using latest technology, design for IMA systems has been approached by designing the hardware around commercial available computer operating systems. This has created economic momentum in the building of IMA systems with less cost and complexity of applications. Much initial design of software and applications has been averted by employing the existing commercial systems that have been available. Ease of software upgrades has thus been also been ensured. Modern airliners invariably use either of two IMA concepts:

COMMON CORE SYSTEM (CCS) CONCEPT

This IMA system consists of a common core System with network components. The core system contains a central computer (dual computers for redundancy purposes) with data processing circuitry that processes many different kinds of information. This processing circuitry, is shared by the various avionics systems that have been integrated. Information from various sensors, controls, and LRU's is brought into the core system for processing, usually in a concentrated form, from data concentration modules. Data is then sent out from the core system to displays, actuators, and other places in the aircraft where that information is used.

The core system uses the same computer processors, or modules, for many different purposes. For example, a core system can use the same processor for such tasks as calculating throttle settings for best fuel economy, calculating the amount of rudder deflection needed to coordinate a turn, monitoring an instrument landing system receiver for malfunctions, determining if the stall warning system needs to be activated, and many others. The core system's processing power is shared among these various functions. This sharing eliminates the need to have a processor in each and every system.

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NETWORK COMPONENTS

Network components are the parts of the IMA system that allow data to be transferred into and out of the core system. These components include input and output devices and data bus wiring. Input devices receive data from a digital data bus and couple it to the core system processing circuits. Output devices prepare core system output data for transmission along

a digital data bus. The exact properties of the input and output devices vary with the particular kind of data transfer system that is being used. As mentioned above, there are several different data transfer systems used in aircraft, some of the more common ones being ARINC 429, ARINC 629, and AFDX. ***Figure 20-8*** contains various data transfer formats that have become standard with specific systems.

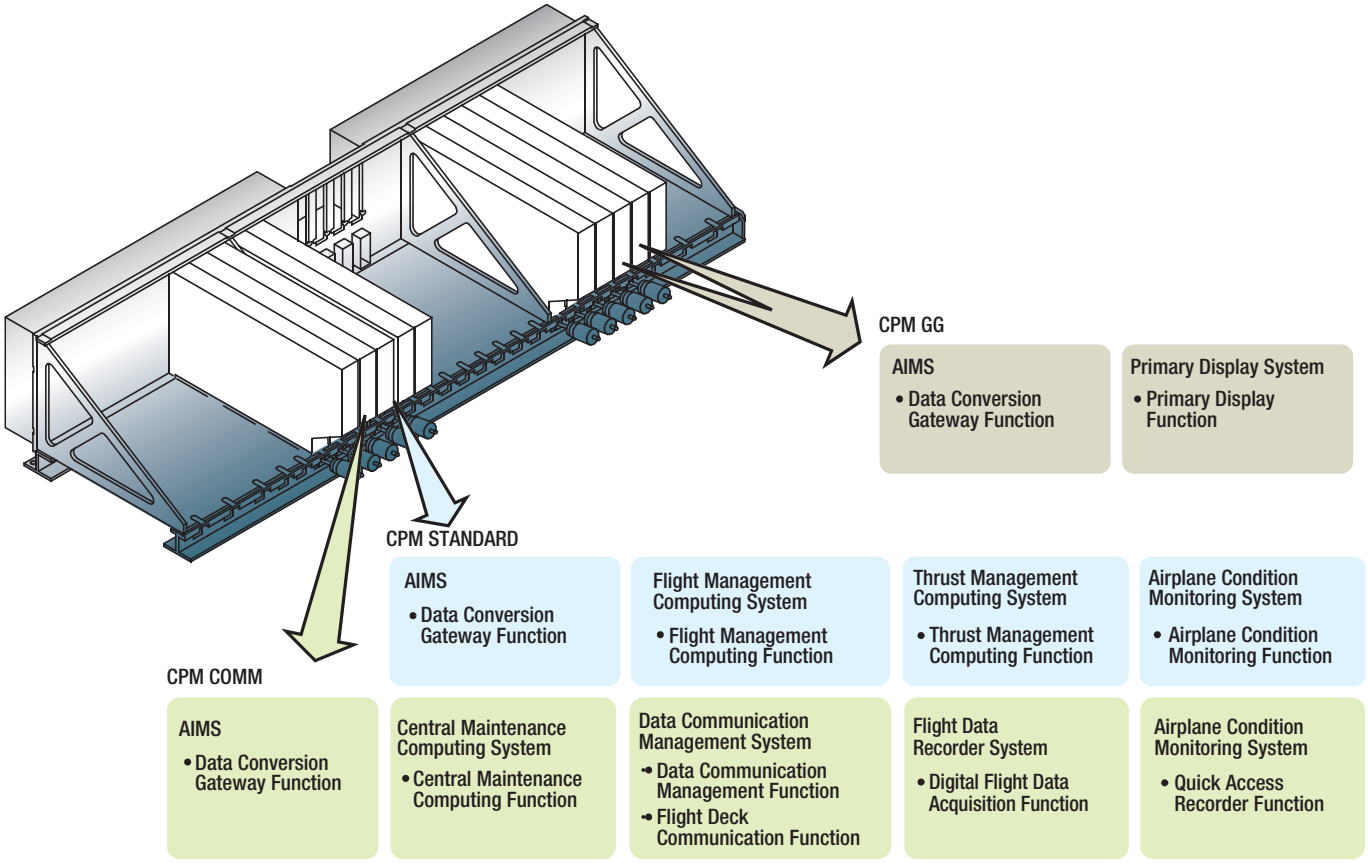


Figure 20-8. Boeing 777 Airplane Information Management System (AIMS) cabinet.

BOEING 777 AIMS

An example of an aircraft that uses integrated modular avionics is the Boeing 777. In this aircraft, Boeing calls its IMA system the airplane information management system (AIMS). The core system of the 777 AIMS system is contained in two cabinets which are located in the aircraft's main equipment center.

Each of the AIMS cabinets has ten line replaceable modules (LRMs) which plug into it. These modules are of four different types. The cabinets themselves contain network components and circuitry that is shared by all the LRMs installed in each cabinet. This common circuitry is contained in a "backplane bus." The cabinets are also networked with other LRUs in the aircraft via digital data buses.

In the 777, the Line Replaceable Modules in the AIMS process information for the following avionics systems:

- Primary Display System (PDS),
- Flight Management Computer System (FMCS),
- Thrust Management Computer System (TMCS),
- Central Maintenance Computer System (CMCS),
- Airplane Condition Monitoring System (ACMS),
- Data Communication Management System (DCMS)
- Flight Data Recorder System (FDRS).

The AIMS is interconnected with many other units in the aircraft. It receives input data from these units, and it provides output data to them. (***Figure 20-9***) The AIMS uses several methods for sending and receiving information. These include six different digital data

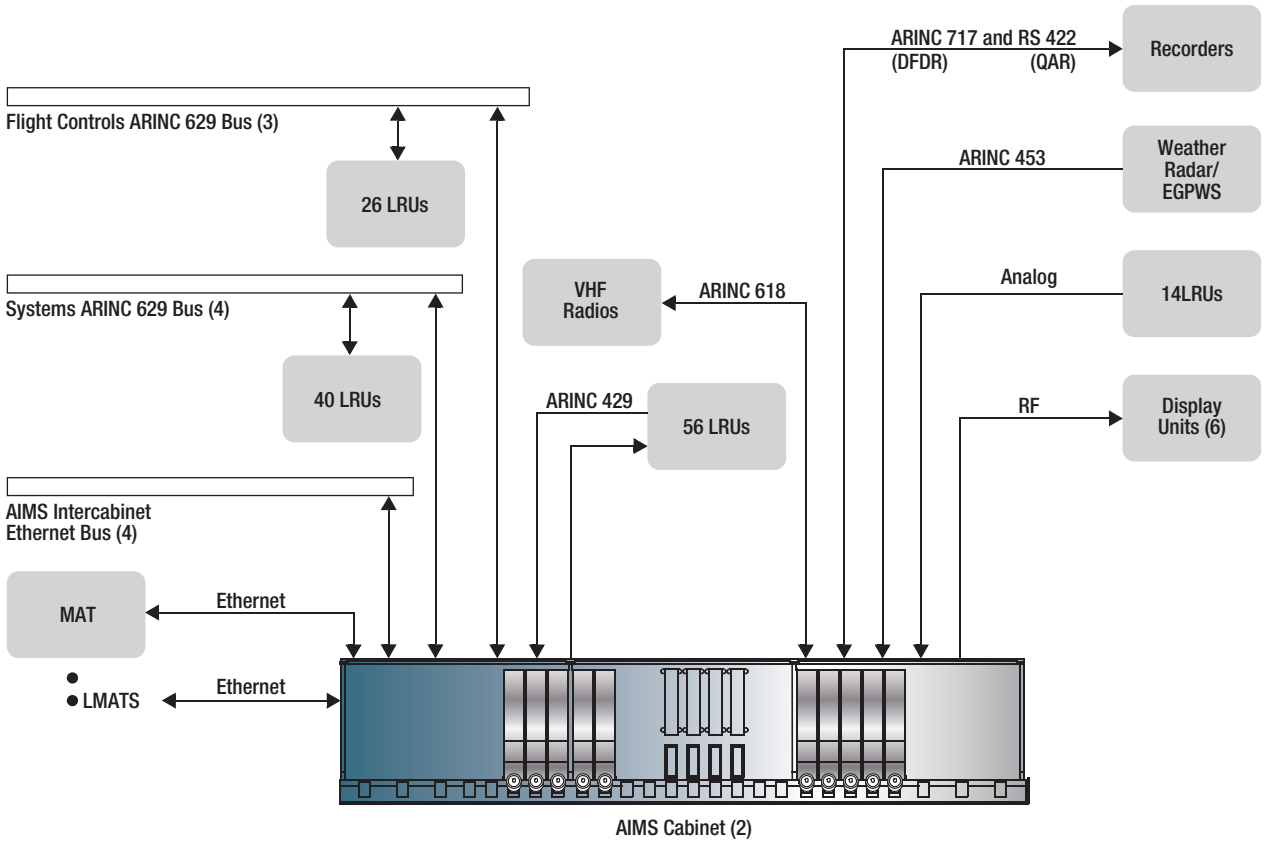


Figure 20-9. Boeing777 AIMS Interfaces.

transfer formats, analog data transfer, and wireless RF links. The primary data transfer format used by the AIMS is ARINC 629. The AIMS is connected to 66 LRUs located throughout the aircraft via ARINC 629. ARINC 629 uses a bi-directional data bus. This means that data can be sent both ways on the same data bus. The AIMS can both send information to an LRU, and receive information from that LRU, using a single data bus connected between the two. ARINC 629 uses a more rapid data transfer rate (up to 2 Mbps) than some other digital data transfer systems.

The AIMS also uses the ARINC 429 data format to communicate with 56 LRUs throughout the aircraft. Data transfer using ARINC 429 is unidirectional. This means that information can only be transferred in one direction on a single ARINC 429 bus. If an LRU needs to send information to the AIMS, one bus is required.

If the AIMS needs to send information back to that LRU, a second bus is required. ARINC 429 data transfer rates are somewhat slower than those used with ARINC 629. The transfer rates range from 12 kbps to 100kbps. ARINC 429 is a widely used data transfer system that was first developed in the 1970s. It is used in many makes and models of aircraft. The later developed Boeing 787 IMA system uses fibre-optic Ethernet that connects all systems with the CCS using AFDX/ARINC 664 protocol. Computer processors use ARINC 653 operating systems.

The later developed Boeing 787 IMA system uses fibre-optic Ethernet that connects all systems with the CCS using AFDX/ARINC 664 protocol. Computer processors use ARINC 653 operating systems.

The AIMS displays information on six flat-panel display units (DUs). (***Figure 20-10***) These units are located on the instrument panel in the cockpit. They show the information that has been gathered from the various systems connected to the AIMS, and which has been processed by the AIMS. These DUs are the main displays used by the flight crew.

