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EMB-120 Brasilia
PILOT TRAINING MANUAL

VOLUME 2
AIRCRAFT SYSTEMS

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NOTICE

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At the time of printing it contained then-current information. In the event of conflict between data provided herein and that in publications issued by the manufacturer or the FAA, that of the manufacturer or the FAA shall take precedence.

We at FlightSafety want you to have the best training possible. We welcome any suggestions you might have for improving this manual or any other aspect of our training program.

FOR TRAINING PURPOSES ONLY

CONTENTS

Chapter 1	AIRCRAFT GENERAL
Chapter 2	ELECTRICAL POWER SYSTEMS
Chapter 3	LIGHTING
Chapter 4	MASTER WARNING SYSTEM
Chapter 5	FUEL SYSTEM
Chapter 6	AUXILIARY POWER UNIT
Chapter 7	POWERPLANT
Chapter 8	FIRE PROTECTION
Chapter 9	PNEUMATICS
Chapter 10	ICE AND RAIN PROTECTION
Chapter 11	AIR CONDITIONING
Chapter 12	PRESSURIZATION
Chapter 13	HYDRAULIC POWER SYSTEMS
Chapter 14	LANDING GEAR AND BRAKES
Chapter 15	FLIGHT CONTROLS
Chapter 16	AVIONICS
Chapter 17	OXYGEN SYSTEMS

WALKAROUND

ANNUNCIATOR PANEL

INSTRUMENT PANEL POSTER

CHAPTER 1
AIRCRAFT GENERAL

CONTENTS

Page

INTRODUCTION 1-1
GENERAL 1-2
STRUCTURES 1-2
General 1-2
Cockpit 1-5
Cabin 1-9
Doors 1-13
EMERGENCY EQUIPMENT 1-17
General 1-17
Emergency Locator Transmitter 1-17
Emergency Exits 1-19
Emergency Lighting 1-19
SYSTEMS 1-19
Electrical Systems 1-19
Fuel System 1-20
Auxiliary Power Unit 1-20
Powerplant 1-20
Fire Protection 1-20
Ice and Rain Protection 1-20
Air Conditioning and Pressurization 1-21
Hydraulic System 1-21

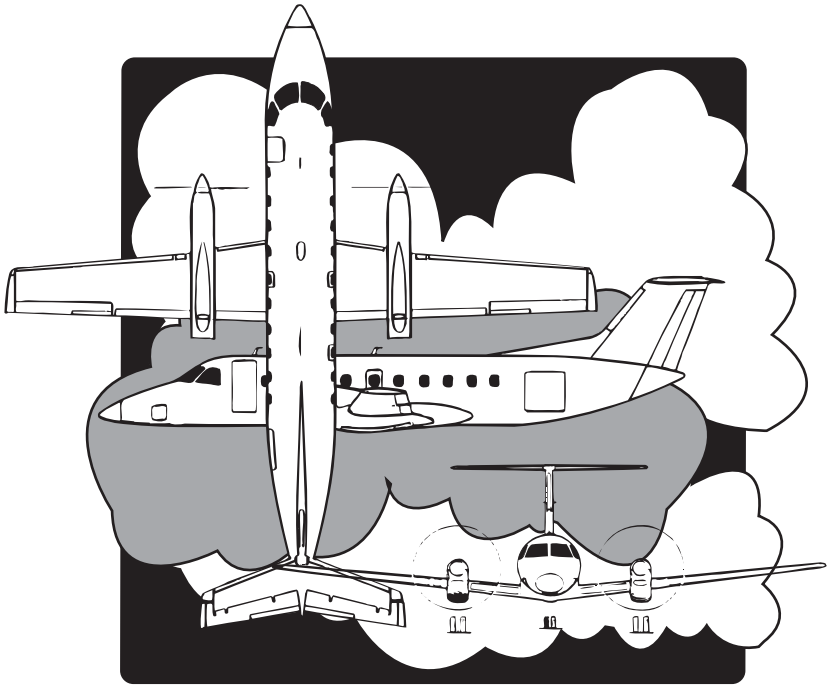
Landing Gear and Brakes 1-21
Flight Controls 1-21
Avionics 1-21
Oxygen 1-22
PUBLICATIONS 1-22
QUESTIONS 1-24

ILLUSTRATIONS

Figure	Title	Page
1-1	EMB 120 RT Brasilia	1-2
1-2	Exterior Three-View Drawing	1-3
1-3	Turning Distance	1-4
1-4	Danger Zones	1-4
1-5	Cockpit Layout	1-5
1-6	Overhead Panel	1-6
1-7	Windshield and Direct Vision Windows	1-7
1-8	Normal and Emergency Window Operation	1-7
1-9	Pilots' Seats	1-8
1-10	Observer's Seat	1-8
1-11	Pedal Adjust Mechanism	1-8
1-12	Pilot's Seat Adjustment	1-9
1-13	Passenger Configuration/Interior Layout (Typical)	1-10
1-14	Attendant's Station (Typical)	1-11
1-15	Attendant's Panel	1-12
1-16	Toilet	1-12
1-17	Galley	1-12
1-18	Forward Door Controls	1-13
1-19	Door Warning Lights	1-13
1-21	Forward Entry Door Operation	1-14
1-20	Forward Door Emergency Valve	1-14
1-22	Cargo/Service Door Location	1-15
1-23	Cargo Door Operation	1-16

1-24 Doors Warning Lights (Overhead Panel) 1-16
1-25 Standard Emergency Equipment Location 1-17
1-26 Life Vest Location and Operation 1-18
1-27 Hand Hold Rope Location and Use 1-18
1-28 Emergency Exit Operation 1-19

CHAPTER 1
AIRCRAFT GENERAL



INTRODUCTION

This training manual provides a description of the major airframe and engine systems installed in the EMB-120 Brasilia. The information contained herein is intended only as an instructional aid. This material does not supersede, nor is it meant to substitute for, any of the manufacturer's maintenance or operating manuals. The material presented has been prepared from current design data.

Chapter 1 covers the structural makeup of the airplane and gives an overview of the systems.

An annunciator section in this manual displays all annunciator and other light indications and should be folded out for reference while reading this manual.

Review questions are contained at the end of most chapters. These questions are included as a self-study aid, and the answers can be found in the appendix section.

GENERAL

The EMB-120 Brasilia is certified in accordance with FAR Part 25 airworthiness standards. It is designed for passenger and cargo transportation on typical commercial air carriers. There are three types: RT, ER, and FC.

The minimum crew requirements for operations in the EMB-120 Brasilia are one pilot and one copilot. The pilot-in-command must have a Brasilia type rating and meet the requirements of FAR 61.58 for two-pilot operation. The copilot shall possess a multiengine rating and meet the requirements of FAR 61.55.

STRUCTURES

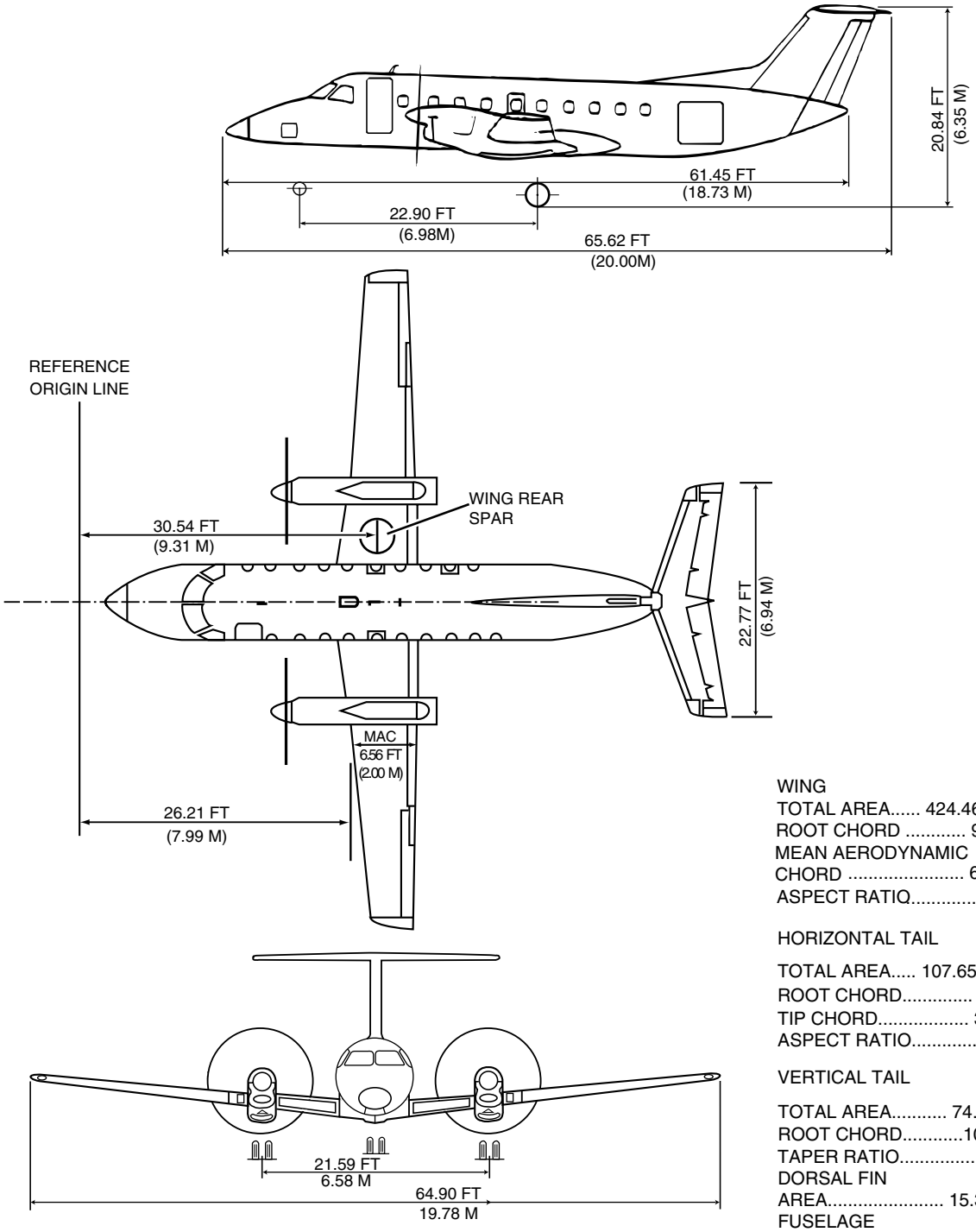
GENERAL

The EMB-120 Brasilia (Figure 1-1) is an all-metal construction, pressurized, low-wing T-tail, monoplane. Two Pratt and Whitney PW118 engines are mounted on the wing, supported by a semimonocoque/tubular nacelle structure. The landing gear is the retractable, twin-wheel type.

Figure 1-2 shows a three-view drawing of the EMB-120 Brasilia with the principal dimensions. Figure 1-3 shows turning distance, and Figure 1-4 is a diagram of danger zones.



Figure 1-1. EMB 120 RT Brasilia



WING
TOTAL AREA..... 424.46 FT ^{2} (39.43 M ^{2})
ROOT CHORD ..... 9.22 FT (2.81 M)
MEAN AERODYNAMIC
CHORD ..... 6.56 FT(2.00 M)
ASPECT RATIQ..... 9.92

HORIZONTAL TAIL

TOTAL AREA..... 107.65 FT^2(10.00 M^2)
ROOT CHORD..... 6.00 FT(1.83 M)
TIP CHORD..... 3.64 FT (1.11 M)
ASPECT RATIO..... 4.63

VERTICAL TAIL

TOTAL AREA..... 74.28 FT^2(6.90 M^2)
ROOT CHORD.....10.43 FT(3.18 M)
TAPER RATIO..... 0.65
DORSAL FIN
AREA..... 15.39 FT^2 (1.43 M^2)
FUSELAGE
OUTSIDE
DIAMETER..... 7.48 FT (2.28 M)

Figure 1-2. Exterior Three-View Drawing

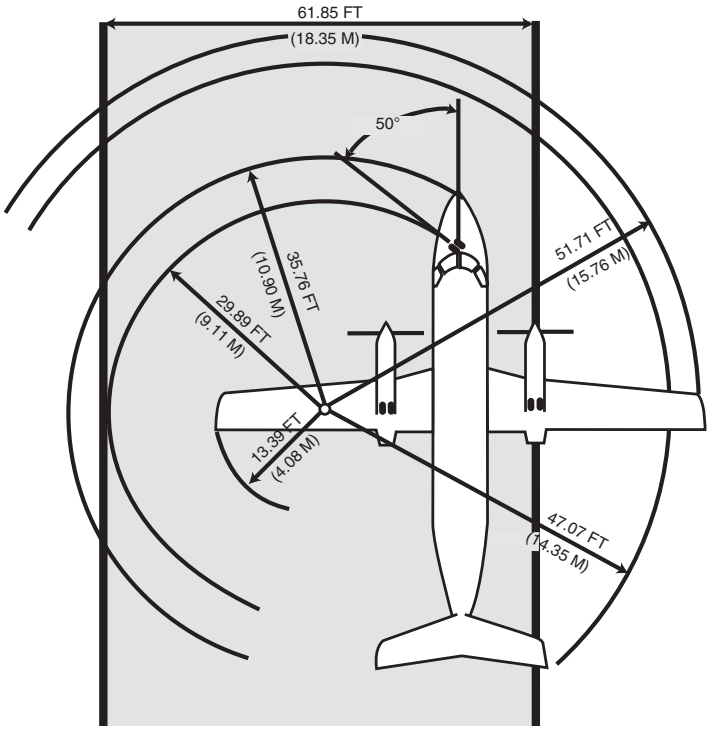


Figure 1-3. Turning Distance

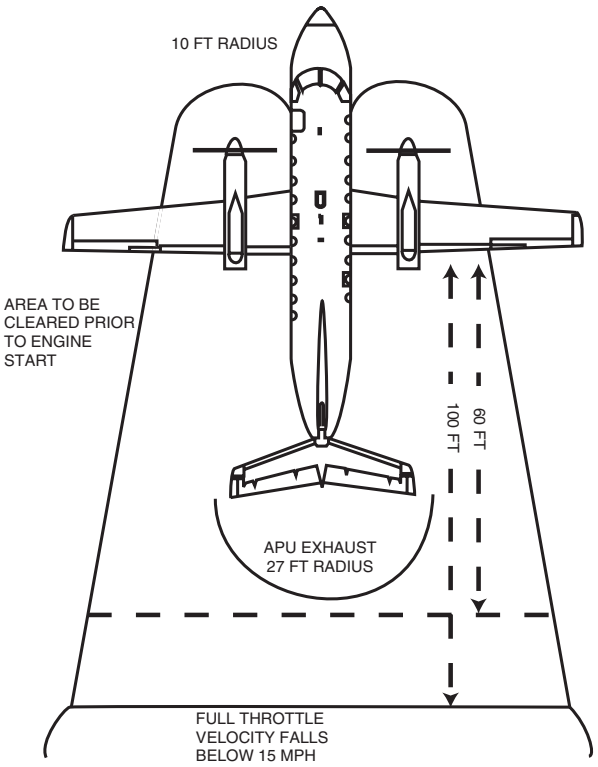


Figure 1-4. Danger Zones

COCKPIT

General

The general layout of the cockpit is shown in Figure 1-5. The overhead panel is shown in more detail in Figure 1-6.

Specific customer requirements may cause some instruments and equipment to vary from standard configuration.

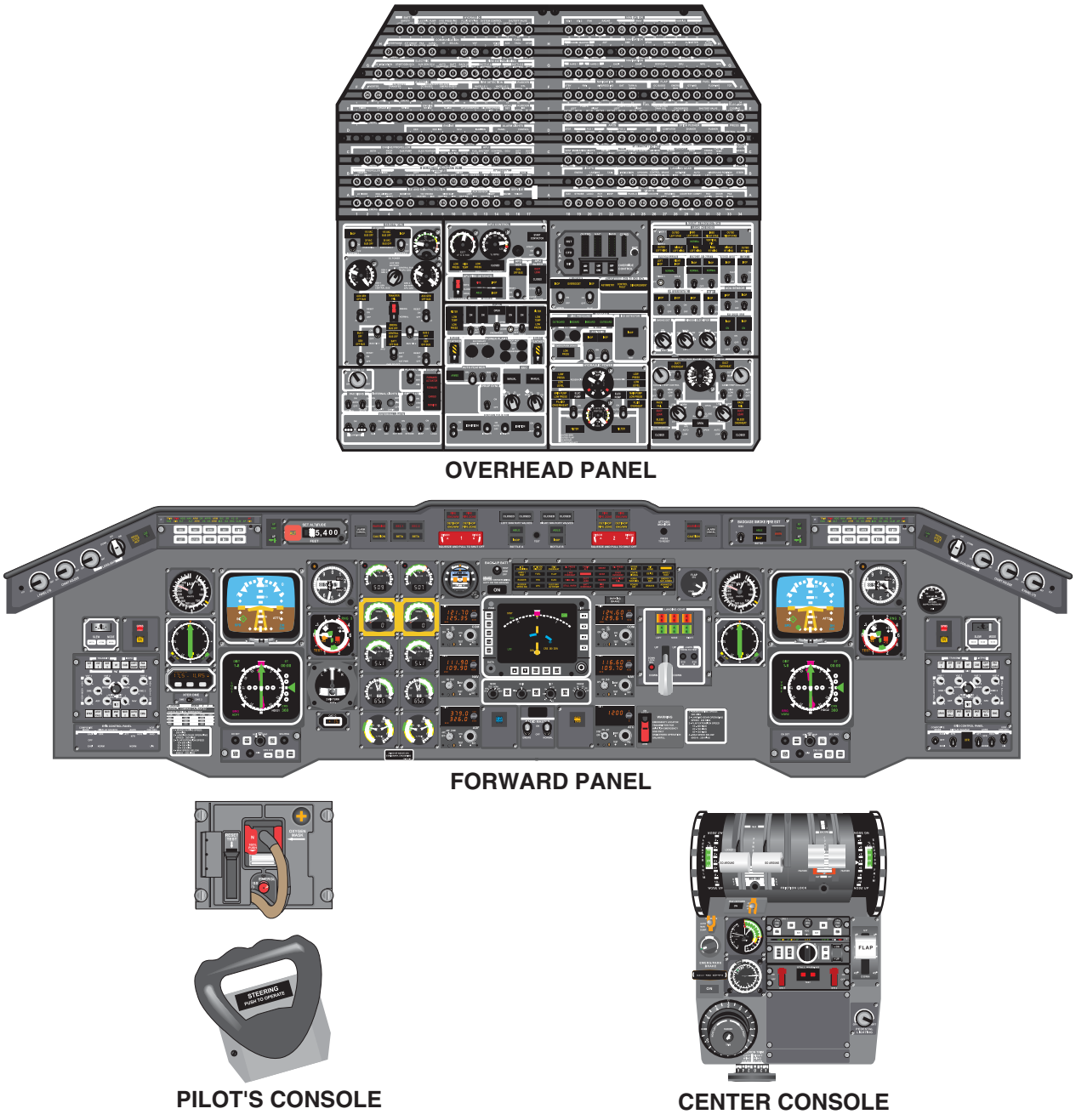


Figure 1-5. Cockpit Layout



Figure 1-6. Overhead Panel



Figure 1-7. Windshield and Direct Vision Windows

Windshield and Windows

The EMB 120 has two windshields and two direct vision windows (Figure 1-7). Only the windshields are heated.

The direct vision windows may be partially opened during normal operation on the ground. They may be totally removed in case of loss of visibility through windshield or for cockpit evacuation. A WINDOW NOT CLOSED inscription, on the window front frame, will be visible when the window is not closed properly. Window operation is shown in Figure 1-8.

Crew Seats

The pilot's seats (Figure 1-9) are fixed to slide tracks which permit fore, aft, and lateral seat movement. They are also equipped with a

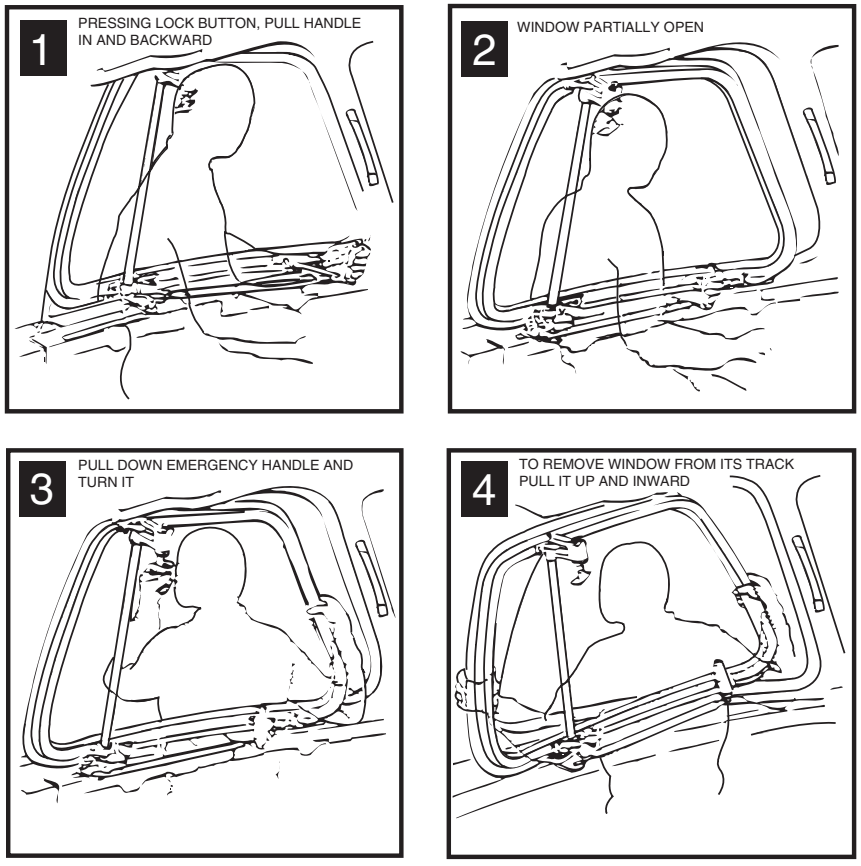


Figure 1-8. Normal and Emergency Window Operation

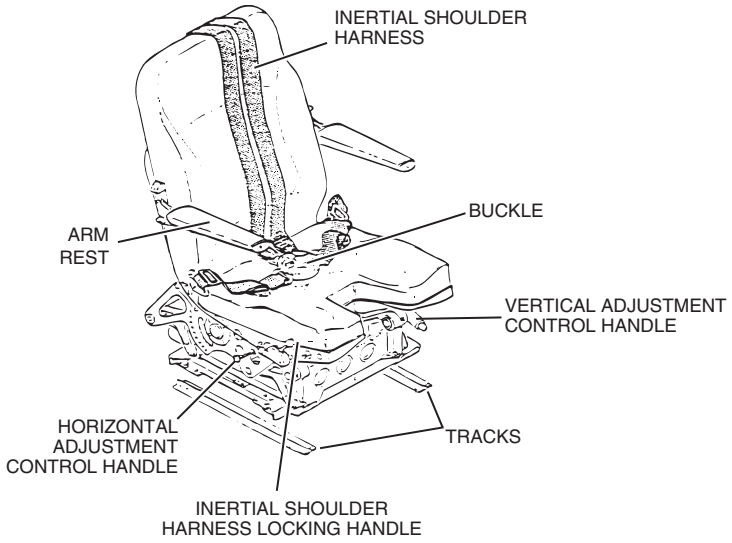


Figure 1-9. Pilot Seats

height adjustment mechanism to lower or raise the seat. The seats include quick-disconnect combination lap belts and shoulder harness with inertial reels. Lateral seat movement is possible only when the seat is in the full AFT position.

Observer's Seat

A foldable jump seat (Figure 1-10), installed in the floor at the cockpit entrance, may be used for an observer or a third crew member. The observer's seat is provided with safety belt and inertia reel.



Figure 1-10. Observer's Seat

Pedal Adjustment

A mechanism under each pilot's front panel (Figure 1-11), allows the pilots to adjust their pedals for optimum position.

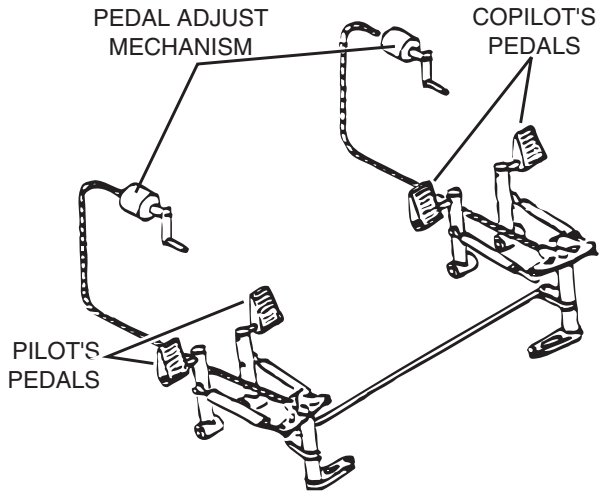


Figure 1-11. Pedal Adjust Mechanism

Seat Adjustment

Each seat may be adjusted to position the pilot for optimum control column operation using the alignment balls as shown in Figure 1-12.

This is accomplished by first moving the seat up or down until the pilot's line of sight reaches the same horizontal plane as the alignment balls. Then, move the seat fore and aft so that the opposite white ball becomes aligned with the black ball.

CABIN

Figure 1-13 shows typical passenger configuration and internal layout.

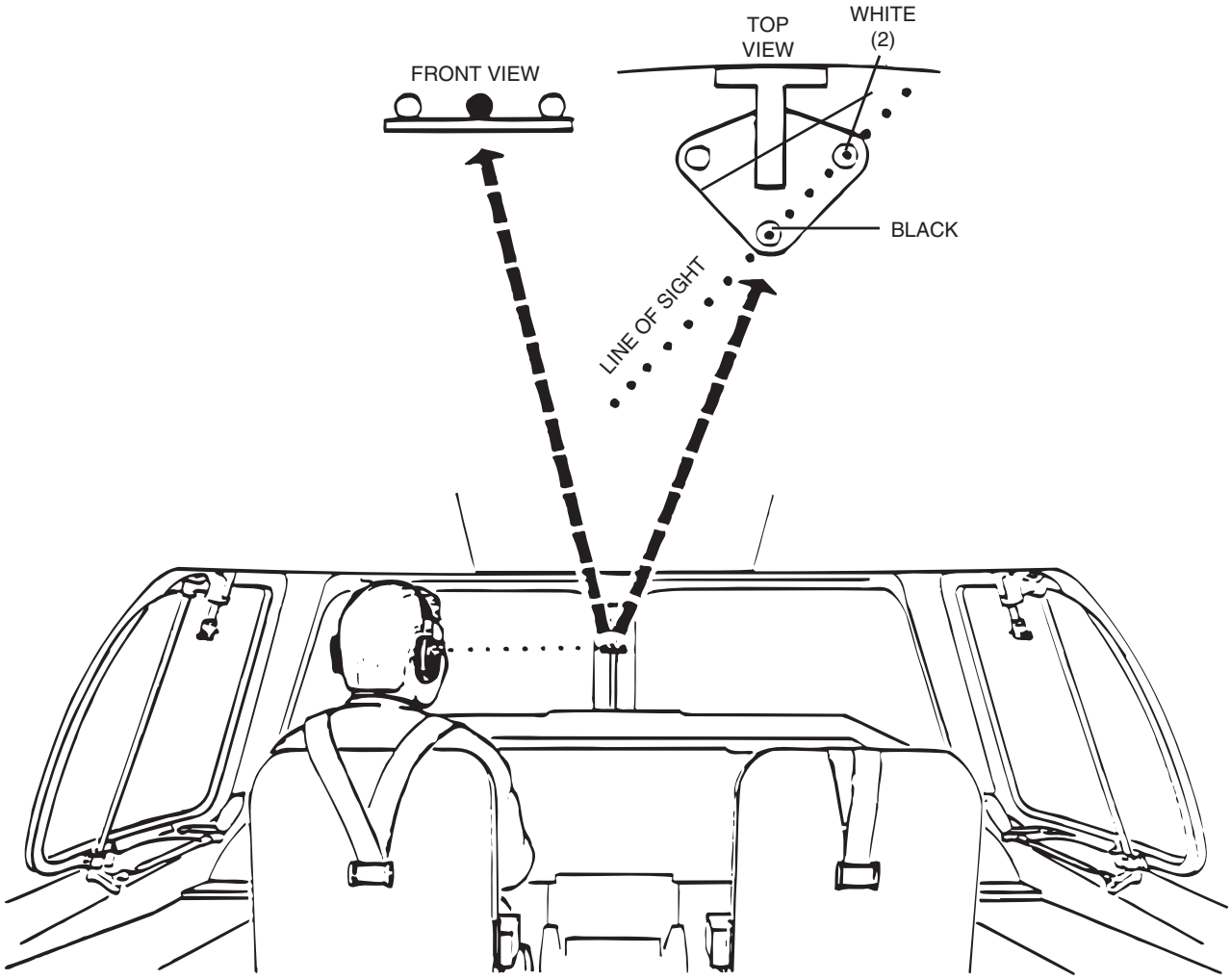


Figure 1-12. Pilot Seat Adjustment

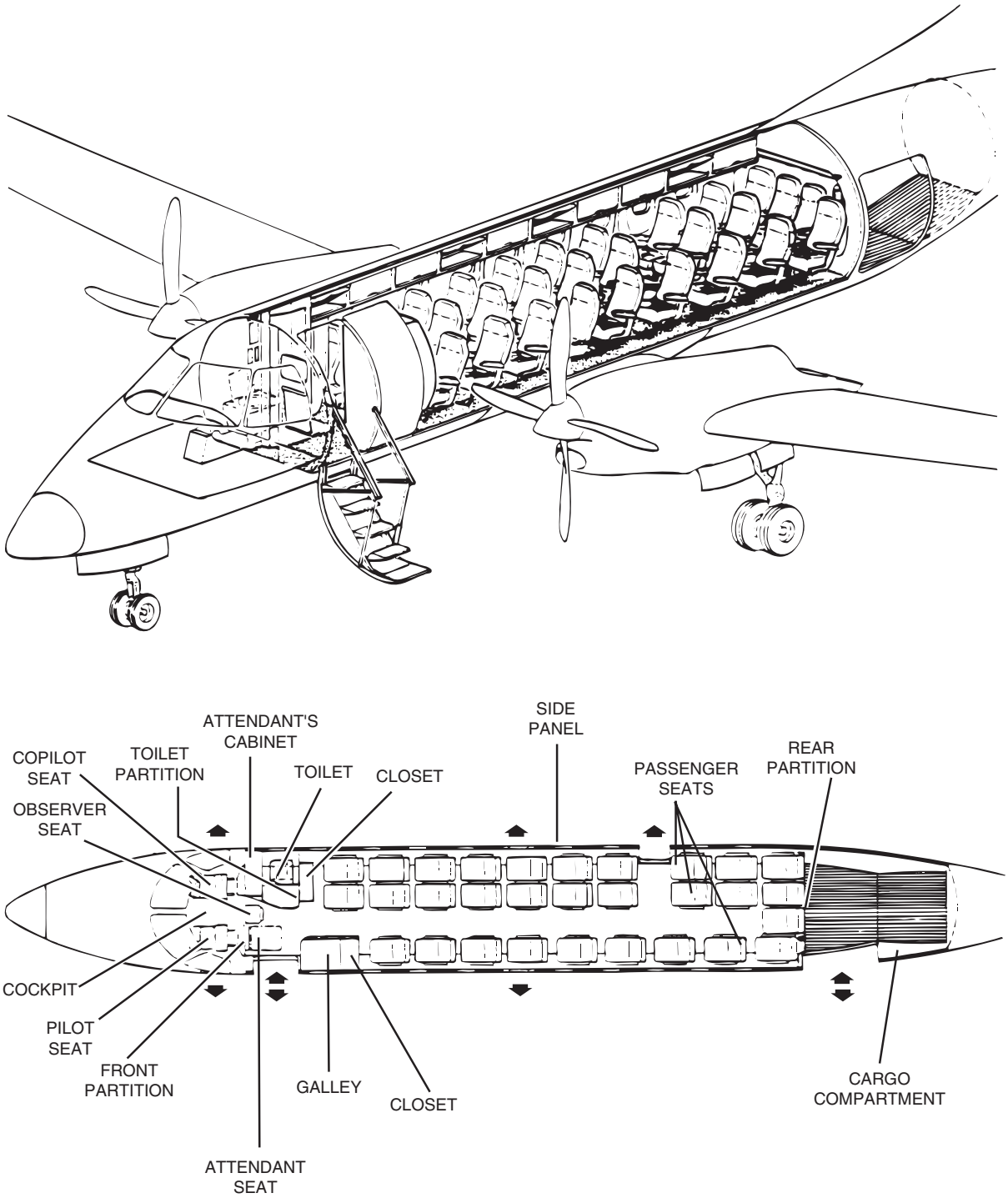


Figure 1-13. Passenger Configuration/Interior Layout (Typical)

Attendant's Station

The attendant's station, shown in Figure 1-14, is located next to the forward entry door. It is provided with interphone, folding seat, fire extinguisher, two life vests, and a flashlight.

The attendant's panel (Figure 1-15) has controls for the emergency lights, cabin lights, cabin temperature, and forward entry door.

Toilet

The toilet, as seen in Figure 1-16, is located opposite the forward entry door. On some earlier aircraft the toilet may be located in the rear of the aircraft. The galley (Figure 1-17) is located just aft of the forward entry door.

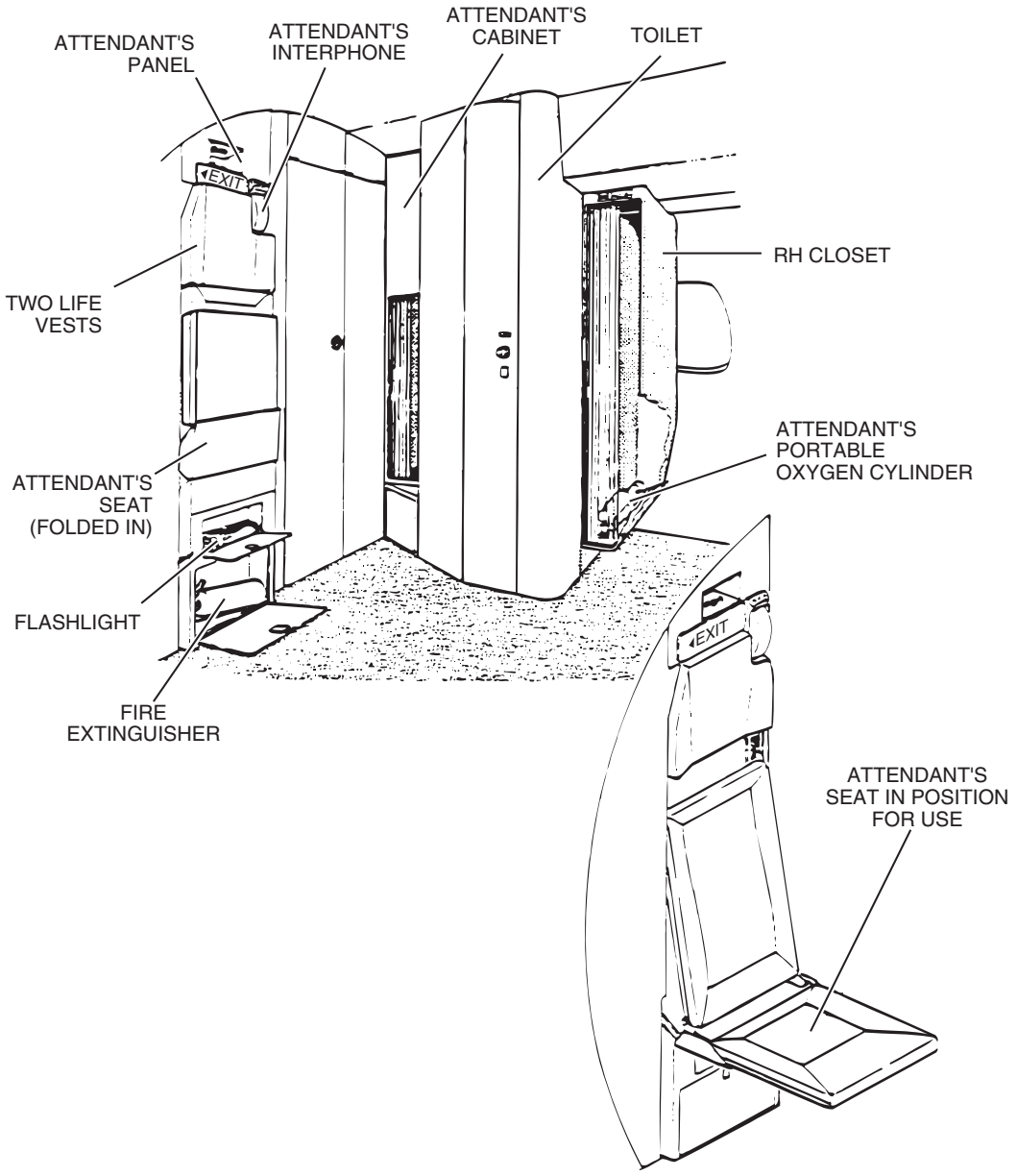


Figure 1-14. Attendant's Station (Typical)

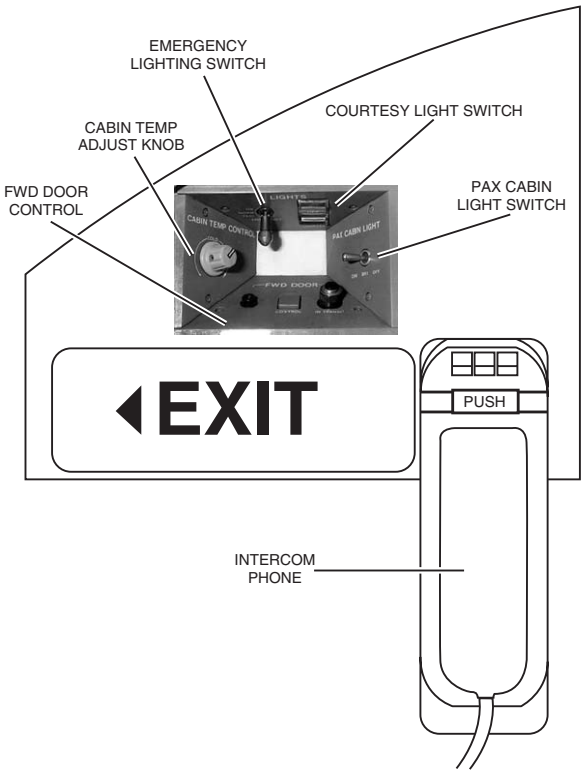


Figure 1-15. Attendant's Panel

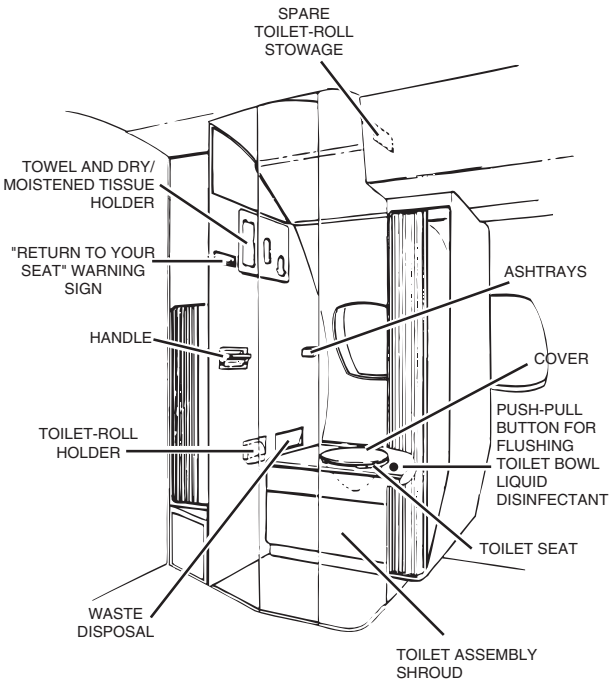


Figure 1-16. Toilet

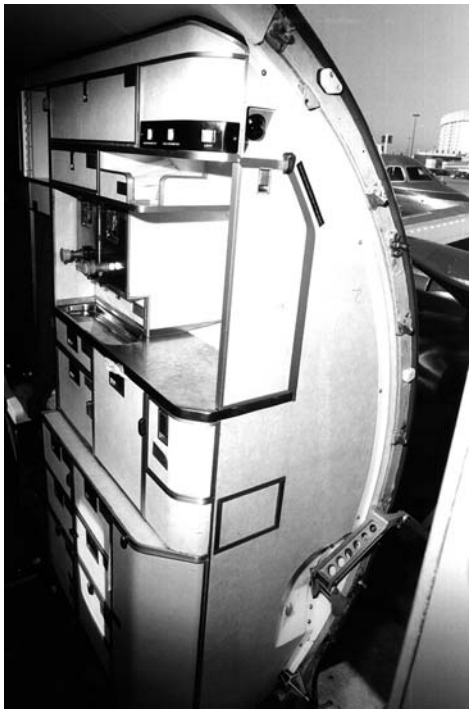


Figure 1-17. Galley

DOORS

Forward Entry Door

The forward entry door, located just aft of the cockpit on the left side, incorporates folding air stairs and is hinged at its lower edge.

In normal operation, the door is closed (raised) by two hydraulic door actuators, and opened (lowered) manually with hydraulic dampening. With no hydraulic pumps operating, an accumulator provides sufficient pressure for four complete operations of the door. The door may also be raised manually from outside by a ground attendant.

The door may be operated from either inside or outside the aircraft. The interior control is located on the flight attendant's panel just inside the door, and the exterior control is on the fuselage at the lower left side of the door (Figure 1-18). Each control panel incorporates a pushbutton which energizes a solenoid valve, allowing hydraulic power to raise the door, and blue light that illuminates while the door is moving up. The interior control panel also incorporates a circuit breaker.

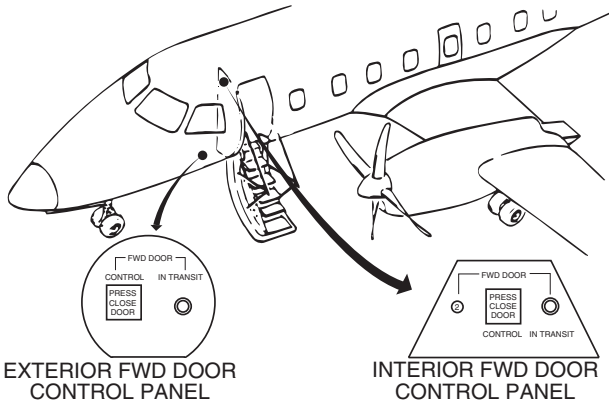


Figure 1-18. Forward Door Controls

With the door in the raised position it is then closed and locked by operation of either the inner or outer door handles.

When the forward door is not closed and locked the FORWARD light on the DOORS panel, shown in Figures 1-19 and 1-24, illuminates.

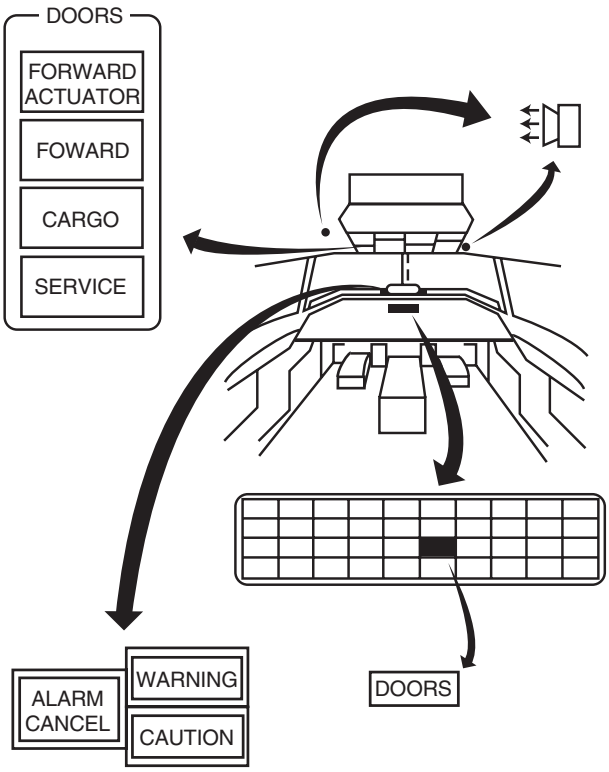


Figure 1-19. Door Warning Lights

If the forward door actuator remains pressurized after closing, blocking the door hydraulically, the FORWARD ACTUATOR light on the DOORS panel illuminates (Figures 1-19 and 1-24). In this event, an emergency valve (Figure 1-20) is provided in the cockpit to allow the door to be lowered

Normal door operation from outside the aircraft is shown in Figure 1-21.

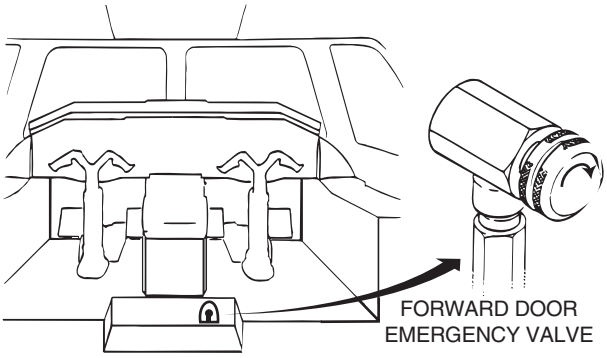


Figure 1-20. Forward Door Emergency Valve



Figure 1-21. Forward Entry Door Operation

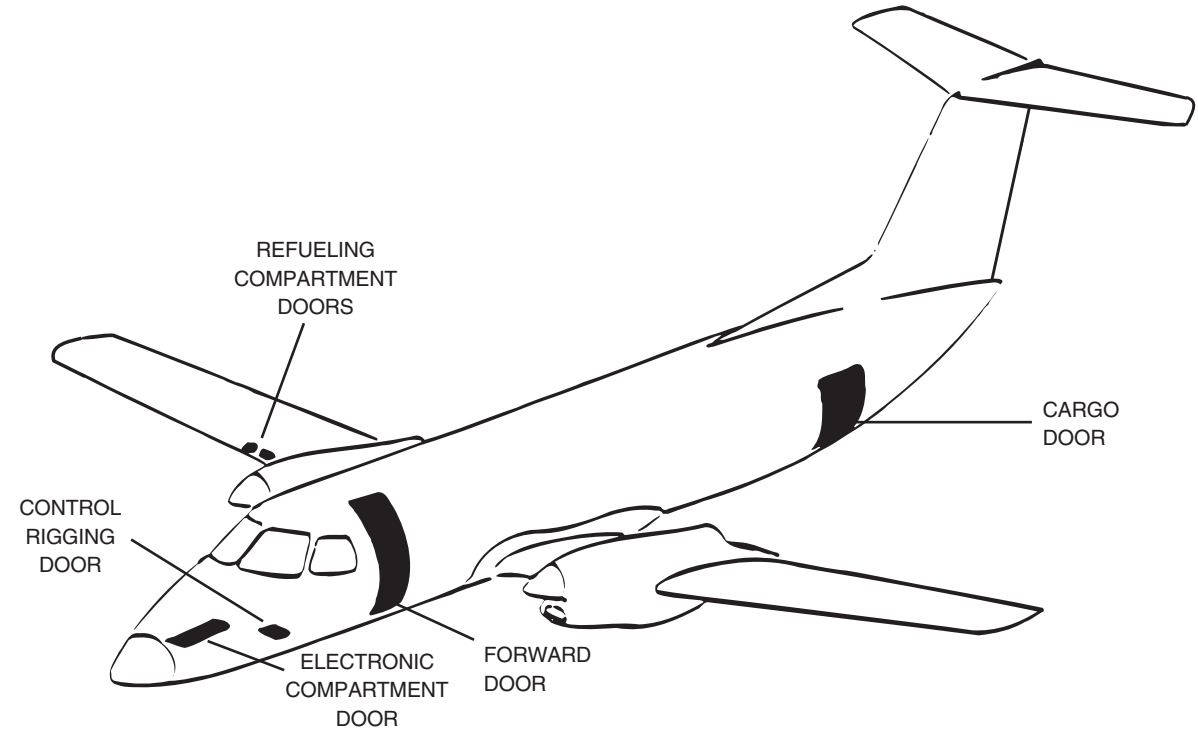


Figure 1-22. Cargo/Service Door Location

Cargo/Service Doors

Location of the cargo and service doors is shown in Figure 1-22.

Cargo Door

Operation of the cargo door is shown in Figure 1-23. The initial opening (displacement of the door inward) and final closing (displacement of the door outward) are controlled by the external handle in the lower half of the door. This handle is also responsible for door locking.

When the cargo door is not closed and locked the CARGO light on the DOORS panel (Figures 1-19 and 1-24) illuminates.

Service Doors

The service doors are the external doors which provide maintenance access to airplane systems and equipment.

The controls rigging door is located on the fuselage beneath the cockpit providing access to the fuselage pressurized compartment. The electronic compartment door is located inside the nose landing gear compartment. The fuel compartment doors are located under the wing outboard of the right engine nacelle.

If any door is not closed and locked the SERVICE light on the DOORS panel (Figures 1-19 and 1-24) illuminates.

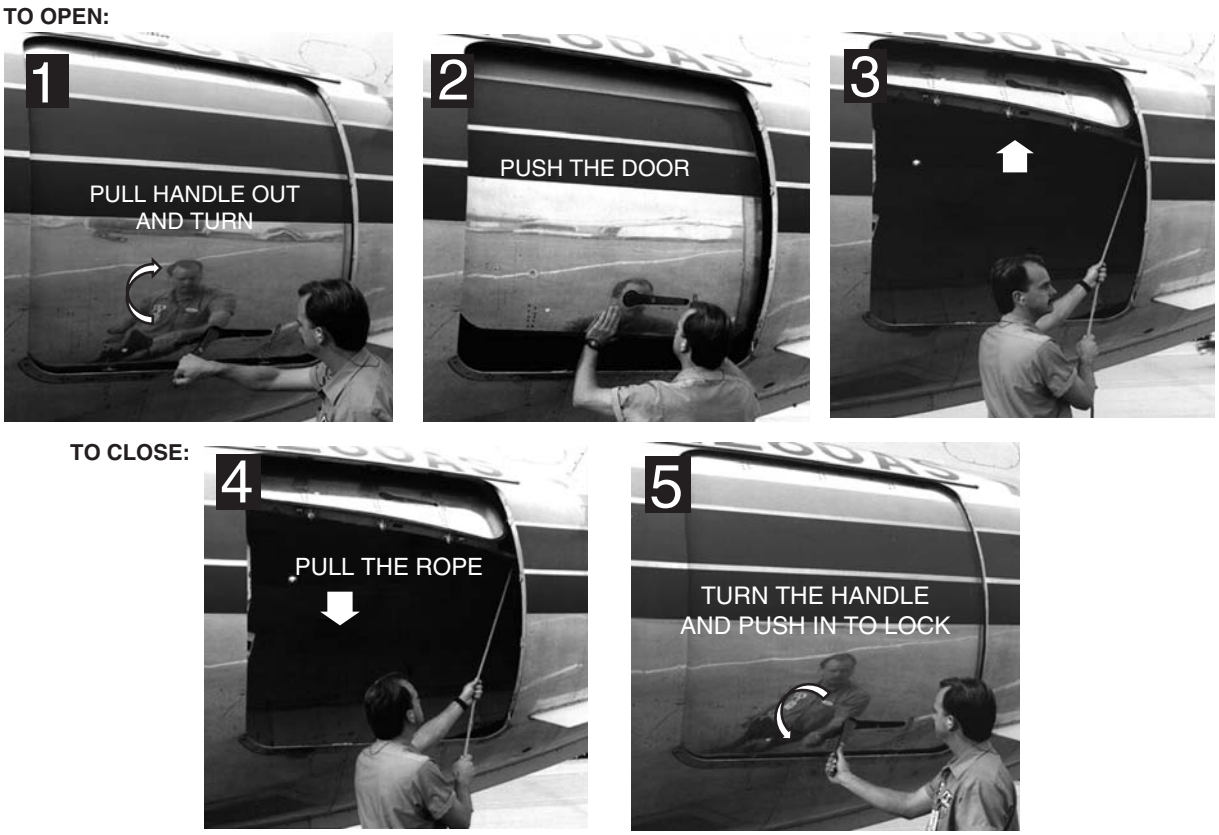


Figure 1-23. Cargo Door Operation

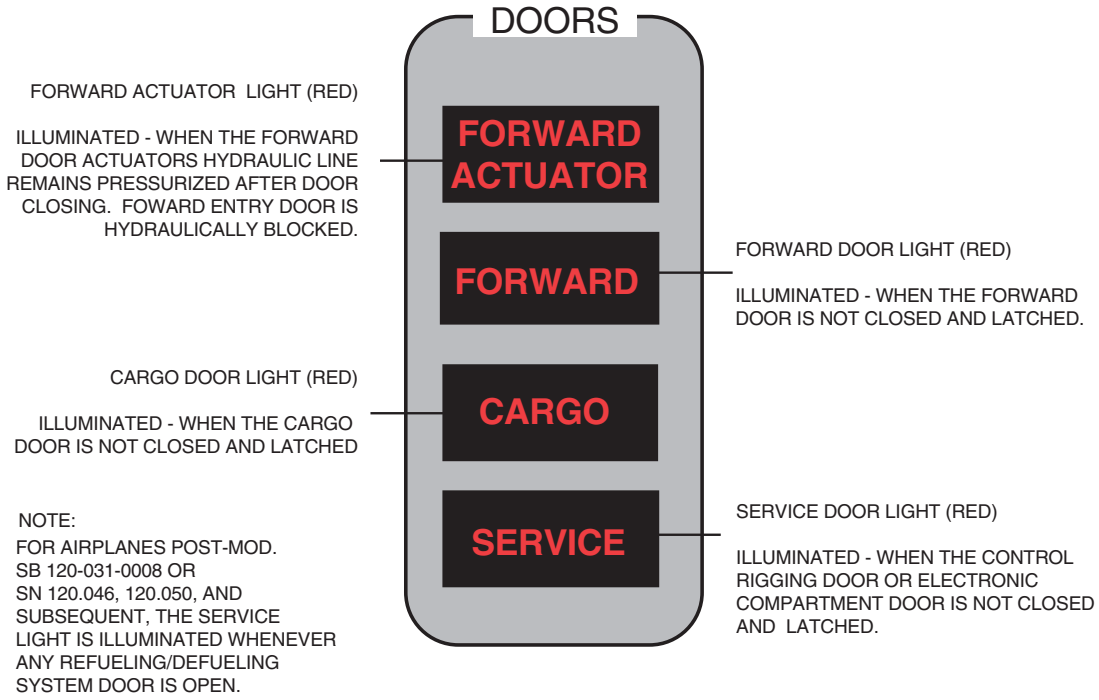


Figure 1-24. Doors Warning Lights (Overhead Panel)

EMERGENCY EQUIPMENT

GENERAL

Location of fire extinguishers, portable oxygen cylinders, oxygen masks, smoke goggles, escape ropes, and first aid kit is shown in Figure 1-25. Emergency flashlights, though not shown, are also provided in the cockpit and cabin.

Passenger seat cushions can serve as a floatation device and are easily removable. Life jackets for the attendant and observer are stowed in the attendant's panel, and for the pilots under their seats (Figure 1-26). Escape rope location and use is shown in Figure 1-27.

EMERGENCY LOCATOR TRANSMITTER

An Emergency Locator Transmitter (ELT) is located in the dorsal fin (Figure 1-25). A remote switch on the copilot's panel selects either automatic (ARM) or manual (ON RESET) activation. With the remote switch in the ARM position, the ELT automatically transmits on 121.5 and 243.0 MHz when the airplane is subjected to a longitudinal deceleration of 5 to 7 G.

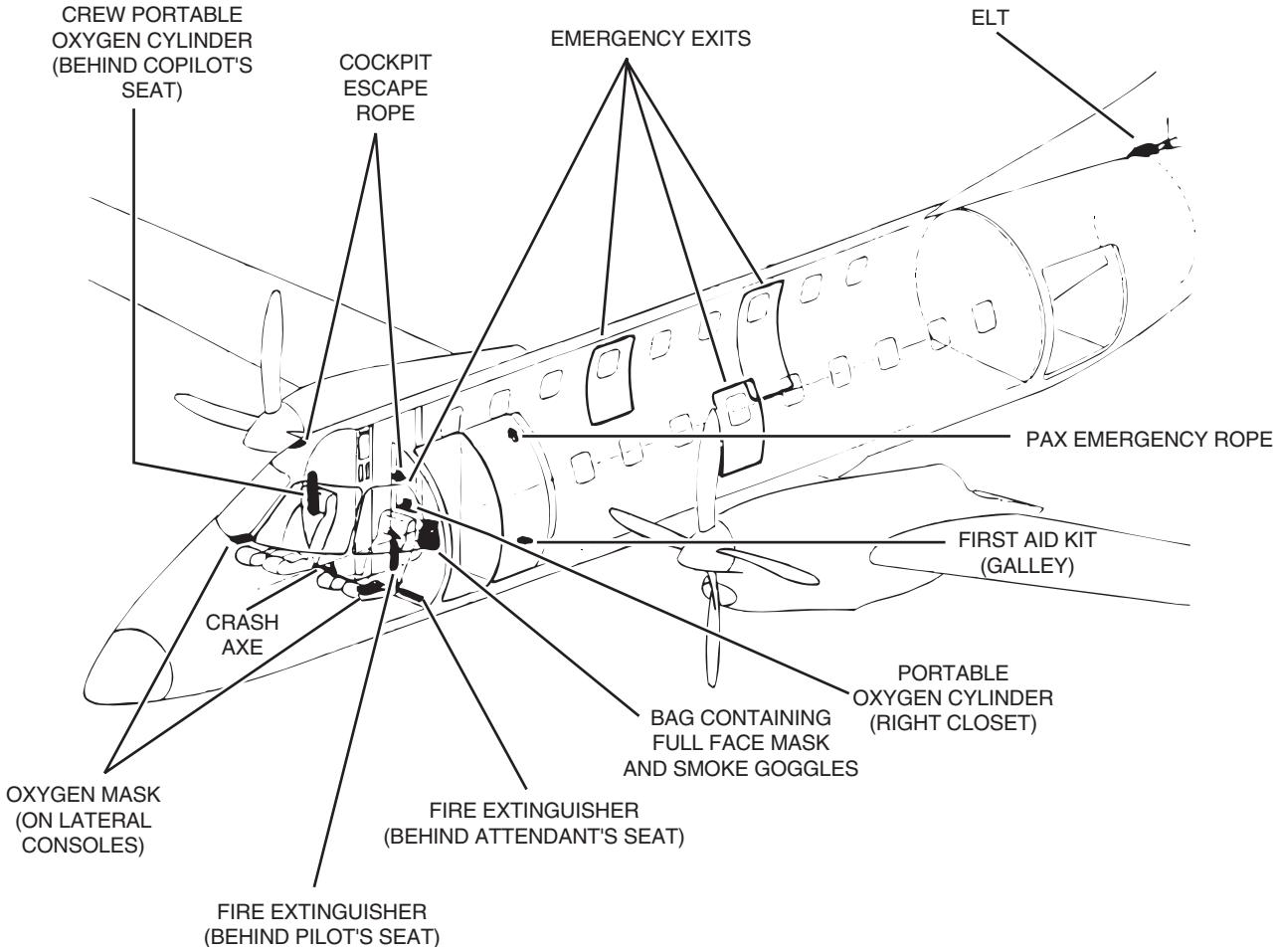


Figure 1-25. Standard Emergency Equipment Location

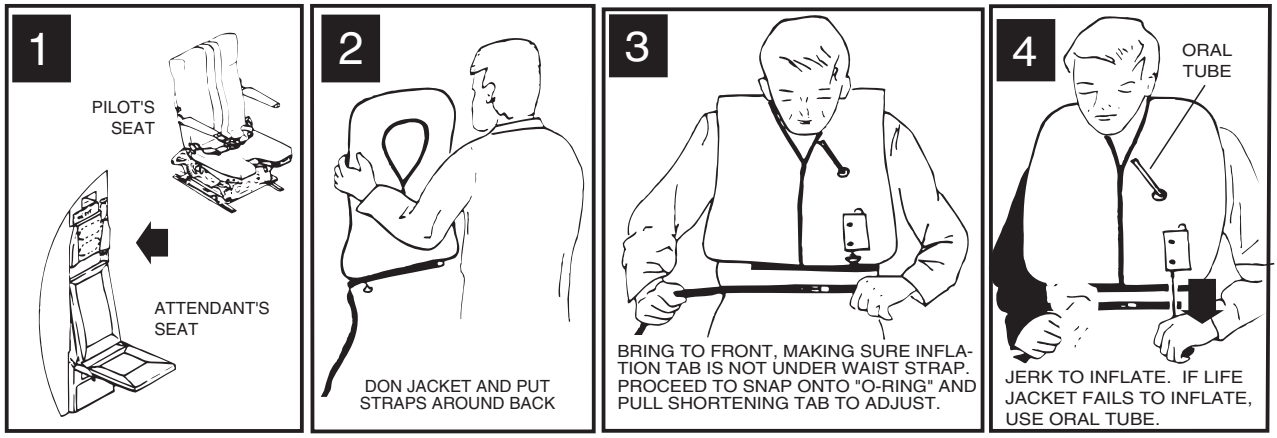


Figure 1-26. Life Vest Location and Operation

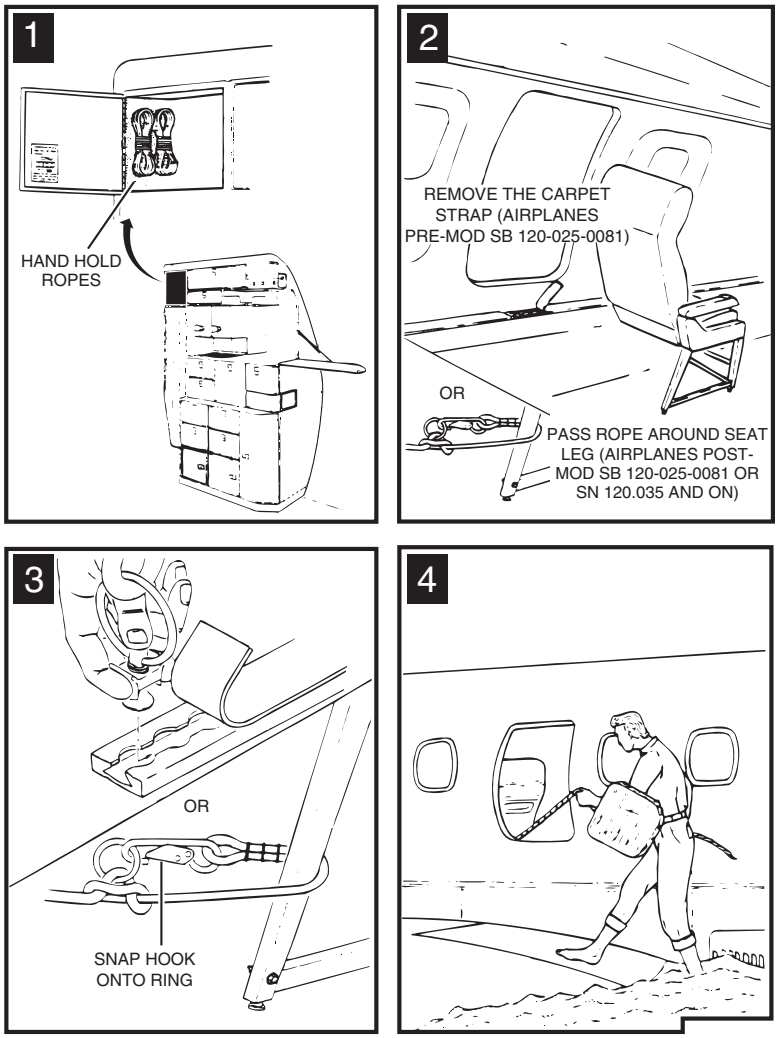


Figure 1-27. Hand Hold Rope Location and Use

EMERGENCY EXITS

In addition to each pilot's direct vision window, there are three emergency exits in the cabin of the aircraft. Two are overwing exits on each side of the fuselage and one is a floor level exit on the right rear side. All are plug-type exits that open to the inside of the cabin from either inside or outside the fuselage. Emergency exit operation is shown in Figure 1-28.

EMERGENCY LIGHTING

Emergency lighting is provided internally for each emergency exit door, the main entry door, and the aisle. External lighting illuminates the wing and ground in the vicinity of each exit.

The emergency lights and their operation are covered in Chapter Three, "Lighting".

SYSTEMS

ELECTRICAL SYSTEMS

Two 400-amp, 28-volt DC starter-generators, one on each engine, are the primary source of electrical power. Two 150-amp, 28-volt DC generators, one each in the propeller reduction gearbox, supply power to essential circuits in case of a complete failure of the primary 28 VDC sources.

The airplane AC power is provided by two static inverters, one being a standby.

A 24-volt nickel-cadmium battery is designed to assist each starter-generator during the engine starting cycle, and supply essential loads in the event of complete generator/engine failure.



Figure 1-28. Emergency Exit Operation

An emergency battery is available as a backup for selected loads and to provide emergency power for both the standby horizon and generator control units (GCUs).

A starter-generator installed on the APU, identical to the main generators, is used to start the APU. It may be used to provide electrical power to all buses on the ground, and for standby power in flight.

FUEL SYSTEM

The fuel system is made up of two tanks, one in each wing. Each wing tank is made up of two independent cells separated by the wheelwell. These inboard and outboard cells are interconnected by tubes for gravity fuel transfer. Fuel is supplied to the engine by pumps installed in a collector tank located in the lowest region of each inboard fuel cell. Each wing tank has a usable capacity of 437 US Gal.

Each engine is supplied independently from its wing tank. A crossfeed line allows either wing tank to supply both engines simultaneously. Gages for monitoring fuel flow and fuel quantity are located on the fuel management panel in the cockpit.

The aircraft may be gravity fueled using overwing fillercaps, and manually defueled. A pressurized system is provided for faster fueling/defueling.

AUXILIARY POWER UNIT

The APU, located in the tail cone, is a gas turbine engine used to supply pressurized air and electrical power to the airplane.

POWERPLANT

Two Pratt and Whitney PW118 or PW118A turboprop engines, both flat rated at 1800 SHP, are mounted on the wings. The engine is a three-shaft, two-spool gas generator with a free power turbine.

Engine airflow is straight-through with air entering an intake below the propeller spinner, then through an S-duct to the engine. This S-duct provides inertial separation and protection in the event of foreign object ingestion.

Propeller

Each engine is equipped with a Hamilton Sundstrand model 14 RF-9, four-blade, constant speed, reversible, full feathering propeller. Automatic feathering and propeller synchronization systems are installed.

FIRE PROTECTION

Fire detectors installed in the engine accessories section, wheelwell, pipe zones and APU compartment provide fire or overheat warning.

The engine fire control panel, installed on the center glareshield panel, is provided with bottle discharge ability and INOP lights, shutoff valve position indicators, fire warning lights, extinguishing handles, and a test button. The APU fire control panel on the overhead APU CONTROL panel, provides the means for APU fire detection and extinguishing.

A smoke detection system is installed for use when the airplane is converted to cargo configuration.

ICE AND RAIN PROTECTION

An electrical anti-icing system protects the left and right windshields, pitot/static tubes, static ports, angle of attack, and side slip sensors.

Electrical deicing system protects the propeller blades to permit unrestricted operation into known icing conditions. The wings, stabilizers, vertical fin, and engine air inlets are protected by inflatable deicers.

The rain removal system consists of two independent two-speed wipers, one on each windshield.

AIR CONDITIONING AND PRESSURIZATION

The air-conditioning and pressurization system provides conditioned pressurized air to the cockpit and passenger cabin. The system is operated using bleed air from either the engines or the APU.

The cockpit and passenger cabin are supplied by separate air ducting with cross-connecting capabilities. Temperature of the two zones is independently controlled.

The cabin pressure control, designed to maintain a 7 psi cabin/ambient pressure differential, maintains the cabin at sea level pressure up to an altitude of 16,800 ft. Control is accomplished by an electropneumatic valve (automatic mode) or by a pneumatic valve (manual mode).

HYDRAULIC SYSTEM

Hydraulic power is provided by two independent systems. Each system is powered by a main pump, driven by the left or right propeller gearbox. DC powered electrical pumps provide backup pressure for each system. Two pressurized hydraulic reservoirs are arranged so that leakage in either of the systems will not affect operation of the other. Each hydraulic reservoir is equipped with transmitters and switches that display system status on the HYDRAULIC POWER PANEL located on the cockpit overhead panel.

LANDING GEAR AND BRAKES

The landing gear is a conventional tricycle, dual-wheel, forward retracting type.

Three modes are provided for operation of the landing gear:

- Normal hydraulic retraction and extension
- Alternate electrical override extension
- Emergency free-fall extension with manual release of the uplocks.

Nosewheel steering is controlled through a steering handle on the pilot's left console. Limited nosewheel steering is available with the rudder pedals.

A normal brake system is actuated by conventional means through the pilot or copilot rudder pedals and controlled by a dual anti-skid system.

An emergency brake system is actuated through a handle and control valve. Pressure to the brakes is proportional to handle displacement.

FLIGHT CONTROLS

Flight controls are operated by conventional control wheels, columns, and rudder pedals for pilot and copilot. They are normally interconnected and jointly operated. In an emergency, control wheels and columns may be disconnected between the pilot and copilot, rendering the airplane controllable by either. Pedals are independently adjustable, allowing comfortable operation.

The elevators and ailerons are mechanically actuated. The rudder is hydraulically actuated with a mechanical back-up.

The flaps, divided into three panels per wing, are hydraulically actuated.

Elevator and aileron trim is by mechanical actuators to trim tabs. Rudder trim is hydraulic.

AVIONICS

Flight instruments

Conventional air data instruments (airspeed, altimeter, and vertical speed indicator) are provided with separate pitot/static sources for pilot and copilot.

An independent standby attitude indicator on the center panel, and a standby compass in the top of the windshield center post, provide back-up attitude and heading information.

Navigation

The navigation system includes the following equipment:

- Electronic flight instrument system (EFIS)
- Two independent attitude and heading reference systems (AHRS)
- Two radiomagnetic indicators (RMI)
- Two distance measuring equipment (DME) systems
- Two VHF/NAV(VOR/ILS/MB) radios
- Two ATC transponders
- One automatic direction finder (ADF)
- One radio altimeter system

The EFIS displays consist of two electronic attitude director indicators (EADI), two electronic horizontal situation indicators (EHSI), and a multifunction display (MFD). The EADI and EHSI are color cathode-ray tube displays.

The AHRS provides attitude and heading signals to the EFIS and autopilot/flight director; pitch and roll angle to the weather radar; and turn rate and normal acceleration data to the autopilot, if required.

Radar is displayed on the multifunction display and each EHSI when in the ARC or map mode.

Autoflight

The autoflight system is a fully integrated three-axis dual flight control system including manual electric trim. It is divided into two general systems:

- Flight director system
- Autopilot system

Available functions include heading, altitude and airspeed control, VOR/ILS approach coupling, glide-slope operation, and go-around mode.

Communication

The communication system includes:

- Interphone for communication between personnel in the cockpit, and flight crew members and cabin/ramp personnel
- Passenger address system for communication between flight crew members and passengers
- VHF for air-to-air and air-to-ground communication.

The airplane is equipped with a cockpit voice recorder.

OXYGEN

The airplane is equipped with a conventional gaseous oxygen system. One oxygen cylinder supplies low-pressure oxygen to both the crew and passenger systems. The crew system consists of three quick-donning masks. The passenger system consists of continuous flow masks in dispensing units, installed in the aisle ceiling. The units open automatically, when cabin altitude exceeds 14,000 feet, or manually by a switch installed on the control stand in the cockpit.

Other related equipment includes portable oxygen cylinders, smoke goggles, and full face masks.

PUBLICATIONS

The FAA-approved Airplane Flight Manual (AFM) is a required flight item. It contains the limitations, operating procedures, performance data pertinent to takeoffs and landings, and weight and balance data. It does not contain enroute performance information. The AFM always takes precedence over any other publication.

The *EMB 120 Operating Manua*l contains expanded descriptions of the airplane systems and operating procedures. It contains enroute flight planning information as well as some takeoff and landing performance information.

The EMB120 checklist (QRH–*Quick Referrence Handbook)* contains abbreviated operating procedures and abbreviated performance data. If any doubt exists or if the conditions are not covered by the checklist, the *AFM* must be consulted.

The *EMB 120 Weight and Balance Manual* contains detailed infomation in the form of tables and diagrams. However, it is not required to be in the airplane as the basic empty weight and moment and means of determining the center-of-gravity location are all contained in the *AFM*.

QUESTIONS

1. The EMB 120 is certified under:

A. FAR Part 25
B. FAR Part 61
C. FAR Part 91
D. Brazilian CAA Paragraph 7

2. The EMB 120 has how many generators?

A. 2
B. 3
C. 4
D. 5

3. The EMB 120 forward entry door may be:

A. Raised and lowered electrically
B. Raised and lowered manually
C. Raised hydraulically—lowered manually
D. Both B and C

4. The EMB 120 direct vision windows may be removed by the crew.

A. True
B. False

5. If the main entry door is not securely closed and locked:

A. An alarm will sound at the flight attendants station.
B. A door open light will illuminate in the cockpit.
C. A door solenoid will prevent engine start.
D. There is no indication of this situation.

6. How many cycles can the forward entry door be operated without recharging its accumulator?

A. 1
B. 2
C. 3
D. 4

CHAPTER 2
ELECTRICAL POWER SYSTEMS

CONTENTS

Page

INTRODUCTION 2-1
GENERAL 2-1
DC POWER 2-2
Components 2-2
Distribution 2-6
Control and Monitoring 2-9
AC POWER 2-12
Components 2-12
Distribution 2-12
Control and Monitoring 2-12
ELECTRICAL SYSTEM OPERATION 2-12
Normal Mode 2-15
Emergency Mode 2-20
Overcurrent Protection 2-22
PRE MOD SB 120-024-00051 ELECTRICAL SYSTEM DIFFERENCES 2-29
ELECTRICAL CONTROL PANEL SUMMARY 2-37
QUESTIONS 2-39

ILLUSTRATIONS

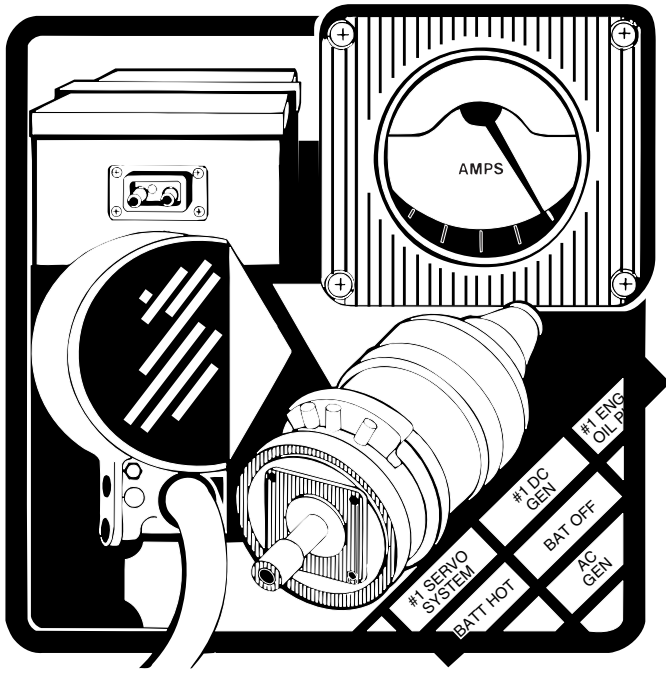
Figure	Title	Page
2-1	Electrical Component Location	2-2
2-2	Battery Location	2-2
2-3	Battery Temperature Gage	2-3
2-4	Backup Battery Switch	2-4
2-5	External Power Receptacle	2-5
2-6	Power Select Switch	2-5
2-7	Central DC Power Distribution	2-6
2-8	Auxiliary DC Power System	2-7
2-9	Backup DC Power System	2-8
2-10	Radio Master System	2-9
2-11	Electrical Control Panel—DC System	2-11
2-12	AC Power System	2-13
2-13	Electrical Power System	2-14
2-14	Electrical System Configuration—Battery Only	2-16
2-15	Electrical System Configuration—External Power	2-17
2-16	Electrical System Configuration—APU Generator	2-18
2-17	Electrical System Configuration—Single Engine	2-19
2-18	Emergency Mode—Both Main Generators and One Auxiliary Generator Failure	2-21
2-19	Emergency Mode—Total Generator Failure	2-23
2-20	Overcurrent Case 1: Short Circuit, Central DC Bus	2-24
2-21	Overcurrent Case 2: Short Circuit, Relay Box DC Bus 1	2-26
2-22	Overcurrent Case 3: Short Circuit, Relay Box DC Bus 2	2-28

2-23 Circuit-Breaker Panel 2-35
2-24 Electrical Control Panel 2-36

TABLE

Table Title Page
2-1 Electrical Bus Equipment Distribution 2-30

CHAPTER 2
ELECTRICAL POWER SYSTEMS



INTRODUCTION

This chapter provides a description of the EMB-120 Brasilia Mod SB 120-024-0008 electrical power system; (Pre Mod differences are covered in the final section.) Included is information on the DC and AC systems. The DC system consists of storage, generation, distribution, and system monitoring. The AC system consists of generation, distribution, and system monitoring. Provision is made for a limited supply of power during emergency conditions in flight, and for connection of an external power unit while on the ground.