The DUs give the following types of displays:

- Primary Flight Display (PFD)
- Navigation Display (ND)
- Engine Indicating/Crew Alerting System (EICAS)
- Multifunction Display (MFD)

Switches are provided in the cockpit that allow the

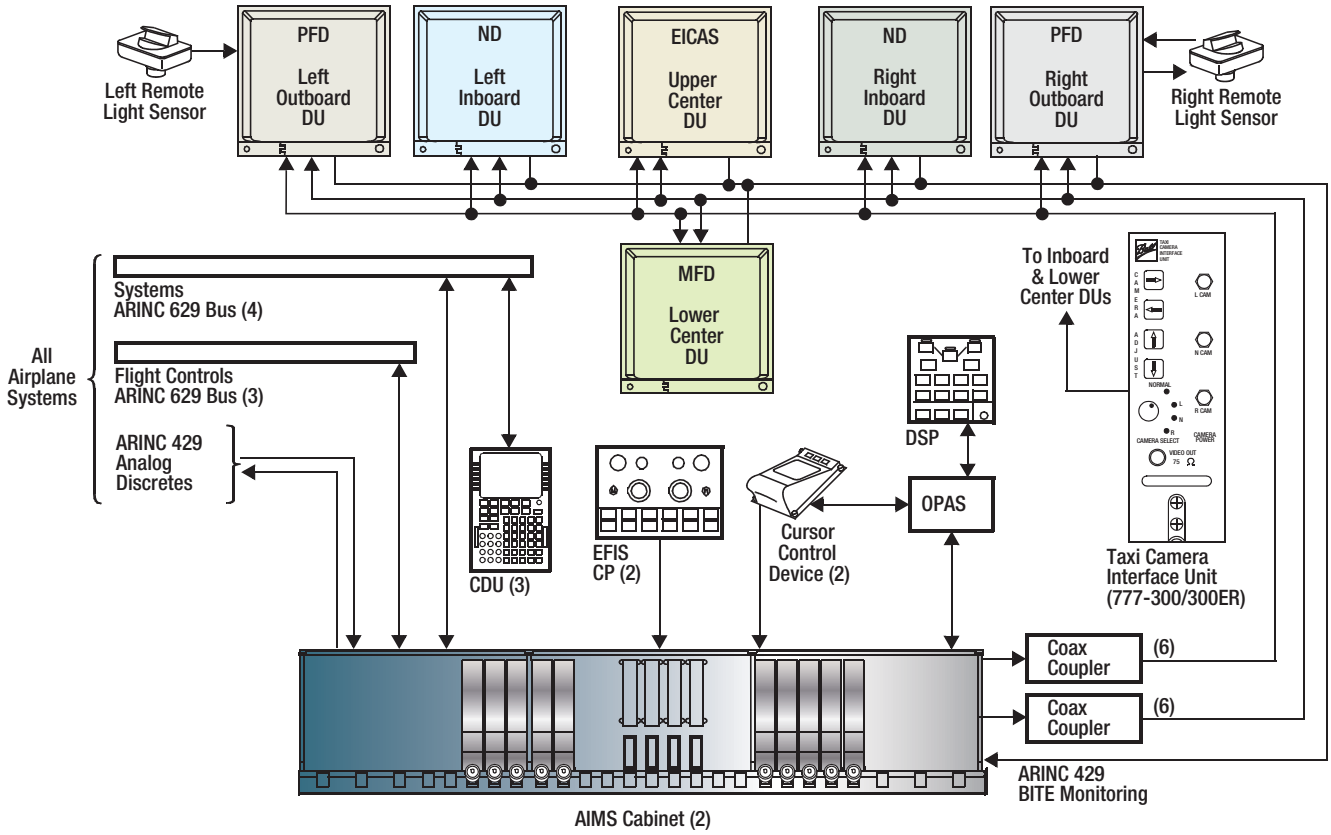


Figure 20-10. Boeing 777 Primary Display System, showing Display Units (DUs).

flight crew to control the DUs. These switches allow a particular type of display to be shown in different positions on the instrument panel. If one of the DUs fails, its function can be taken over by another functional DU. The system thus provides a great deal of flexibility.

Figure 20-11 shows a DU operating as a PFD. It shows aircraft pitch and roll attitude, airspeed, altitude, vertical speed, heading, and other information.

Figure 20-12 shows a DU operating as an ND. The ND can display data in four different modes. These modes are VOR, Plan, Map, and Approach. The displays are laid out differently for each mode, and different information is displayed in different modes. The ND shows heading, VOR, DME, localizer, glideslope, TCAS, along with other information.

Figure 20-13 shows a DU operating as an EICAS display. This display shows engine operating parameters such as EPR, N1, EGT, etc. It shows fuel quantity, landing gear position, flap/slat position, and other aircraft system information. It also provides warnings, cautions and advisories to the flight crew.

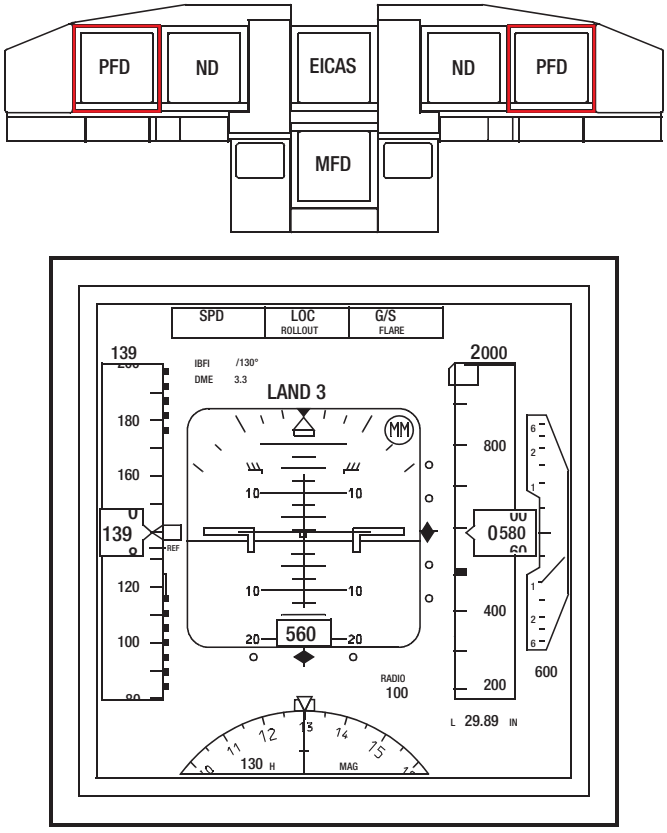


Figure 20-11. Display Unit operating as a Primary Flight Display (PFD).

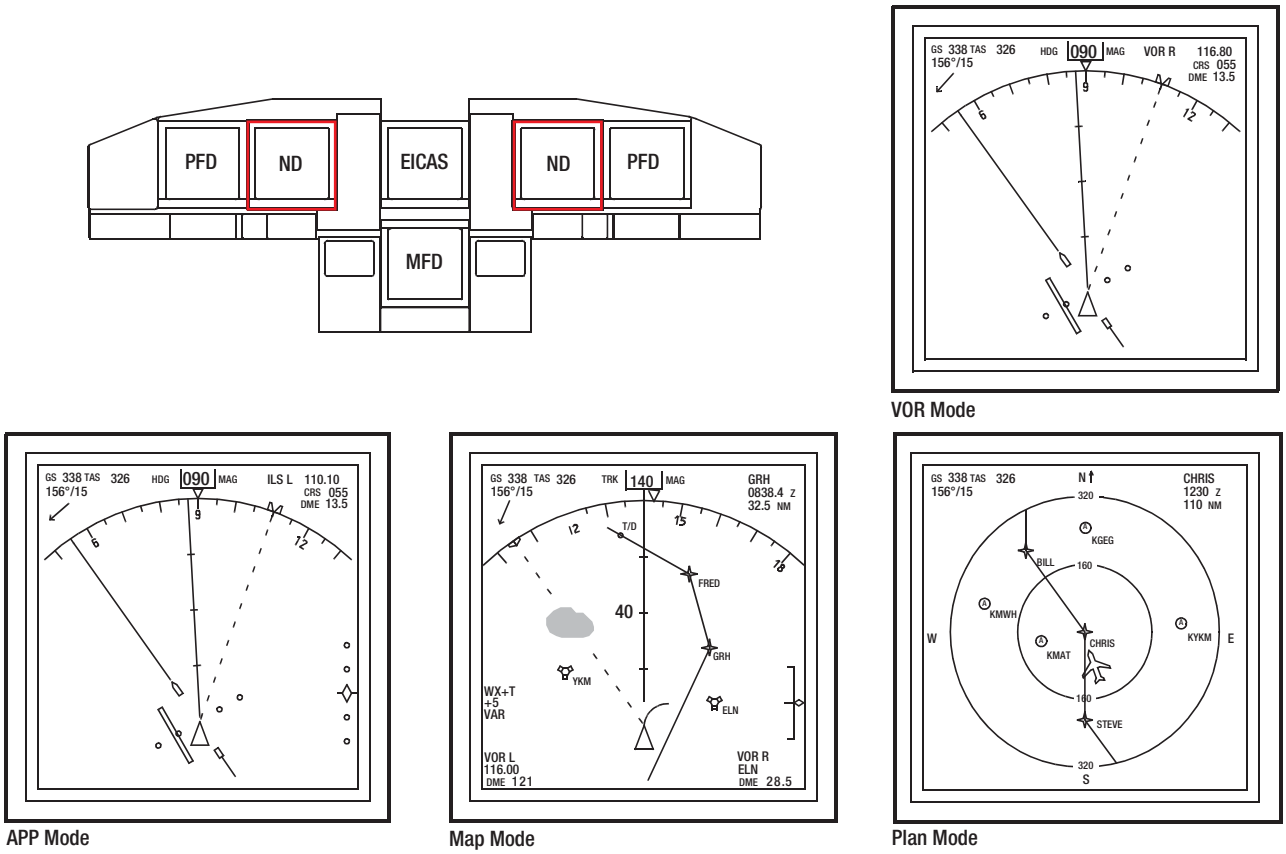


Figure 20-12. Display Unit operating as a Navigation Display (ND).

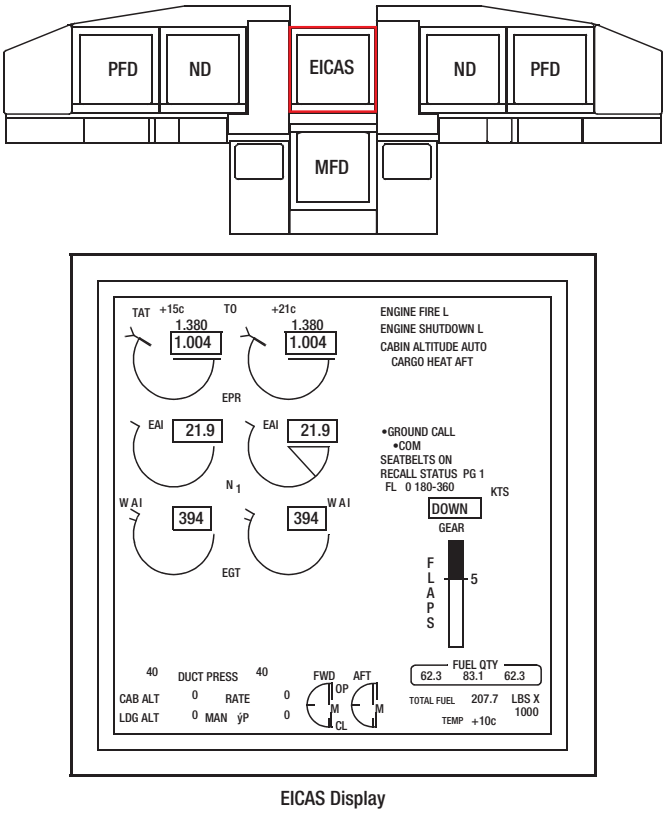
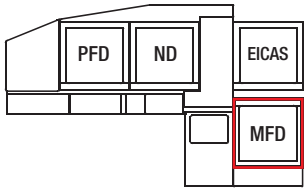
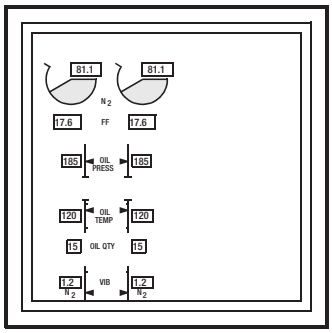


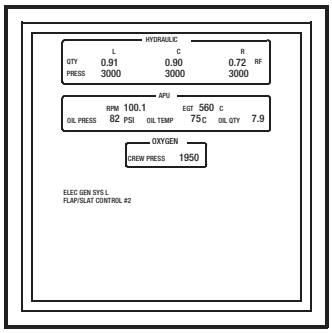
Figure 20-13. Display Unit operating as an Engine Indicating and Crew Alerting System (EICAS) display.

Figure 20-14 shows a DU operating as an MFD. The MFD can display many different kinds of information. Different screens (formats) can be selected. These formats are Secondary Engine Display, Status Display, Synoptic Display, Maintenance Page, Communication Display, Electronic Checklist, and Ground Maneuver Camera Display. The Boeing 777 AIMS is one implementation of an integrated modular avionics system.

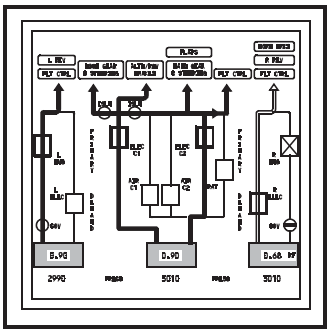




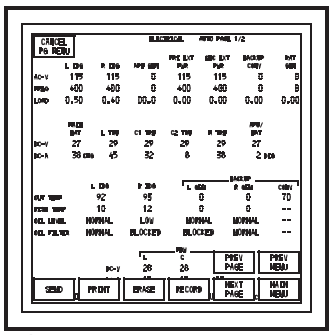
Secondary Engine Display



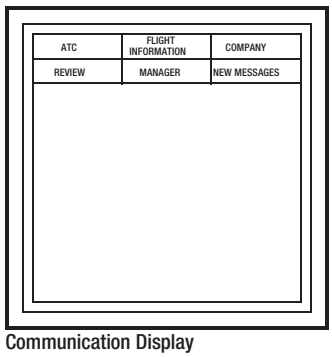
Status Display



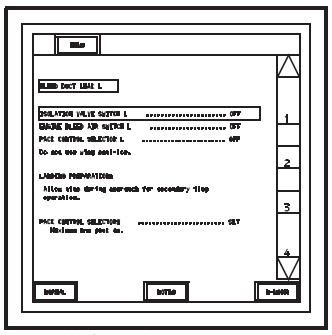
Synoptic Display
(Typical)



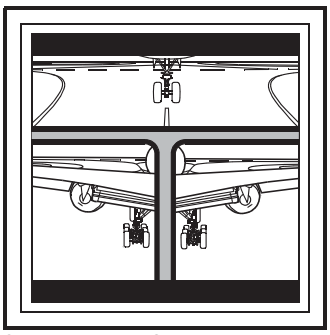
Maintenance Page
(Typical)



Communication Display



Electronic Checklist



Ground Maneuver Camera
System Display (777-300/300ER)

Figure 20-14. Multi-Function Displays (MFDs).