GENERAL

The primary electrical power for the EMB-120 Brasilia is 28 VDC. Two main and two auxiliary generators are the primary power sources. Secondary sources that may be utilized are: an external power source; the auxiliary power unit (APU) generator; or the aircraft's nickel-cadmium battery. An additional battery supplies backup power to essential instruments and the standby horizon.

Distribution of DC power is primarily via two groups of buses, normally connected by a tie bus (the central DC bus). Each group of buses may be isolated from the central DC bus and powered by its respective generator. In flight, three buses are normally connected to the auxiliary generators. The emergency buses, normally connected to the right main generator, may be switched automatically or manually to other power sources in the event of main generator power loss.

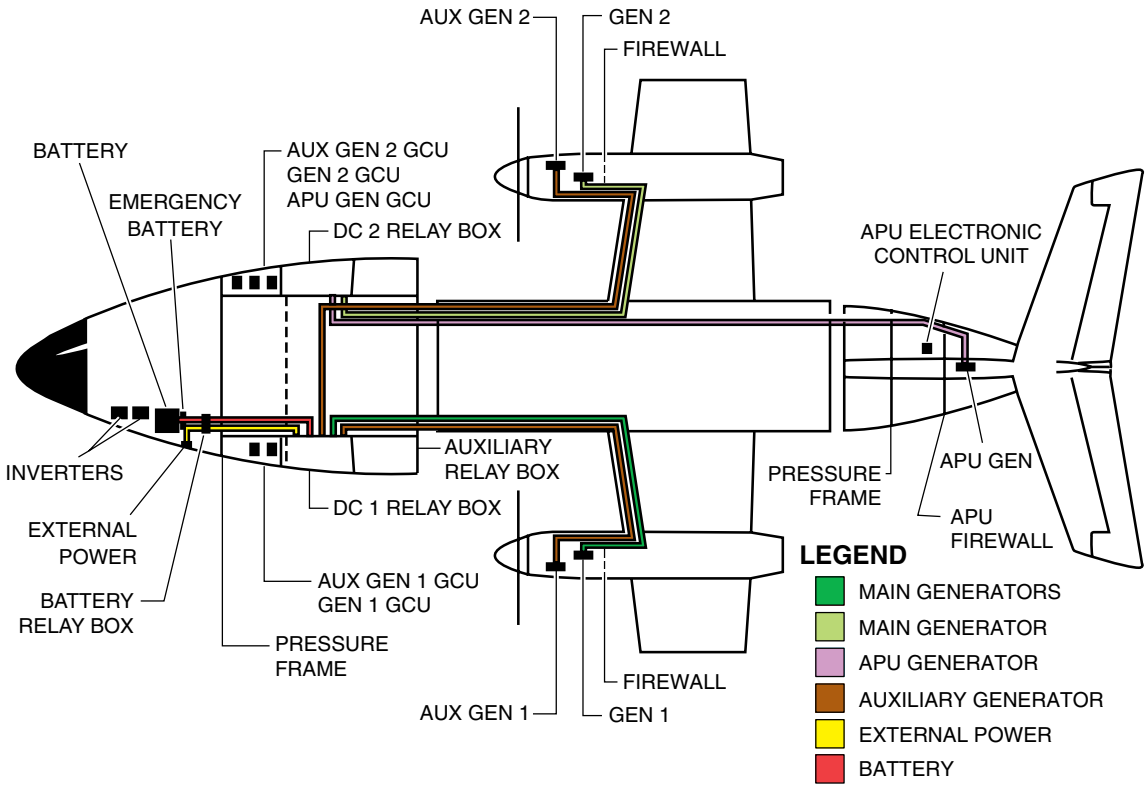


Figure 2-1. Electrical Component Location

AC electrical power is provided by either of two 250-VA, 400 Hz static inverters which convert 28 VDC power into 115 and 26 VAC. In normal operation, inverter No.1 powers the four AC buses, with inverter No.2 as standby power.

DC POWER

COMPONENTS

Many of the electrical system components are located in the nose of the aircraft (Figure 2-1). They are accessible through either the electronic compartment door inside the nose landing gear compartment, or individual panels on the outside of the aircraft.

Batteries

There are two batteries installed on the EMB 120: A 24-VDC, 36 ampere-hour, nickel-cadmium main battery, and a 24-VDC, 5 ampere-hour, sealed lead-acid, backup battery. Both are located in the nose of the aircraft and accessible through a panel just forward of the cockpit on the left side (Figure 2-2).



Figure 2-2. Battery Location

In flight, forced airflow is provided to the battery compartment to ensure suitable ventilation for the main battery.

Main Battery

The main battery is connected in parallel with the main generators. It is designed to power each generator during the engine starting cycle when external power is not available. It will also supply essential loads for approximately 30 minutes in the event of loss of all generators.

The battery is always connected to the hot battery bus.

Positioning the PWR SELECT switch to BATT energizes the battery contactor closed, connecting the battery to the central DC bus. The battery contactor is provided with a protective device that opens when the current from the central DC bus to the battery exceeds 500 amps.

On the ground with the battery as the only power source, a safety circuit inhibits power to the recirculation fans, gasper fan, and ground cooling fans.

Battery Temperature Monitoring System. A battery temperature monitoring system is used to warn the crew of a battery overheat condition.

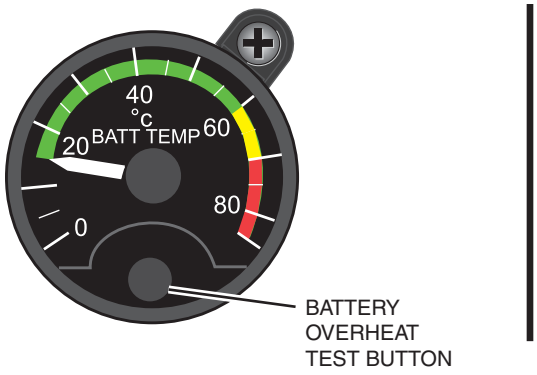


Figure 2-3. Battery Temperature Gage

A gage on the cockpit overhead electrical panel (Figure 2-3), displays battery temperature in the following ranges:

- 15 to 60°C . . . . . Green arc
- 60 to 70°C . . . . . Yellow arc
- 70 to 85°C . . . . . Red arc

If the battery temperature exceeds 70°C, the "Battery" voice warning sounds, the BATT OVERHEAT light on the multiple alarm panel illuminates, and the master WARNING lights flash.

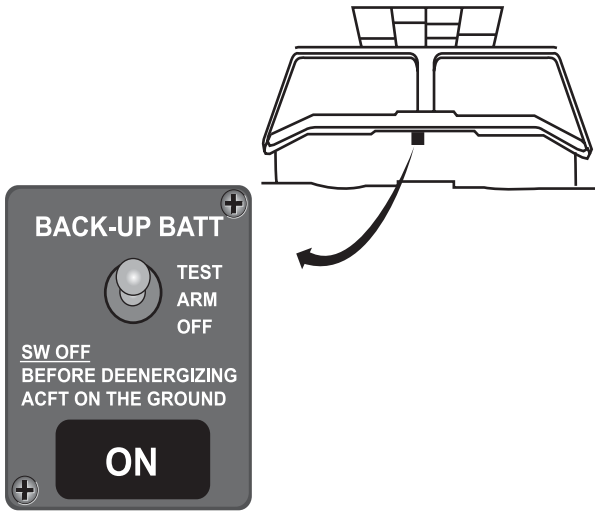


Figure 2-4. Backup Battery Switch

The system incorporates two individual temperature sensing elements in the battery case. One provides a signal for the gage and the other a signal for the warning system.

To test the sensing system, press and hold the battery overheat test button on the battery temperature gage. Heating elements heat the two sensors, causing the temperature indication to rise. When the temperature reaches 70°C, the alarm and warnings are triggered. Release the test button.

After the test button is released, the temperature will continue to rise briefly. It should then decrease and the alarm and warnings stop. Holding the test button beyond the warning activation point may cause damage to the sensing elements.

During the test the two sensors are heated by separate heating elements. This may cause a variation between the warning activation and the gage temperature. Minor differences are not a problem.

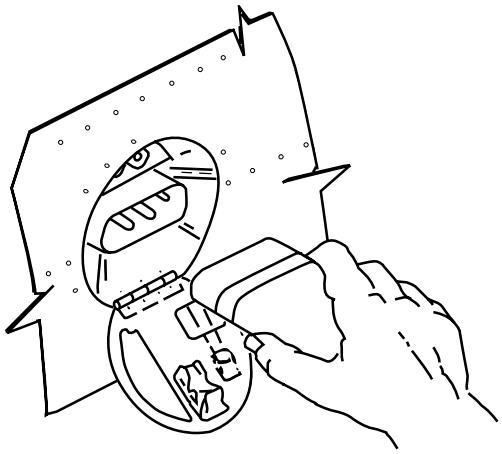


Figure 2-5. External Power Receptacle

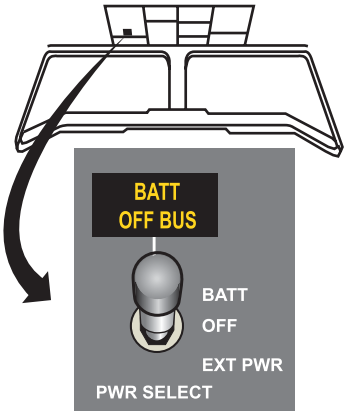


Figure 2-6. Power Select Switch

Backup Battery

The primary purpose of the backup battery is to provide an uninterrupted source of power to the standby artificial horizon should the normal power supply fail.

Its secondary purpose is to provide a stabilized power supply to sensitive electronic components during power transients such as engine starts.

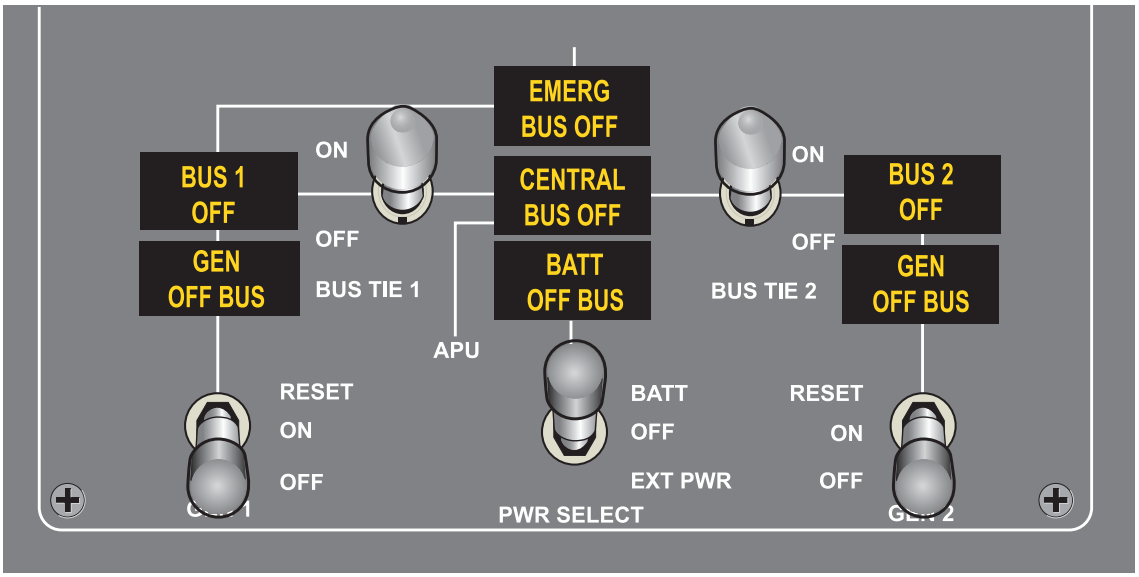
The backup battery is controlled by a three-position (OFF–ARM–TEST) switch on the center panel (Figure 2-4). The normal in-flight position for this switch is ARM.

Starter-Generators

With one installed in each engine, two 28-VDC, 400-amp (600-amp during start) engine-driven starter-generators are the primary power source for the electrical system. They may be connected independently or in parallel to the main distribution buses.

During engine start, the starter-generators act as motors and are powered by the central DC bus through the respective engine start contactor. At 50% N_H, (high-pressure compressor speed) when the starting cycle is completed, the generator control unit (GCU) automatically opens the engine start contactor allowing the starter-generator to operate as a generator.

The engine-driven starter-generators are connected to the aircraft electrical system by the main generator contactors. The GCUs command the



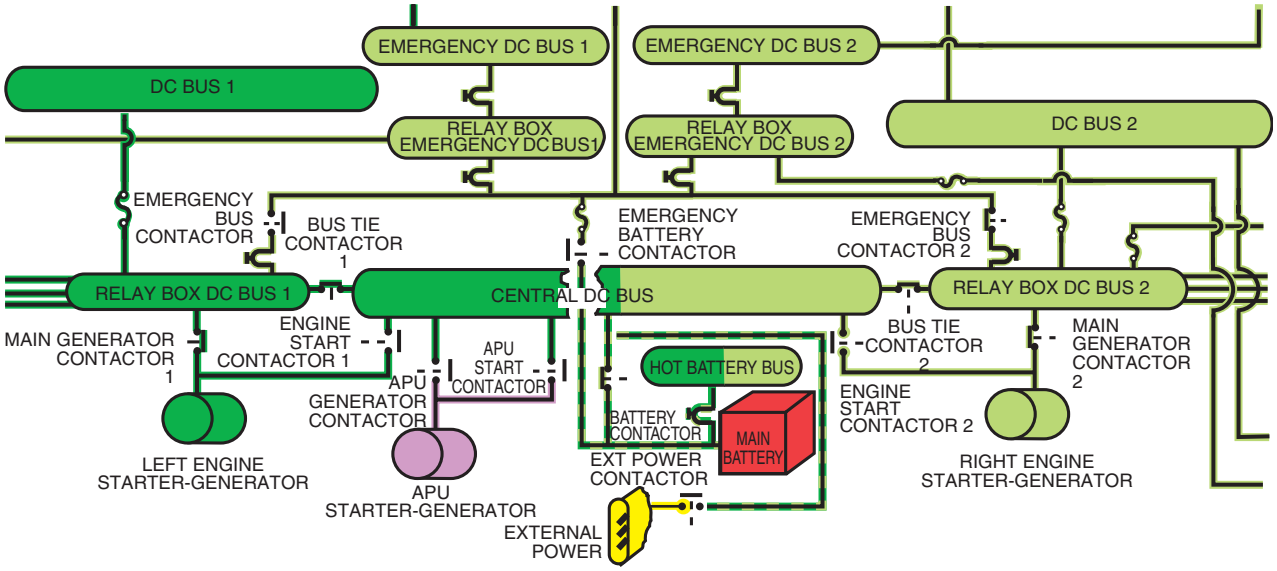


Figure 2-7. Central DC Power Distribution

closure of these contactors when the main generator switches, on the overhead electrical panel, are positioned to ON.

Auxiliary Generators

Two 28-VDC, 150-amp generators, each driven by its respective propeller reduction gearbox, supply power to the auxiliary power system.

The auxiliary generators are not connected to the aircraft electrical system unless the propeller speed (N_P) is greater than 70%.

APU Starter-Generator

The APU starter-generator is identical to the main starter-generators.

When the APU turbine reaches 95% rpm, the APU generator is able to provide its nominal electrical power output.

The main purpose of the APU generator is to substitute for the external power unit on the ground. It may also be used in parallel with the battery to power a starter-generator during engine start.

In flight, if required, the APU generator may be used in parallel with the main generators.

External Power

An external power unit may be connected to the aircraft DC system through a receptacle located just aft of the battery compartment panel (Figure 2-5).

Placing the PWR SELECT switch on the overhead electrical control panel (Figure 2-6) to the EXT PWR position closes the external power contactor, allowing the external source to power the central DC bus.

A protective circuit prevents the APU generator and the external power source from simultaneously supplying the central DC bus; priority is external

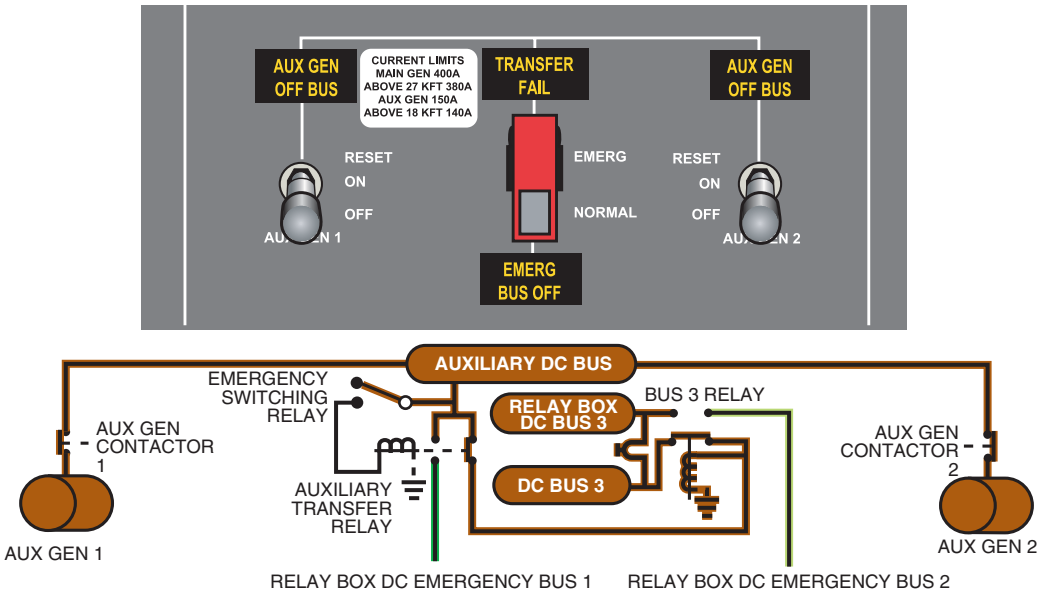


Figure 2-8. Auxiliary DC Power System

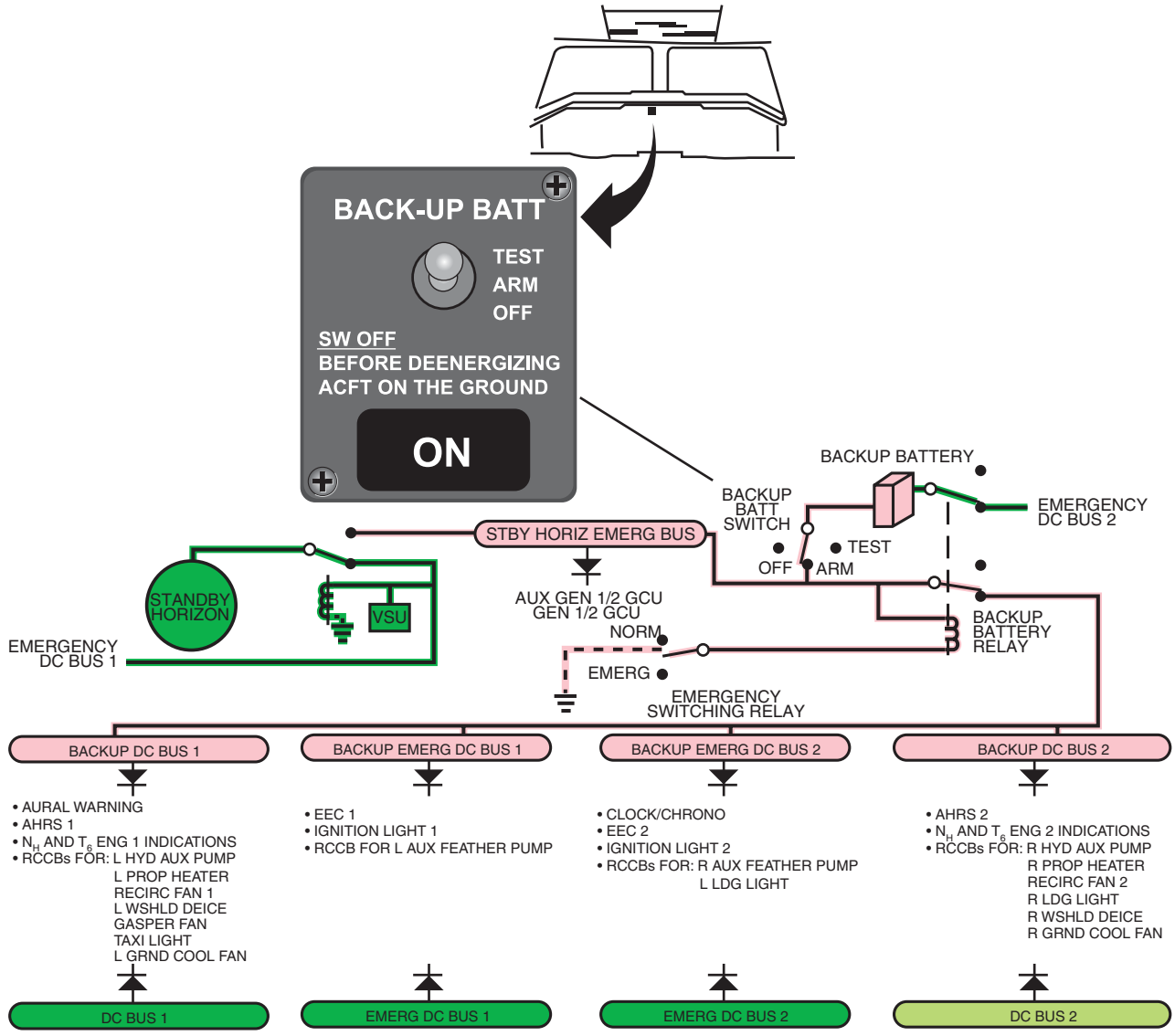


Figure 2-9. Backup DC Power System

power. A similar circuit also prevents the main generator and the external power source from simultaneously supplying the central DC bus; priority in this case is the main generator.

An overvoltage relay protects the central DC bus if external power exceeds 32 VDC. In this condition, the external power contactor will open, disconnecting the external power from the central DC bus.

The external power voltage may be monitored on the left voltammeter by setting the voltammeter selector to the CENTRAL BUS/APU GEN position.

DISTRIBUTION

Central DC System

Direct current is distributed throughout the aircraft as shown in Figure 2-7.

Each main generator is normally connected to the

EMB-120 PILOT TRAINING MANUAL

central DC bus, which functions as a tie bus. This allows either generator to supply the entire aircraft electrical load.

The battery is always connected directly to the hot battery bus. During normal operation, the power source energizing the central DC bus powers the hot battery bus and charges the battery.

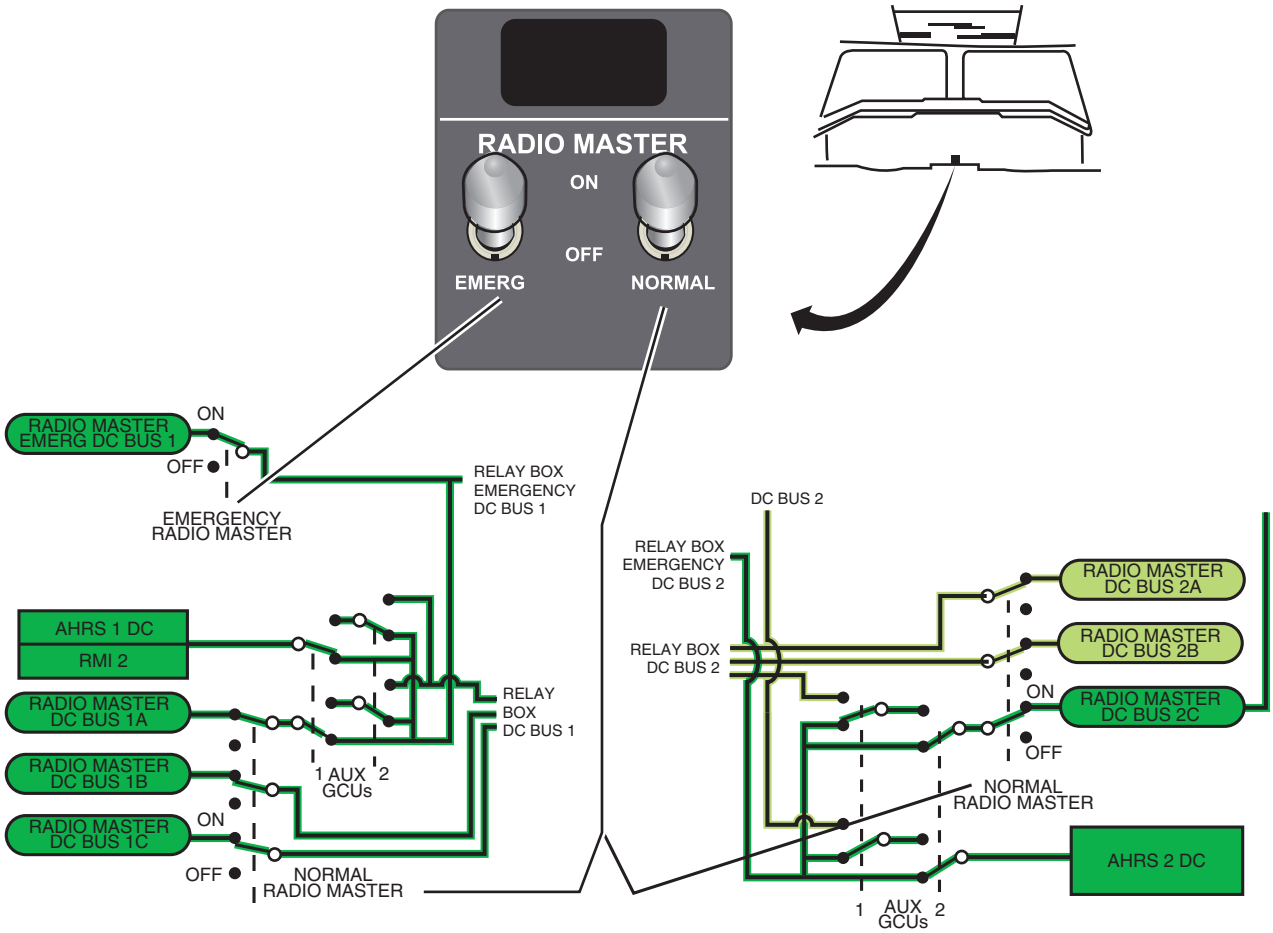


Figure 2-10. Radio Master System

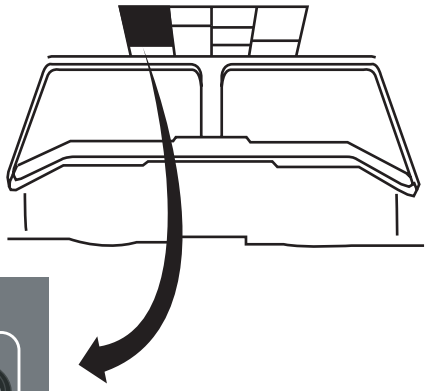
Auxiliary DC System

When propeller speed (N_P) reaches 70%, the auxiliary generator control unit (GCU) connects the auxiliary generator to the auxiliary DC bus (Figure 2-8). (The auxiliary generators are the only power source for the auxiliary DC bus.)

When powered, the auxiliary DC bus, energizes a relay that shifts the bus 3 contactor, switching both relay box DC bus 3 and DC bus 3 from relay box DC bus 2 to the auxiliary DC bus.

When N_P drops below 70%, the auxiliary GCU disconnects the generator from the auxiliary DC bus. If the auxiliary DC bus loses power (both auxiliary generators below 70%), both relay box DC bus 3 and DC bus 3 switch back to relay box DC bus 2.

The auxiliary GCUs also switch the primary navigation equipment (AHRS 1, AHRS 2, RMI 2, and radio master DC bus 1A and 2C) to the emergency buses.



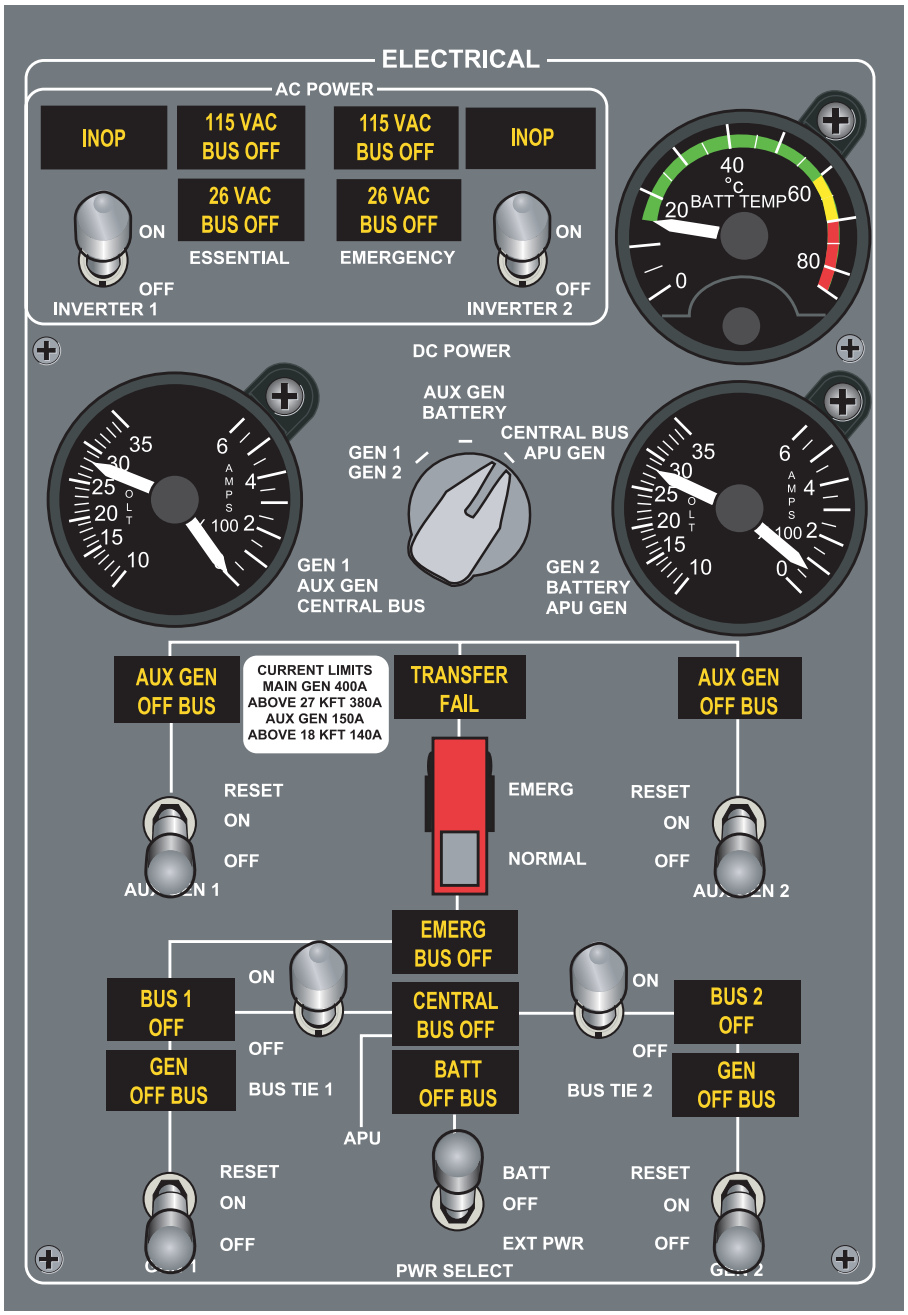


Figure 2-11. Electrical Control Panel—DC System

Backup DC Power System

The backup power system (Figure 2-9) consists of the backup battery, the BACKUP BATT switch on the center panel, and the following buses:

- Backup DC bus 1
- Backup DC bus 2
- Backup emergency DC bus 1
- Backup emergency DC bus 2
- Standby horizon emergency bus

Through one-way diodes, each backup bus is jointly connected to essential instruments normally powered by one of the four main DC buses (DC bus 1, DC bus 2, emergency DC bus 1, emergency DC bus 2).

The backup battery system operates in the normal mode whenever the BACKUP BATT switch is in the ARM position (and the electrical DC system is operating in the normal mode). In this mode, all five buses are in a powered, standby status. The backup battery is charged by emergency DC bus 2, and the standby attitude indicator is powered by emergency DC bus 1.

If the voltage on any of the four main DC buses drops below the voltage of their respective backup buses, power will be supplied to essential instruments through the backup buses.

The backup bus circuit breakers are all rated one amp higher than the corresponding main DC bus circuit breakers. As a result, the main bus circuit breakers on the overhead panel open before the backup bus circuit breakers. This feature assists fault detection by the pilot.

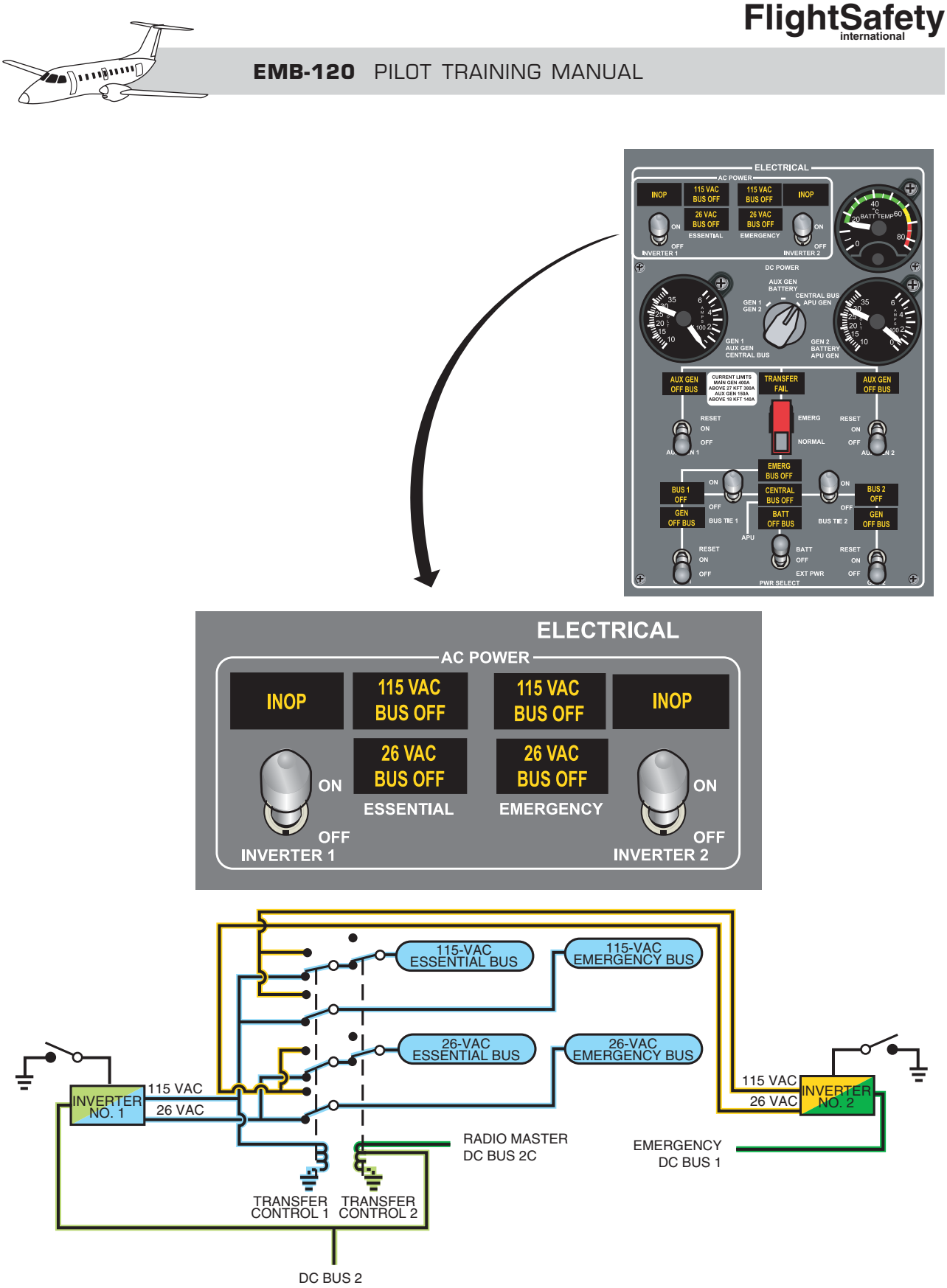


Figure 2-12. AC Power System

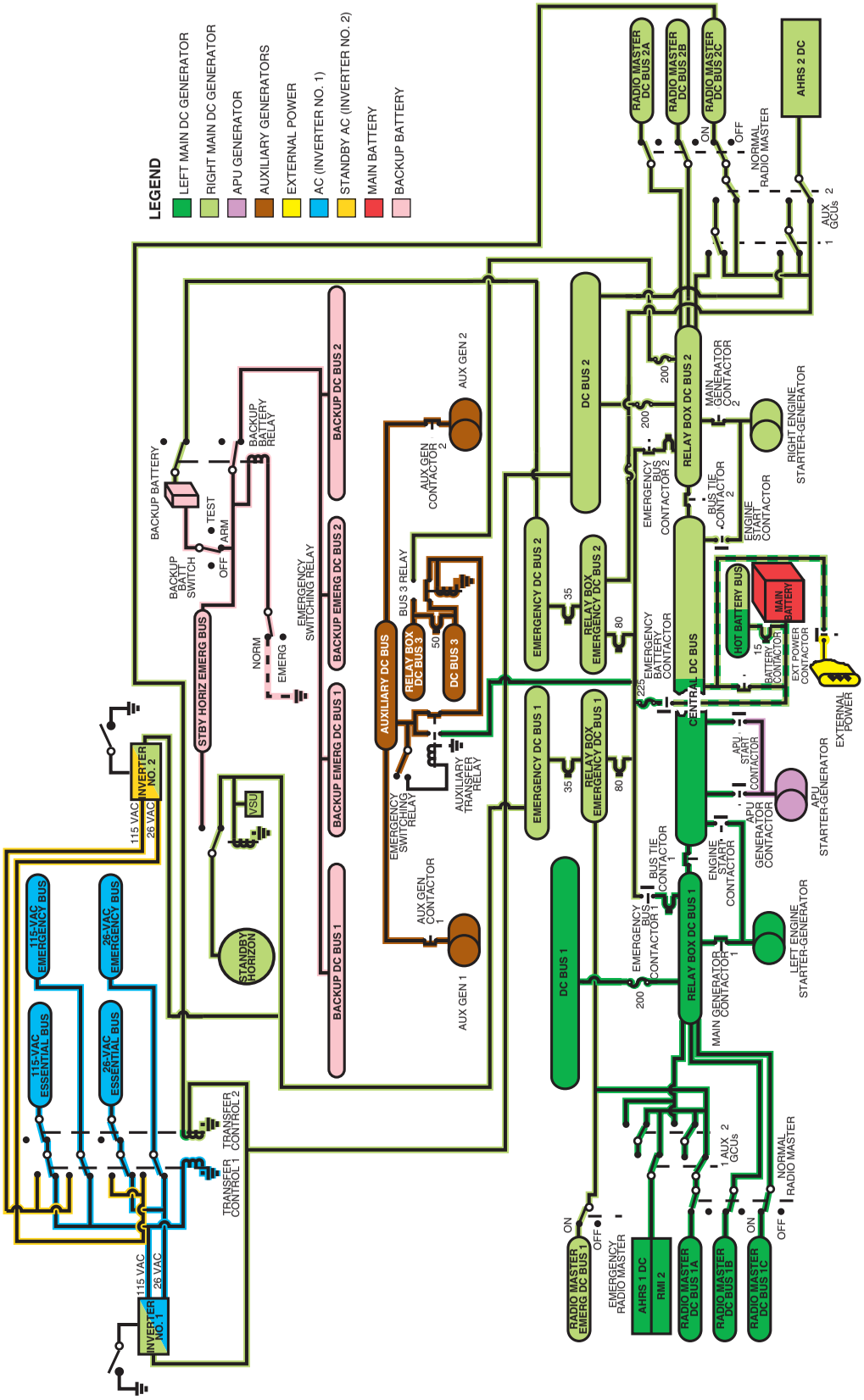


Figure 2-13. Electrical Power System

Radio Master System

During the engine starting cycle, bus voltage transients may cause damage to communication and navigation instruments. To avoid this, the radio master switches allow the pilot to turn off the avionics during engine start.

There are two radio master switches, one labeled EMERG and the other labeled NORMAL (Figure 2-10).

The radio master system consists of the radio master switches, six normal radio master buses, and one emergency radio master bus.

With both radio master switches in the ON position, the three corresponding relays are deenergized. This connects DC power to the radio master system.

When the switches are in the OFF position, the relays are energized and power is removed from the radio master system.

CONTROL AND MONITORING

Control of the DC electrical system is accomplished automatically through the five generator control units (GCUs), and manually by switch position on the overhead electrical control panel.

Monitoring of the electrical system is through the gages and indicator lights on the overhead electrical control panel.

GCUs

The main/APU starter-generators and the auxiliary generator are controlled by individual GCUs. These multifunction units also provide electrical fault protection.

Functions of the starter-generator GCUs are as follows:

- Initiates start cycle:
  - Closes start contactor
  - Energizes ignition circuit
  - Energizes return solenoid to empty drain collector (EPA) tank
  - Allows up to 600 amps for generator-aided cross start
  - Provides overspeed protection in event of sheared shaft
- Cancels start cycle at 50% RPM:
  - Opens start contactor
  - Deenergizes ignition circuit
  - Deenergizes EPA return solenoid
  - Enables the generator function
- Regulates generator output between 26 and 30 VDC (Normal 28.5 VDC)
- Overvoltage protection (> 32 VDC)
- Overcurrent protection (> 400 amps)
- Reverse current protection
- Generator field over excitation protection
- Paralleling protection (maintains generator loads within 10%)

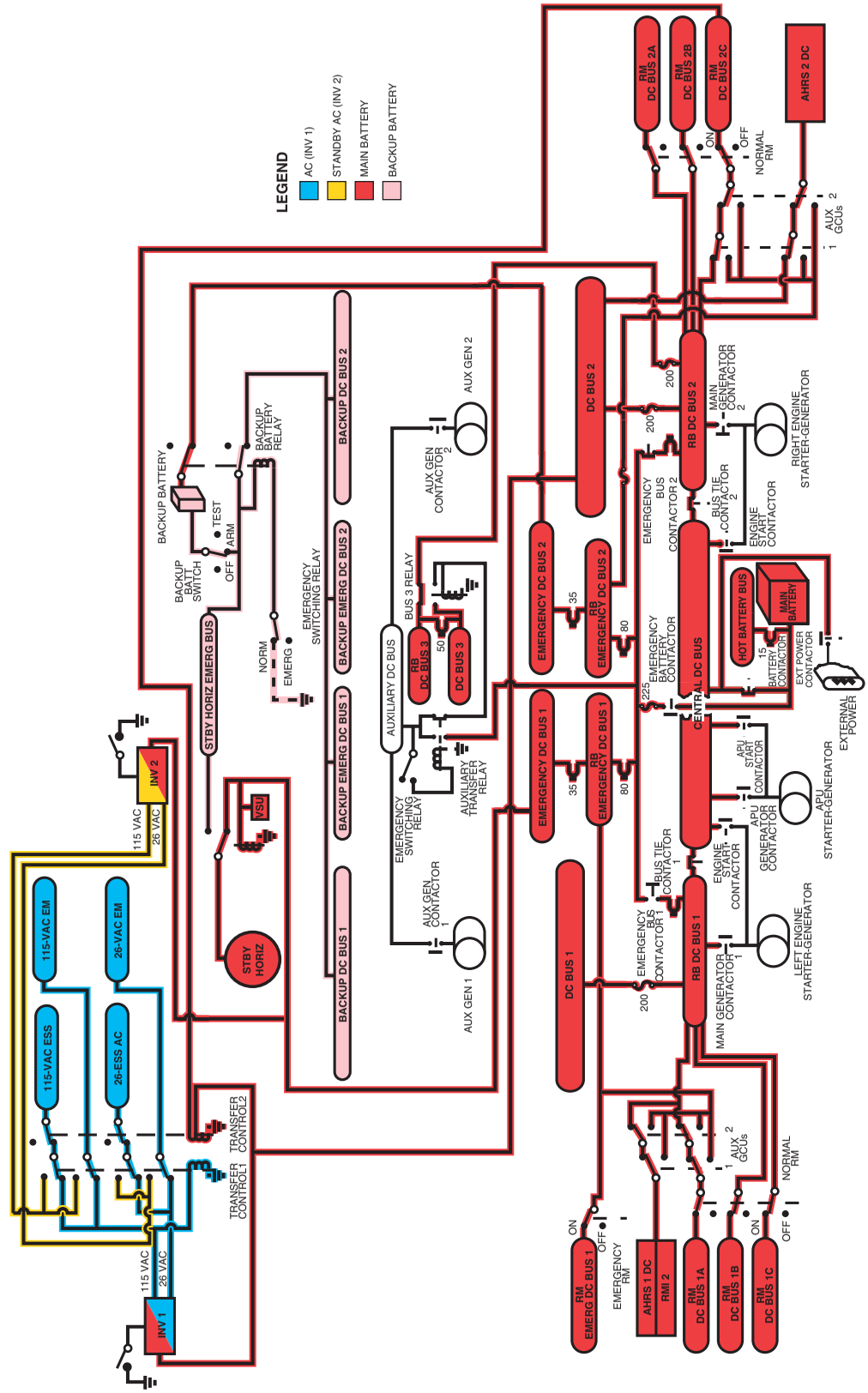


Figure 2-14. Electrical System Configuration—Battery Only

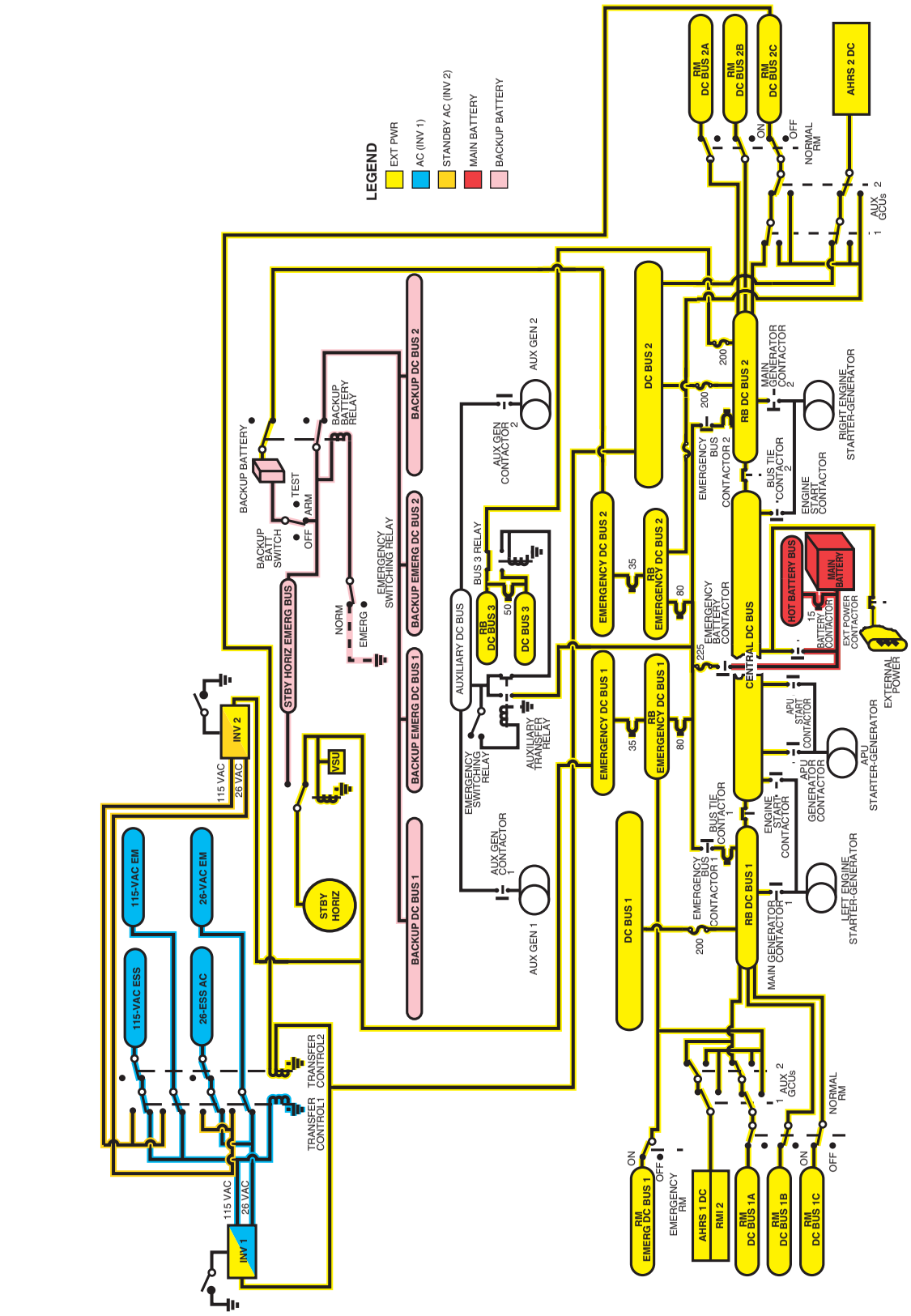


Figure 2-15. Electrical System Configuration—External Power

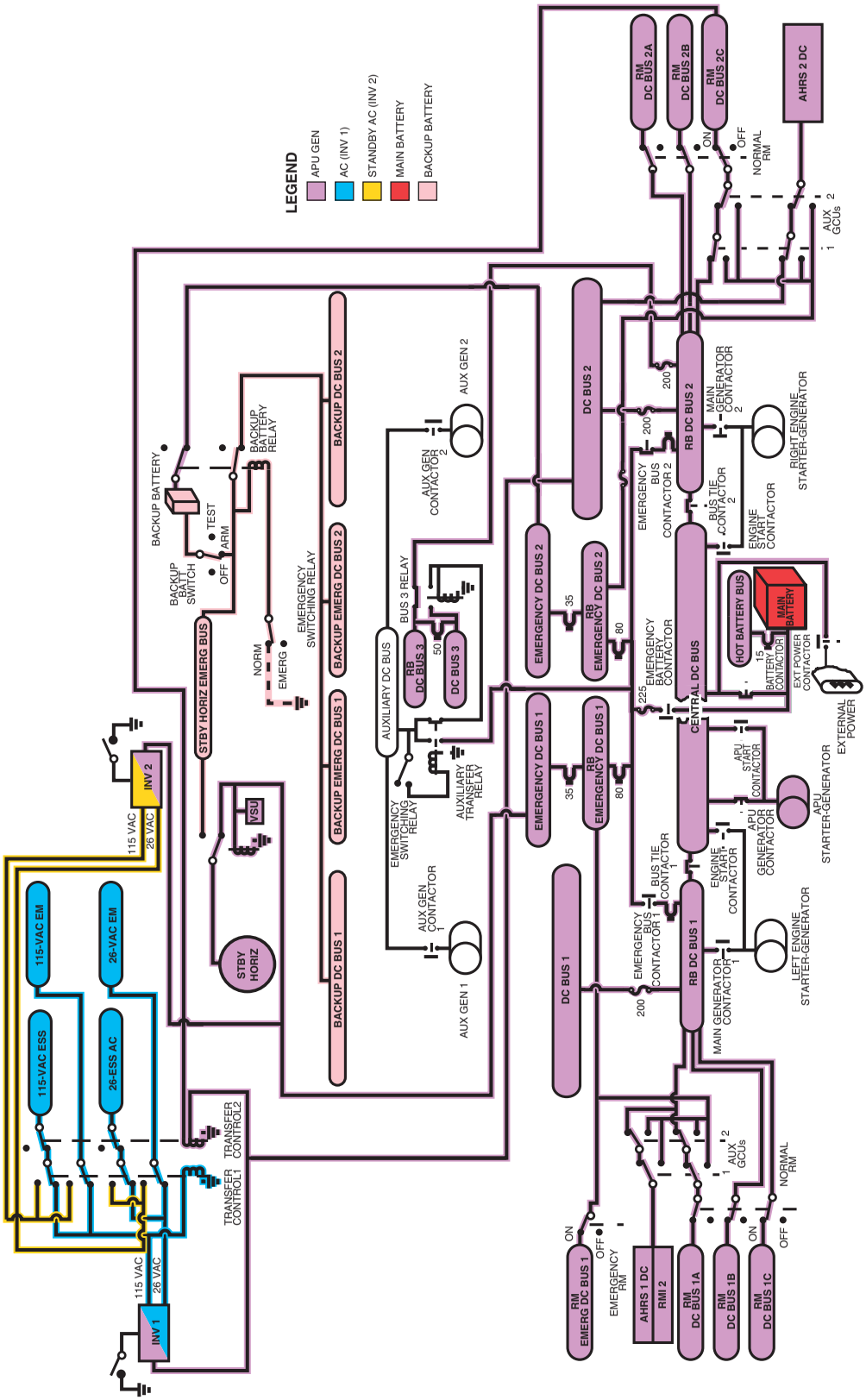


Figure 2-16. Electrical System Configuration—APU Generator

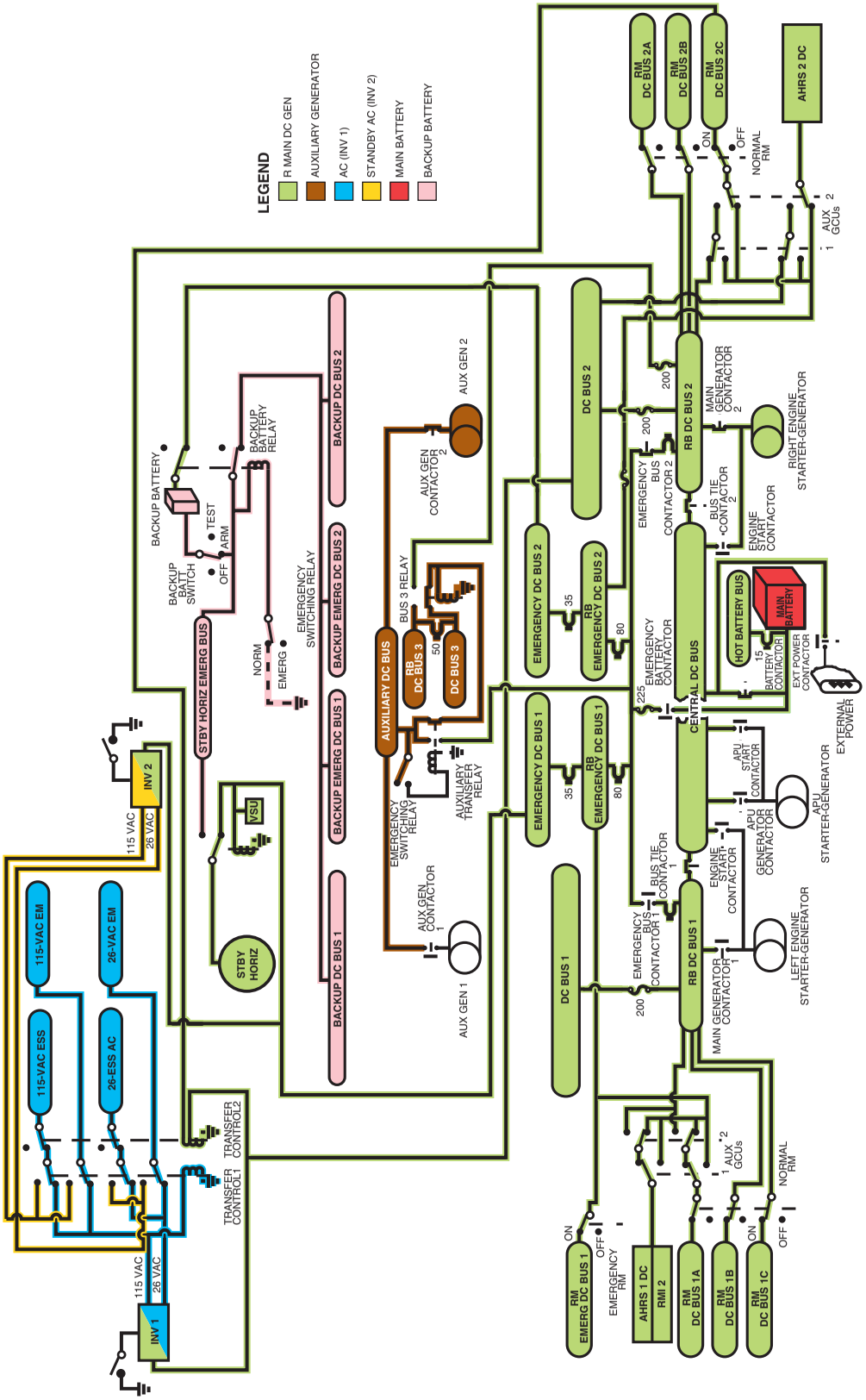


Figure 2-17. Electrical System Configuration—Single Engine

Functions of the auxiliary generator GCUs are as follows:

- At 70% propeller RPM:
  - Connects generator to auxiliary DC bus
  - Switches relay box DC bus 3 and DC bus 3 from relay box DC bus 2 to auxiliary DC bus
- Voltage regulation same as main generators
- Overvoltage protection (> 32.5 VDC)
- Undervoltage protection (< 23 ±1 VDC)
- Overcurrent protection (220 amps)
- Field excitation protection same as main generators
- Paralleling same as main generators
- Underspeed protection when propeller speed drops below 70%

Electrical Control Panel

The DC electrical system is monitored on the overhead electrical control panel (Figure 2-11) by two voltammeters, the battery temperature gage, and the following annunciator warning lights:

- GEN OFF BUS (left and right main generators)
- BUS 1 OFF
- BUS 2 OFF
- EMERG BUS OFF
- CENTRAL BUS OFF
- BATTERY OFF BUS
- AUX GEN OFF BUS (left and right auxiliary generators)
- TRANSFER FAIL

A voltammeter selector switch permits monitoring of the volts and amps of each main generator, the auxiliary DC bus, main battery, central bus, and APU generator.

AC POWER

COMPONENTS

Inverters

The alternating current system (Figure 2-12) consists of two independently powered inverters located in the nose section of the aircraft and accessible through the electronic compartment door in the nose wheel well. Each inverter produces 115- and 26-VAC, 400-Hz power.

Normally both inverters are energized, but only one is used to power the AC buses. Inverter No. 1, powered from DC bus 2, is the primary AC power source.

Inverter No. 2, powered from emergency DC bus 1, is kept in a powered backup status for inverter No. 1.

Transfer Control Relays

Two transfer control relays direct the output of the inverters to the appropriate AC buses.

Transfer control relay 1 is powered by the 115-VAC output of inverter No. 1. Transfer control relay 2 is powered by DC bus 2 (the power source for inverter No. 1) and radio master DC bus 2C.

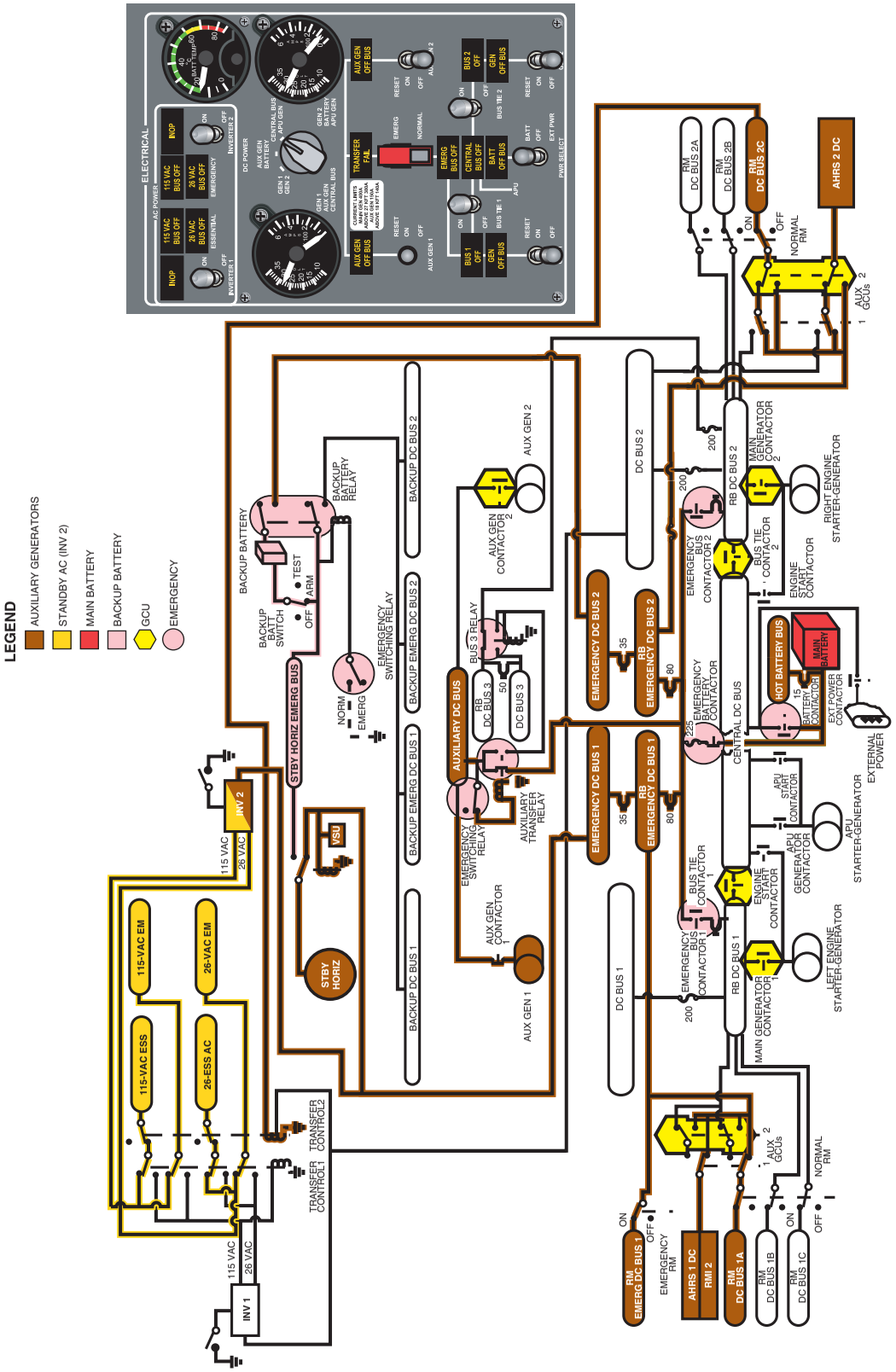


Figure 2-18. Emergency Mode—Both Main Generators and One Auxiliary Generator Failure

DISTRIBUTION

The 115/26-VAC inverter output is distributed to four buses: two 26-VAC (essential and emergency) and two 115-VAC (essential and emergency) buses.

In the event of inverter No. 1 failure, the loss of 115-VAC output causes the transfer control relay to automatically switch the four AC buses to inverter No. 2. (A failure of only the 26-VAC output of inverter No. 1 does not lead to automatic system transfer; thus, it is necessary to manually deenergize inverter No.1. This condition is indicated by the illumination of both essential and emergency 26 VAC BUS OFF lights).

CONTROL AND MONITORING

Each inverter is controlled by its respective switch on the electrical panel (Figure 2-12). An INOP annunciator light illuminates in the event of inverter failure or if the switch is turned off. Each AC bus has an annunciator light to indicate loss of power to that bus.

ELECTRICAL SYSTEM OPERATION

There are two modes of operation for the aircraft electrical system:

- Normal
- Emergency

The normal configuration for the aircraft electrical system is the normal mode.

Under normal operating conditions and if both main generators fail while operating in the normal mode, the system automatically switches, after a 2 second delay to the emergency mode.

The electrical system may be manually switched to the emergency mode by positioning the guarded electrical emergency switch to EMERG.

NORMAL MODE

Before Engine Start

Prior to engine start, the battery or external power supplies electrical power to all DC buses (except the auxiliary DC bus) and to both inverters (Figures 2-14 and 2-15). If the APU is available and provided the PWR SELECT switch is set to BATT, the APU generator will supply the same buses and charge the battery (Figure 2-16).

Takeoff and Normal Flight

When N_P reaches 70%, the auxiliary GCUs connect the auxiliary generators to the auxiliary DC bus.

Once the auxiliary DC bus is powered the bus 3 contactor shifts, switching relay box DC bus 3 and DC bus 3 from relay box DC bus 2 to the auxiliary DC bus.

The auxiliary GCUs also switch the primary navigation equipment (AHRS 1, AHRS 2, RMI 2, and radio master DC buses 1A and 2C) to the emergency DC buses.

The main generators continue to power the remaining buses and charge the battery.

Normal electrical system configuration (N_P > 70%), is shown in Figure 2-13.

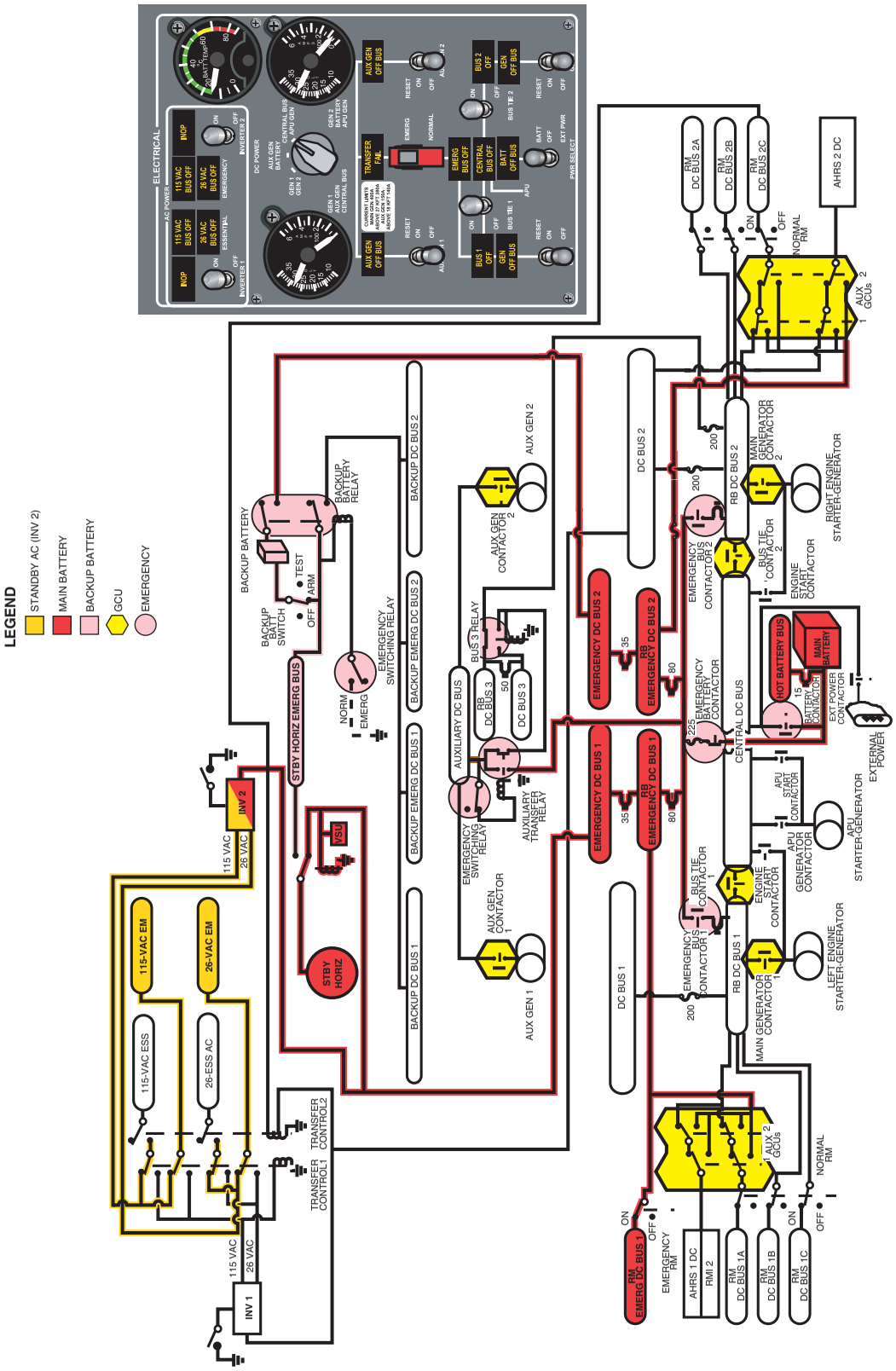


Figure 2-19. Emergency Mode—Total Generator Failure

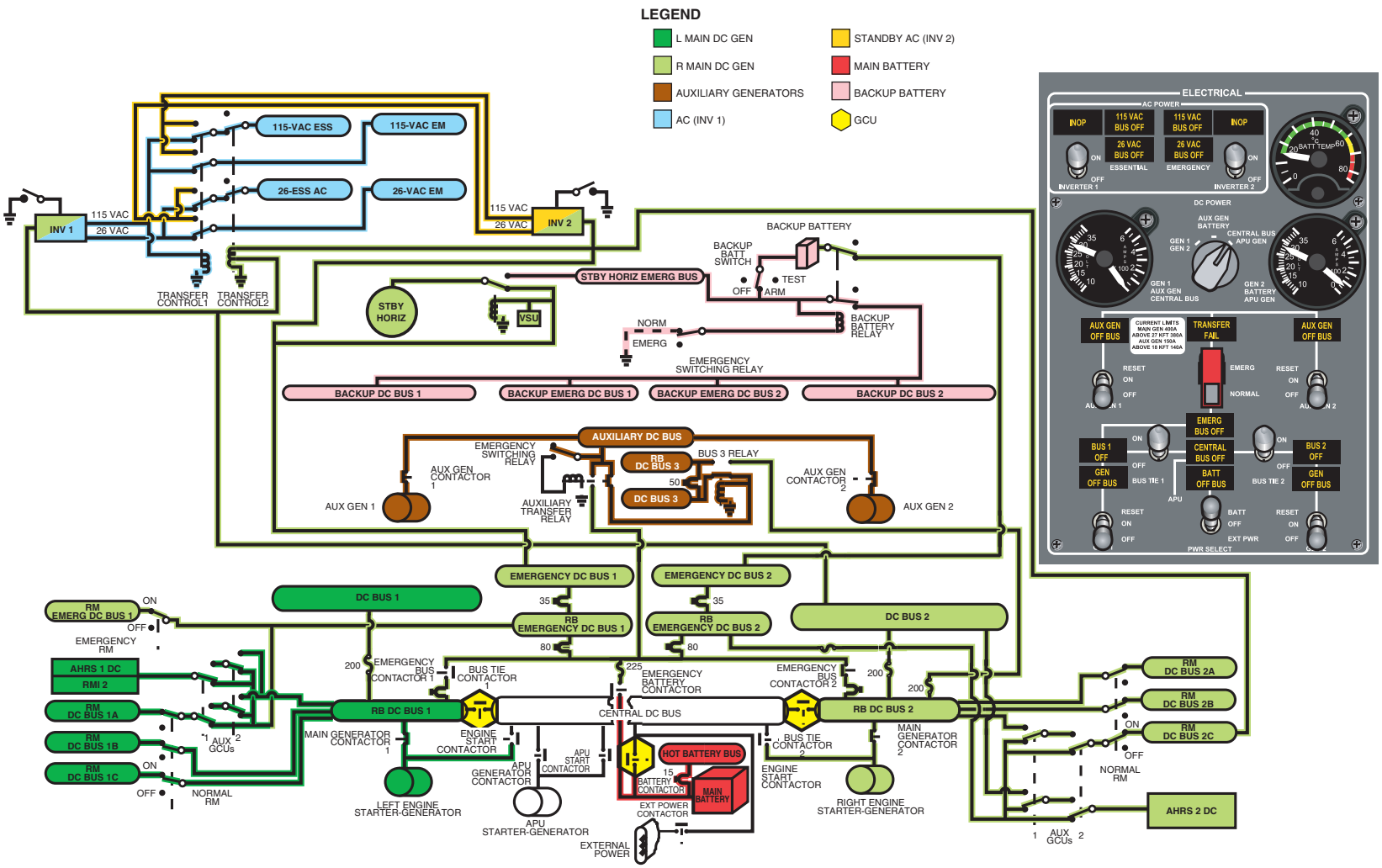


Figure 2-20. Overcurrent Case 1: Short Circuit, Central DC Bus

Single-Engine Flight

The electrical power distribution for single-engine flight (Figure 2-17) is the same as the normal flight configuration.

The DC load on the remaining main generator should be reduced to 400 amps if the APU generator is not available.

The remaining auxiliary generator continues to supply the loads of the auxiliary DC bus, relay box DC bus 3 and DC bus 3 with no restriction.

EMERGENCY MODE

The electrical system may be switched to the emergency mode of operation manually; or it may switch automatically with the loss of both main generators (after a 2-second delay).

For the switch to occur automatically, the following conditions must be met:

- At least one main generator switch ON
- PWR SELECT switch in BATT
- Both main generators loss – (Line contactors open)

NOTE

To prevent a reset to the normal mode after an automatic switch to the emergency mode, select EMERG on the electrical emergency switch prior to turning both main generators switches OFF.

In the emergency mode:

- The emergency bus contactor opens, disconnecting the emergency DC buses from relay box DC bus 2.
- The auxiliary transfer relay shifts, connecting the emergency DC buses to the auxiliary DC bus.
- The battery contactor opens, disconnecting the battery from the central DC bus.
- The emergency battery contactor closes, connecting the battery to the emergency DC buses.
- The backup battery relay is deenergized, disconnecting the backup buses from the backup battery.