OPEN IMA CONCEPT

This IMA concept was developed for the Airbus 380. It incorporates numerous computers with individual functions, connected in a network, hence it's not reliant on a single CCS to operate the aircraft's equipment. Eight computer processing modules, having similar architecture, memory and power supply characteristics, are utilized in a network connection using AFDX (Avionics Full Duplex Switched Ethernet) - ARINC 664. AFDX is capable of 100 Mbits/sec. and it's operation is based on a star topology using common switched modules to transfer data between systems.

CPIOM (core processor input/output modules) using ARINC 653 for the internal data transfer format, are interfaced with each computer to host various applications, and provide signal acquisition and transmission. Each CPIOM is different, depending on the aircraft system it's specific computer interfaces with. A single computing module requires levels of partitioning so that the performance of each application system is not affected by any other. (***Figure 20-15***)

These partition boundaries are achieved by:

- Timing; each application has strict periodic allocation timing dependent on its importance
- Memory; a specific application can only access parts of memory that have been configured for it. Internal monitoring detect and prevent violations.
- I/O segregation provides allocation of channels and communication paths for each application.

Advantages of partitioning include:

- System independence; thus ease of modifications and testing
- Ease of configuration

Other aircraft's IMA systems vary somewhat in their specifics. The exact system architecture and the terminology used varies from one manufacturer to another, and from one model to another. However, the basic features of IMA found in any IMA system are: (***Figure 20-16***)

- Modular design
- Integration of modules within the system
- Use of digital data buses for interconnection
- Fault tolerance
- Flexibility

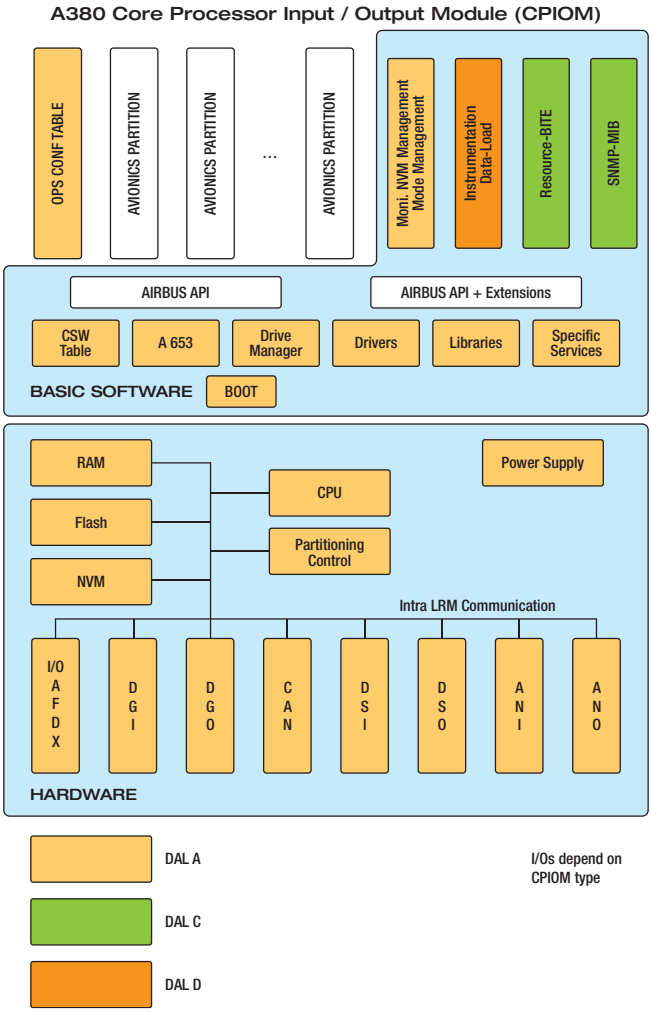


Figure 20-15. A380 basic netwwork system.

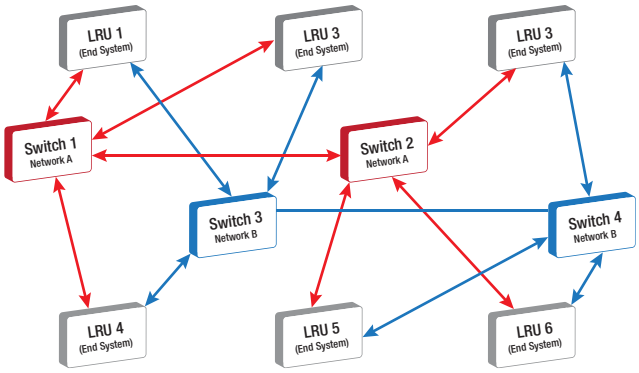


Figure 20-16. A380 basic network system and AFDX star topology system.

QUESTIONS

Question: 20-1
AFDX in comparison to other data transfer formats, such as ARINC 629 and ARINC 429, has?

Question: 20-2
AFDX data buses are used on?

Question: 20-3
A format that uses duplex switching for data transfer between modules is?

Question: 20-4
Partitioning of Core-processor computers in IMA systems is?

Question: 20-5
An advantage of partitioning in an open loop concept IMA system is?

Question: 20-6
A "Backplane" is a?

Question: 20-7
The internal data transfer format for core-processor computers in IMA systems is?

Question: 20-8
IMA systems can be categorized into functions of?

ANSWERS

Answer: 20-1
Much higher data capability, reliability and fault tolerance.

Answer: 20-2
Open systems and CCS systems.

Answer: 20-3
AFDX

Answer: 20-4
Isolation of each application system to prevent interaction.

Answer: 20-5
Ease of configuration.

Answer: 20-6
System that is used for inter-module communication within a cabinet.

Answer: 20-7
ARINC 653.

Answer: 20-8
Cabin, utilities and cockpit.



PART-66 SYLLABUS **LEVELS**

CERTIFICATION CATEGORY → **B2**

Sub-Module 21

CABIN SYSTEMS (ATA 44)

Knowledge Requirements

13.21 - Cabin Systems (ATA 44)

The Cabin Network Service may host functions such as: 3

Access to predeparture/departure reports,

E-mail/intranet/internet access,

Passenger database;

Cabin Core System;

In-flight Entertainment System;

External Communication System;

Cabin Mass Memory System;

Cabin Monitoring System;

Miscellaneous Cabin System.

INTRODUCTION

Cabin Systems include those used to communicate both within the aircraft cabin and between the cabin and the cockpit. They also include systems for in-flight entertainment (IFE) of passengers, and systems used by passengers to communicate with the ground.

Like most aircraft systems, cabin systems have evolved and upgraded over the years. Older aircraft used analog electronics, while newer designs use digital electronics. In the case of IFE video, early systems used video projectors and large screens visible to all passengers. Video was stored on reels, and later on videocassettes. There was only one choice of video to watch. Newer aircraft are equipped with individual flat-panel video displays at each seat, and video is stored digitally. Passengers can select from many different video sources. Early systems used pneumatic headsets for passenger IFE audio; modern aircraft use electronic headsets.

As with all complex systems, there are variations in cabin systems. Different models of aircraft have different systems and different features. Some of the features that can be included in cabin systems are:

- Passenger Address - for addressing passengers through overhead cabin speakers.
- Cabin Interphone - for communication among the cabin crew members, and also between the cabin and the flight deck.
- Flight Attendant Calling - allows passengers to call for assistance from the cabin crew, Audio and Video Entertainment.
- Moving Map Displays - showing the aircraft's current location, altitude, and airspeed,
- Telephone, Fax service, and Internet Service.

CABIN INTERCOMMUNICATION DATA SYSTEM

A cabin intercommunication data system is used by flight attendants, pilots, and passengers. Typical user interfaces for the system are flight attendant panels, cabin handsets, cockpit handsets, and flight attendant call buttons at passenger seats, here are also speakers and passenger information lights/signs.

Flight attendants can use the system to call each other in various parts of the cabin. For example, a flight attendant located at the galley at the front of the cabin can call another flight attendant who is at the rear of the cabin. Also, the pilots on the flight deck can call the flight attendants at their stations in the cabin, and vice versa. Flight attendants can make general announcements to the passengers. Passengers can activate flight attendant call lights. An example of a cabin intercommunication system is described below under 777 Cabin Services System.

PA announcements, made through Cabin speakers and passengers headsets, over-ride all other passengers audio and entertainment. Pilot's PA announcements have priority over Cabin Crew PA announcements for obvious safety reasons. "Chimes" are a sound similar to a low frequency bell and are used to gain the attention of Cabin Crew through the cabin speakers.

CABIN NETWORK SERVICE

A cabin network service is a digital system that is typically hosted on a server within the aircraft. It provides services, such as access to email accounts and the internet. It may also provide access to data stored in databases. A cabin network service is essentially a computer Local Area Network (LAN) within the aircraft, one that can interface with external networks.

The interconnections within the LAN can be wired, or they can be wireless. The design trend is toward wireless connections. Because of the potential for interference with other systems, the use of wireless is restricted to certain phases of flight. Operation of wireless systems is not permitted during takeoffs and landings (refer to Wi-Fi section in this sub-module).

Although systems vary, and new features are added to each new version, typical uses of the cabin network service are for passengers to connect to the internet while in flight, and to access In Flight Entertainment (IFE). In fact, on some aircraft the cabin network service is integrated with the IFE into a single system. Some systems also permit interconnection to satellite communications systems for in-flight telephone calls.

Cabin Networks can transfer large volumes of data to many different locations by using multiplexing and demultiplexing techniques, which are described below under In Flight Entertainment. Multiplexing reduces the quantity of wiring needed, and allows advanced

IFE generation systems to be easily developed and installed in aircraft.

All IFE systems fitted to modern airliners must be designed in accordance with relevant safety regulations. In the advent of failure, IFE systems must not interfere with the function of any other aircraft systems; they are isolated from other electrical systems and it is common for an IFE system master switch to be located in the cockpit. Installation of IFE systems must therefore be carried out to ensure that the protection and integrity of all other aircraft systems is maintained and that failure of IFE systems will not affect any safety aspect or function of the aircraft. Future concepts for IFE include live television reception with on demand capabilities.

CABIN CORE SYSTEM

777 CABIN SERVICES SYSTEM

A specific example of an aircraft with a multi-featured cabin system is the Boeing 777. In the 777, the system is called the cabin services system (CSS). It includes both the cabin intercommunication data system and the cabin network service. ***Figure 21-1*** shows a block diagram of the 777 Cabin Services System.

As can be seen in ***Figure 21-1***, the system is digital and makes extensive use of digital data buses. The system uses three different data bus systems: ARINC 629, the CSS Intersystem Bus, and ARINC 628. The ARINC 629 bus connects to the aircraft information management system (AIMS), which is the primary information management system for the aircraft. The CSS intersystem bus is a data transfer bus internal to the cabin system. ARINC 628 is a data interface specifically designed for use with in-flight entertainment (IFE) systems. There are several available IFE systems for an operator (airline) to choose from, and the 777 can accommodate any IFE system that uses the ARINC 628 interface.

The central processor of the system is the cabin systems management unit (CSMU). This unit is connected to the aircraft's main ARINC 629 data bus. The CSMU connects to the IFE system through a different data bus - the ARINC 628 bus. There is also a passenger address /cabin interphone (PA/CI) controller, which the cabin crew uses to control the passenger address (PA) and cabin interphone (CI) functions. The cabin systems control panel (CSCP) is an interface used to load data into the system. The aircraft's cabin is divided into zones. Each zone has a zone management unit (ZMU), which is a processor connected to the rest of the system through a data bus called the "CSS Intersystem Bus." Each ZMU is controlled by a cab in area control panel (CACP).

Cabin attendant handsets (CAHs) are connected to the ZMUs. These handsets are similar to telephones. They are used by the flight attendants to communicate with each other, and with the flight crew. They are also used for making announcements to the passengers. (***Figure 21-2***)

Each ZMU is connected to a series of overhead electronic units (OEUs) within its zone. At each passenger seat there are reading lights and attendant call lights. These lights are connected to the OEUs, which in turn are connected to the ZMU for the entire zone.

IN-FLIGHT ENTERTAINMENT (IFE) – BOEING 777

Each passenger seat in the 777 also has IFE equipment. The IFE equipment typically includes a video display, an audio headphone jack, and controls for selecting specific video or audio content to be enjoyed by the passenger. It can also include other items, such as a credit card reader, or a port for connecting the passenger's own personal equipment to the system. The system may allow for connection of a device to the internet.

These functions are provided by a central source. This central source may be called the main IFE computer. As stated previously, various versions of IFE systems are available from various manufacturers, and their terminology and features differ somewhat.

In the 777 system, the main IFE computer is connected to the Cabin Services System via an ARINC 628 data bus. The data routes from the IFE computer, through the CSMU and the ZMUs, to the individual passenger seats. This is a two-way connection. Passenger requests for particular content (e.g. specific audio, video, games, etc.) are sent to the main IFE computer, and the content is then sent from the main IFE computer to the passenger.