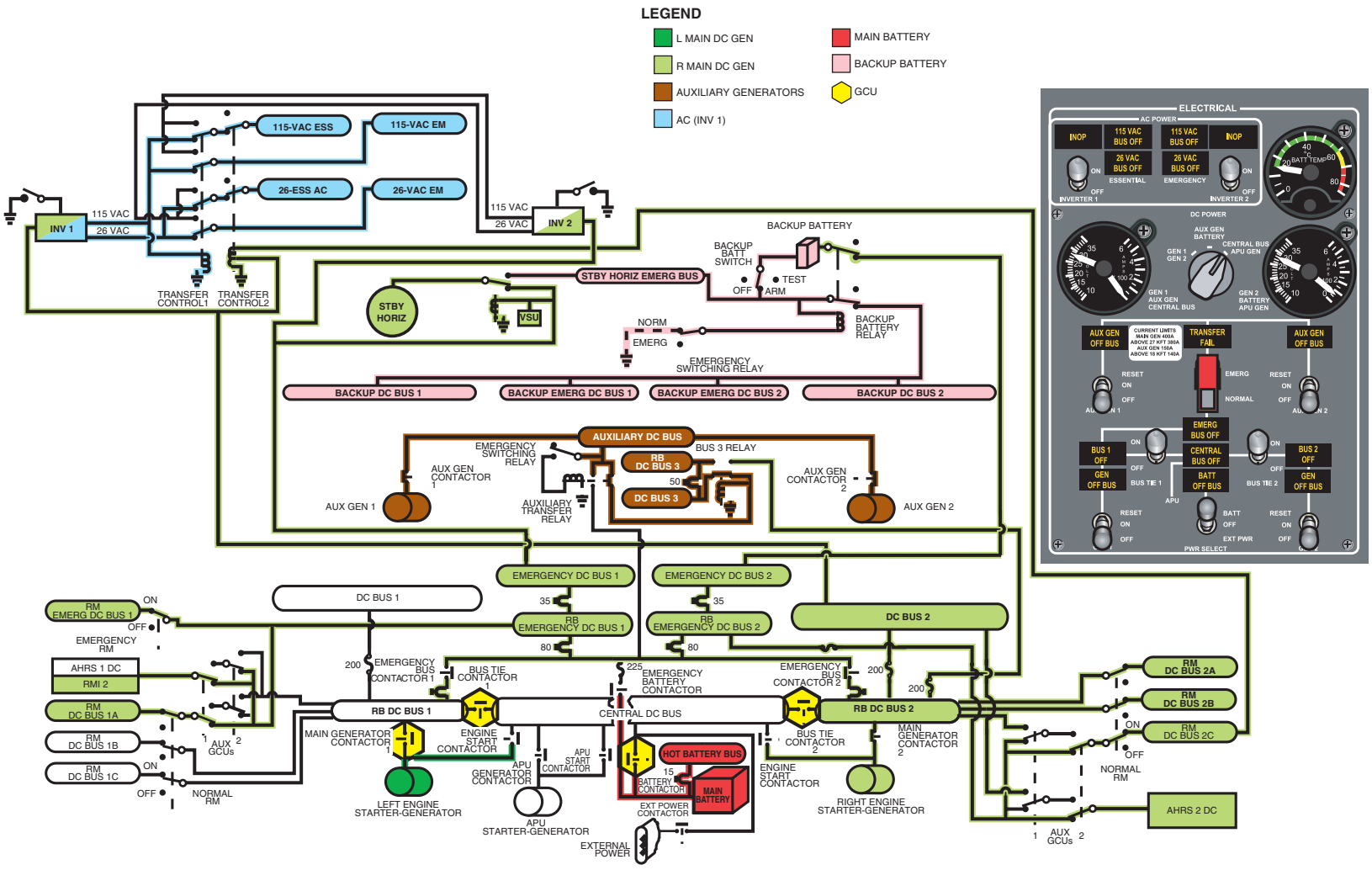


Figure 2-21. Overcurrent Case 2: Short Circuit, Relay Box DC Bus 1

Emergency Mode with APU/Auxiliary Generator Available

NOTE

If the APU is running, it is connected to the central DC bus to replace the main generators. The electrical system should be switched to the normal mode to reconnect the battery and emergency buses to the central DC bus.

If the APU is not running, the electrical system must be switched to the normal mode to energize the central DC bus for APU airstart.

To switch the electrical system to the normal mode:

- Both main generators switches—OFF
- Electrical emergency switch—NORMAL

Power Source—APU Generator Only

After connecting the APU to the central DC bus and switching the electrical system to the normal mode, the electrical power distribution is the same as normal APU only.

Power Source—Auxiliary Generator(s) Only

All emergency DC buses are powered, the main buses are isolated, and inverter No. 2 powers all AC buses. Distribution is shown in Figure 2-18.

Power Source—APU Generator and Auxiliary Generator(s)

All emergency buses are powered. The main buses feed from the APU.

Emergency Mode with Loss of All Generators

Power Source—Battery Only

The battery is automatically connected to the emergency DC buses and supplies power for approximately 30 minutes.

Only the emergency AC buses powered.

Distribution is shown in Figure 2-19.

NOTE

The loads connected to the emergency DC buses should be reduced at crew discretion.

OVERCURRENT PROTECTION

A short circuit in the DC buses could result in an overcurrent condition in the electrical system. The aircraft is protected from this condition by:

- Automatic intervention of the generator control units (APU, auxiliary, or main GCUs).
- Circuit breakers
- Fuses

The buses protected by GCUs are those that have a direct connection to a generator (i.e., the central DC bus, relay box DC buses 1 and 2, and the auxiliary DC bus).

Except for the auxiliary DC bus, a short circuit in any of the other buses protected by GCUs will result in operational limitations to the aircraft, especially in flight.

The GCUs first priority is to disconnect the battery.

CAUTION

It is strongly recommended not to try an electrical system reset following a short circuit in the central DC bus or

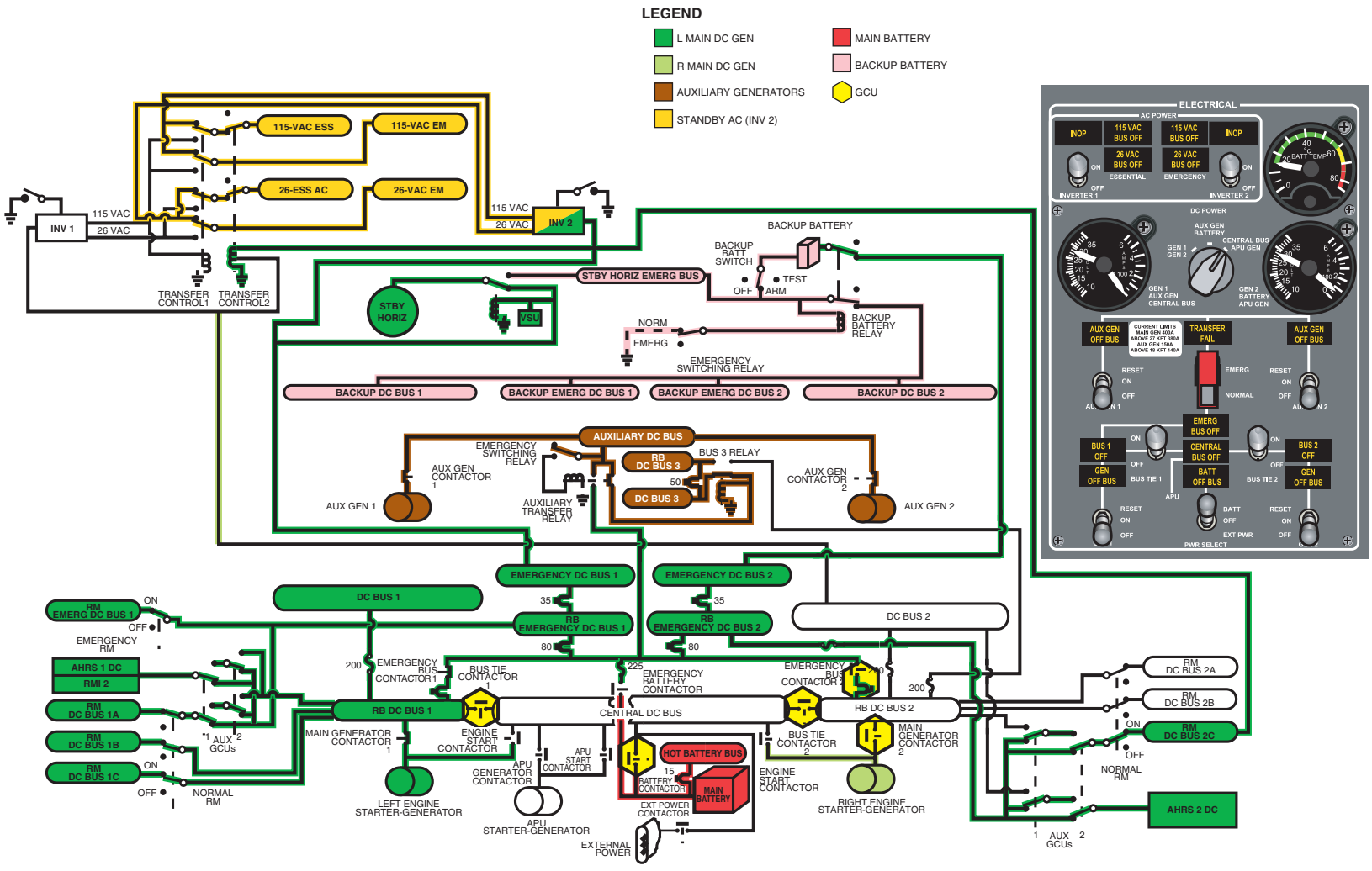


Figure 2-22. Overcurrent Case 3: Short Circuit, Relay Box DC Bus 2

relay box DC buses 1 or 2.

NOTE

In the following cases, it is assumed that the aircraft is in flight and the electrical system was in normal mode prior to the short circuit.

Case 1—Short Circuit in the Central DC Bus

In the event of a short circuit in the central DC bus (Figure 2-20), both main GCUs isolate the bus from the electrical system by opening the battery contactor and bus tie contactors 1 and 2.

If the APU is connected to the central DC bus at the time the short circuit occurs, it will be disconnected by its GCU.

The following lights illuminate on the electrical panel:

- CENTRAL BUS OFF—Central DC bus is not powered
- BATT OFF BUS—Battery disconnected from the central DC bus

On APU panel (if connected):

- GEN OFF BUS—APU is not powering the central DC bus.

Loss of the central DC bus results in loss of the following:

- Engine/APU airstart capability
- Electrical crossfeed (operative main generator supplying the faulty generator buses)
- Battery charging

To regain battery charging, the electrical system must be switched to the emergency mode by positioning the electrical emergency switch to EMERG.

In the emergency mode, the auxiliary DC bus switches from relay box DC bus 3 and DC bus 3 to the emergency DC buses and battery.

Table 2-1. ELECTRICAL BUS EQUIPMENT DISTRIBUTION

Central DC Bus

Hot Battery Bus (sec)
Relay Box DC Bus 1
Relay Box DC Bus 2
APU start
No. 1 Engine start
No. 2 Engine start

Hot Battery Bus

Autotransfer
Clock/chronograph
Courtesy light
DC indication 1
DC indication 2
Fire extinguishing, left
Fire extinguishing, right
Forward entry door
Fuel shutoff valve, left
Fuel shutoff valve, right
Hydraulic shutoff valve, blue
Hydraulic shutoff valve, green

Relay Box DC Bus 1

Central DC bus
DC Bus 1
Radio Master DC Bus 1A (sec)
Radio Master DC Bus 1B
Radio Master DC Bus 1C
Relay Box Emerg DC Bus 1 (pri)
Relay Box Emerg DC Bus 2 (pri)
AHRS 1 DC (sec)
Ground cooling fan, left
Propeller heater, left
Recirculation fan 1
RMI 2 (sec)
Storm light
Electric hydraulic pump, blue
Taxi lights

DC Bus 1

ADS 1
Air cond indicating lights, left
Air cond pack shutoff valve, left
Altitude alerter
Antiskid, outboard
AOA sensor heater, left
AOA sensor indicator, left
AP transfer
Attendant handset/observer interphone
Aural warning
Auxiliary generator GCU 1
Aux pitot/static heater (Note 4)
Aux pitot/static indicator (Note 4)
Battery safety, left
Bus off relay DC 1
Cabin air temperature indicator
Deicing system monitor 1
Door warning
Electric bay cooling, right
Electric fuel pump control, left rear
Electric fuel pump indicator, left rear
Engine bleed-air shutoff valve, left
FGC 1 servos
Flap DC bus 1
Fuel collector tank solenoid 1
Fuel flow indicator, left
Fuel quantity indicator, left
Fuel shutoff valve indicator
Hydraulic pressure indicator, blue
Hydraulic quantity indicator, blue
Hydraulic shutoff valve indicator
Hydraulic system control, blue
N_H indicator 1
N_L indicator 1
N_P indicator 1
N_P overspeed 1
OAT
Oil temp/press indicator 1
Panel lights, pilot
Pax cabin light
Propeller synchro
Propeller timer 1
Radio master 1
Reading lights, left
Reading lights, right
Stick pusher 1
Starter-generator GCU 1
Strobe light
Turn and bank indicator 1
Turn and bank indicator 2 (optional)
Windshield wiper, left

Table 2-1. ELECTRICAL BUS EQUIPMENT DISTRIBUTION (Cont)

Relay Box DC Bus 2

Central DC Bus
DC Bus 2
Radio Master DC Bus 2A
Radio Master DC Bus 2B
Radio Master DC Bus 2C (sec)
Relay Box DC Bus 3 (sec)
Clock/chronograph
Electric hydraulic pump, green system
Galley
Ground cooling fan, right
Landing light, right
Propeller heater, right
Recirculation fan 2
Windshield heater, right

DC Bus 2

ADS 2
Air cond indicating lights, right
Air cond pack shutoff valve, right
Altimeter 2
Antiskid, inboard
AOA sensor heater, right
AOA sensor indicator, right
APU control unit
APU duct leakage
APU EGT/RPM indicator
APU GCU
Auxiliary generator GCU 2
Battery safety, right
Bus off relay DC 2
Cockpit light
Deicing system monitor 2
Electric fuel pump control, right rear
Electric fuel pump, right rear
Engine bleed shutoff valve, right
FGS 2 servos
Flap DC bus 2
Fuel collector tank solenoid 2
Fuel flow indicator, right
Fuel quantity indicator, right
Fuel totalizer
Hydraulic fluid quantity indicator, green
Hydraulic system control, green
Hydraulic system pressure indicator, green
Inspection lights
Inverter 1
Navigation lights
NH indicator 2
NL indicator 2
NP indicator 2
NP overspeed 2
Oil temp/press indicator 2
Panel lights, copilot
Pitot static 2 heater
Pitot static 2 indicator
Pressurization control
Propeller timer 2
Radio master 2
Rotating beacon
Side slip heater
Side slip indicator
Smoke detector
Starter-generator GCU 2
Stick pusher computer 2
Stick pusher 2
Stick shaker 2
Transfer control relay 2 (pri)
Toilet service
Windshield control, right
Windshield wiper, right

Relay Box Emergency DC Bus 1

Emergency DC Bus 1
Radio Master DC Bus 1A (pri)
Radio Master Emerg DC Bus 1
AHRS 1 DC (pri)
Propeller aux feather pump 1
RMI 2 (pri)

Table 2-1. ELECTRICAL BUS EQUIPMENT DISTRIBUTION (Cont)

Emergency DC Bus 1

AC bus transfer indicator
AC bus transfer 2
Air inlet deicing, left engine
Air/ground position, left
Airspeed indicator 1
Alarm lights control 1
Altimeter 1
Battery temp monitor
Beta 1
EEC 1
EEC 1 indicator
Electric feather 1
Electric fuel pump control, left front
Electric fuel pump, left front
Emergency DC bus 1 off relay
Emergency lights
Fire detection, nacelle 1
Fire detection inop indicator, nacelle 1
Fire extinguisher bottle A inop indicator
Flap emergency bus
Fuel crossfeed indicator
Fuel crossfeed valve
Ignition 1
Interphone 1
Inverter 2
Landing gear indicator B
Landing gear down control override
Leading edges timer 1
Oxygen system
Panel alarm lights 1
Pressurization alarm
Radio master emergency
RMI 1DC
Rudder green system control
Rudder green system indicator
SCU 1
Stby horizon (pri)
Steering
Stick pusher computer 1
Stick shaker 1
Torque indicator 1
T_6 indicator, left engine
VHF 1
Voice recorder

Relay Box Emergency DC Bus 2

Emergency DC Bus 2
Radio Master DC Bus 2C (pri)
AHRS 2 DC (pri)
Floodlight
Landing light, left
Propeller aux feather pump 2

Emergency DC Bus 2

Air conditioning vent valve
Air/ground position, nose
Air/ground position, right
Airspeed indicator 2
Air inlet deicing, right engine
Alarm lights control 2
APU fire detection
APU fire extinguishing
APU fire inop indication
APU fuel shutoff valve
APU fuel shutoff indication
Backup battery
Beta 2
Brake lights
EEC 2
EEC indicator 2
Electrical feather 2
Electric fuel pump control, right front
Electric fuel pump, right front
Emergency DC bus 2 off relay
Fire detection, nacelle 2
Fire detection inop indicator, nacelle 2
Fire extinguisher bottle B inop indicator
Ignition 2
Interphone 2
Landing gear control
Landing gear indicator A
Leading edges timer 2
Panel alarm lights 2
Pax signs
Pitot static 1 heater
Pitot static 1 indicator
Rudder blue system control
Rudder blue system indicator
SCU 2
Torque indicator 2
T_6 indicator, right engine

Table 2-1. ELECTRICAL BUS EQUIPMENT DISTRIBUTION (Cont)

Radio Master Emerg DC Bus 1

ADF 1
TDR 1

Radio Master DC Bus 1A

DCP/CHP 1
DPU 1
EADI 1
EHSI 1
VOR/ILS 1DC

Radio Master DC Bus 1B

DME indicator
DME
FGS 1 DC
Passenger address
Radio altimeter 1

Radio Master DC Bus 1C

TDR 2
VHF 2 (Note 1)
MFD
SELCAL

Radio Master DC Bus 2A

MPU 1
Radar DC
DME 2
FGS 2DC
MPU 2
VHF 2 (Note 2)

Radio Master DC Bus 2B

Radio Master DC Bus 2C

AC bus transfer 1
AC transfer control relay 2 (sec)
DCP/CHP 2
DPU 2
EADI 2
EHSI 2
VOR/ILS 2DC

Auxiliary DC Bus

Hot Battery Bus (ter)
Relay Box DC Bus 1 (sec)
Relay Box DC Bus 2 (sec)
Relay Box EC Bus 3 (pri)
DC Bus 3

Relay Box DC Bus 3

Gasper fan
Windshield heater left
Air cond cross bleed

DC Bus 3

Air cond distribution valves
Aux pitot/static heater (Note 3)
Aux pitot/static indicator (Note 3)
Compartment lights
Electric bay cooling, left
Logo-type lights
Overhead panel light
Windshield control, left
Entertainment system

115-VAC Essential Bus

Entertainment system
MK-II
GPWS
(Omega)
Radar AC

26-VAC Essential Bus

AHRS 2 AC
FGS 2 AC
VOR/ILS 1AC
VOR/ILS 2 AC

Table 2-1. ELECTRICAL BUS EQUIPMENT DISTRIBUTION (Cont)

115-VAC Emergency Bus

FDR
TAS—for barber pole
Voice Recorder

115-VAC Emergency Bus

AHRS 1 AC
FGS 1 AC

Backup DC Bus 1

Aural warning
AHRS 1
N_H and T_6 indicators, eng 1
RCCBs for
- Left hydraulic aux pump
- Left propeller heater
- Recirculation fan 1
- Taxi light
- Left ground cooling fan

Backup Emergency DC Bus 1

EEC 1
Ignition light 1
RCCB for:
- Left aux feather pump

Backup DC Bus 2

AHRS 2
N_H and T_6 indicators, eng 2
RCCBs for
- Right hydraulic aux pump
- Right propeller heater
- Recirculation fan 2
- Right landing light
- Right windshield heater
- Right ground cooling fan

Backup Emergency DC Bus 2

Clock/chronograph
EEC 2
RCCB for:
- Right aux feather pump
- Left landing light

Standby Horizon Emergency Bus

Standby horizon (sec)
GEN 1 GCU
GEN 2 GCU
APU GCU

NOTES:
1. For aircraft Pre Mod SB 120-023-0014
2. For aircraft Mod SB 120-023-0014
3. For aircraft Pre Mod SB 120-023-0005
4. For aircraft Mod SB 120-023-005 or SN 120.029 and subsequent

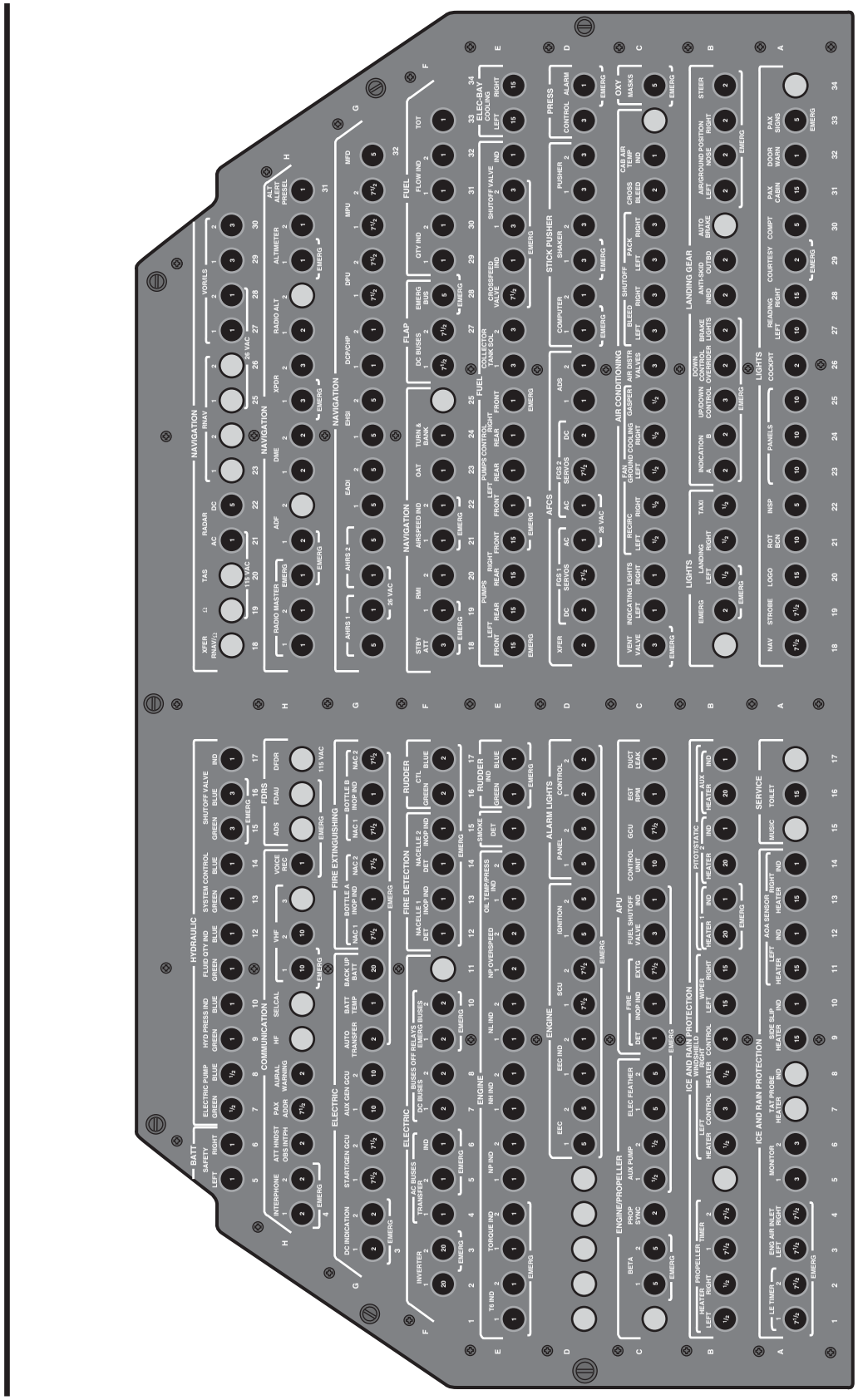


Figure 2-23. Circuit-Breaker Panel

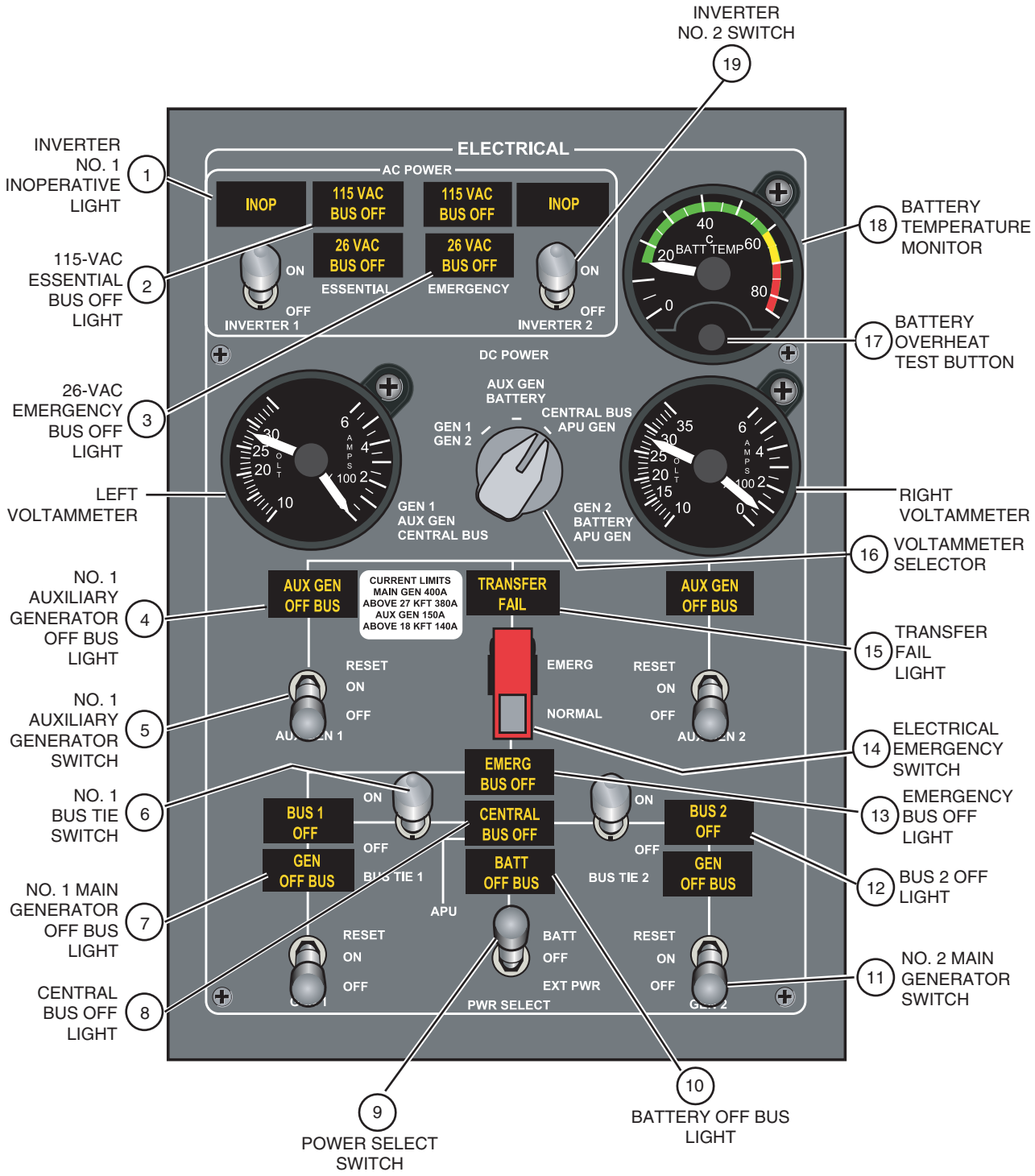


Figure 2-24. Electrical Control Panel

The electrical distribution is as follows:

- Generator 1 continues to power relay box DC bus 1 and DC bus 1.
- Generator 2 continues to power relay box DC bus 2 and DC bus 2, and picks up relay box DC bus 3 and DC bus 3 via relay box DC bus 2.
- Through the auxiliary DC bus, the auxiliary generators power all emergency DC buses and charge the battery.
- The central DC bus remains isolated from the electrical system.

The flight may be continued at the pilot's discretion; however, engine/APU airstart and electrical crossfeed are not available.

NOTE

The CENTRAL BUS OFF and the BATTERY OFF BUS lights remain on.

Case 2—Short Circuit in Relay Box DC Bus 1

In the event of a short circuit in relay box DC bus 1 (Figure 2-21), the main GCUs isolate both relay box DC bus 1 and the central DC bus from the electrical system by opening the main generator contactor 1, the battery contactor, and bus tie contactors 1 and 2.

The following lights illuminate on the electrical panel:

- GEN 1 OFF BUS—Main generator 1 is isolated from the system.
- BUS 1 OFF—DC bus 1 is not powered.
- CENTRAL BUS OFF—Central DC bus is not powered.
- BATTERY OFF BUS—Battery is disconnected from the central DC bus.

NOTE

In some cases, the CENTRAL BUS OFF light will not illuminate.

Loss of the central DC bus results in loss of the following:

- Engine/APU airstart capability
- Electrical crossfeed
- Battery charging

Loss of relay box DC bus 1 results in loss of the following:

- All equipment connected to:
- DC Bus 1
- Radio master DC buses 1B and 1C.

The aircraft is limited to 25,000 ft, since the left engine bleed (DC bus 1) is closed.

To regain the emergency buses and battery charging, the electrical system must be switched to emergency mode by positioning the electrical emergency switch to EMERG. In the emergency mode, the auxiliary DC bus switches from relay box DC bus 3 and DC bus 3 to the emergency DC buses and battery.

The electrical distribution is as follows:

- The main generator 2 continues to supply relay box DC bus 2 and DC bus 2, and picks up relay box DC bus 3 and DC bus 3.
- The auxiliary generators, through the auxiliary DC bus, power all emergency DC buses and charge the battery.
- The following buses remain deenergized:
  - Central DC bus
  - Relay box DC bus 1
  - DC bus 1
  - Radio master DC bus 1B and 1C

The left engine bleed is still closed (DC bus 1) limiting the aircraft to 25,000 ft.

The flight may be continued at the pilot's discretion; however, engine/APU airstart and electrical crossfeed are not available, and equipment connected to the lost buses is also out. Therefore, it is recommended to land as soon as practical.

NOTE

The GEN 1 GEN OFF BUS, BUS 1 OFF, CENTRAL BUS OFF, and BATTERY OFF BUS lights remain on.

Case 3—Short Circuit in Relay Box DC Bus 2

In the event of a short circuit in relay box DC bus 2 (Figure 2-22), the main GCUs isolate both relay box DC bus 2 and the central DC bus from the electrical system by opening the main generator contactor 2, the battery contactor, and bus tie contactors 1 and 2, and emergency bus contactor 2.

The following lights illuminate on the electrical panel:

- (GEN 2) GEN OFF BUS—Main generator 2 is isolated from the system.
- BUS 2 OFF—DC bus 2 is not powered.
- CENTRAL BUS OFF—Central DC bus is not powered.
- BATTERY OFF BUS—Battery is disconnected from the central DC bus.
- (INVERTER 1) INOP—Inverter 1 is not powered.

NOTE

In some cases, the CENTRAL BUS OFF light will not illuminate.

Loss of the central DC bus results in loss of the following:

- Engine/APU airstart capability
- Electrical crossfeed
- Battery charging

Loss of relay box DC bus 2 results in loss of all equipment connected to:

- DC bus 2
- Radio master DC bus 2A and 2B

The right engine bleed (DC Bus 2) is closed, limiting airplane to 25,000 ft.

Emergency bus contactor 2 opens and emergency bus contactor 1 closes precluding the loss of emergency busses.

To regain battery charging, the electrical system must be switched to the emergency mode by positioning the electrical emergency switch to EMERG. In the emergency mode, the auxiliary DC bus switches from relay box DC bus 3 and DC bus 3 to the emergency buses and battery.

NOTE

Due to the loss of DC bus 2, this procedure results in the loss of relay box DC bus 3 and DC bus 3.

QUESTIONS

The electrical distribution is as follows:

- The main generator 1 continues to power relay box DC bus 1 and DC bus 1.
- Through the auxiliary DC bus, the auxiliary generators power all emergency DC buses and charge the battery.
- The following buses remain deenergized:
  - Central DC bus
  - Relay box DC bus 2
  - DC bus 2
  - Relay box DC bus 3
  - DC bus 3
  - Radio master DC bus 2A and 2B

The flight may be continued at the pilot's discretion; however, engine/APU airstart and electrical crossfeed are not available, and equipment connected to the lost buses is also out. Therefore, it is recommended to land as soon as practical.

NOTE

The GEN 2 GEN OFF BUS, BUS 2 OFF, CENTRAL BUS OFF, BATTERY OFF BUS, and inverter 1 INOP lights remain on.

Case 4—Short Circuit in the Central DC Bus, Relay Box DC Bus 1, or Relay Box DC Bus 2 on the Ground

A short circuit with the airplane on the ground results in the same GCU logic of operation as in flight. On the ground, however, the auxiliary DC bus is normally deenergized and relay box DC bus 3 and DC bus 3 are normally powered by relay box DC bus 2.

PRE MOD SB 120-024-0051 ELECTRICAL SYSTEM DIFFERENCES

This manual does not cover Pre Mod electrical differences. Refer to appropriate Embraer *AFM* and *OPS* manuals.

ELECTRICAL CONTROL PANEL SUMMARY

1. Inverter 1 INOP Light—Illuminates when Inverter No. 1 is not operating due to inverter failure, loss of power, or switch in the OFF position.

2. Essential 115 VAC BUS OFF Light—Illuminates when the 115-VAC essential bus is deenergized.

3. EMERGENCY 26 VAC BUS OFF Light—Illuminates when the 26-VAC emergency Bus is deenergized.

4. No. 1 AUX GEN OFF BUS Light—Illuminates when the auxiliary generator contactor No. 1 is open, isolating the generator from the electrical system.

5. AUX GEN 1 Switch—

   - OFF: Disconnects auxiliary generator No. 1 from the auxiliary DC bus.

   - ON: Connects the No. 1 auxiliary generator to auxiliary DC bus.

   - RESET: This momentary position reconnects the No. 1 auxiliary generator.

6. BUS TIE 1 Switch—

   - Off: Disconnects relay box DC bus 1 from the central DC bus.

   - ON: Connects relay box DC bus 1 to the central DC bus.

7. No. 1 GEN OFF BUS Light—Illuminates when the main generator contactor No. 1 is open, isolating the generator from the electrical system.

8. CENTRAL BUS OFF Light—Illuminates when the central DC bus is not powered.

CHAPTER 3
LIGHTING

CONTENTS

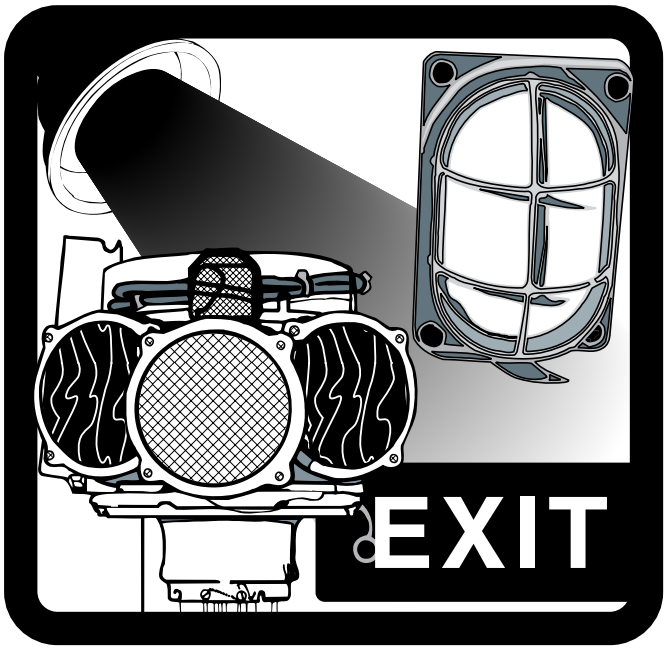
Page

INTRODUCTION 3-1
GENERAL 3-1
INTERNAL LIGHTING 3-2
Cockpit Lighting 3-2
Passenger Cabin Lighting 3-5
Compartment Lighting 3-7
EXTERNAL LIGHTING 3-8
Landing Lights 3-8
Taxi Lights 3-9
Navigation Lights 3-9
Rotating Beacons 3-9
Strobe Lights 3-9
Wing Inspection Lights 3-9
Logo Lights 3-9
EMERGENCY LIGHTING 3-10
Internal Emergency Lights 3-10
External Emergency Lights 3-10
Operation 3-11
QUESTIONS 3-12

ILLUSTRATIONS

Figure	Title	Page
3-1	Cockpit Lighting Controls	3-2
3-2	Fluorescent, Utility, and Dome Lighting	3-3
3-3	Observer Lights	3-4
3-4	Passenger Cabin Lighting	3-5
3-5	Passenger Advisory Lights	3-6
3-6	Flight Attendant Call Lights	3-6
3-7	Sterile Light	3-7
3-8	External Lights	3-8
3-9	External Lights Control Panel	3-8
3-10	Emergency Illuminated Areas	3-10
3-11	Cockpit Emergency Lighting Switch	3-11
3-12	Flight Attendant Emergency Lighting Switch	3-11

CHAPTER 3
LIGHTING



INTRODUCTION

Lighting on the EMB-120 includes lighting for both the interior and exterior of the aircraft. Internal lighting consists of cockpit area and instrument lights, cabin area lights, and the emergency lighting system. External lighting consists of landing, taxi, navigation, strobe, rotating beacon, wing inspection, and logo lights. The baggage compartment, the toilet, nose and tail cone compartments, the entry and cargo door are also provided with individual lights.

GENERAL

Aircraft lighting is divided into internal and external lighting. Internal lighting is further divided into cockpit, cabin, and emergency lighting. Cockpit lighting consists of instrument panel lights, floodlights, and map lights. Cabin lighting consists of fluorescent lights along the ceiling, passenger reading lights, and lighted signs. The emergency lighting system illuminates the cabin area, the emergency exits, and the emergency evacuation routes.