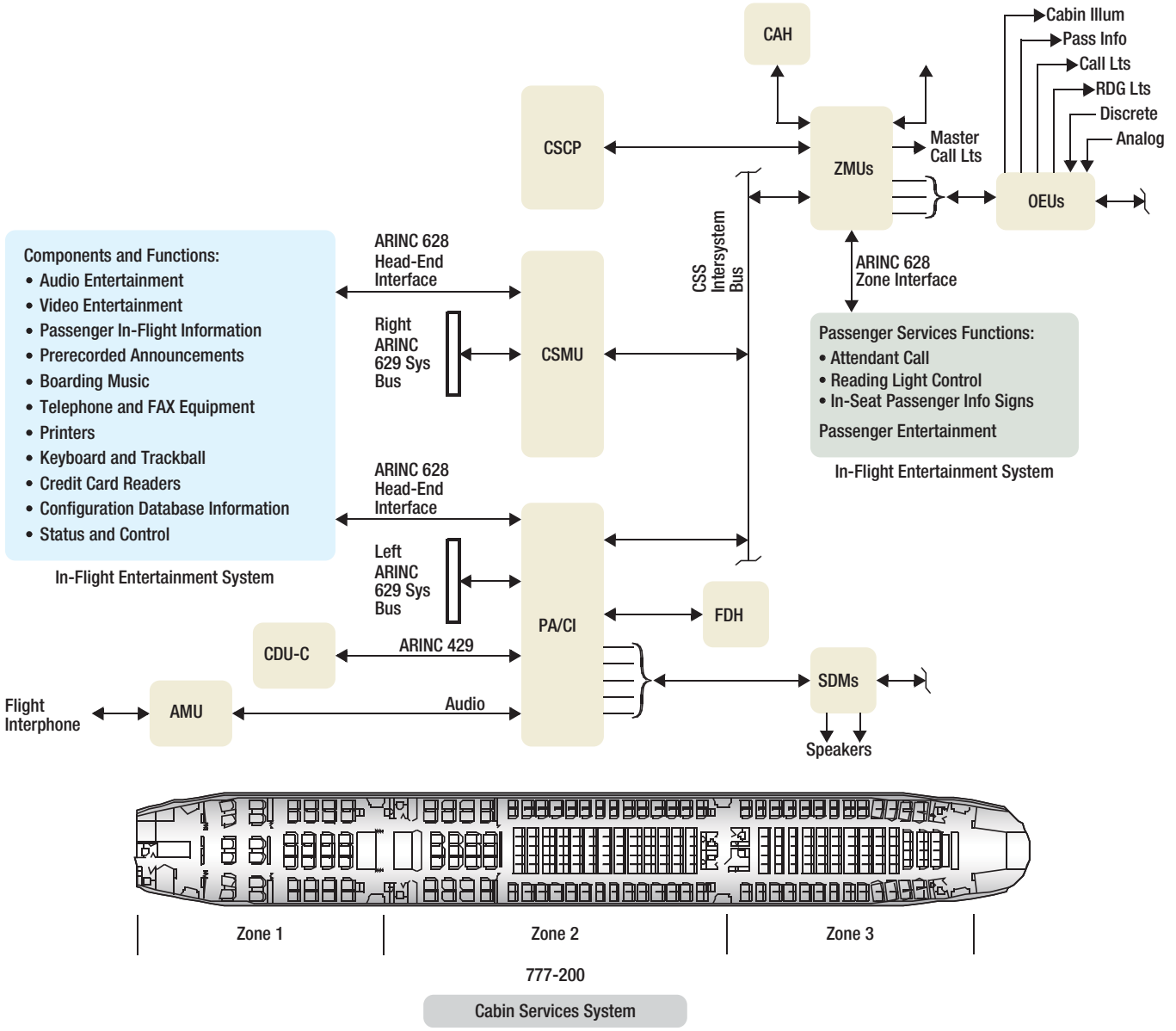


Figure 21-1. Cabin services system.

In flight, the main IFE computer is constantly receiving requests for different content from different passengers, and it must route the requested content to the correct passenger. This content, which may be audio, video, text, or something else, has been stored as digital data. It will be transferred as a data stream. The computer may be called upon to send out a large number of different data streams to the various passenger seats, all at the same time. This is accomplished by breaking up the data into small packets, attaching an address to each packet, then sending the packets at high speed down the data bus. The addresses indicate which passenger seat will receive which data packets. The data packets for all the seats are placed onto the data bus. The order in which they are sent does not matter, because the addresses will allow them to be sorted out at the receiving end. This process is known as multiplexing.

At the receiving end, the data packets are sorted by address. All the data packets labeled with a particular address (for a particular passenger) are reassembled. This is known as de-multiplexing. The data is converted back into audio, video, text, or whatever content it originally represented and delivered to the passenger. Each passenger receives only the data that was addressed to his or her seat. A high-speed digital data transfer system can multiplex and de-multiplex many channels, and carry information to many destinations simultaneously.

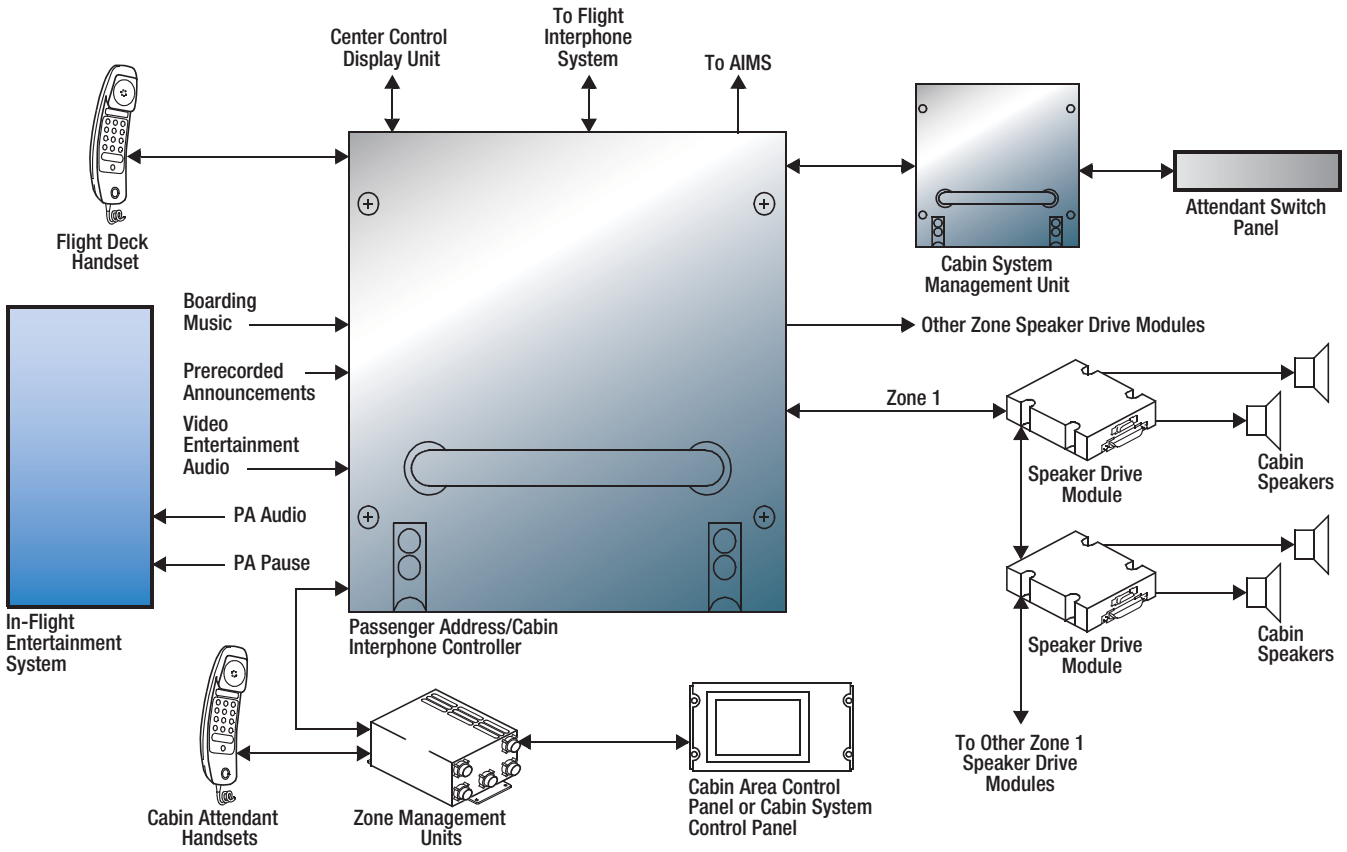


Figure 21-2. Cabin interphone and passenger address functions.

EXTERNAL COMMUNICATION SYSTEM

Although a large amount of data can be stored on board the aircraft and accessed by the cabin system, it is also useful to communicate with points outside the aircraft. This involves connecting with the internet, and with the telephone system. To do this, the aircraft's external communication system is used to connect with either a ground-based network or a satellite-based network. There are several commercial networks available, and each of them has advantages and disadvantages. Changes and upgrades to these networks occur frequently.

Direct communication from the aircraft to the ground is used if the sender/receiver links are within line-of-sight range. Network systems automatically decide the routing at specific time communication is taking place. There are systems that are designed for just direct ground to air, for example, if an aircraft is operated just in one defined geographical location. Depending on the systems employed, ground based is faster than Satellite – it also must be expected that, generally, all air Internet services will be slower than ground domestic/business systems. However as technology concepts expand, it can be expected that this difference in data speed, between systems, will progressively become less. Using two antenna on aircraft, one up-link and one down-link, enables the data speed to be dramatically increased.

For connecting with the external networks, the aircraft uses a router that is connected to dedicated radio transmitting and receiving equipment. This radio equipment uses various frequencies, depending on which external network is being accessed. Antenna size and shape also varies according to the network being used. For satellite-based systems, the antennas must be mounted to the upper surface of the aircraft, typically on the top of the fuselage or in the tail section. For ground-based systems, they will be mounted on the lower surface of the aircraft, usually on the belly of the fuselage.

Ground-based networks use radio towers to communicate with the aircraft. These towers have a limited range, so ground-based systems can provide coverage only over limited geographical areas. Ground-based systems tend to have a wide bandwidth, allowing for higher-speed data transfer. This makes web pages load faster, and

results in good-quality audio and streaming video. Satellite networks, while providing a larger coverage area than ground-based networks, tend to operate at somewhat slower speeds, and with narrower bandwidths than ground-based networks. Several satellite systems are available for use by aircraft systems. The coverage areas and capabilities of these systems vary.

The Inmarsat system uses 4 geostationary satellites, which orbit at an altitude of approximately 22 000 miles (approximately 36 000 km) in equatorial orbits. The coverage is over about 95% over the earth, with minimal coverage in the polar regions. It was the first system to be put into operation for voice/data links. Until it's uniform world-wide coverage is able to be utilized, other systems, such as Iridium, are generally preferred. However the Inmarsat system has very good reliability and connectivity and is relatively cheap.



Figure 21-3. Iridium satellite network.

The Iridium system uses low-earth orbit (LEO) satellites, which orbit at around 500 miles (800 km) altitude, and has world-wide coverage capability. Its characteristic low earth orbit ensures minimal voice/data delay, in comparison to geostationary higher orbit satellites.

Iridium utilizes a constellation of 66 satellites, evenly spaced in 6 orbital planes (11 satellites equally spaced in each plane) in near-polar orbits. Horizon-to-horizon time for each satellite is approximately 10 minutes – automatic linking, or hand-over, between satellites ensures continual data connectivity for users. The system operates in the L-band: 1616 to 1625.5 MHz for aircraft to satellites links. Cross-links between satellites is in the K Band: 23.18-23.38 GHz. Data is linked to the nearest satellite, and then relayed between satellites until the optimum, or overhead, satellite is able to link directly to the aircraft. A similar linking procedure occurs in the opposite direction. (***Figure 21-3***)

The main advantage of the Iridium system is the world-wide coverage capability, but it can have disruptive connections and is more expensive, relative to the Inmarsat system.

CABIN MASS MEMORY SYSTEM

The cabin mass memory system is where the data that is used in the cabin is stored. The mass memory system can be programmed with cabin configuration information. This is information about the way the seats are configured, for example, which seats are first class, and how many zones the cabin is divided into. The cabin configuration programming varies, even the same make and model of aircraft can have several possible cabin configurations.

The mass memory system can store many types of data files. This includes audio and video files that are used for passenger briefing and announcements, and can include IFE such as pre-recorded music. An aircraft's mass memory system is accessible to the cabin crew through user interfaces (control panels). Although it is stored in a central location, data from the mass memory system can be transferred to access points around the cabin using digital data buses. The system can be updated when necessary through a data loading system.

CABIN MONITORING SYSTEM

Cabin monitoring systems are used to monitor various conditions in the passenger cabin. These conditions can include (among others): Cabin temperature, lighting, status of passenger warning lights such as "Fasten Seat Belts" and "No Smoking" lights, cabin entry and exit door status (open or closed), Status of smoke and fire detectors in lavatories and elsewhere in the cabin, and Status of galley equipment, such as water tanks. Information about the conditions being monitored can be accessed by flight attendants in real time. It can also be stored and accessed later for purposes such as troubleshooting and trend monitoring.

MISCELLANEOUS CABIN SYSTEMS

Cabin systems, particularly in-flight entertainment systems, are the subject of intensive commercial development. New systems are being marketed, and some existing aircraft that were built without modern cabin systems are being retrofitted with them. Because these systems are highly visible to an airline's customers (its passengers), they can affect the customers' choice of airline. Since the airline industry is highly competitive, further refinements and improvements have recently occurred in the areas such as Cabin WI-FI.

AIRCRAFT WI-FI

Cabin Wi-Fi, allows passengers to connect their own personal devices to an on-board Wi-Fi system. Such Wi-Fi systems are becoming more common, through installation on new aircraft and also through retrofitting older aircraft. Secure connections are achieved using encryption for passengers that require access to their own specific sensitive data. Satellite links are utilized for world-wide coverage. (***Figure 21-4***)

Advantages of using WI-FI inside an aircraft cabin in comparison to hard-wired aircraft systems include:

- Higher speed, easier and improved access to data and information.
- Better accessibility throughout the aircraft cabin.
- Hard points are eliminated (Ethernet) for connection to the internet.

However, aircraft Wi-Fi service to the World Internet is still in its growing and developing phase, in technology and acceptable commercial models.

Limited bandwidth is a major consideration, typically only 10Mbps, using a ground based system with dual modems and antennas. Note that this bandwidth is for the entire aircraft, not individual passengers, so obviously the general concept of video streaming is not yet an acceptable consideration. The bandwidth reduces considerably when satellite based systems are required, such as flying over the ocean. Future developments would include Satellites having expanded bandwidth capabilities in the Ka Frequency Band to accommodate data.

EXTERNAL CAMERAS

Outside video cameras are becoming a very popular feature on modern airliners. They essentially provide a real-time presentation of the aircraft's position for passengers. "Forward looking" and "down looking" cameras are commonly used and the video is usually incorporated in other flight data as part of the individual passenger network displays.

Figure 21-5 shows a typical external camera arrangement which is used on a rotorcraft for cargo sling operations. This underside cowling covers two separate cameras that can be switched for views of the: cargo attachment, (lens shown); or looking directly downwards, (lens hidden from view).

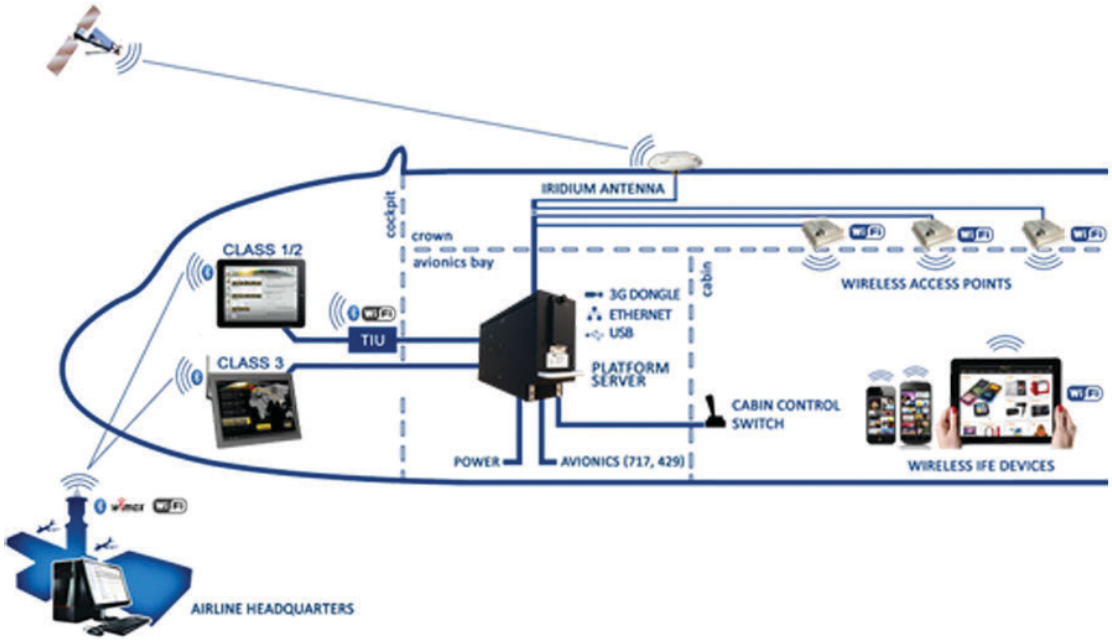


Figure 21-4. A typical components required for an installed aircraft Wi-Fi system.

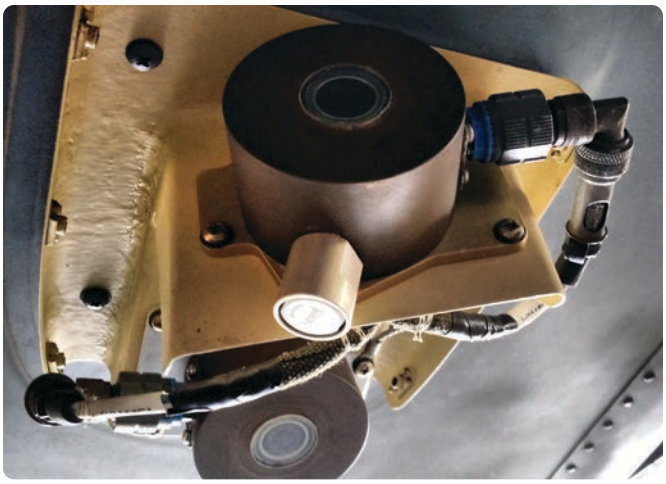


Figure 21-5. A typical FLIR external camera arrangement.



Figure 21-6. FLIR module on the underside of a Rotorcraft.

FLIR

Other applications of external cameras are FLIR (Forward Looking Infra Red). This equipment is able to take still photos, video, and utilize infra-red pictures for night vision. A dedicated crew member (FLIR operator) is invariably required for the operation of the equipment. Additional support equipment employed can be digital recording for onboard disc cards for analysis of pictures and video after flights. Applications for FLIR is mainly for law enforcement and defense agencies, for obvious reasons.

Figure 21-6 shows a FLIR module on the underside of a Rotorcraft near the nose. When switched off the camera lens is rotated back within the module for protection.

Figure 21-7 shows the operators position and associated equipment with viewing screen. Hand held controls (similar to commercial video game apparatus complete the operating system for manipulating the FLIR.

PICTORIAL MOVING MAP DISPLAYS

Other information available, as part of the flight data network, include Moving Map displays. These provide a real-time position of the aircraft relative to it's geographical location, with additional information such as: altitude, speed, direction, relative prominent ground features and similar data.



Figure 21-7. The FLIR operators position and associated equipment with viewing screen.

Question: 21-1
A function test of passenger audio reception, for fault diagnosis or routine maintenance, would ensure that the priority reception order is?

Question: 21-2
An aircraft, utilizing both a Wi-Fi and hard wired network services, is observed to have an abnormal slower data speed. The most probable cause is?

Question: 21-3
Total failure of a Cabin Services System is likely to be caused by?

Question: 21-4
After the installation of a modern IFE system, the most important post installation test would be to ensure that?

Question: 21-5
Testing of a suspect ARINC 628 data bus would require?

Question: 21-6
Passenger Wi-Fi cabin network service, in comparison to hard wired data connections, have _____ data speed.

Question: 21-7
Typical data speed for an aircraft Wi-Fi linked to a ground based system is ___ Mbps.

Question: 21-8
The aircraft to satellite frequency band for the Iridium satellite system operates in the _____ band.

ANSWERS

Answer: 21-1
Pilot PA, cabin crew PA, entertainment.

Answer: 21-2
The Wi-Fi system has failed.

Answer: 21-3
The cabin systems management unit.

Answer: 21-4
Complete isolation of the IFE system and integrity of other aircraft systems is maintained.

Answer: 21-5
Simulating data flow in both directions.

Answer: 21-6
higher.

Answer: 21-7
10.

Answer: 21-8
L.



PART-66 SYLLABUS **LEVELS**

CERTIFICATION CATEGORY → **B2**

Sub-Module 22

INFORMATION SYSTEMS (ATA 46)

Knowledge Requirements

13.22 - Information Systems (ATA 46)

The units and components which furnish a means of storing, updating and retrieving digital information traditionally provided on paper, microfilm or microfiche. Includes units that are dedicated to the information storage and retrieval function such as the electronic library mass storage and controller. Does not include units or components installed for other uses and shared with other systems, such as flight deck printer or general use display. Typical examples include Air Traffic and Information Management Systems and Network Server Systems. **3**

Aircraft General Information System;
Flight Deck Information System;
Maintenance Information System;
Passenger Cabin Information System;
Miscellaneous Information System.

INFORMATION SYSTEMS INTRODUCTION

Aircraft operations and maintenance involve dealing with large quantities of information. This information must be stored in some manner, and, ideally, it should be capable of being accessed as efficiently as possible. Also, the information changes from time to time. To keep up with the changes, there must be a way to update the information stored in an information system.

Aircraft Information systems have evolved over the years. Initially, paper was the storage medium used. Manuals, drawings, charts, and other publications were available only in printed form. Information was retrieved by physically locating and reading the pages containing the information needed. Paper information sources have the advantage of being self-contained. No special equipment is needed to access the information. When revisions are required, new pages are printed and distributed. The old pages are removed and discarded, and are replaced by the new, revised pages.

Because large amounts of paper are bulky and heavy, other methods for storing large quantities of information were developed. Microfilm and microfiche were methods that saved space and weight. These methods involved using tiny photographic images of the pages contained in manuals, drawings, charts, etc. A disadvantage of microfilm and microfiche was that special equipment was needed to magnify the images so that they could be viewed. The microfilm or microfiche was useless if the viewing equipment malfunctioned, or if it was not available when the information needed to be accessed. Updates to microfilm and microfiche came in the form of new rolls of microfilm, or new microfiche sheets. After the new rolls or sheets were received, the old media was removed and discarded.

Digital computers represent a major advance in information technology. Modern computers can store and process very large amounts of information. They are compact, and they are very lightweight compared to other storage media, such as printed books. Computers store information in a type of memory known as RAM (randon access memory) and ROM (read-only memory). This RAM and ROM can be internal to the computer, as in a hard drive. It can also be portable, contained in compact disks, flash drives, and other forms. The information contained in the digital memory can be accessed through display screens. It can be transferred in and out of the computer on wired data buses and wirelessly, and it can be printed. When updates are needed to the information contained in the system, the memory can be electronically erased and written over. There is no need to physically remove and replace paper or film. Aircraft information systems can be used to store and retrieve many kinds of information for various users. Examples are flight deck information systems, maintenance information systems, and passenger cabin information systems.

BOEING INFORMATION SYSTEMS

FLIGHT DECK B777 ELECTRONIC FLIGHT BAG

An example of a flight deck information system is the Electronic Flight Bag (EFB), an optional system used on the Boeing 777 and on other aircraft. (***Figure 22- 1***)

The flight crew uses the EFB to access information that would traditionally have been found in various printed publications and carried aboard in a flight bag. Such publications include sectional charts and approach plates. The system also provides advanced capabilities beyond those available in printed publications.

On a 777 equipped with the EFB, the system displays information on the two Display Units (DUs) that are installed on the flight deck. One DU is for the Captain and one is for the First Officer. These two DUs are touch-screen devices, and operate independently of each other. The system can be navigated by the touch-screen DUs, and through keyboards and cursor-control devices. The DUs are connected to two electronics units (EUs) located on the airplane information management system (AIMS) rack in the main equipment center. The AIMS is an integrated system for processing information from many sources in the aircraft.

In the 777 system, the EUs send information stored in databases to the DUs through fiber-optic cables. The two DUs are connected to each other through wired connections. Databases accessible through the EFB

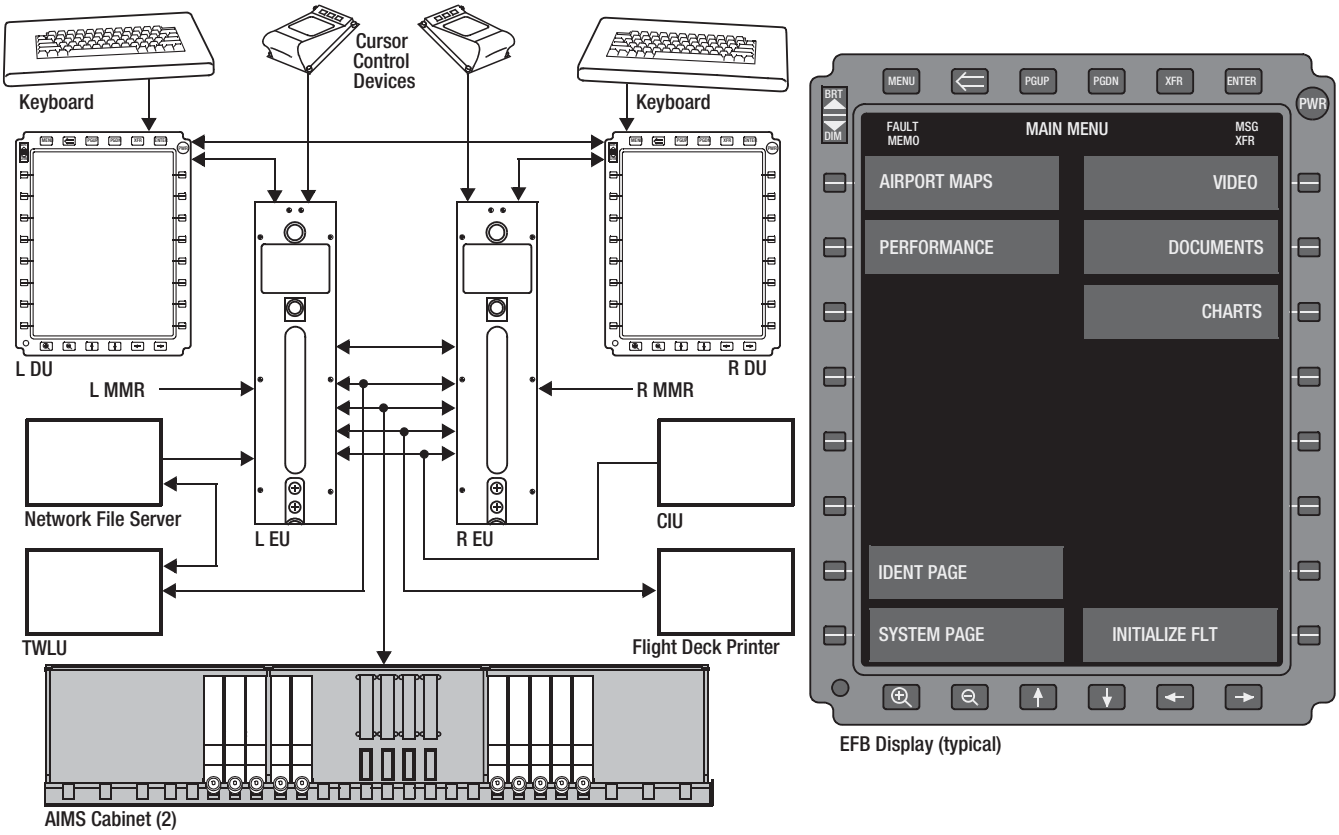


Figure 22-1. Electronic flight bag.

include aeronautical charts, airport maps and charts with real-time position monitoring, manuals, minimum equipment lists, and logbooks. The system can also display video. The EFB system receives data from the AIMS and from multi-mode receivers (MMRs). MMRs are GPS and Instrument Landing System receivers designed for use in instrument landings.