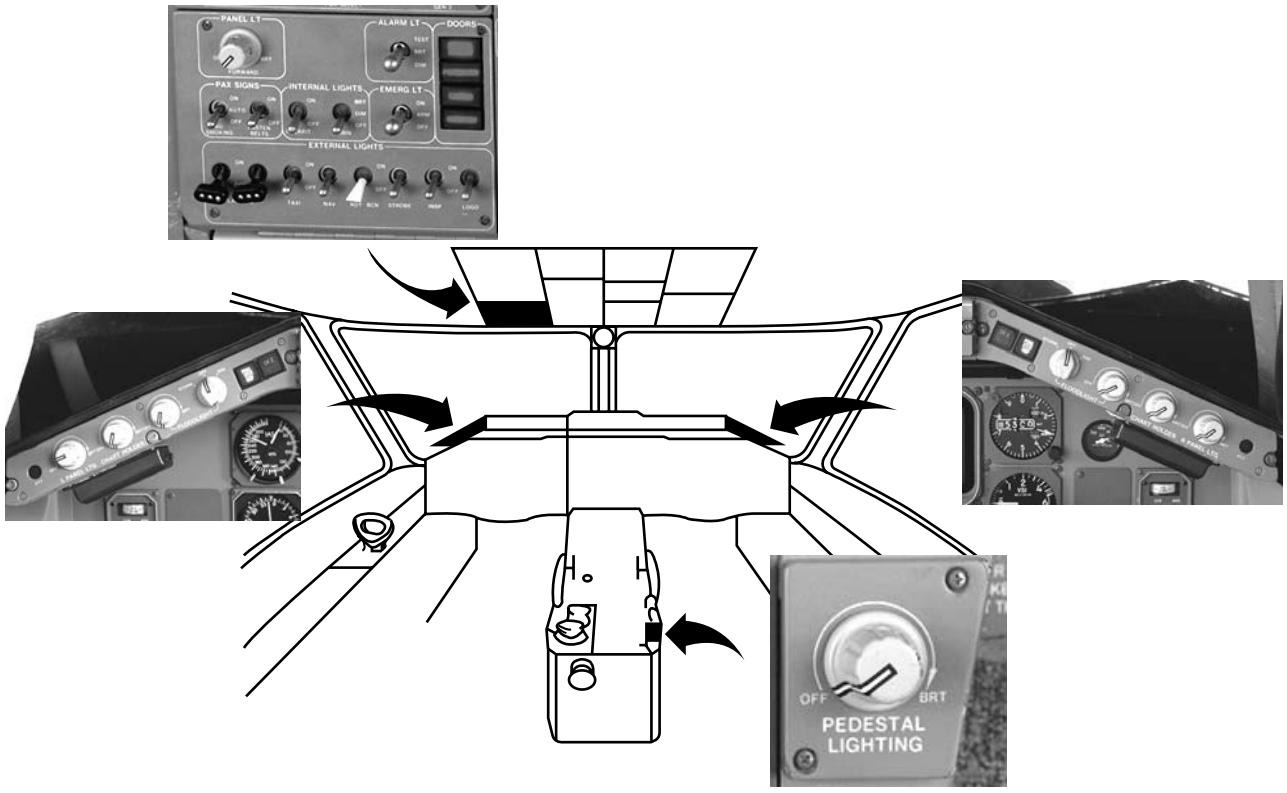


Figure 3-1. Cockpit Lighting Controls

INTERNAL LIGHTING

Internal lighting is provided for the cockpit, cabin, nose and tail cone compartments, and the baggage compartment. Lighting controls are located in the cockpit, on the attendant's panel just inside the forward entry door, and in each compartment.

COCKPIT LIGHTING

The cockpit lighting system consists of instrument and panel lighting, specific area lighting, and general area lighting. All instrument and panel lighting are white and the brightness is adjustable.

The brightness control of all cockpit lighting is adjusted through rheostat control by the applicable control knobs. The counterclockwise position is the OFF position. Turning the knobs clockwise increases the brightness of the related lights.

The cockpit lighting consists of the following lights:

- Integral instrument lighting/ instrument panel and pedestal panel lighting
- Chart holder lights
- Panel fluorescent floodlighting
- Utility (map) lights
- Dome lights
- Observer lights

Cockpit lighting controls are located in the following areas (Figure 3-1):

- Overhead panel, to control overhead panel lights and dome lights
- Left and right side of the glareshield, to control the pilot's and copilot's panel/instrument lights, chart lights, and the fluorescent lights

- Aft end of the center pedestal, for control of the center pedestal integral lighting.

Other cockpit light controls are located:

- Adjacent to each utility (map) light on the ceiling, for control of the respective left and right light
- On the aft bulkhead, for observer control of the aft left and right floodlights (when the respective pilot/copilot's floodlight control is OFF).

Panel Lights

The instrument panel acrylic masks on the forward instrument panels, overhead panel, and center pedestal are all back-lighted by miniature incandescent lamps internally installed on printed circuit boards. Each panel is controlled by its own rheostat (Figure 3-1):

- The forward instrument panel controls, labeled L PANEL LTG and R PANEL LTG, are located with the other lighting controls on the left and right side of the glareshield.
- The overhead panel control, labeled OVERHEAD, is in its own PANEL LT section on the overhead lighting control panel.
- The pedestal panel control, labeled PEDESTAL LIGHTING, is located on the right aft portion of the center pedestal.

Instrument Lights

All instruments are integrally white-lighted. The instrument lights are controlled by the same rheostats used for the panel lights.

Chart Holder Lights

The chart holders, mounted on each control yoke, are provided with reading lights. The light sets are made up of seven miniature incandescent lamps powered by DC bus 1.

The control for each chart holder light (and yoke mounted clock), is labeled CHART HOLDER and located on the left and right glareshield next to the instrument panel lighting control (Figure 3-1).

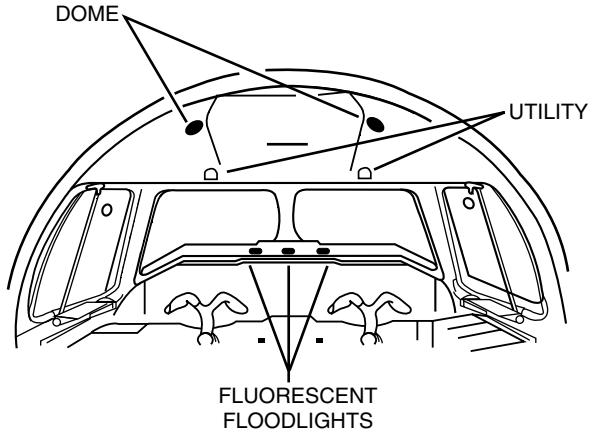


Figure 3-2. Fluorescent, Utility, and Dome Lighting

Fluorescent Floodlights

The main instrument panel is illuminated by three fluorescent light assemblies, each containing two fluorescent tubes mounted under the center portion of the glareshield (Figure 3-2).

These lights are used for general instrument panel floodlighting. They also operate automatically during electrical emergencies in the absence of the normal instrument lights. A STORM position is used to provide maximum panel floodlighting as a protection against crew temporary flash blindness due to lightning. The floodlights are normally powered by DC bus 1, but switch to emergency DC bus 2 during electrical emergency conditions.

The floodlights are operated by a set of two FLOODLIGHT controls located on the left and right glareshield. One is a selector knob labeled STORM/OFF/ DIM, the other is a rheostat labeled OFF/BRT. These controls may be operated together or independently depending on the lighting desired.

The controls on the left glareshield panel operate the left and center light assemblies; the controls on the right panel operate the center and right assemblies.

Floodlight Partial Lighting

To use floodlight partial lighting, with the STORM/OFF/DIM knob in the OFF position, turn the rheostat knob towards the BRT position. This will cause two fluorescent lamps (one per assembly being operated) to illuminate.

Floodlight Full Lighting

To use floodlight full lighting, place the STORM/OFF/DIM knob to DIM. This causes the remaining lamps to illuminate with their intensity controlled by the respective rheostat.

Placing either STORM/OFF/DIM knob to STORM causes all six fluorescent lamps to illuminate at full brightness.

Floodlight Automatic Lighting

Whenever an electrical emergency condition occurs, one fluorescent tube in each of the three light assemblies (and the two incandescent pedestal/observer lights on the cockpit bulkhead) automatically illuminate to provide panel/pedestal lighting.

Utility (Map) Lights

Two incandescent utility lights, powered by DC bus 1, are mounted in the ceiling on either side of the overhead panel (Figure 3-2). Each is controlled by an adjacent rocker switch labeled OFF/DIM/BRT. The light's beam may be oriented by the crewmembers.

Dome Lights

The cockpit general illumination is provided by two incandescent dome lights mounted in the ceiling on either side of the overhead panel. They are controlled by a two-position (ON/OFF) COCKPIT switch located in the INTERNAL LIGHTS section of the overhead lighting control panel. The dome lights are powered by DC bus 1.

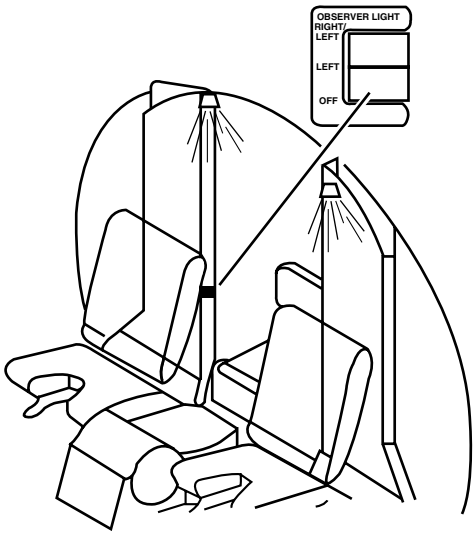


Figure 3-3. Observer Lights

Observer Lights

Two incandescent lights mounted on the cockpit bulkhead, normally used by the emergency lighting system, may be used as reading lights for the observer (Figure 3-3). The left or right observer lights are turned on when the respective FLOODLIGHT switch on the glareshield panels are moved out of the OFF position. With the FLOODLIGHT switch in the OFF position, the observer spotlights are controlled by a rocker switch, labeled OFF, LEFT/RIGHT, LEFT, located to the right of the observers seat.

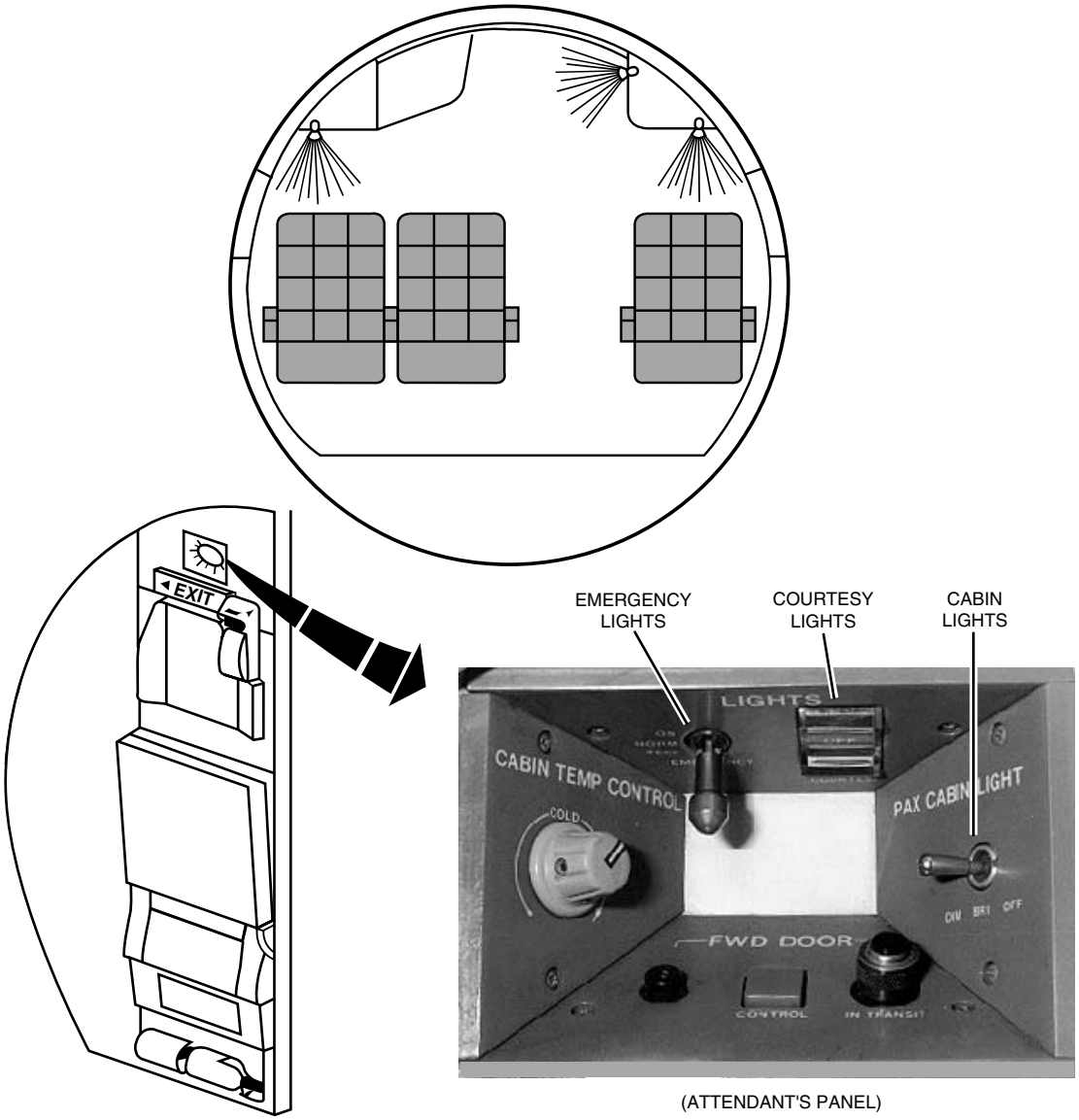


Figure 3-4. Passenger Cabin Lighting

PASSENGER CABIN LIGHTING

The passenger cabin lights illuminate the cabin, toilet, and galley areas. In addition, they provide illumination of passenger advisory signs, reading lights, and attendant call lights.

Cabin lighting is divided into the following groups:

- General cabin lighting
- Advisory sign lights
- Passenger reading lights
- Discrete call lights
- Toilet lights

General Cabin Lighting

Passenger cabin lighting is provided by three rows of fluorescent lights (Figure 3-4). The two side rows, installed just above the sidewall lining panels, provide direct lighting. One row installed in the ceiling over the aisle provides subdued illumination.

The cabin lights use 190 VAC from three inverters powered by DC bus 1. The lights are controlled by three-position (DIM/BRT/OFF), PAX CABIN LIGHT switch located on the flight attendant's panel (Figure 3-4).

Passenger Advisory Lights

No smoking and fasten seat belt signs (Figure 3-5) are installed in the passenger service units (PSUs) located over each row of seats and in the galley area. A return to your seat sign installed in the toilet illuminates with the fasten seat belt signs.

The passenger advisory signs, powered by emergency DC bus 2, are controlled by two switches in the cockpit. The switches are located in the PAX SIGNS section of the overhead lighting panel. The NO SMOKING switch is labeled ON/AUTO/OFF, and the FASTEN BELTS switch is labeled ON/OFF.

With the NO SMOKING switch in the AUTO position, the signs will illuminate when the landing gear is selected down or the oxygen masks are deployed. Some airplanes do not have an AUTO position.

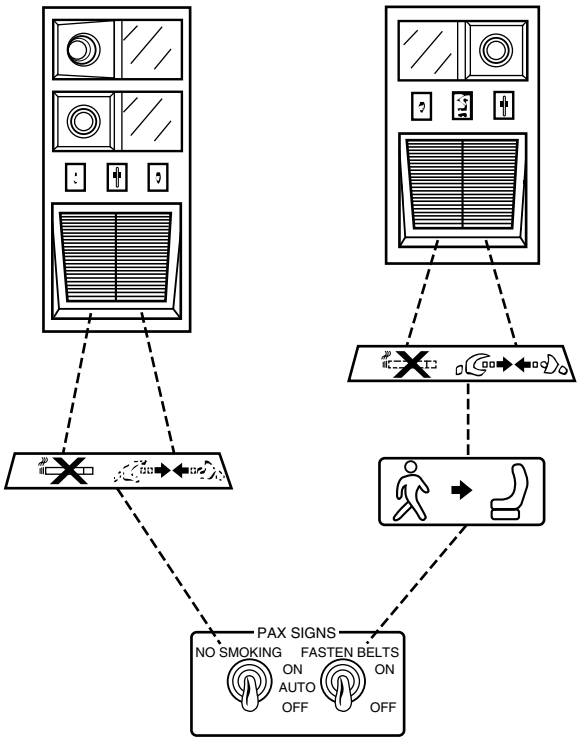


Figure 3-5. Passenger Advisory Lights

Activation of either switch sounds a single chime over the passenger address (PA) system which is powered by radio master DC bus 1A.

Passenger Reading Lights

Passenger reading lights are provided for each passenger seat. These lights, powered by DC bus 3, operate independently from the general cabin lighting. The lights are controlled by reading light buttons located on the PSUs.

Attendant Call

An attendant call button, also located on the PSU, is provided for each passenger seat. When the call button is pressed, it illuminates the button itself, the blue light on the flight attendant's discrete call light panel, and generates a single hi-low chime on the PA system The lights are powered by DC bus 3.

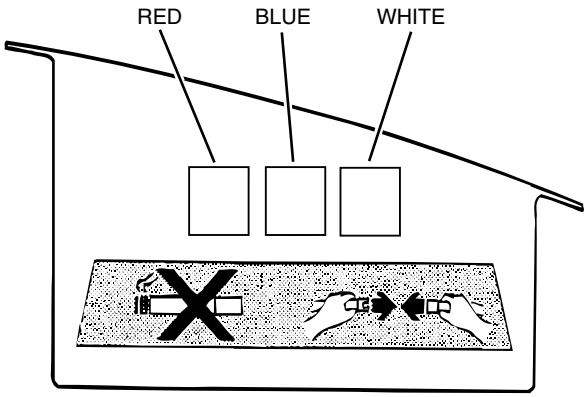


Figure 3-6. Flight Attendant Call Lights

Flight Attendant Discrete Call Lights

A flight attendant discrete call panel (Figure 3-6) is located in the cabin to alert the flight attendant to the origin of the call.

The panel consists of three colored lights and is mounted on the aft cabin bulkhead and/or opposite the flight attendants seat in the forward cabin.

The red light indicates a call from the cockpit, the blue light indicates a call from a PSU, and the white light indicates a call from the lavatory.

When the call originates from a PSU, the PSU call button also illuminates to indicate which unit is calling. The blue panel light and the PSU light extinguish when the PSU call button is pressed a second time. The toilet light operates in a similar fashion.

The cockpit call light extinguishes when the flight attendant hangs up the interphone.

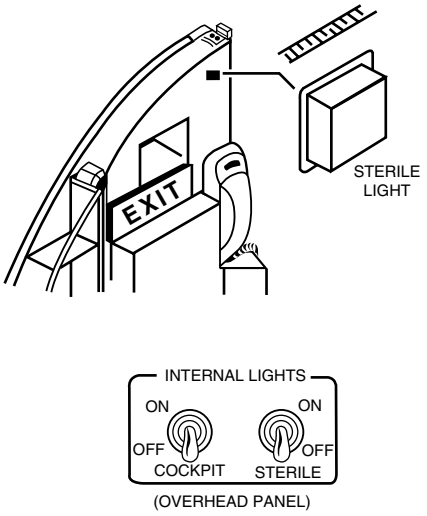


Figure 3-7. Sterile Light

Sterile Light

The sterile light (Figure 3-7) is installed as an option to inform the flight attendant that the aircraft is below 10,000 feet and the cockpit is off limits. It is located on the forward left bulkhead and is controlled by the STERILE switch in the INTERNAL LIGHTS section of the cockpit overhead lighting panel.

Lavatory Lights

The lavatory lights illuminate the lavatory compartment. They are controlled by an ON/OFF switch in the compartment ceiling. The lights are powered by DC bus 3.

Courtesy Light

The courtesy lighting system consists of two lights:

- A white light, located over the door, illuminates the entrance door stairs and the aisle towards the cockpit.
- A red light, located on the step between the passenger cabin and the cockpit, illuminates the step.

The lights are controlled by a three-position COURTESY switch on the flight attendants panel. The functions of the switch positions are as follows:

ON— Lights illuminate, regardless of main door condition, provided PWR SELECT switch (on the electrical control panel in the cock pit) is positioned to BATT. The lights are powered by the hot battery bus through the battery off bus relay. This prevents the battery from discharging if the COURTESY light switch is left ON when the PWR SELECT switch is OFF.

OFF— Lights are out regardless of door position.

AUTO—Lights illuminate when the main door is open. The lights are powered by the hot battery bus regardless of the PWR SELECT switch position. When the door is closed the lights go off automatically.

COMPARTMENT LIGHTING

Incandescent lamps are installed in the nose and tail compartments. Fluorescent lamps are used in the baggage compartment. All compartment lights are switched on or off in the compartment itself and are powered by DC bus 3.

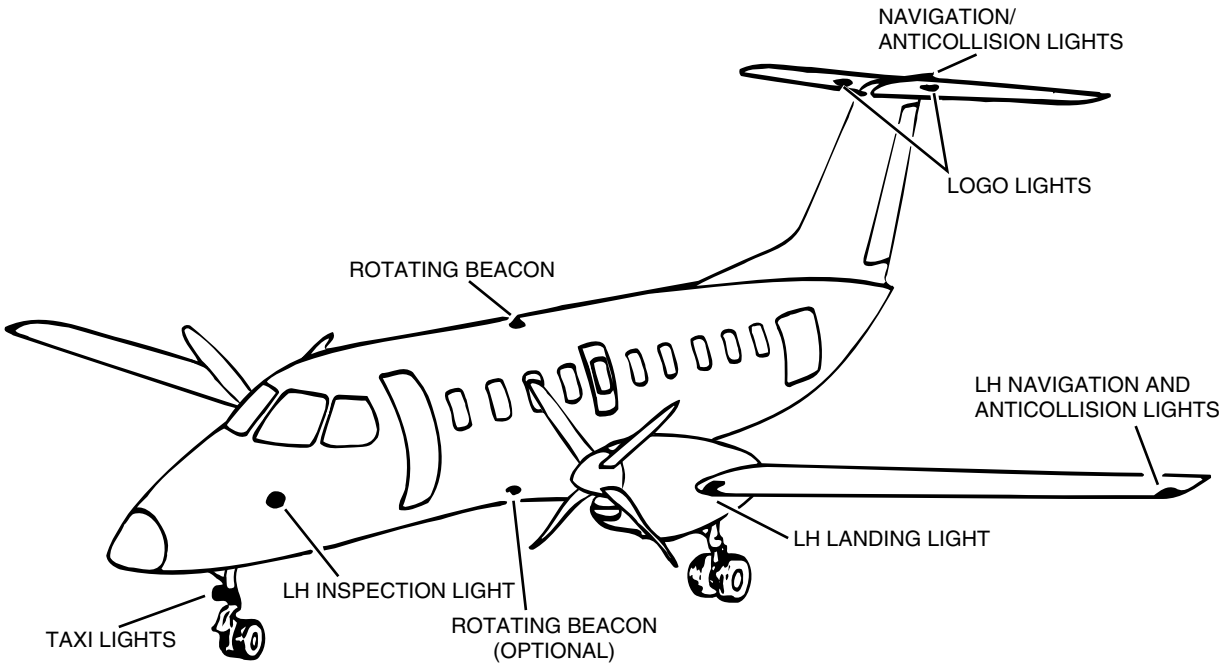


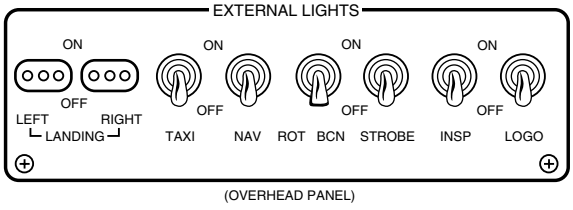
Figure 3-8. External Light

Fueling station lights are controlled by microswitches on the quick-disconnect latch of the compartment door and are powered by DC bus 2.

EXTERNAL LIGHTING

The external lighting system consists of the following lights (illustrated in Figure 3-8):

- Landing
- Taxi
- Navigation
- Anticollision
- Strobe
- Wing inspection
- Logo lights



(OVERHEAD PANEL)

Figure 3-9. External Lights Control Panel

All external light control switches are located in the EXTERNAL LIGHTS section of the cockpit overhead lighting control panel (Figure 3-9).

LANDING LIGHTS

The landing light system consists of two sealed-beam, high intensity lamps, one in each wing leading edge. They are installed just outboard of the engine nacelles to shield the cockpit from the glare.

The left landing light is powered by emergency DC bus 2 to ensure lighting is available in the event of an electrical emergency condition. The right is powered by DC bus 2.

The landing lights are controlled by two LANDING switches individually labeled LEFT and RIGHT (Figure 3-9).

NOTE

Normally, the MEL will allow dispatch with one landing light inoperative if both taxi lights are operative.

TAXI LIGHTS

The taxi light system consists of two sealed-beam lamps mounted on the nose landing gear oleo strut. One lamp has a wide beam and the other a narrow beam to ensure that all reference angles are illuminated while taxiing. The narrow beam lamp may also be considered a third landing light. The landing gear must be locked down for the lights to illuminate.

The taxi lights are powered by DC bus 1 and are controlled by the TAXI switch (Figure 3-9).

NAVIGATION LIGHTS

The navigation lights system consists of six lights, two colored (aviation red and green) and four white.

The red light is located in the left wing tip and the green in the right wing tip.

The four white lights are installed at the upper most center point on the vertical tail (two above and two below the strobe light assembly). Regulatory requirements for view angles require that, because the strobe assembly is mounted between the upper and lower navigation lights, both an upper and lower white navigation light must be illuminated for night operations.

The navigation lights are powered by DC bus 2 and are controlled by the NAV switch (Figure 3-9).

ROTATING BEACONS

The auxiliary anticollision light system consists of two red rotating beacons. One beacon is mounted on the upper surface of the fuselage center section and the other is mounted on the underside of the fuselage center section.

These 100-candle power lights are mounted on rotating platforms and enclosed under a red transparent cover. The lights and the motors which turn the platforms are powered by DC bus 2.

The rotating beacons are controlled by the ROT BCN switch (Figure 3-9).

STROBE LIGHTS

The main anticollision light system is the three 400-candle power, white strobe lights, installed in each wing tip and in the tail. The flash rate is 50 per minute.

The strobes are normally off for ground operation as a courtesy to other aircraft.

NOTE

Turn off the strobe lights when operating in clouds at night.

The strobe lights are powered by DC bus 1 and controlled by the STROBE switch (Figure 3-9).

WING INSPECTION LIGHTS

The wing inspection lights are two incandescent lights installed on each side of the aircraft nose. They are used to inspect the wing leading edges and engine air inlets for ice formation. They are also used to visually inspect the main landing gear.

The lights are powered by DC bus 2 and controlled by the INSP switch (Figure 3-9).

LOGO LIGHTS

The logo light system consists of two sealed-beam lights mounted in the horizontal stabilizer lower surface. They are used to illuminate a company logo or name on both sides of the vertical stabilizer.

The logo lights are powered by DC bus 3 and controlled by the LOGO switch (Figure 3-9).

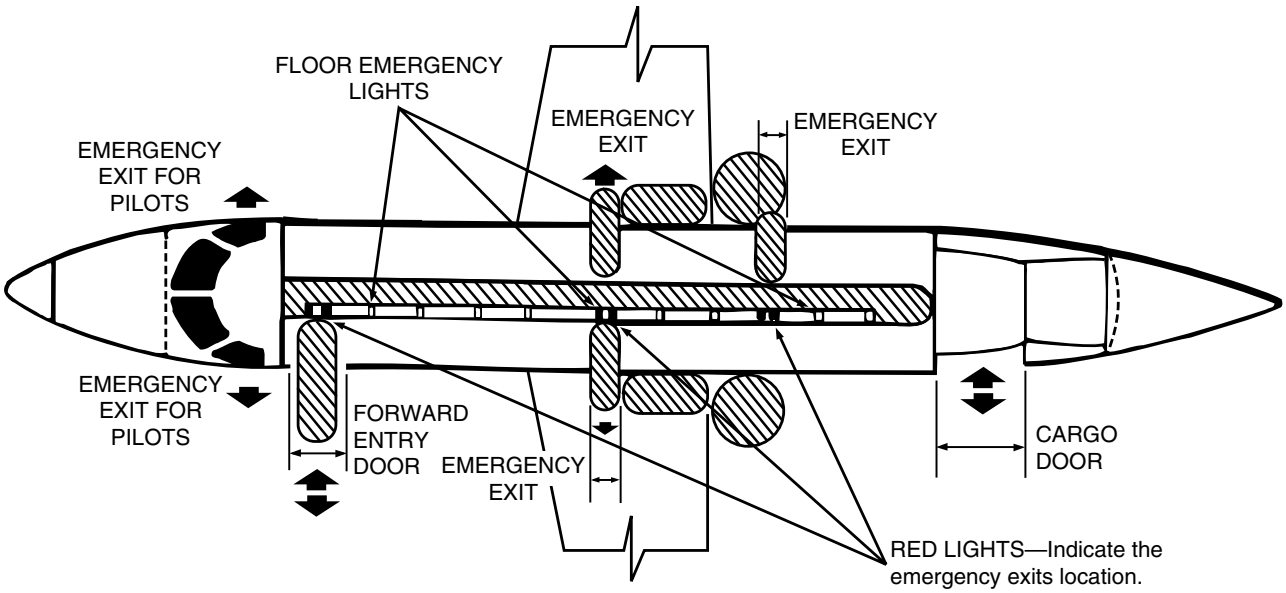


Figure 3-10. Emergency Illuminated Areas

EMERGENCY LIGHTING

Emergency lighting is a separate system designed to operate independently of the aircraft primary lighting system. It provides enough lighting to assure safe crew and passenger night evacuation under emergency conditions (Figure 3-10).

Power for the system is provided by three rechargeable battery packs with a maximum life of 17 minutes. The system can be controlled from either the cockpit or the cabin.

Components of the emergency lights system are:

- Rechargeable battery packs
- Control switch in the cockpit.
- Control switch on the attendant's panel in the cabin
- Emergency lights in the cabin
- EXIT signs in the cabin
- Aisle strip lighting
- External lights

INTERNAL EMERGENCY LIGHTS

Cabin Emergency Lights

Cabin illumination is provided by white lights mounted over the exits and in the ceiling. The lights over the exits illuminate the area immediately in front of each exit. The ceiling lights illuminate the aisle.

Installed on the aisle itself is a lighted strip assembly. These white lights are more closely spaced in the vicinity of the exits to facilitate exit location in the event the cabin fills with smoke.

Red EXIT signs mounted adjacent to each of the three emergency exits and the main entry door identify the location of the exits.

EXTERNAL EMERGENCY LIGHTS

There are two lights mounted in each wing fillet to illuminate the wing in the vicinity of the overwing emergency exits, and the ground aft of the wing.

OPERATION

The emergency exit lights are operated by two switches: one in the cockpit on the overhead lighting control panel, and the other on the attendant's panel in the cabin.

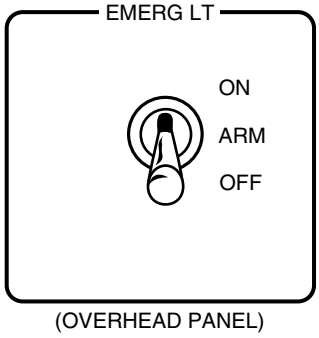


Figure 3-11. Cockpit Emergency Lighting Switch

Cockpit Emergency Lighting Switch

The EMERG LT switch on the lighting control panel in the cockpit (Figure 3-11) is a three-position switch labeled ON/ARM/OFF. Functions of each position are as follows:

OFF— Disconnects the battery packs from the emergency lights, turning them off. Battery packs are recharged by emergency DC bus 1. Only in the OFF position can the aircraft electrical system be turned off without the emergency lights coming on.

ARM—The normal flight position. Battery packs are recharging. If emergency DC bus 1 fails, all emergency lights come on automatically.

ON— Power from emergency DC bus 1 is cut off, simulating an electrical failure. Battery packs are connected to the emergency lights, turning them on.

NOTE

If the cockpit EMERG LT switch is in the ON or ARM position when the aircraft is deenergized, the battery packs will supply the emergency lights and will deplete the batteries in approximately 17 minutes. The battery packs must then be removed from the aircraft to be charged (approximately 22 hours).

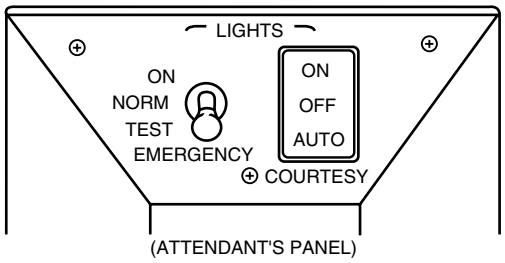


Figure 3-12. Flight Attendant Emergency Lighting Switch

Cabin Emergency Lighting Switch

The EMERGENCY switch on the attendant's panel in the cabin (Figure 3-12) is a three-position switch labeled ON/NORM/TEST. Functions of each position are as follows:

ON— Power from emergency DC bus 1 is cut off, simulating an electrical failure. Battery packs are connected to the emergency lights, turning them on, even if the cockpit overhead panel emergency lighting switch is selected to OFF.

NORM—Normal flight position. Allows control of the emergency light system from the cockpit.

TEST— Momentary position. Checks the system for proper operation by simulating an emergency DC bus 1 failure. Battery packs are connected to the emergency lights, turning them on. (Cockpit switch must be in the ARM position).

QUESTIONS

1. The EMB-120 has how many landing lights?
   A. 1
   B. 2
   C. 3
   D. 4

2. The EMB-120 has how many taxi lights?
   A. 1
   B. 2
   C. 3
   D. 4

3. Passenger cabin lighting is controlled by:
   A. A "PCL" switch located on the FO's side panel
   B. A "PCL" switch located on the captain's side panel.
   C. A three-position switch located on the attendant's panel
   D. All of the above

4. Taxi lights turn off automatically on landing gear retraction.
   A. True
   B. False

5. The flight attendant may be called from:
   A. The cockpit
   B. The cabin
   C. The lavatory
   D. All of the above

6. The emergency lights come on if:
   A. The cockpit switch is in ARM and emergency DC bus 1 fails.
   B. The cockpit switch is placed to ON.
   C. The attendant's switch is placed to ON, regardless of the position of the cockpit switch.
   D. All of the above

CHAPTER 4
MASTER WARNING SYSTEM

CONTENTS

Page

INTRODUCTION 4-1
GENERAL 4-1
VISUAL WARNING SYSTEM 4-2
Multiple Alarm Panel 4-2
WARNING and CAUTION Lights 4-3
Alarm Cancel 4-3
Alarm/Indication Lights 4-3
Alarm Lights Switch 4-3
AURAL WARNING SYSTEM 4-4
Power On Self-Test 4-5
Synthetic Voice Warnings 4-5
Additional Aural Alerts 4-5
Speakers and Audio System 4-6
WARNING SYSTEM OPERATION 4-6
QUESTIONS 4-11

ILLUSTRATIONS

Figure	Title	Page
4-1	Multiple Alarm Panel	4-2
4-2	WARNING/CAUTION Lights	4-3
4-3	Alarm Lights Switch	4-3
4-4	Aural Warning System (AWS) Interfaces	4-4
4-5	Multiple Alarm Panel Emergency Faults	4-9
4-6	Multiple Alarm Panel Abnormal Faults	4-10

TABLE

Table	Title	Page
4-1	Emergency Faults	4-7

CHAPTER 4
MASTER WARNING SYSTEM



INTRODUCTION

The master warning system on the EMB-120 includes a visual warning system and an aural warning system to alert the crew of equipment malfunctions, unsafe operating conditions requiring immediate attention, as well as indication that a system is in operation. A stall warning system is also installed.

A color representation of all the panel annunciator lights is located in the ANNUNCIATOR PANEL section in the back of this manual.

GENERAL

The visual warning system consists of two pair of master warning/caution lights on the glareshield, a center multiple alarm panel (MAP) to indicate individual system fault, and discrete alarm/indication lights. The aural warning system consists of voice warnings and a wide variety of other distinct aural alerts. The capability is provided to dim lights and to test and cancel most annunciation and aural warnings.

The stall warning system that provides continuous visual information concerning the angle of attack; also aural warnings, control column shakers, and control column pushers is covered in Chapter 15, Flight Controls.

VISUAL WARNING SYSTEM

The visual warning system consists of alarm/indication lights distributed on cockpit panels to provide information on a specific system failure or advisory information on system operations.

The major components of the system are:

- Multiple alarm panel
- Master WARNING and CAUTION lights
- ALARM CANCEL buttons
- ALARM LT switch

MULTIPLE ALARM PANEL

The annunciator panel, or multiple alarm panel (MAP), is a display of 40 alarm lights with red or amber colored lenses and applicable system captions (Figure 4-1). It is located on the center instrument panel, and monitors most of the aircraft systems. The captions are colored red for warnings and amber for cautions. There are 16 red captions that make up the center section of the MAP, with 12 amber captions on either side.

When a fault occurs the appropriate MAP light flashes to alert the pilots. The caption informs the pilot of a specific fault or directs attention to the proper system control panel for fault identification and correction.

When a red MAP light flashes, a signal is also generated to activate both red master WARNING lights on the glareshields. An amber MAP light activates both amber master CAUTION lights.

Upon pilot recognition of the alarm, the MAP must be reset by pressing one of the two ALARM CANCEL buttons located on the glareshield panel. When reset, the flashing alarm light turns steady and remains on for as long as the fault exists.

Once reset, the MAP is able to sense another fault and flash another alarm. Intermittent fault signals may trigger repeat alarms after having once been reset. In this case, the MAP light begins flashing again and must again be reset. The reset operation may be repeated as many times as necessary.

There are two types of captions displayed on the MAP:

- Specific
- System

Specific captions enable the pilot to determine the nature of the fault without having to look elsewhere (e.g., BATT OVERHEAT).

System captions indicate a general system fault. In this case, it is necessary to refer to the respective system control panel to identify the fault (e.g. ELEC). All control panel captions are the same color as the associated caption on the MAP.

Each MAP light and its associated fault is summarized in Figures 4-5 and 4-6 at the end of this chapter.

The multiple alarm panel is powered by emergency DC buses 1 and 2. Should one of these fail, the amber POWER OFF alarm light on the MAP will flash. When reset with the ALARM CANCEL button the light becomes steady; it will only extinguish if that power source is restored. If both power sources are lost, the MAP will not function.

Two light levels for the alarm lights are provided by the ALARM LT switch on the overhead lighting control panel. BRT for daylight operation and DIM for night.