The MMRs provide aircraft position information that is extremely precise. This allows the EFB to pinpoint the aircraft's position on an airport map. The system also receives video from the Camera Interface Unit (CIU). This video is from the flight deck entry surveillance camera.

The 777 Electronic Flight Bag's databases can be updated through the aircraft's data loader, but they can also be updated wirelessly. For wireless updates, the system uses a Terminal Wireless LAN Unit (TWLU). The TWLU contains a radio transmitter and receiver that creates a Local Area Network (LAN) between the airplane and a ground-based network. This allows the databases to be updated wirelessly while the airplane is parked at the gate.

MAINTENANCE INFORMATION SYSTEM

The 777 EFB is one example of an aircraft information system. That particular system is used by the flight crew. However, the same kind of technology is also used by maintenance personnel. Flight crews need to access the EFB's information while in flight, but maintenance personnel need to access different kinds of information while the aircraft is on the ground.

Maintenance Information Systems work along similar lines as a flight deck information system (such as the EFB), but the information being stored is different. Instead of maps and charts, maintenance crews use maintenance manuals, illustrated parts catalogs, wiring diagram manuals, service bulletins, and other technical data. Maintenance Information Systems provide access to these publications electronically, just as Flight Deck Information Systems electronically provide the flight crew with access to maps and charts.

An advantage of using an electronic format, in addition to the space and weight savings it provides, is the ability to quickly locate the desired information. Instead of leafing through a large, paper maintenance manual, for example, a technician using a maintenance information system uses hyperlinks which allow easy navigation within the system. A typical method of achieving this is to use a menu containing links to each of the ATA 100 chapters within the manual. Within each chapter, the table of contents contains links that will quickly access a particular page. This allows the technician to locate the desired page with a few "clicks."

Aircraft manufacturers, which previously published their manuals only on paper or microfilm/microfiche, now offer their manuals in electronic format. Laptop computers are very well suited for storing and retrieving maintenance information in the aviation maintenance environment. Laptops can be taken practically anywhere on the aircraft that the technician might need to go while performing maintenance. For this reason, laptop computers are widely used to access maintenance information. (***Figure 22-2***)

PASSENGER CABIN INFORMATION SYSTEM

Flight attendants need to be able to access information to do their jobs, just as pilots and maintenance technicians do. In paper form, a typical flight attendant's manual weighs about five pounds (2.3 kg). Each flight attendant is required to carry the manual while serving as cabin crew. The flight attendant's manual contains vital information for cabin crew members, such as checklists, procedures, security information, and information about safety devices and systems on the aircraft. Passenger cabin information systems provide the same advantages to the cabin crew that Flight Deck Information Systems provide to the flight crew, and that maintenance information systems provide to the maintenance crew. The use of the digital electronics allows flight attendants to carry a small device, such as a tablet, that contains the manual they are required to have on hand. (***Figure 22-3***)

The device weighs much less than a paper manual. This saves the airline money, as even small reductions in weight affect fuel economy. The airline also saves money on the cost of printing revisions to the flight attendant's manual.

AIRBUS INFORMATION SYSTEMS

The information system developed for the Airbus A380 is based on different system architecture than that used on Boeing airplanes however, its primary function is similar in that, it provides efficient data management for flight deck, maintenance and passenger cabin operations. The information system comprises two primary sub systems.

A Network Server System (NSS) serves as the primary platform hosting On-board Information System (OIS) software. The NSS divided into two parts, a secured area of data management known as the "Avionics world" and a less secure area of data management and exchange outside of the aircraft, such as communication with airline operations and service providers, known as the "Open world".



Figure 22-2. Laptop computers are used to access maintenance information.

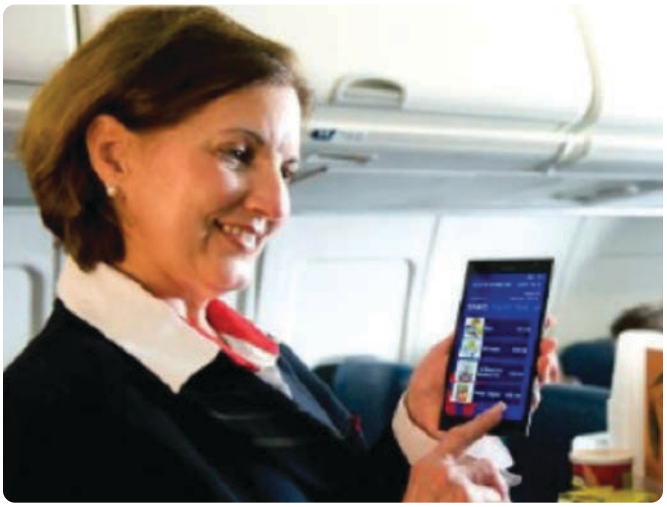


Figure 22-3. Passenger Cabin Information can be accessed with a computer tablet.

Data can not flow from the Avionics World section into the less secure Open World. However Open World can flow into the secured Avionics World section of the NSS.

The second system is the Air Traffic Control (ATC) system which uses data link communications to transfer communication, navigation and surveillance data between the airplane and air traffic control services.

NETWORK SERVER SYSTEM (NSS)

The primary function of the NSS is to host the On-board Information System (OIS) software using a high speed protocol to communicate with a network of servers and routers. The servers and routers provide a means to manage data to and from user interfaces for flight crews, maintenance personnel, cabin crew and passengers. The user interfaces may be flight crew control panels and displays, maintenance terminals, cabin terminals and in-flight entertainment systems. The network server system also provides a means of receiving data from the secure avionics area via a firewall and communications externally with Airline control centers, such as flight operations and maintenance control, and with contracted service providers such as telecoms and e-mail.

The On-board Information System (OIS) software consists of a number of applications and document templates to replace paper documents and to increase operational efficiency both on the ground and in the air. The applications provide a means for the flight crews, maintenance crews and cabin crews to access data quickly and efficiently to allow for smooth operations and quick turn-around times for the airplane.

In addition, passengers are provided with access to e-mail, internet services and a number of entertainment channels including movies and games. The Network Server System (***Figure 22-4***) is divided into three areas known as 'Domains'. These are the Avionics, Flight Operations and Communication & Cabin Domains. The OIS software applications are also shared across these three domains.

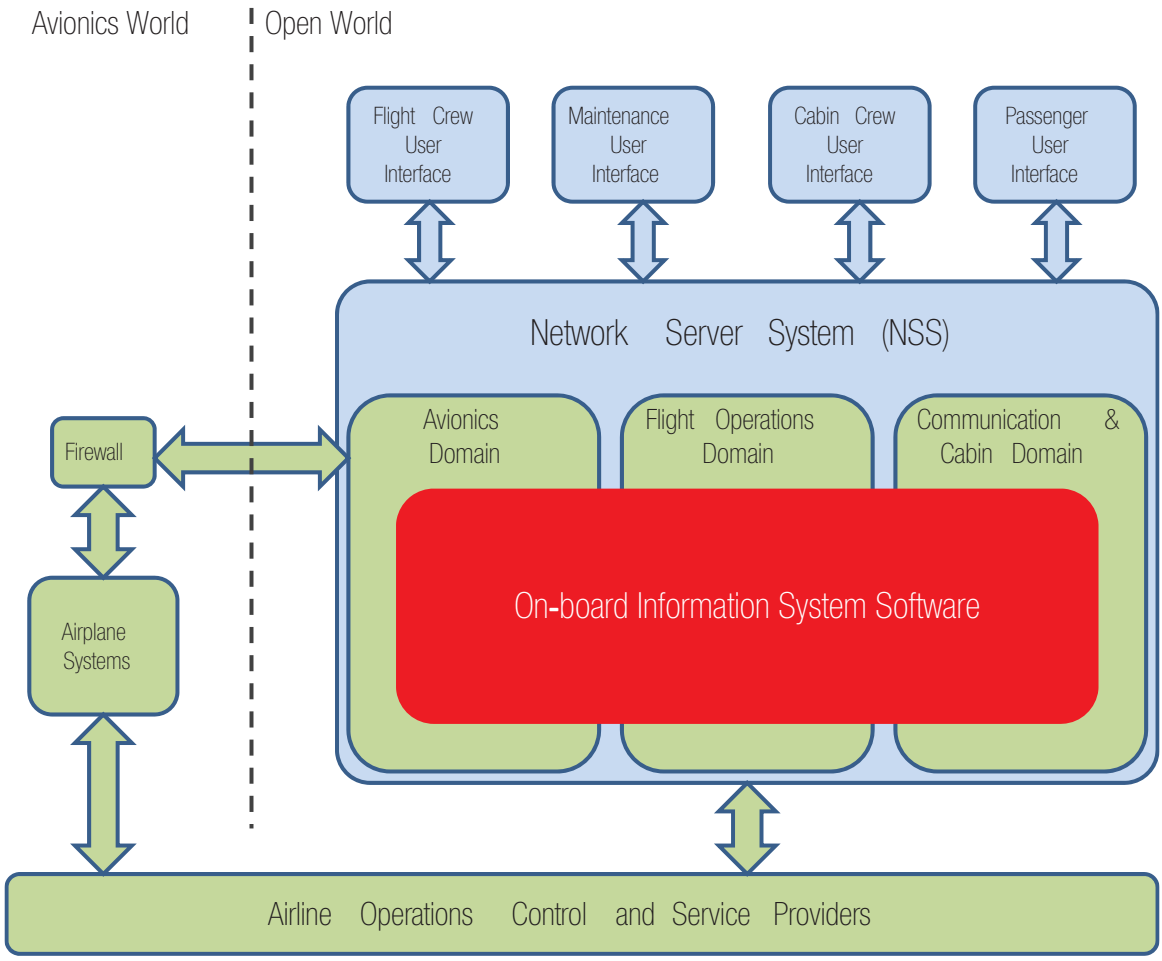


Figure 22-4. Network server system and on-board information system.

AVIONICS DOMAIN

The avionics domain (***Figure 22-5***) consists of Network Servers, Router Units, Input/Output interfaces, Data Acquisition Units and a unidirectional data gate.

Dedicated processing units and memory devices provide computing and storage functions for the relevant OIS applications operating in the Avionics Domain. Data communication and acquisition of data from the 'Avionics world' is achieved via the input/output interfaces and the data acquisition units. This data is then routed to user interfaces, the NSS and recording devices where necessary. Given the critical and sensitive nature of the data used in the 'Avionics world' any data processed and used within the avionics domain must be checked and 'cleaned' to ensure no malware or virus is allowed to corrupt the system therefore, an additional function of the avionics domain is to 'virus check' and decontaminate any data received from and transmitted to the 'Avionics world' via the avionics domain.

The unidirectional gate ensures that data flows from the avionics domain to other domains but prevents data flowing into the avionics domain from the NSS, thus protecting aircraft avionic systems from any malware or virus that may exist in the 'Open world' environment.

Access to the avionics domain is achieved through Flight Deck terminals (Captain and First Officer), Maintenance terminals, printers and data recorders. Limited access is also provided for Cabin Crew for communication and systems status (Doors armed/not armed, smoke and fire detection, etc.).

The On-board Information System applications used in the avionics domain provide on-board and air-to-ground communication functions, recording of aircraft data, electronic documents, maintenance and servicing operations within the 'Avionics world' and avionic domain.

FLIGHT OPERATIONS DOMAIN

The flight operations domain (***Figure 22-6***) consists of a server, router and terminals in the flight deck. As in the avionics domain the flight operations domain server contains the processor and memory devices for computation and storage facilities to support the relevant applications in the On-board Information System software suite. The flight deck terminals used by the captain and first officer consist of portable laptops connected via a docking station.

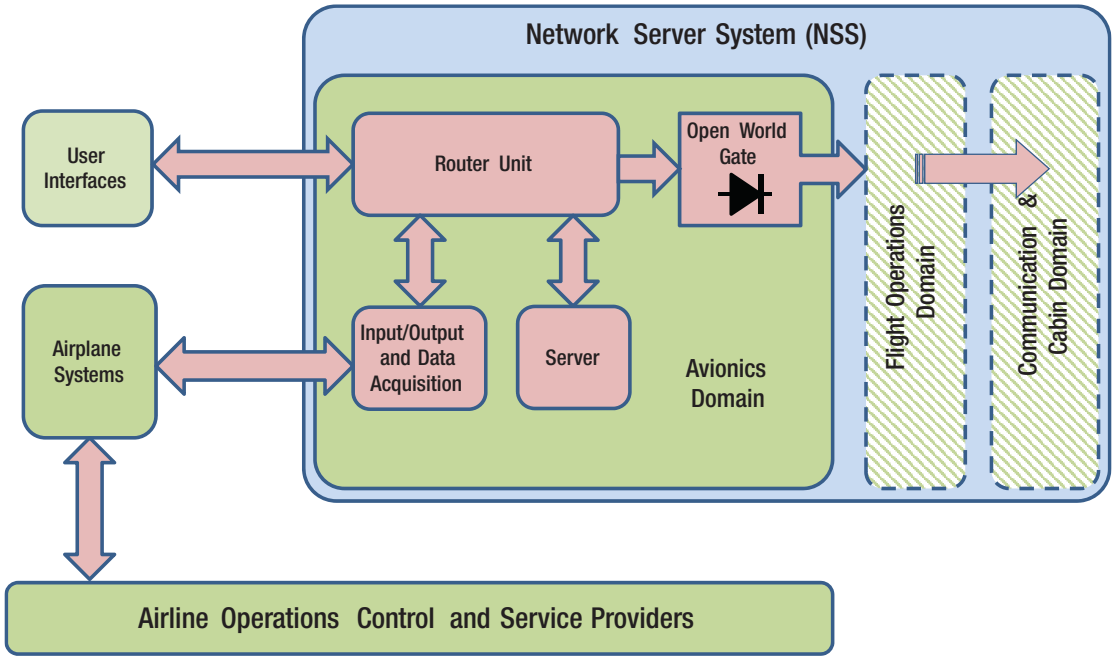


Figure 22-5. Avionics domain.