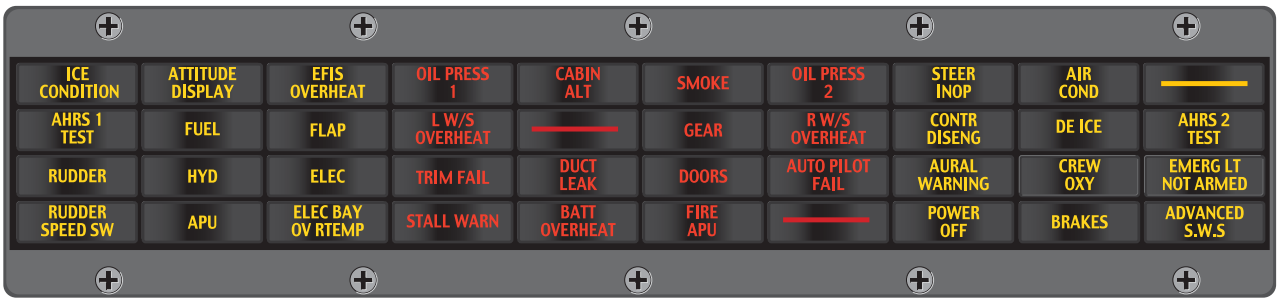


Figure 4-1. Multiple Alarm Panel



Figure 4-2. WARNING/CAUTION Lights

WARNING and CAUTION LIGHTS

The master WARNING and CAUTION lights are located on the pilot's and copilot's glareshield panels (Figure 4-2). They direct the crews attention immediately to an alarm condition on the MAP, and allow the degree of importance of the fault to be assessed through the color of the light.

ALARM CANCEL

The ALARM CANCEL buttons (Figure 4-2), located adjacent to the master WARNING and CAUTION lights, cancel the master WARNING or CAUTION light, stop MAP light flashing, and silence most voice warnings and aural alerts.

ALARM/INDICATION LIGHTS

All alarm/indication lights are unreadable when not illuminated . When illuminated, they remain on as long as the fault or condition exists.

MAP alarm lights provide information on a specific failure identified by the inscribed caption, or by a combination of the MAP caption and system panel caption.

Alarm light colors are used to represent the seriousness of the fault and the importance of the subsequent corrective action.

Indication light colors are used to represent the status and condition of system operations.

Red Alarm Lights

The red alarm lights advise of an emergency condition or fault usually requiring immediate attention and action by the crew. Failure to take the appropriate corrective action in a timely manner may lead to an unsafe flight condition.

Amber Alarm Lights

The amber alarm lights advise of an abnormal condition or fault requiring crew attention but not immediate action.

Green Indication Lights

Provide normal operating advisory indication of systems that are not normally used continuously (e.g., deicing boots).

White Indication Lights

Provide operating advisory condition of system components which are not usually in operation, or standby systems which operate when the main system has failed. (e.g., electric hydraulic pump ON lights).

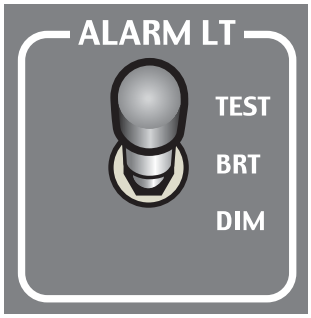


Figure 4-3. Alarm Lights Switch

ALARM LIGHTS SWITCH

The ALARM LT switch (Figure 4-3) is a three-position switch (TEST/BRT/DIM) located on the pilot's overhead panel. It provides for bulb testing and brightness control for the MAP and almost all of the indication lights. The exceptions are:

- APU/engine fire protection system lights
- Engine firewall shutoff valve advisory lights
- Flap annunciator light bars
- Backup battery indication light

- Landing gear position indication lights
- GPU available light
- T_6 lights
- Flight director/autopilot annunciator lights
- Main parking brake light
- Red stall warning lights

The functions of the three positions are as follows:

TEST — Checks the MAP and the master WARNING and CAUTION alarm light illumination by flashing the lights; illuminates almost all indication lights. During this test, a single-chime alert is produced by aural warning system.

BRT/DIM — Provides selection of bright or dim illumination modes for nearly all alarm lights. The exceptions are:

- Red alarm lights
- Backup battery ON light
- Master WARNING and CAUTION lights

AURAL WARNING SYSTEM

The aural warning system (AWS), operating in conjunction with the visual warning system (Figure 4-4), presents aural alerts in the form of specific voice warning messages and advisory sound effects to alert the crew that a fault has occurred.

The aural alerts are self-canceling when proper crew action results in correction of the condition causing the alert.

The AWS consists of an aural warning unit which processes sensor signals and generates synthesized voice messages and tones.

The unit contains two channels. In case of failure of the primary channel, the secondary channel is activated. In the case of a failure of both channels, the unit shuts down.

If the power is interrupted for more than 3 minutes and then restored, the unit will restart and perform the power ON self-test.

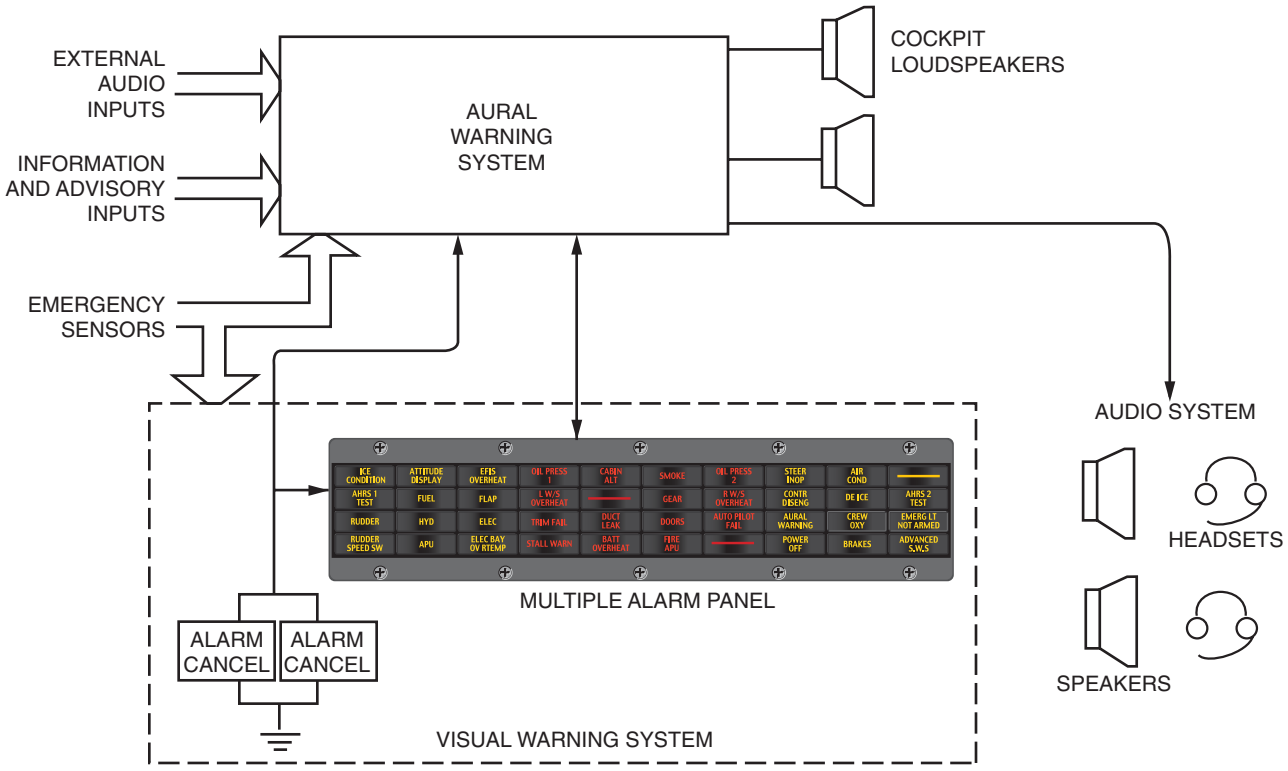


Figure 4-4. Aural Warning System (AWS) Interfaces

POWER ON SELF-TEST

When the aural warning unit is powered up (DC Bus 1/back-up DC Bus 1), both channels automatic ally go through a functional test. If both channels are fully operational, the unit announces "Aural unit OK". If either channel failed its power on test, the unit announces "Aural unit one channel".

Failures in both channels disconnects the unit and illuminates the amber AURAL WARNING light on the MAP.

The unit also performs periodic self tests to determine if any internal failure has occurred.

SYNTHETIC VOICE WARNINGS

A synthesized female voice, sometimes referred to as "Natasha" announces voice warnings associated with visual alarms to the crew. Most voice warnings are preceded by a three-chime alert.

The messages are:

- "Engine control" (EEC failure)
- "Doors" (not closed)
- "Stall warning" (computer fail)
- "Battery" (overheat)
- "Landing gear" (not extended)
- "Trim fail" (electric pitch)
- "Smoke" (detection)
- "Takeoff brakes" (parking brake set)
- "Takeoff trim" (out of takeoff range)
- "Takeoff flaps" (not set for takeoff)
- "Takeoff autofeather" (not armed)
- "Oil" (pressure low)
- "High speed" (exceeding V_{MO})
- "Cabin" (altitude above 10,000 feet)
- "Windshield" (overheat)
- "Duct leak" (bleed air)
- "Autopilot" (computer fail)
- "T_6" (over temperature limit)

ADDITIONAL AURAL ALERTS

In addition to the synthetic voice warnings, the six different sound effects listed below are also generated as aural alerts by the AWS.

Three-Chime Alert

A three-chime alert signals an emergency condition requiring immediate attention/corrective action by the crew, and:

- Precedes and takes priority over a voice warning.
- Accompanies all red MAP lights
- Is repeated at 1-second intervals until the fault has been cleared or canceled by the crew with the ALARM CANCEL button.
- With the exception of the landing gear alert, all emergency three-chime alerts may be canceled.

Multiple emergency faults result in a single three-chime alert followed by the voice messages prioritized in order of severity.

One-Chime Alert

A one-chime alert signals an abnormal condition requiring attention/corrective action by the crew, and:

- Accompanies all amber MAP lights.
- Is repeated at 5-second intervals until the fault has been cleared or canceled by the crew with the ALARM CANCEL button.
- All one-chime aural alerts may be canceled. (The one-chime alert stops regardless of the number of pending abnormal faults.)

Other Aural Alerts

- **Clacker**—A continuous clacker sound is produced whenever the stall warning system is activated.
- **Bell**—A continuous bell sound is produced whenever a fire warning condition is detected.
- **Beep**—The single-beep sound is produced whenever the ALARM CANCEL button is pressed.
- **2,900 Hz Tones**—Three short tones for altitude alerter

NOTE

Aural alarms are not provided for the nosewheel PEDAL STEER INOP lights, or the propeller BETA lights.

SPEAKERS AND AUDIO SYSTEM

The cockpit contains four speakers in addition to the pilot's and copilot's headphones.

Two of the four speakers are driven exclusively by the aural warning unit and are called the AWS pilot's and AWS copilot's loudspeaker. There is no provision for crew adjustment of the AWS speaker volume, nor can they be turned off.

The remaining two speakers, along with the crewmember headphones, are driven by their respective audio panel.

Aural alerts are always heard through the pilot's and copilot's headphones and can be heard through the audio system provided that SPKR is selected on the pilot's and/or copilot's audio panels.

WARNING SYSTEM OPERATION

Operation of the warning system groups the alarms into four priority levels. These levels, listed in order of priority, are:

Level 3 (1st Priority—Emergency). Operational or airplane systems condition requiring immediate corrective or compensatory action by the crew. (See table 4-1.)

- Red alert light on MAP
- Red WARNING light on glareshield
- Three-chime alert tone
- Voice warning
- Takes priority over all alerts except:
- Stall warning (clacker)
- Fire warning (bell)

NOTE

If stall and fire faults occur simultaneously, only the stall clacker is heard until the stall condition has been corrected. Thereafter, the fire bell sounds if the fire condition still exists.

If a stall or fire fault occurs at the same time as another emergency fault, the amplitude of the clacker or bell is attenuated to allow the crew to hear the voice message.

Level 2 (2nd Priority—Abnormal). Operational or airplane systems condition requiring immediate crew awareness and subsequent corrective or compensatory action by the crew.

- Amber alert light on MAP
- Amber CAUTION light on glareshield
- One-chime alert tone

Level 1 (3rd Priority—Advisory). Operational or airplane systems condition requiring crew awareness and may require crew action.

- White lights on system control panels
- Indicates operation of systems not usually required, or standby systems.

Level 0 (4th Priority—Informational). Operational or airplane systems condition requiring flight deck indication, but not necessary as a part of the integrated alerting system.

- Green lights
- Provides normal operating indication of commonly used systems.

Most aural warnings and alerts may be manually canceled with the ALARM CANCEL button. The exceptions are:

- Stall warning clacker
- "Glide slope" (below)
- "Landing gear" (not down locked with flap position greater than 17°)
- "High speed" (exceeding V_{MO})
- Takeoff configuration warnings

Table 4-1. Emergency Faults

FAULT	AURAL ALARM	VISUAL ALARM
STALL	Clacker tone output for 2 seconds out of every 4 seconds—Non-cancellable	Indications (not alarm) on the fast/slow indicators
ENGINE-WHEEL WELL or TAILPIPE FIRE	Fire bell sounds until fault has been cleared or manually cancelled.	FIRE ENG/WW and/or FIRE PIPE ZONE red lights (fire control panel)
APU FIRE		FIRE APU red light (multiple alarm panel) FIRE red light (APU fire control panel)
DESCENT BELOW GLIDE SLOPE (OPTIONAL)	GLIDE SLOPE voice message generated every 1.4 seconds	INDICATION (not alarm) on ADI or EADI
TRIM FAIL	TRIM FAIL voice message	TRIM FAIL red light (multiple alarm panel) TRIM amber light (autopilot and flight control panels)
AUTOPILOT FAILURE	AUTOPILOT voice message (*)	AUTOPILOT FAIL red light (multiple alarm panel) AP red light (autopilot and flight control panel)
AUTOPILOT DISENGAGEMENT (Airplanes Post-Mod. SB 120–022–0010 or SNs 120.047 and subsequent)		Proper annunciation on autopilot and flight control panels
EEC FAILURE	ENGINE CONTROL voice message	EEC 1 or EEC 2 red lights (glareshield panel)
SPEED EXCEEDING V_{MO}	HIGH-SPEED voice message – Non-cancellable	Indication (not alarm)on either airspeed indicator through the V_{MO} indicator
BATTERY OVERHEAT	BATTERY voice message	BATT OVERHEAT red lights (multiple alarm panel)
LANDING GEAR NOT DOWN-LOCKED	LANDING GEAR voice message – Non-cancellable when flaps ≥ 17°	GEAR red light (multiple alarm panel)

(*) AUTOPILOT voice message for AP disengagement is generated just once. On airplanes Post-Mod. SB 120–031–0003 or SNs 120.035, and subsequent, the AUTOPILOT voice message is inhibited when the airplane is on the ground.



Table 4-1. Emergency Faults (CONT)

FAULT	AURAL ALARM	VISUAL ALARM
LOW ENGINE OIL PRESSURE	OIL voice message	OIL PRESS 1 OR OIL PRESS 2 red lights (multiple alarm panel)
APU OR ENGINES AIR BLEED DUCT LEAKAGE	DUCT LEAK voice message	DUCT LEAK red light (APU control panel, air conditioning panel, and multiple alarm panel)
DOORS NOT CLOSED AND LOCKED OR MAIN DOOR ACTUATOR HIGH PRESSURE AFTER DOOR IS CLOSED	DOOR voice message (*)	FORWARD ACTUATOR, FORWARD, SERVICE, or CARGO (doors panel), and DOORS red light (multiple alarm panel)
WINDSHIELD OVERHEAT	WINDSHIELD voice message	LW/S or RW/S OVERHEAT red lights (multiple alarm panel)
CABIN ALTITUDE ABOVE 10,000 FT	CABIN voice message	CABIN ALT red light (multiple alarm panel)
AIRPLANE NOT IN TAKEOFF CONFIGURATION Preconditions: Airplane on ground and at least one power lever set for takeoff	TAKEOFF voice message, plus TRIM, or FLAPS, or BRAKES, or AUTOFEATHER, or a combination of them depending on which condition caused the fault—Non-cancellable for airplanes Post-Mod. SB 120–031–0018 or SNs 120.114 and subsequent.	
STALL WARNING FAIL	STALL WARNING voice message(*)	STALL WARN red light (multiple alarm panel)
SMOKE IN THE CARGO COMPARTMENT	SMOKE voice message	Failure warning red light (stall warning panel) SMOKE red light (multiple alarm panel)
INTERTURBINE OVERTEMPERATURE (Airplanes Post-Mod. SB 120–031–0006 or SNs 120.064, 120.066, 120.067, 120.070, 120.071, 120.073 thru 120.076, 120.079, and subsequent).	T_6 voice message	Red warning light on T_6 indicator

(*) For the airplanes Post-Mod. SB 120-31-0003 or S/N 120.035 and on: a) the STALL WARNING voice message is inhibited on the ground and, b) the DOOR voice message and the DOORS red light (on the MAP) associated with forward and cargo door not closed and locked are inhibited on the ground when the left condition lever is set to FUEL CUTOFF.

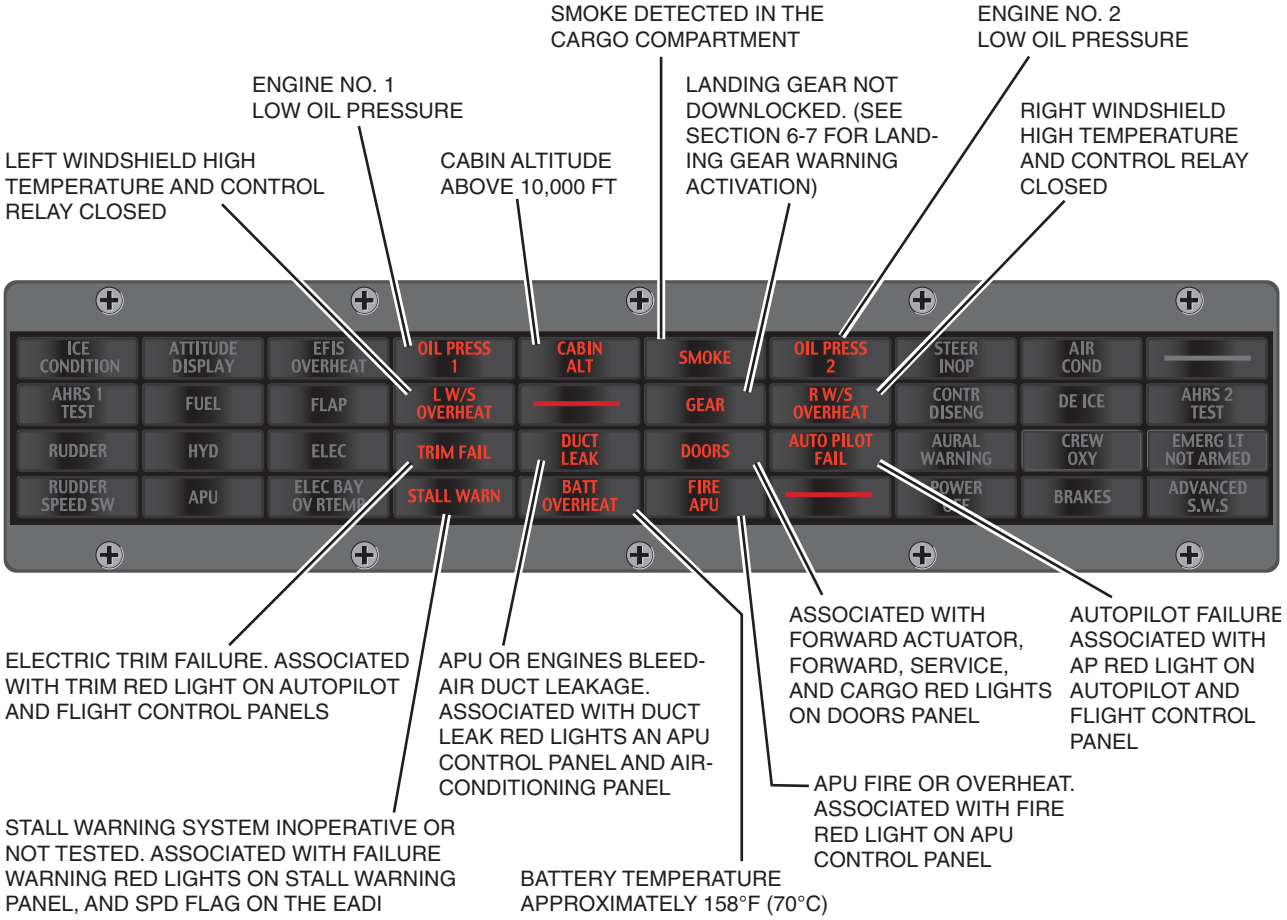


Figure 4-5. Multiple Alarm Panel Emergency Faults

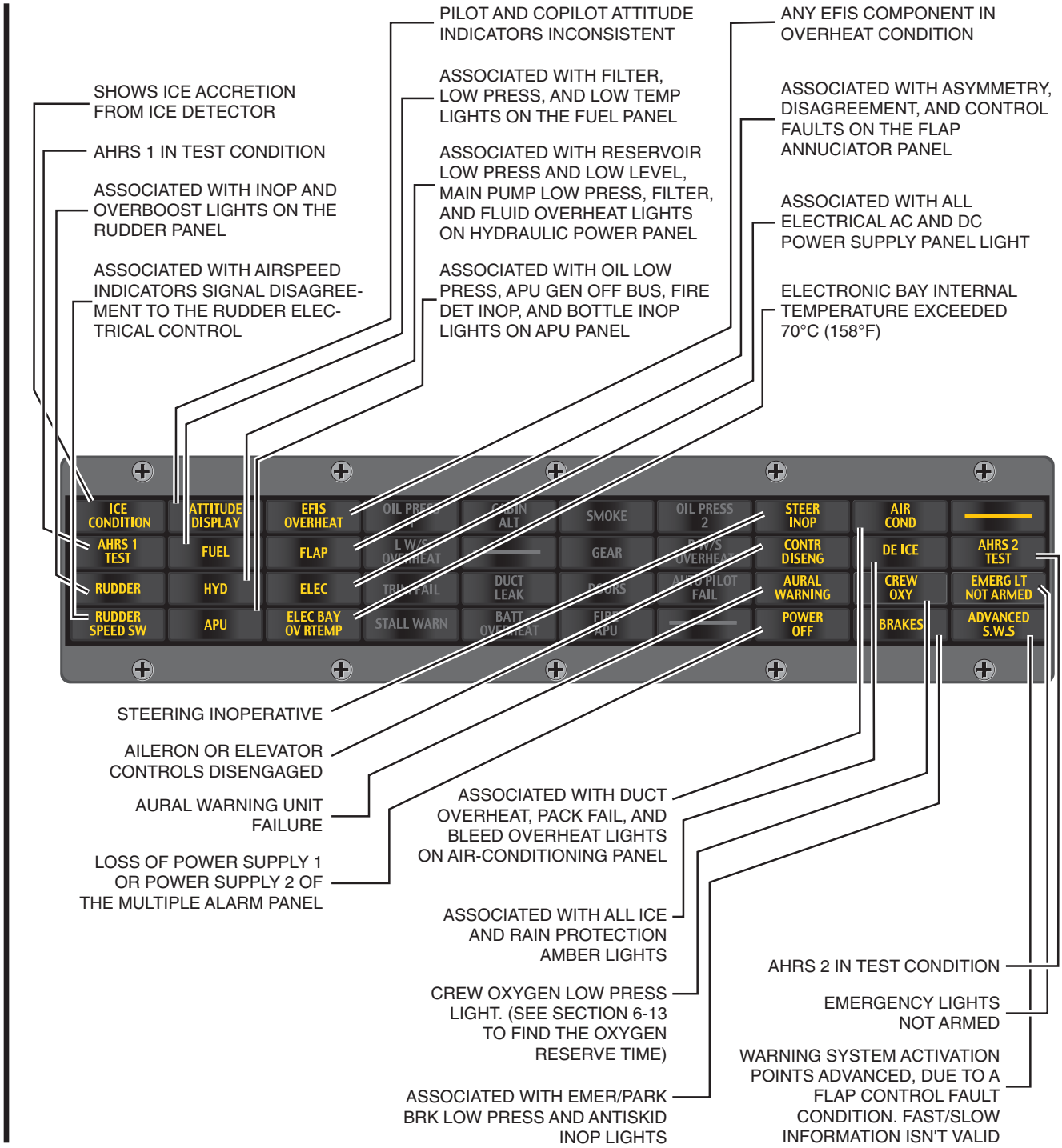


Figure 4-6. Multiple Alarm Panel Abnormal Faults

QUESTIONS

1. The EMB-120 has a:

A. Visual alert warning system
B. Aural alert warning system
C. Vocal alert warning system
D. All of the above

2. The color of the alarm/indication system lights are:

A. Red and green
B. Amber and white
C. Red and purple
D. Both A and B

3. Visual alarm light dim levels are controlled automatically.

A. True
B. False

4. The multiple alarm panel has:

A. One source of power
B. Two sources of power
C. Three sources of power
D. Does not need a source of power

5. The aural warning system volume may be controlled by:

A. A switch on the captain's lower sub panel
B. A switch on the FO's lower subpanel
C. A switch on the top right overhead panel
D. None of the above

CHAPTER 5
FUEL SYSTEM

CONTENTS

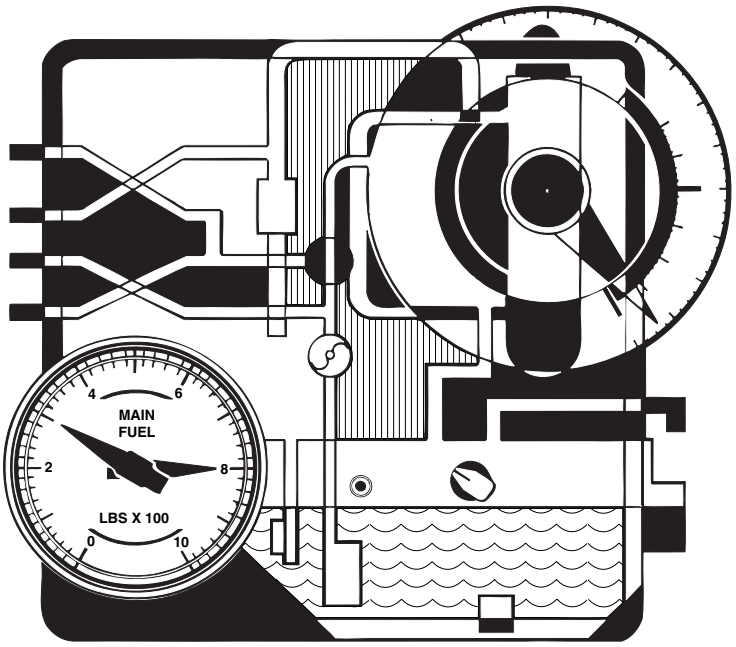
Page

INTRODUCTION 5-1
GENERAL 5-1
FUEL STORAGE SYSTEM 5-2
General 5-2
Fuel Tanks 5-2
Fuel Tank Venting 5-3
FUEL FEED SYSTEM 5-3
General 5-3
Fuel Pumps 5-5
Fuel System Valves 5-6
Engine Fuel Feed System 5-7
APU Fuel Feed System 5-9
FUEL SYSTEM CONTROL AND MONITORING 5-9
Fuel Control Panel 5-9
Fuel Management Panel 5-9
Fuel Quantity Indication 5-9
Fuel Flow Indicators 5-11
Fuel Totalizer 5-11
OPERATION 5-12
Normal 5-12
Crossfeed 5-13
FUEL SERVICING 5-14
Refueling 5-14
Defueling 5-19
QUESTIONS 5-23

ILLUSTRATIONS

Figure	Title	Page
5-1	Fuel Storage	5-2
5-2	Fuel Feed System	5-4
5-3	Jet Pump	5-5
5-4	Engine Fuel Feed System	5-7
5-5	Fuel Control Panel	5-9
5-6	Fuel Management Panel	5-9
5-7	Fuel Quantity Measuring Stick	5-10
5-8	Fuel Totalizer	5-11
5-9	Fuel System Operation During Start	5-12
5-10	Fuel System Normal Operation	5-12
5-11	Fuel System Crossfeed	5-13
5-12	Fueling Panel	5-14
5-13	Refueling System	5-15
5-14	Refueling Adapter	5-15
5-15	Refueling Vent Valve	5-16
5-16	Pilot Valve	5-16
5-17	Refueling Shutoff Valve	5-16
5-18	Refueling System Operation	5-17
5-19	Defueling System	5-19
5-20	Overhead Fuel Panel	5-21
5-21	Underwing Fueling Panel	5-22

CHAPTER 5
FUEL SYSTEM



INTRODUCTION

The EMB-120 Brasilia fuel system provides a means for storing fuel and distributing it to the engines and the auxiliary power unit. The fuel system is actually two seperate and identical systems, one located in each wing. During normal operation each system supplies fuel to its respective engine; however, fuel crossfeed capability is provided.

GENERAL

This chapter covers the operations of the airframe fuel system up to the engine fuel control, referred to as the hydromechanical metering unit (HMU). As fuel enters the HMU, fuel system operation becomes an engine fuel system function and is covered in Chapter 7, “Powerplant.”

Low-pressure fuel flow from the wing tank to the engine-driven fuel pump is provided by electrically driven boost pumps and, once the engine is operating, an ejector jet pump. The engine-driven fuel pump provides high-pressure fuel to the HMU for the engine and for operation of the various ejector jet pumps.

The fuel system is made up of the following subsystems:

- Fuel storage
- Fuel feed
- Fuel indicating
- Fuel servicing

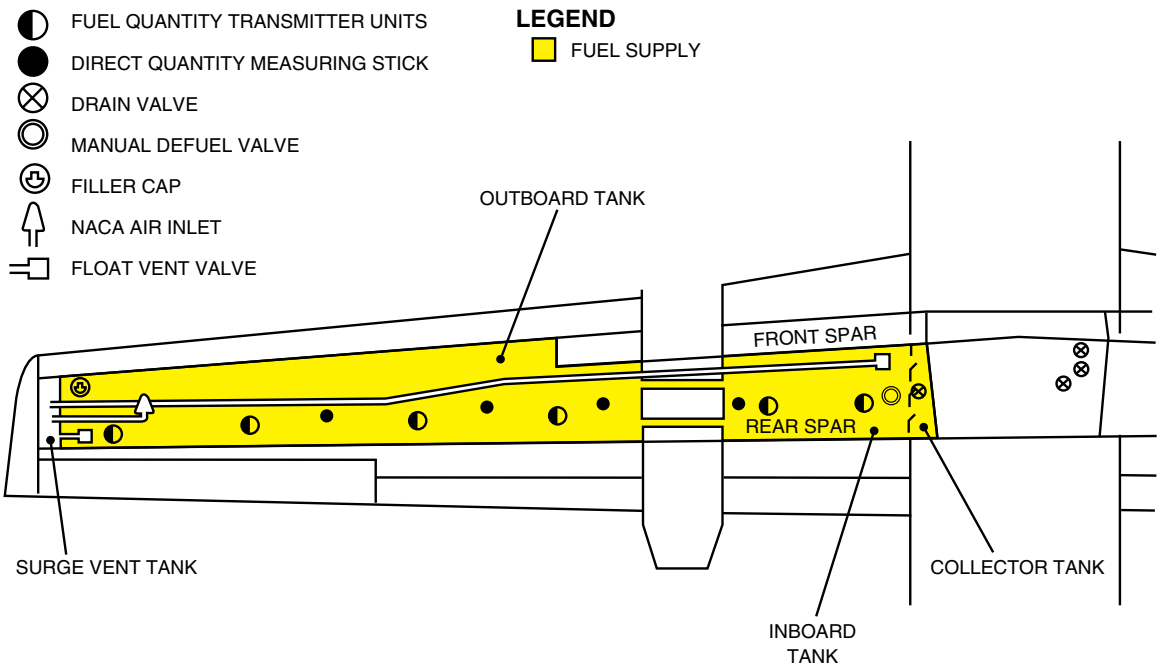


Figure 5-1. Fuel Storage

FUEL STORAGE SYSTEM

GENERAL

The fuel storage system is made up of four integral wing fuel tanks and a fuel tank vent system. Each tank is an integral part of the wing and is formed by the upper and lower wing skin and the wing's box beam.

The usable fuel capacity of each wing is 437 U.S. gallons or 2,866 pounds. The total usable fuel capacity is 874 U.S. gallons or 5,732 pounds. These values are based on 6.55 pounds per U.S. gallon.

FUEL TANKS

Each wing contains two tanks, an outboard and an inboard (Figure 5-1). The outboard tank is located between the landing gear nacelle and the wing tip. The inboard tank is located between the landing gear nacelle and the fuselage.

The outboard and inboard tanks are interconnected by a rectangular fuel duct located in the aft part of the main landing gear compartment, and a vent duct located in the upper main landing gear compartment. The fuel duct provides for gravity transfer of fuel between the inboard and outboard tanks, while the vent duct ensures pressure equalization. Both tanks, therefore, act as a single-fuel reservoir.

A third "tank" in each wing, the collector tank, is an integral part of the inboard tank. It is located at the lowest inboard portion of the tank and is where fuel is introduced into the fuel feed system to supply the engines.

Fuel is gravity fed from the inboard tank into the collector tank through two flap-type check valves. Fuel is also pressure-fed into the collector tank from the inboard tank by two transfer jet pumps located at the inboard tank low points. The transfer jet pumps reduce the unusable fuel and ensure a constant fuel level in the collector tank during normal airplane maneuvers or attitude changes.

A constant fuel level is necessary because, as previously mentioned, the collector tank is the point from which fuel is fed to the engine.

The electric booster pumps and the main jet pump for the fuel feed system are mounted in the collector tank, and a drain valve is located at its lowest point.

FUEL TANK VENTING

The fuel tank vent system provides a means to equalize pressure differentials between the wing tanks and the atmosphere. These pressure differentials may be created by fuel pump suction, airplane altitude variation, or fuel expansion due to temperature increase.

The fuel tank vent system is identical in both wings and consists of the following components:

- Tank interconnecting vent duct
- Surge vent tank
- NACA air intake
- Flame arrestor
- Float vent valves

The vent duct interconnects the outboard and inboard tanks. This duct is located in the upper part of the landing gear compartment and interconnects the upper part of both tanks to insure pressure equalization and vent system efficiency.

The surge vent tank is located at the wing tip end of the outboard tank (Figure 5-1), and provides the means for eliminating internal and external pressure differentials.

The NACA air intake (Figure 5-1) connects the surge vent tank with the atmosphere. Located on the underside of the wing, it is mounted along with a sedimentation tank in such a way that, in case of rain, water is separated and drained overboard.

The flame arrestor allows air/fuel vapors to vent to the atmosphere but prevents external flames from propagating through the vent system tubing into the fuel tanks. It is located in the vent tube between the NACA air inlet and the surge vent tank.

A float vent valve, located at the highest point in each of the tanks (Figure 5-1), prevents fuel from flowing from the fuel tank into the surge vent tank during slips, skids or taxiing turns.

NOTE

The float vent valve ***is not*** the same as the refueling vent valve explained in the refueling/defueling section. The normal tank vent lines ***are not*** large enough to allow for a safe single-point refueling operation.

FUEL FEED SYSTEM

GENERAL

The fuel feed system (Figure 5-2) is divided into two identical left and right subsystems which provide fuel to the main engines and APU. The main engines are normally fed by the fuel system in their respective wing. The APU is normally fed from the fuel system in the right wing.

The fuel feed system in each wing consists of the following components:

- Electric boost pumps
- Main jet pump
- Transfer jet pumps
- Check valves
- Firewall shutoff valve
- Crossfeed
- Motive flow shutoff valve
- Relief valve

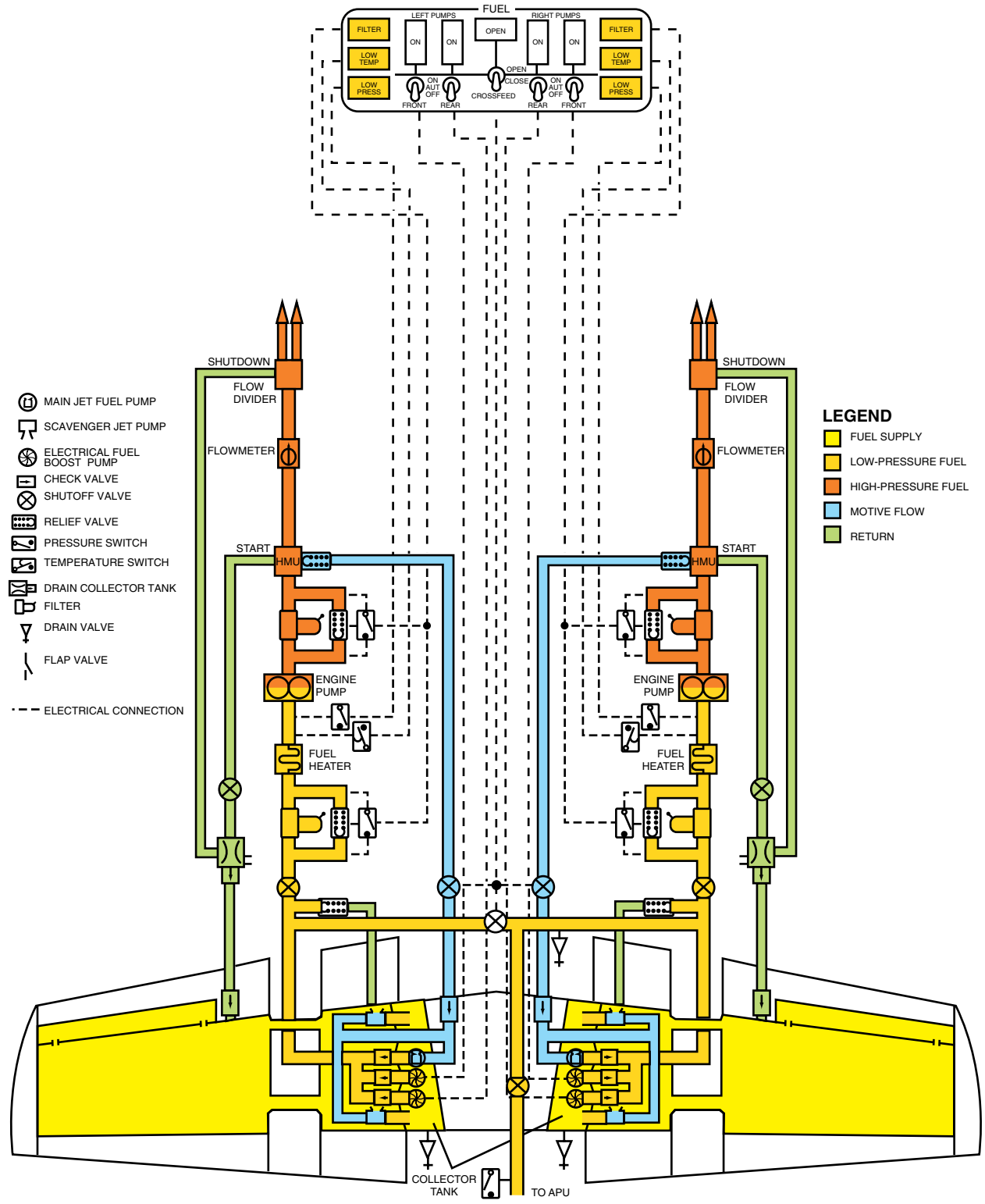


Figure 5-2. Fuel Feed System

Operation and monitoring of each fuel feed system is accomplished using the:

- Fuel control panel
- Fuel management panel

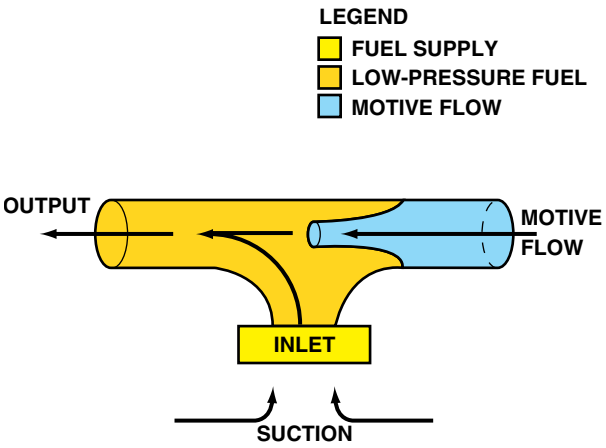


Figure 5-3. Jet Pump

FUEL PUMPS

The three types of fuel pumps on the EMB-120 are:

- Engine-driven fuel pump
- Electric boost pumps
- Ejector or "jet" pumps

Operation of the engine-driven fuel pump, mounted on the engine, will be discussed later in more detail.