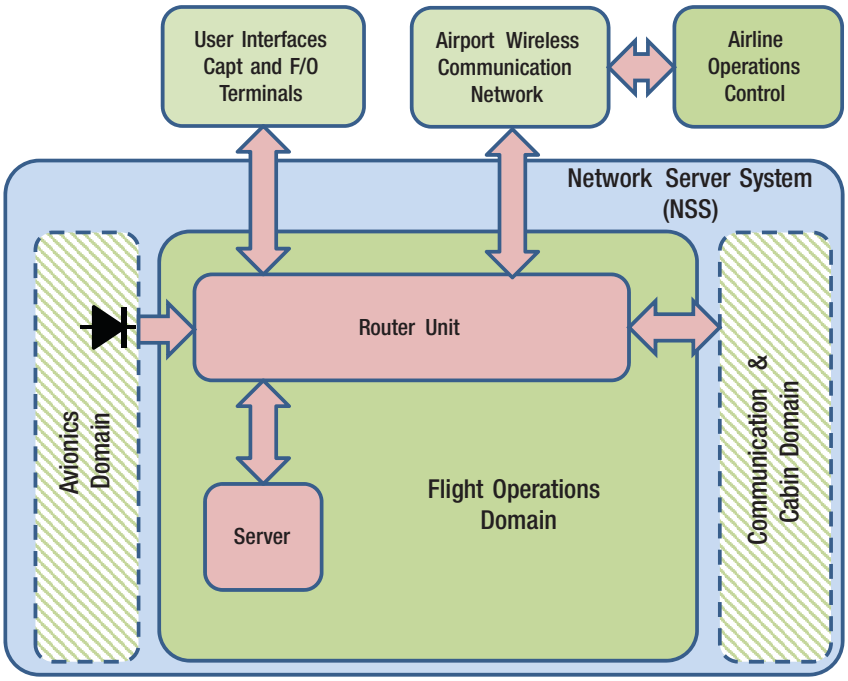


Figure 22-6. Flight operations domain.

An airport wireless communication system also provides a link with operational control centers whilst the airplane is on the ground. Access to the flight operations domain is provided by use of the Flight Deck terminals and/or portable laptop computers for captain, first officer and maintenance crews. Universal Serial Bus (USB) ports are also available for loading data. Outputs can also be provided to an on-board dedicated printer for flight and maintenance crew use. The On-board Information System applications installed in the flight operations domain provide flight operations electronic documentation, airplane performance computation functions, navigation and communication functions, data management and maintenance operations associated with the flight operations domain.

COMMUNICATION AND CABIN DOMAIN

Like the other two domains the communication & cabin domain consists of network servers and routers but, unlike the other domains, it also controls a number of wireless units in the cabin area. The servers provide the processing and memory functions needed to support the OIS applications installed in the domain. (***Figure 22-7***)

The router manages data between the communications & cabin domain and the flight operations domain, server, on-board video surveillance systems, satellite communications systems (SATCOM) and passenger in-flight entertainment systems. The cabin installed wireless units are connected to antenna in the cabin and provide a wireless connection for portable electronic devices (laptops, tablets, smartphones, etc.) to enable access to electronic mail and internet services. The communication & cabin domain is accessed from the airplane cabin via a cabin terminal used by cabin crew and maintenance crews. OIS applications hosted by the communication & cabin domain include data management for data transferred between the airplane and operations centers, maintenance functions and passenger services (internet and electronic mail).

AIR TRAFFIC AND INFORMATION MANAGEMENT SYSTEM

Until recently the primary means of communication between flight crews and ground air traffic services have been by the use of voice communication systems such as High Frequency (HF) and Very High Frequency (VHF) Amplitude Modulated (AM) radio. Although effective, these systems have traditionally suffered from the effects

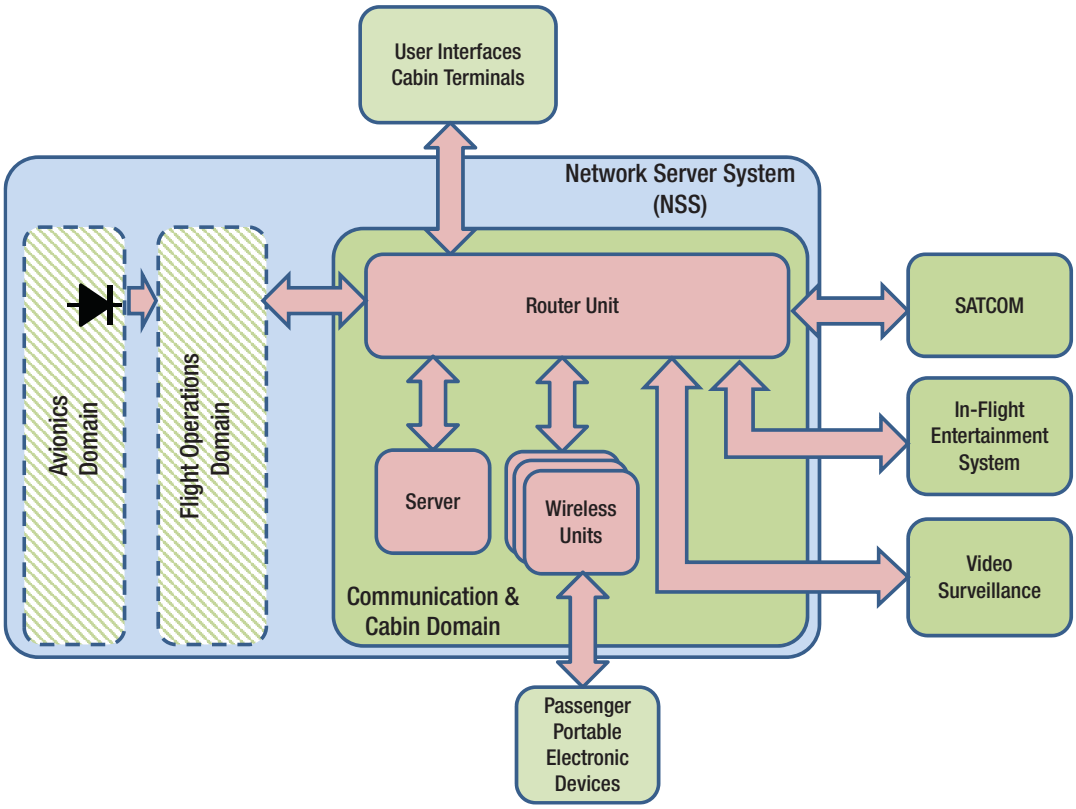


Figure 22-7. Communication & cabin domain.

of atmospheric interference/disturbance resulting in poor and unreadable communications. The bandwidths used by these systems were such that the number of available channels was limited. As air traffic increased to the level we see today it was only inevitable that the 'system' would become overburdened and there would not be enough channels to support an efficient data exchange in todays congested airspace.

The development of High Frequency Data Link (HFDL) and Very High Frequency Data Link (VDL) systems with much narrower bandwidths means that there are many more channels available which are not prone to the levels of interference as the traditional AM systems. These systems provide the means to transfer data more efficiently, accurately and quickly. This also enables air traffic services to handle busier and more congested airspaces.

The Air Traffic Information Management System (ATIMS) (***Figure 22-8***) developed by Airbus Industrie provides a means for a swift transfer of data via datalink between airplane and ground stations. Air traffic controllers are able to communicate with flight crews using a datalink 'texting' facility to ensure accurate data transfer for air traffic management.

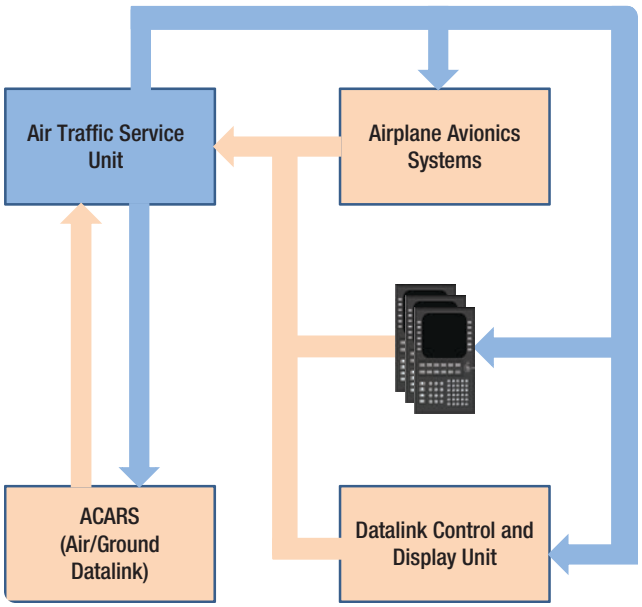


Figure 22-8. Air traffic information management system.

Airline operations centers benefit from more efficient operation, and thus reduced costs, with more effective communication with their airplanes.

Automatic surveillance of airplanes by air traffic centers also provides controllers with much more accurate airplane position, intended flight path, airplane attitude and flight details. The ATIMS consists of an Air traffic

Service Unit (ATSU), two Datalink Control and Display Units (DCDU) located in the flight deck for captain and first officer, two air traffic control message pushbutton switches and a switch to reset the ATSU. The airplane Multipurpose Control and Display Units (MCDU) are used by maintenance crews to interface with the ATIMS.

AIR TRAFFIC SERVICE UNIT (ATSU)

The Air Traffic Service Unit is loaded with various software applications used to manage data to and from the airplane for air traffic control purposes. The software packages are loaded and updated via a Multipurpose Disk Drive Unit (MDDU). The software loaded into the ATSU includes ATSU Airplane Interface software, ATSU configuration software, ATSU router software, Airline Operation Control software, Airline Operation Control databases, ATC FANS A+ applications.

The ATSU Airplane Interface software provides functionality for monitoring the ATIMS, acquiring airplane parameters and controlling air/ground communications. The ATSU configuration software contains files for the Air Traffic Control and Airline Operations Control communications settings. The ATSU router software holds the worldwide Datalink Service Provider database.

Airline Operation Center datalink applications provide airlines with means to send and receive operational data. This data would include flight plans, system status for maintenance and general airplane management. The ATC FANS A+ application package provides applications associated with the control and transfer of data for Air Traffic Control purposes. These applications enable communications with the ATC ground station using datalink instead of voice communications. Uplink messages are displayed on the Datalink Control and Display Units in the flight deck which are managed in order of priority and, where applicable, are loaded into the airplane avionics systems. Downlink messages are configured through the MCDU and displayed on the DCDU for checking by the crew before transmission.

The FANS A+ application package also enables the Air Traffic Control center to receive airplane identification and four dimensional position reports for surveillance purposes.

- A Departure Clearance (DC) application automatically generates requests for departure information and clearance.
- An Oceanic Clearance (OC) application automatically generates requests for oceanic clearance prior to entering oceanic airspace.

The Air Traffic Service Unit communicates with airplane avionics systems using an ARINC 429 bus interfacing with:

- Datalink Control and Display Units
- Flight Management Guidance and Envelope Computers
- Multipurpose Control and Display Units
- VHF Datalink communications
- HF Datalink communications
- SATCOM
- Cabin Terminals
- Radio Management Panels
- Data Management Units
- Flight Warning Computers
- Data Acquisition Concentrators
- ATC Transponders
- Central Maintenance Computers
- Multipurpose Disk Drive Units
- Flight Deck Clock

DATALINK CONTROL AND DISPLAY UNITS (DCDU)

The information displayed on the DCDU (***Figure 22-9***) provided to the flight crew from the ATC ground centres and enables the crew to send responses to ATC messages received. All information displayed on the DCDU is managed and controlled by the ATSU.

Flight crew are alerted of a message by the illumination of ATC message pushbutton switches and are accompanied by an audible alert generated by the Flight Warning Computer. The audible alerts may be single chime or a repetitive chime depending on the message priority.



Figure 22-9. Datalink control and display unit.

MULTIFUNCTION CONTROL DISPLAY UNIT (MCDU)

Communication with Airline Operations is achieved through use of the MCDU where messages are prepared and/or edited before sending. ATIMS software applications can also be configured using the MCDU.

Question: 22-1
What is the purpose of an electronic flight bag?

Question: 22-2
The two areas of the Network Server System a re k nown a s _____ and _____.

Question: 22-3
The three domains in the Network Server System are known as _____, _____ and _____.

Question: 22-4
The primary purpose of securing information in the Avionics World section of the On-Board Information System is what?

Question: 22-5
State three examples of the benefits to maintenance crew of an electronic information storage system compared to printed publications.

Question: 22-6
State four examples of information which might be stored in an on-board Maintenance Information System.

Question: 22-7
What signal alerts the flight crew of a high priority message from air traffic control?

Question: 22-8
What software system generates and transmits a request for an in-flight oceanic clearance?

ANSWERS

Answer: 22-1
Electronic information storage system for pilots, replacing traditional printed navigation charts, etc.

Answer: 22-2
Avionics World containing secured data management information; Open World containing outside communications information and unsecured data.

Answer: 22-3
Avionics domain; Flight operations; Communications and cabin domain

Answer: 22-4
To protect critical navigation data from corrupting malware or viruses.

Answer: 22-5
Saves weight, saves space, easier and faster information retrieval.

Answer: 22-6
Maintenance manuals, wiring diagrams, parts catalogs, service bulletins.

Answer: 22-7
A repetitive audible chime generated by the flight warning computer.

Answer: 22-8
Air traffic control FANS A+ software via a multi-purpose disk drive.

AC	/	Alternating Current
ACARS	/	Aircraft Communications Addressing And Reporting System
ACIPS	/	Airfoil And Cowl Ice Protection System
ACM	/	Air Cycle Machine
ACMS	/	Airplane Condition Monitoring System
ACP	/	Audio Control Panel
ACU	/	Alternator Control Unit
ADC	/	Analog To Digital Converter
ADC	/	Air Data Computer
ADF	/	Automatic Direction Finding
ADI	/	Attitude Director Indicator
AFS	/	Auto Flight System
AFCS	/	Auto Flight Control System
AFDC	/	Autopilot Flight Director Computer
AFDS	/	Autopilot Flight Director System
AFDX	/	Avionics Full Duplex
AGL	/	Above Ground Level
AHRS	/	Attitude Heading Reference System
AIMS	/	Aircraft Information Management System
AIS	/	Audio Integration System
ALNA	/	Airline Network Architecture
ALT ACQ	/	Altitude Acquisition
AM	/	Amplitude Modulation
AMC	/	Acceptable Means Of Compliance
AMSL	/	Above Mean Sea Level
AMU	/	Audio Management Unit
AOA	/	Angle Of Attack
AP	/	Autopilot
APU	/	Auxiliary Power Unit
APUC	/	Auxiliary Power Unit Controller
ARCDU	/	Audio And Radio Control Display Units
AS	/	Aileron Station
ASCPC	/	Air Supply Cabin Pressure Unit
ATA	/	Air Transport Association
ATC	/	Air Traffic Control
A/THR	/	Auto Throttle System
ATIMS	/	Air Traffic Information Management System
ATSU	/	Air Traffic Service Unit
AVGAS	/	Aviation Gasoline
AVMSCU	/	Airborne Vibration Monitor Signal Conditioner Unit
AWAS	/	Automated Weather Advisory Service
BCM	/	Backup Control Module
BFO	/	Best Frequency Oscillator
BITE	/	Built In Test Equipment
BL	/	Buttock Line (Butt Line)
BPCU	/	Bus Power Control Unit

BS	/	Body Station
BSCU	/	Brake System Control Unit
BSL	/	Best Straight Line
BTB	/	Bus Tie Breakers
CAH	/	Cabin Attendant Handset
CAS	/	Command Augmentation Systems
CBM	/	Condition Based Maintenance
CCS	/	Common Core System
CDI	/	Course Deviation Indicator
CDS	/	Central Diagnostic System
CDU	/	Control Display Unit
CG	/	Center Of Gravity
CGCC	/	Center Of Gravity Control Computer
CHT	/	Cylinder Head Temperature
CIDS	/	Cabin Intercommunication Data System
CMC	/	Central Maintenance Computer
CMCS	/	Central Maintenance Computer System
CMS	/	Central Maintenance System
CO	/	Carbon Monoxide
CPF	/	Corrosion Preventative Fluids
CPIOM	/	Core Processor Input/Output Module
CPU	/	Central Processing Unit
CSD	/	Constant Speed Drive
CSDS	/	Cargo Smoke Detection System
CSMU	/	Crash Survivable Memory Unit
CSMU	/	Cabin System Management Unit
CSS	/	Cabin Service System
CT	/	Current Transformer
CTC	/	Cabin Temperature Controller
CVM	/	Comparative Vacuum Monitoring
CVR	/	Cockpit Voice Recorder
CWS	/	Control Wheel Steering
DADC	/	Digital Air Data Computer
DC	/	Direct Current
DC	/	Departure Clearance
DCDU	/	Datalink Control Display Unit
DCMS	/	Data Communication Management System
DFCS	/	Digital Flight Control System
DG	/	Directional Gyro
DH	/	Decision Height
DLODS	/	Duct Leak And Overheat Detection
DLS	/	Data Loader Selector
DMC	/	Display Management Computer
DME	/	Distance Measuring Equipment
DOT	/	Department Of Transportation
DTI	/	Damage Tolerance Inspection

DU	/	Display Units
EADI	/	Electronic Attitude Director Indicator
EAFR	/	Enhanced Airborne Flight Recorder
EAI	/	Engine Anti Ice
ECAM	/	Electronic Centralized Aircraft Monitor
ECAS	/	Emergency Crew Alerting System
ECS	/	Environmental Control System
ECU	/	Engine Control Unit
ECHP	/	Electronically Controlled Hydraulically Powered
EDIU	/	Engine Data Interface Unit
EDS	/	Electronic Pulse Demand Oxygen System
EDV	/	Electronic Drain Valve
EEC	/	Electronic Engine Controller
EFB	/	Electronic Flight Bag
EFIS	/	Electronic Flight Instrument System
EGT	/	Exhaust Gas Temperature
EICAS	/	Engine Indicating And Crew Alert System
ELMS	/	Electrical Load Management System
ELS	/	Electronic Library System
ELT	/	Emergency Locator Transmitter
EM	/	Electromagnetic
EPA	/	Environmental Protection Agency
EPD	/	Electronic Pulse Demand
EPR	/	Engine Pressure Ratio
EPIRB	/	Emergency Position Indicating Radio Beacons
ETA	/	Estimated Time Of Arrival
EE	/	Technical Standard Order Technical Standard
EXP	/	Expedite
FAA	/	Federal Aviation Administration
FADEC	/	Full Authority Digital Engine Control
FAPS	/	Flight Attendants Panel
FAR	/	Federal Aviation Regulation
FbL	/	Fly By Light
FBW	/	Fly By Wire
FC	/	Flight Computer
FCC	/	Flight Control Computer
FCC	/	Federal Communications Commission
FCOC	/	Fuel Cooled Oil Cooler
FCS	/	Flight Control System
FCU	/	Flight Control Unit
FD	/	Flight Director
FDAU	/	Flight Data Acquisition Unit
FDR	/	Flight Data Recorder
FDRS	/	Flight Data Recorder System
FGC	/	Flight Guidance Computer
FLCH	/	Flight Level Change

FLIR	/	Forward Looking Infrared
FMCS	/	Flight Management Computer System
FMS	/	Flight Management System
FOD	/	Foreign Object Damage
FPM	/	Feet Per Minute
FPV	/	Flight Path Vector
FQIS	/	Fuel Quantity Indication System
FS	/	Fuselage Station
FSEU	/	Flap Slat Electronic Unit
FSO	/	Full Scale Output
GA	/	General Aviation
GB	/	Generator Breaker
GCB	/	Generator Control Breaker
GCU	/	Generator Control Unit
GEOSAT	/	Geostationary Satellites
GNNS	/	Global Navigation Satellite System
GPIB	/	General Purpose Interface Bus
GPIRB	/	Global Position Indicating Radio Beacon
GPS	/	Global Navigation System
GPWS	/	Ground Proximity Warning System
GS	/	Glide Slope
GSE	/	Ground Support Equipment
HCP	/	Helisas Control Panel
HF	/	High Frequency
HFDL	/	High Frequency Data Link
HIRF	/	High Intensity Radio Frequency
HMDG	/	Hydraulic Motor Driven Generator
HOT	/	Hold Over Time
HPFAC	/	High Pressure Fan Air Controller
HSI	/	Horizontal Situation Indicator
HRD	/	High Rate Discharge
HSI	/	Horizontal Situation Indicator
HSS	/	Horizontal Stabilizer Station
HYDIM	/	Hydraulic Interface Module
ICAO	/	International Civil Aviation Organization
IDG	/	Integrated Drive Generator
IFE	/	In Flight Entertainment
ILS	/	Instrument Landing System
IMA	/	Integrated Modular Avionics
IR	/	Infrared
IVHM	/	Integrated Vehicle Health Monitoring
KS	/	Flap Station
KVA	/	Kilovolt Amp
LAN	/	Local Area Network
LCD	/	Liquid Crystal Display
LED	/	Light Emitting Diode

LEO	/	Low Earth Orbit
LEOSATS	/	Low Earth Orbiting Satellites
LOC	/	Localizer Beam Intercepts And Hold
LORAN	/	Long Range Area Navigation
LOX	/	Liquid Oxygen
LRM	/	Line Replaceable Module
LRU	/	Line Replaceable Unit
LUTS	/	Local User Terminals
LV	/	Lower Voice Single Sideband
MAC	/	Mean Aerodynamic Chord
MAT	/	Maintenance Access Terminal
MCC	/	Mission Control Center
MCP	/	Mode Control Panel
MCDU	/	Multifunction Control Display Unit
MDDU	/	Multipurpose Disk Drive Unit
MEMS	/	Micro-Electro-Mechanical System
MFD	/	Multi Function Display
MFP	/	Multi-Function Probe
MIAH	/	Minimum Interruption Altitude Height
MLS	/	Microwave Landing System
MMR	/	Multi Mode Receiver
MSL	/	Mean Sea Level
NC	/	Nacelle Station
ND	/	Navigation Display
NDB	/	Non Directional Beacon
NiCd	/	Nickel Cadmium
NOAA	/	National Oceanic And Atmospheric Administration
NSS	/	Network Server System
NVM	/	Non Volatile Memory
OAT	/	Outside Air Temperature
OPBC	/	Overhead Panel Bus Controller
OBOGS	/	Onboard Oxygen Generating System
OBS	/	Omni-Bearing Selector
OC	/	Oceanic Clearance
OEU	/	Overhead Electronics Unit
OIS	/	On-Board Information System
OOK	/	On-Off Keying
OPAS	/	Overhead Panel Arinc 629 System
PA	/	Passenger Address
PAA	/	Passenger Address Amplifier
PCU	/	Power Control Unit
PDS	/	Primary Display Unit
PFC	/	Primary Flight Computer
PFD	/	Primary Flight Display
PLA	/	Power Lever Angle
PTT	/	Push To Talk

PM	/	Phase Modulation
POST	/	Power-On Self Test
PRSOV	/	Pressure Regulating And Shut-Off Valve
PSE	/	Principle Structural Element
PSEU	/	Proximity Sensor Electronic Unit
PSI	/	Per Square Inch
PSU	/	Passenger Service Unit
PTU	/	Power Transfer Unit
PVS	/	Personal Video Screens
PWA	/	Printed Wire Assembly
QAR	/	Quick Access Recorder
RAM	/	Random Access Memory
RAMP	/	Radio And Audio Management Panel
RAT	/	Ram Air Turbine
RCAU	/	Remote Control Audio Unit
RDF	/	Radio Direction Finder
RF	/	Radio Frequency
RLG	/	Ring Laser Gyros
RMI	/	Radio Magnetic Indicator
RMLA	/	Roll Maneuver Load Alleviation
RMP	/	Radio Management Panel
RMS	/	Root Mean Square
RNAV	/	Area Navigation
ROM	/	Read Only Memory
RTD	/	Resistance Temperature Detector
RTU	/	Radio Tuning Unit
RVR	/	Runway Visual Range
SAS	/	Stability Augmentation System
SAT	/	Static Air Temperature
SATCOM	/	Satellite Communication
SCR	/	Silicon Controlled Rectifier
SELCAL	/	Selective Calling Radio System
SEU	/	Seat Electronics Unit
SHF	/	Super High Frequency
SHM	/	Structural Health Monitoring
SRAM	/	Static Random Access Memory
SSB	/	Single Sideband
SSB	/	Split Systems Breaker
SSFDR	/	Solid State Flight Data Recorder
SSI	/	Structurally Significant Item
STOL	/	Short Takeoff And Landing
TA	/	Terrain Awareness
TAT	/	Total Air Temperature
TAWS	/	Terrain Awareness Warning System
TCAS	/	Traffic Alert And Collision Avoidance System
TCF	/	Terrain Clearance Floor

TDZE	/	Touch Down Zone Elevation
THS	/	Trimmable Horizontal Stabilizer
TIT	/	Turbine Inlet Temperature
TMCS	/	Thrust Management Computer System
TOGA	/	Take-Off/Go-Around
TR	/	Temperature Rise
TRU	/	Transformer Rectifier Unit
TVOR	/	Terminal Very High Frequency Omnidirectional Range
TWLU	/	Terminal Wireless Lan Unit
VSI	/	Vertical Speed Indicator
UHF	/	Ultra High Frequency
ULB	/	Underwater Locator Beacon
ULD	/	Underwater Locator Device
ULD	/	Unit Load Device
USB	/	Universal Serial Bus
UV	/	Ultraviolet
UV	/	Upper Voice Single Sideband
VDL	/	Very High Frequency Data Link
VHF	/	Very High Frequency
VLF	/	Very Low Frequency
V-NAV	/	Vertical Navigation
VOR	/	Very High Frequency Omnidirectional Range
VOT	/	Vor Test Facilities
VRLA	/	Valve Regulated Lead Acid Battery
VSI	/	Vertical Speed Indicator
VSS	/	Vertical Stabilizer Station
VVI	/	Vertical Velocity Indicator
WACS	/	Wireless Airport Communication System
WAI	/	Wing Anti-Ice
WEU	/	Warning Electronic Unit
WL	/	Water Line
WOW	/	Weight On Wheels
WXR	/	Weather Radar
YDS	/	Yaw Damper System
ZMU	/	Zone Management Unit

# #

777 Cabin Services System 21.3

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