The electric boost pumps and the ejector or "jet" pumps, located in the fuel tanks, provide fuel to the engine-driven pump. The electric boost pumps operate whenever electric power is applied to them. Jet pumps, which operate on the venturi principle, (Figure 5-3) operate only when fuel under pressure from another source flows through them. This operating fuel flow, called "motive flow," is supplied by the high-pressure engine-driven fuel pump through a bypass valve on the HMU.

Electric Boost Pumps

There are two electric boost pumps, designated front and rear, located in each collector tank. In normal operation, these pumps are used during engine start and when crossfeeding fuel. They are also used with a main jet pump failure and when operating the APU with the right engine shut down.

The output of each boost pump, 1,800 pph at 20 psi, is sufficient to feed both engines under any condition.

The 28 VDC boost pumps are controlled by two three-position (OFF/AUTO/ON) switches, labeled FRONT and REAR, located on the overhead FUEL panel. In the AUTO position, the boost pump is energized automatically when fuel pressure drops below 9 psi. A white ON light above the switch indicates when the boost pump is energized.

Main Jet Pump

A main jet pump, located in each collector tank, supplies fuel at 1,660 pph and 35 psi to the engine-driven fuel pump during normal operations.

High-pressure, "motive flow," fuel is used to operate the main jet pump. If motive flow is lost, the main jet pump will not operate. This failure may be recognized by the electric boost pumps cycling on and off (in AUTO), alternating with the LOW PRESS lights on the overhead FUEL panel. The FUEL light on the MAP and the master CAUTION lights will also flash.

Transfer Jet Pumps

Two transfer (scavenger) jet pumps, also operated by motive flow, are located in each inboard tank. Their purpose is to keep the collector tank at a level sufficient to provide a constant source of fuel for the main jet pump and the electric boost pumps.

When operating, the transfer jet pumps increase the usable fuel by taking fuel from the lowest portions of the inboard tank and transferring it into the collector tank. When the transfer jet pumps are not operating, usable fuel in the corresponding wing tank is reduced by 79.6 lb (12.15 US gallons).

If motive flow is lost, the transfer jet pumps will not operate. There is no failure annunciation for the transfer jet pumps. However, with the loss of motive flow, the main jet pump no longer operates either and the failure is recognized by the LOW PRESS indications previously discussed.

FUEL SYSTEM VALVES

Fuel Pump Check Valves

These check valves are designed to prevent the return of fuel from the supply lines to the collector tank through an inoperative pump. The valves are placed in the fuel lines at the outlets of each electric boost pump and main jet pump.

Firewall Shutoff Valve

The firewall shutoff-valve is an electrically actuated gate valve that provides a means to stop the flow of fuel to the engine.

During normal operation, the valve is in the open position allowing fuel to flow from the tanks to the HMU.

The valve closes when the respective engine FIRE HANDLE is pulled. A white fuel CLOSED light on the glareshield fire protection panel illuminates when the shutoff valve is closed.

Crossfeed Valve

The crossfeed shutoff valve is located in the crossfeed manifold that connects the right and left fuel systems. This crossfeed system enables the engines and the APU to receive fuel from either or both wings.

The crossfeed valve is physically and functionally identical to the firewall shutoff valves. It controls the flow of fuel through the crossfeed manifold. During normal operation this valve is closed.

The valve, powered by emergency DC bus 1, is controlled by the CROSSFEED switch on the overhead fuel control panel. When the crossfeed valve opens, both a white light above the switch (labeled OPEN) and a white light on the fuel quantity indication panel (labeled CROSS FEED OPEN) illuminate.

When the crossfeed valve is opened, both motive flow shutoff valves close. Therefore, during crossfeed operations, an electric boost pump must be used. Also, because motive flow is lost and the transfer jet pumps no longer operate, the unusable fuel is increased by 79.6 lbs (12.15 U.S. gallons) in each tank.

Motive Flow Shutoff Valve

The motive flow shutoff valves control the fuel flow from the HMU to the jet pumps. They are ball-type, electrically actuated valves and operate in conjunction with the crossfeed valve.

These valves are normally in the open position. When the crossfeed switch is placed in the OPEN position, the motive flow shutoff valves close, blocking fuel flow to the main and transfer jet pumps.

A check valve is located at the outlet of each motive flow shutoff valve to prevent reverse fuel flow into the engine compartment if a fuel line should rupture.

Relief Valve

Each fuel line is equipped with a pressure relief valve. The relief valve will divert fuel from the fuel line to the inboard tank when fuel line pressure exceeds 50 psi.

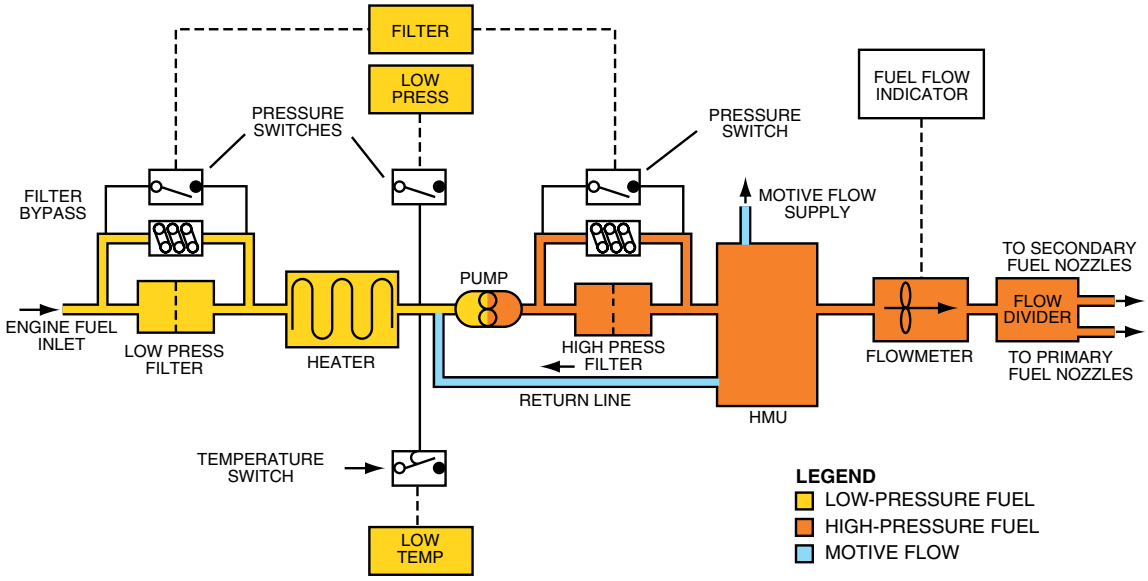


Figure 5-4. Engine Fuel Feed System

ENGINE FUEL FEED SYSTEM

The engine fuel feed system (Figure 5-4) delivers fuel to the combustion section with enough fuel flow to sustain continuous combustion under all operating conditions.

The engine fuel feed system receives its fuel from the electric boost pumps or the main jet pump in the fuel tank.

The discussion which follows only concerns fuel system components up to the hydromechanical metering unit (HMU). For further discussion of engine fuel system past the HMU, consult the engine fuel system section of Chapter 7, "Powerplant."

Components discussed in this section are:

- Low-pressure filter
- Filter impending bypass warning
- Fuel heater
- Low fuel temperature warning
- Low fuel pressure warning
- Fuel pump
- High-pressure filter
- Filter impending bypass warning

Low-Pressure Filter

The low-pressure filter is mounted with the fuel heater and is replaceable. A bypass valve is installed on the low-pressure filter assembly to ensure continuous fuel flow in the event of a clogged filter.

Filter Impending Bypass Warning

If the fuel pressure differential across the filter exceeds 1.5 psid, the amber FILTER light on the overhead fuel control panel illuminates. Also, the amber FUEL light on the MAP illuminates, triggering a single-chime aural alert, and the master CAUTION lights flash. If the differential reaches 3 psid, the bypass opens.

Fuel Heater

The fuel heater assembly is located on the left side of the Turbo Machinery Module (TMM) accessory gearbox. The fuel heater uses engine oil to heat the fuel to prevent ice formation and assist in keeping the fuel temperature within a specific range. Accumulation of ice particles in the fuel system (from water suspended in the fuel) could restrict the fuel flow and interfere with engine operation.

A thermal element in the fuel heater/low-pressure filter assembly controls a sliding valve that regulates the oil flow to maintain the fuel temperature at approximately 20°C.

Low Fuel Temperature Warning

When the fuel temperature drops into the freezing range, the amber LOW TEMP light on the overhead fuel control panel illuminates. Also, the amber FUEL light on the MAP illuminates, triggering a single-chime aural alert, and the master CAUTION lights flash. The warning is canceled when the fuel temperature rises to approximately 6°C.

Low Fuel Pressure Warning

Fuel from the fuel tank electric boost pumps or main jet pump is supplied to the engine fuel system at 25 to 35 psi.

If the fuel pressure drops below 9 psi, the amber LOW PRESS light on the overhead fuel control panel illuminates. Also, the amber FUEL light on the MAP illuminates, triggering a single-chime aural alert, and the master CAUTION lights flash.

During engine start, voltage may decrease, resulting in electric fuel pump output lower than 9 psi. Therefore, during the start cycle, a relay inhibits the low fuel pressure warning circuit.

Fuel Pump

The engine fuel pump assembly is located on the front portion of the TMM accessory gearbox and is driven by the Nh spool. The assembly consists of the following components:

- Ejector jet pump
- Fuel pump inlet filter
- Gear pump

Ejector Jet Pump—The ejector jet pump is installed in the fuel pump assembly to increase the fuel pressure coming from the fuel heater. It is operated by fuel returning from the HMU pressure regulating valve.

Fuel Pump Inlet Filter—A filter is installed in the gear pump inlet. Should clogging occur, the filter is removed from its seat and unfiltered fuel is supplied to the pump. No warning is provided to the crew if this filter clogs; however, maintenance can detect the problem on routine inspections.

Gear Pump—The engine fuel pump is a high-capacity, positive displacement, mechanical gear pump. Output volume and pressure are directly proportional to engine speed. The gear pump provides high-pressure fuel for the operation of the HMU and the fuel tank ejector jet pumps.

High-Pressure Filter

The high-pressure filter is mounted at the fuel pump outlet. A bypass valve is installed on the filter assembly to ensure continuous fuel flow to the HMU in the event of a clogged filter.

Filter Impending Bypass Warning

If the fuel pressure differential across the high-pressure filter exceeds 25 psid, the amber FILTER light on the overhead fuel control panel illuminates. Also, the amber FUEL light on the MAP illuminates, triggering a single-chime aural alert, and the master CAUTION lights flash. Should the differential pressure across the filter reach 50 psid, the bypass valve opens and releases unfiltered fuel to the HMU.

NOTE

Both the high-pressure and low-pressure impending bypass switches on an engine illuminate the same FILTER light on the fuel control panel. Maintenance must determine which filter is about to bypass.

APU FUEL FEED SYSTEM

The APU fuel feed system consists of an electrically actuated shutoff valve and fuel supply lines.

Also included in the supply line is a low-pressure switch and appropriate engine fuel components.

The APU fuel shutoff valve is identical to the firewall fuel shutoff valves. It is electrically powered, through the APU master switch, from emergency DC bus 2.

FUEL SYSTEM CONTROL AND MONITORING

FUEL CONTROL PANEL

The fuel control panel (Figure 5-5) is located on the pilot's overhead panel and is labeled FUEL. It contains fuel system annunciator lights and the control switches for the electric boost pumps and crossfeed shutoff valve.

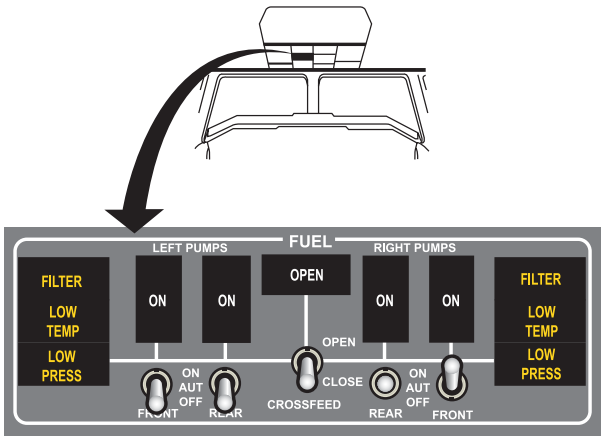


Figure 5-5. Fuel Control Panel

FUEL MANAGEMENT PANEL

The fuel indicating system provides information on the fuel quantity, flow, and consumption for each wing. All of this information is displayed on the fuel management panel (Figure 5-6) located on the forward center console.

This panel contains the master quantity indicators, fuel flow indicators, totalizer/detotalizer, and CROSS-FEED OPEN indicator light.

FUEL QUANTITY INDICATION

The quantity of fuel in the tanks is measured by fuel quantity transmitter units. A signal is transmitted to the master quantity indicators in the cockpit and to the repeater quantity indicators on the external fueling panel. Fuel quantity may also be manually checked using the direct reading, magnetic, dripless measuring sticks.

The electrical fuel quantity indicating system is a capacitance-type system. It indicates the amount of fuel, in pounds or kilograms, stored in each wing. Each wing has an identical and independent measuring system consisting of a master indicator and six tank units.

Master Quantity Indicators

The master fuel quantity indicators, one for each wing's fuel supply, provide the flight crew with the indication of total fuel quantity in that wing. The indicators are located on the fuel management panel.

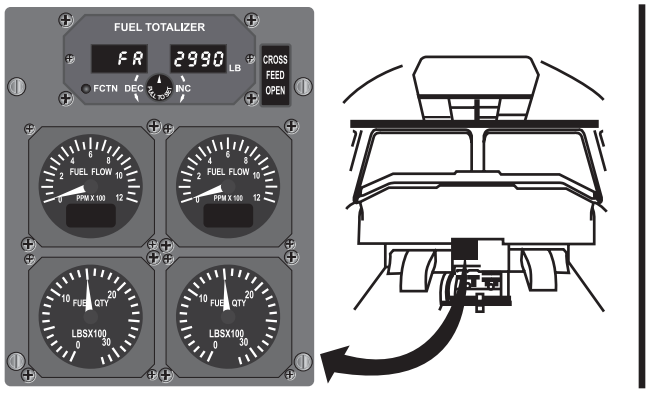


Figure 5-6. Fuel Management Panel

The back-lighted indicator has an analog scale that displays fuel quantity from 0 to 3,200 pounds in 100 pound increments.

The gage indications are "true" fuel quantity, (i.e., the fuel quantity indication is compensated for fuel temperature and density). Also, a zero indication on the gage represents zero usable fuel in level flight and does not represent any unusable fuel in the tanks. The indicator reads "0" if electrical power is lost.

Repeater Quantity Indicators

The repeater fuel quantity indicators are located on the external fueling panel to provide an indication of fuel quantity in each wing during refueling operations.

Fuel Quantity Tank Unit

The tank unit is a sensor which provides a DC signal, proportional to the fuel mass, to the master indicator.

There are six tank units in each wing, four in the outboard tank and two in the inboard tank. Each tank unit is a different size and not interchangeable.

Fuel Quantity Measuring Sticks

The fuel quantity measuring sticks (Figure 5-7) provide for manual measurement of the fuel quantity in each wing. The system consists of four measuring stick assemblies located on the underside of each wing. Three assemblies are located in the outboard tank and one in the inboard tank.

All fuel quantity readings should be done with the airplane laterally leveled. Each measuring stick assembly consists of a magnet floating on the surface of the fuel in the tank which attracts the upper end of a calibrated stick. When the end of the measuring stick aligns with the floating magnet, the stick remains in that position indicating the fuel quantity.

The procedure to manually determine the amount of fuel is:

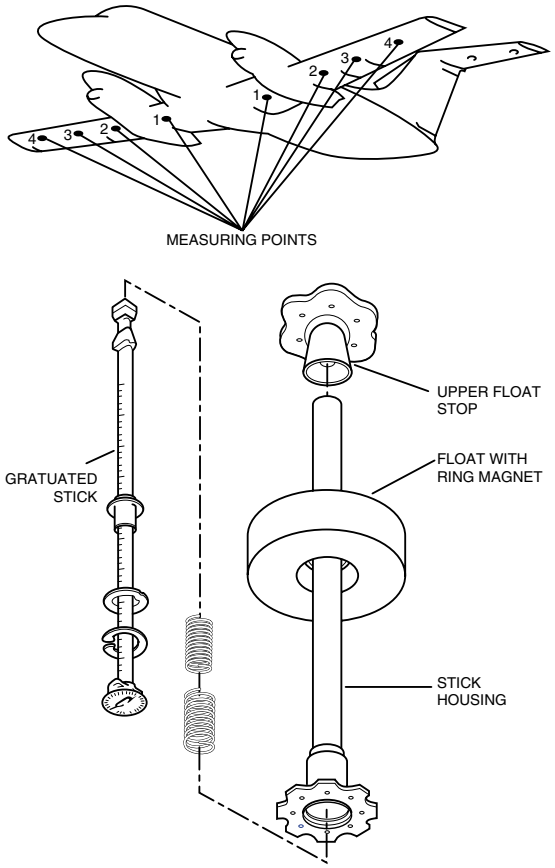


Figure 5-7. Fuel Quantity Measuring Stick

1. Start at the wingtip measuring stick (point No. 4).

2. Open the measuring stick by pressing and turning the stick to unlock it. Allow it to lower smoothly until it is held suspended by the magnetic float.

3. Move inboard checking the remaining sticks until a stick reading greater than zero is found. Read and record the value on the stick. Read the sticks in the other wing and record the results.

4. Consult the scale conversion table in Volume 2 of the *Operations Manual* to determine fuel on board.

FUEL FLOW INDICATORS

The flow display for each engine is a combined analog and digital display. It indicates mass fuel flow, compensated for temperature and density, in pounds per hour or kilograms per hour.

The indicators are mounted on the fuel management panel. If a power failure occurs, the pointers return to zero and the digital display will be blank.

No fuel flow indicator is provided for APU. Refer to the APU section for programmed APU fuel consumption rates provided to the totalizer for various load conditions.

FUEL TOTALIZER

The fuel totalizer (Figure 5-8) has two microprocessor-controlled digital displays. The function window displays FU (fuel used) or FR (fuel remaining), and the corresponding quantity is shown in the adjacent LB window (the last digit is fixed at zero).

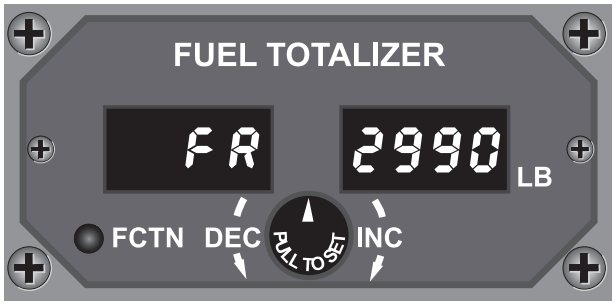


Figure 5-8. Fuel Totalizer

The desired display is selected by pressing the FUNC button.

The set knob is used to initialize or reset the system.

When power is applied to the electrical system, the fuel totalizer display should be compared to the values shown on the fuel quantity indicators.

This is required because the FR display presented on power-up, is the previous fuel remaining from memory. If FU is selected, it is the fuel used from the previous flight.

If the values do not match the fuel quantity indicators, the fuel totalizer must be reset and initialized.

To reset the total fuel used display to zero:

1. Press the FUNC button to select FU.
2. Pull and hold the set knob for three seconds.

Total fuel remaining initialization may be accomplished automatically or manually.

To automatically initialize the fuel remaining:

1. Press the FUNC button to select FR.
2. Pull and hold the set knob for three seconds.
3. The display initializes to the total fuel quantity on board as indicated on the fuel quantity master indicators.

If a fuel quantity indicator is inoperative, initializing FR displays the total fuel on board based on the operative fuel quantity indicator. In this case, the fuel remaining must be set manually.

To manually set the fuel remaining:

1. Press the FUNC button to select FR.
2. Rotate the set knob until the desired quantity is set.

Clockwise rotation of the knob increases the indicated value, while counter-clockwise decreases the value. The rate of change of the value occurs at two speeds, FAST and SLOW, depending on how far the knob is rotated.

The fuel totalizer has a fail-safe check capability. If a “sum check” error is detected, ER is displayed in the function window.

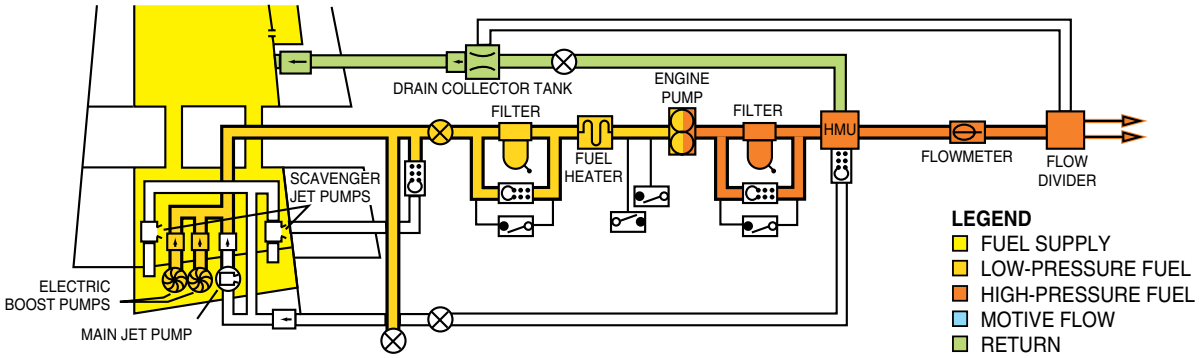


Figure 5-9. Fuel System Operation During Start

To reset the error condition, press the FUNC button. The function window now displays FR, and the LB display indicates "0000". To reinitialize the totalizer, pull the set knob.

If a power failure or signal loss occurs, the function and value displays will be blank.

OPERATION

NORMAL

During engine start (Figure 5-9), fuel pressure is initially provided by the electric boost pumps. Fuel is pumped from the collector tank (through a check valve, firewall shutoff valve, fuel filter, and fuel heater) to the engine-driven fuel pump.

The engine-driven fuel pump sends the fuel, under much greater pressure, to the HMU for distribution to the engine combustion chamber.

Excess fuel not used for starting is routed, via the return solenoid valve, to the drain collector tank.

This bypass fuel purges the HMU of air, evacuates the drain collector tank of fuel from the previous shutdown, and returns to the main fuel tank.

During normal engine operation (Figure 5-10), fuel to the engine-driven pump is supplied by the main jet pump.

Excess fuel not used for engine operation is sent back to the tanks by the HMU via the motive flow shutoff valve. This fuel provides the motive flow to operate the main jet pump and transfer (scavenger) jet pumps.

During engine shutdown, fuel from the engine flow divider is dumped to the drain collector tank. The tank holds the fuel until the next engine start, when the fuel is evacuated to the main tanks.

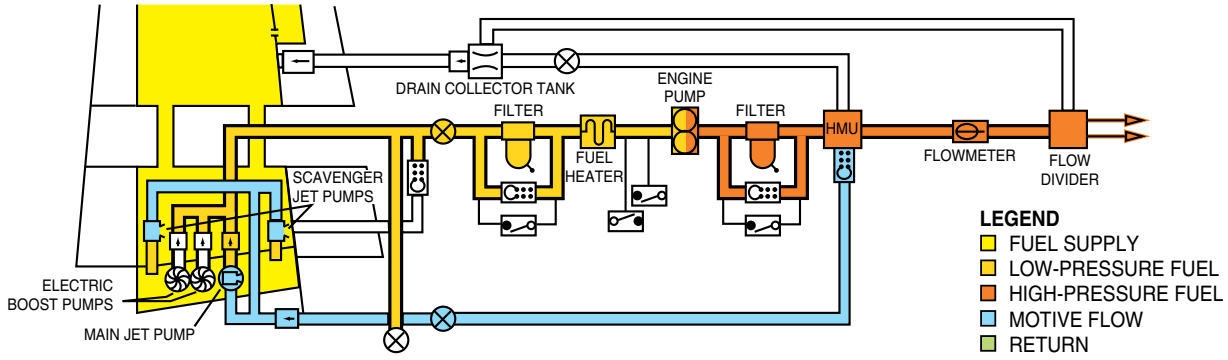


Figure 5-10. Fuel System Normal Operation

CROSSFEED

A crossfeed line, connecting the left and right systems, allows the feeding of fuel to either engine from the opposite tank or both engines from either wing tank. The APU may also be fed from either wing tank.

To crossfeed fuel:

1. Turn on the electric boost pump in the tank from which fuel is to be used.

2. Select the crossfeed switch to OPEN.

When the crossfeed valve opens, the two motive flow valves close to permit proper crossfeed operation. Otherwise, main jet pump output of 35 psi would override 20 psi boost pump output.

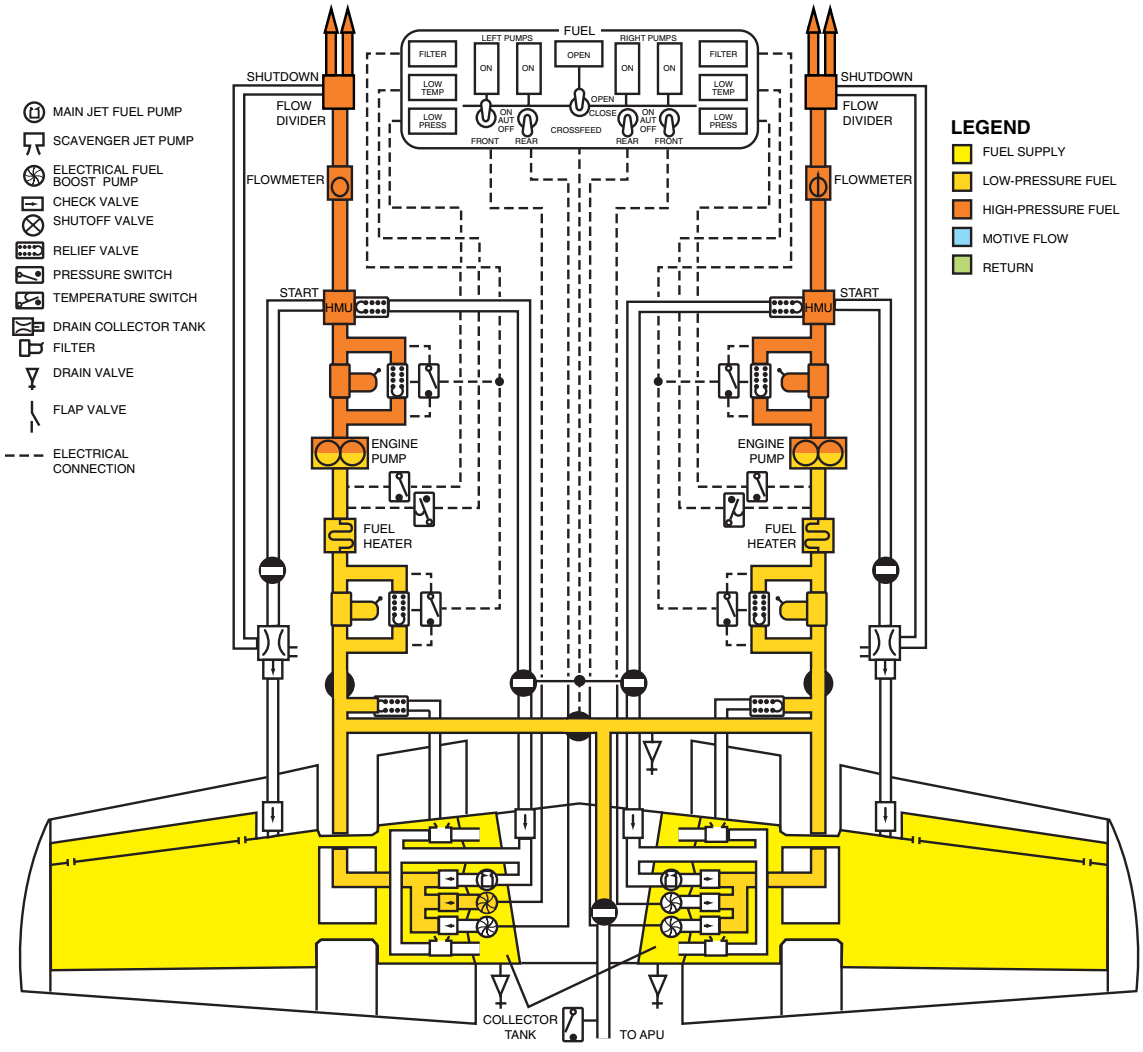


Figure 5-11. Fuel System Crossfeed

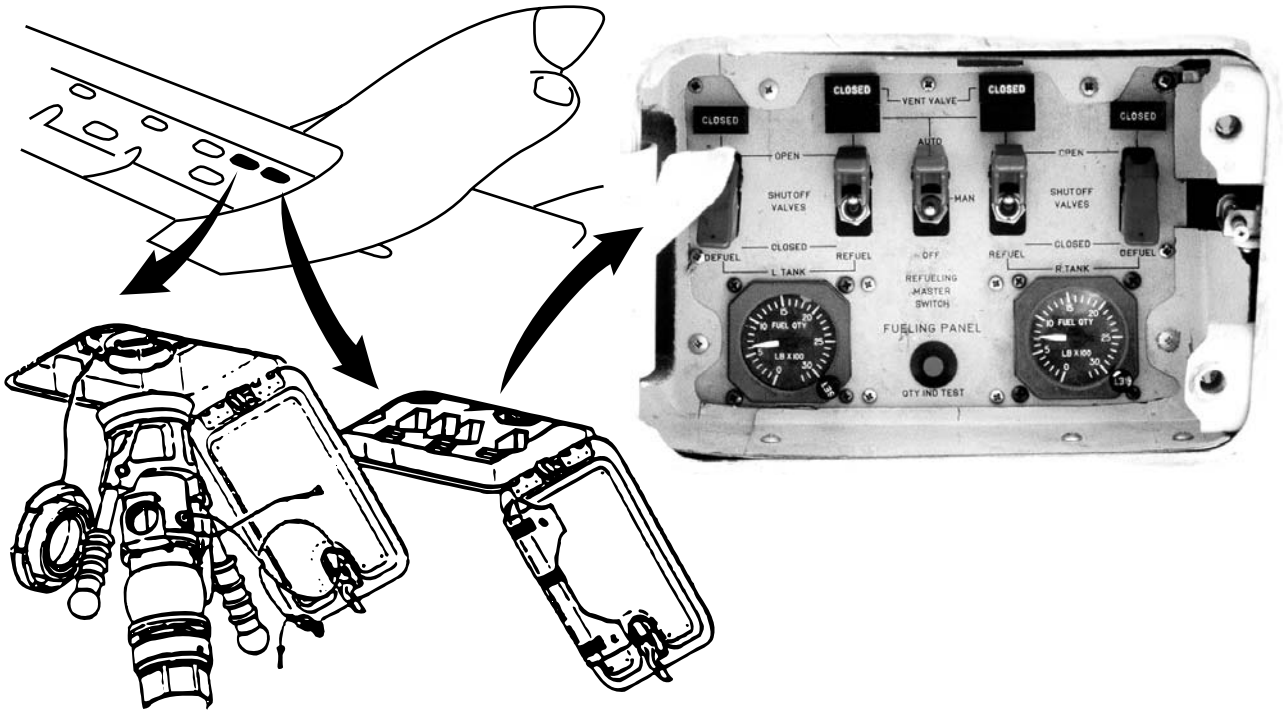


Figure 5-12. Fueling Panel

The selected electric boost pump supplies fuel to its engine and, through the crossfeed valve, to the opposite engine. Because motive flow is not available to operate the transfer jet pumps during crossfeed operation, the usable fuel in each tank is reduced by 79.6 pounds (12.15 U.S. gallons).

FUEL SERVICING

The refueling system installed in the EMB-120 permits refueling of the aircraft by either gravity or pressure methods. The refueling system may also be used to gravity or pressure defuel the aircraft if required for maintenance, fuel contamination, or overfueling.

REFUELING

Although a gravity refueling capability is provided, the primary means of refueling the aircraft is single-point pressure refueling. Pressure refueling of both the left and right fuel system is controlled by a fueling panel located on the underside of the right wing (Figure 5-12).

Gravity Refueling

Gravity refueling is accomplished through a standard fuel cap located on the top of each wing near the wing tip. These caps provide access to the outboard tanks only.

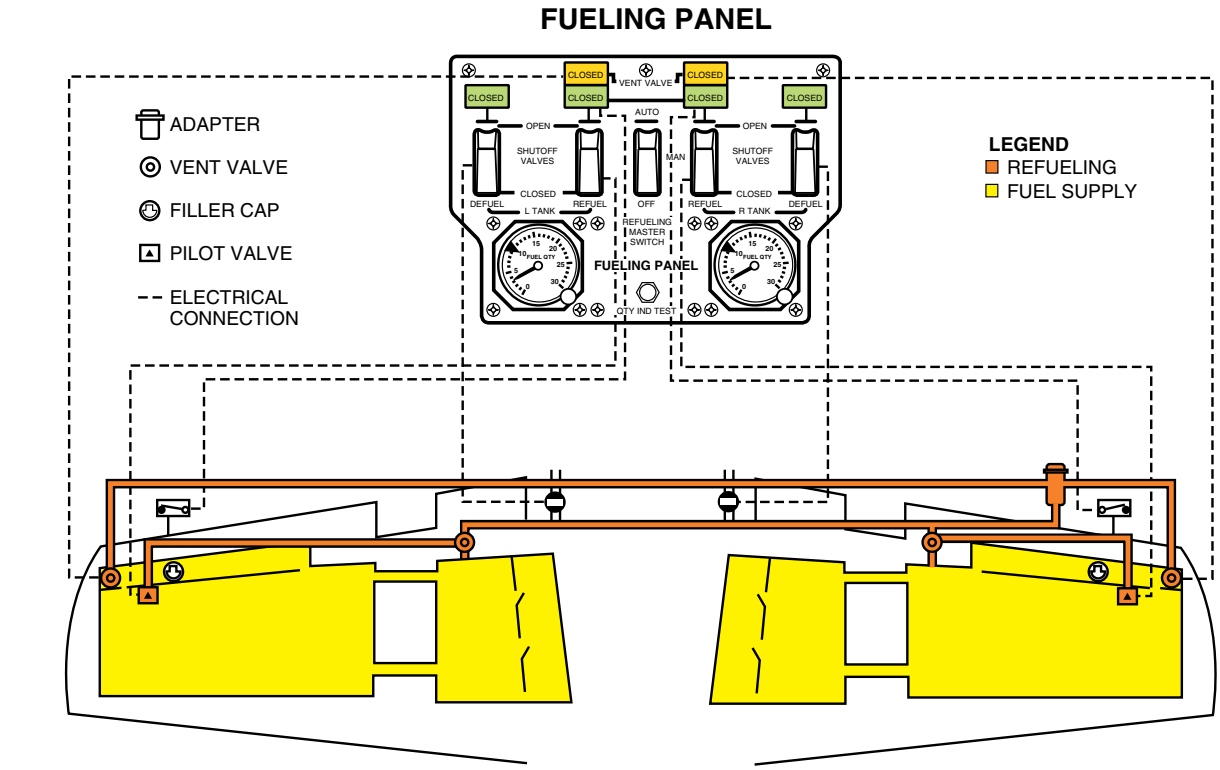


Figure 5-13. Refueling System

Pressure Refueling System

The single-point pressure refueling system (Figure 5-13) provides a rapid means of refueling the airplane. The system has two refueling modes, automatic and manual, and consists of the following components:

- Fueling panel
- Refueling adapter
- Refueling vent valves
- Pilot valves
- Refueling shutoff valves

Fueling Panel

The fueling panel, located on the underside of the right wing, is used to control the pressure refueling and defueling of the airplane.

Refueling Adapter

The refueling adapter (Figure 5-14) located next to the refueling panel, is the connector between a fuel supply (e.g., fuel truck) and the airplane refueling system. It is a poppet-type valve normally kept closed by a spring.

The poppet valve in the adapter opens when a refueling nozzle is connected and its shutoff lever is opened. The adapter also has a port connecting it to the refueling vent valves.

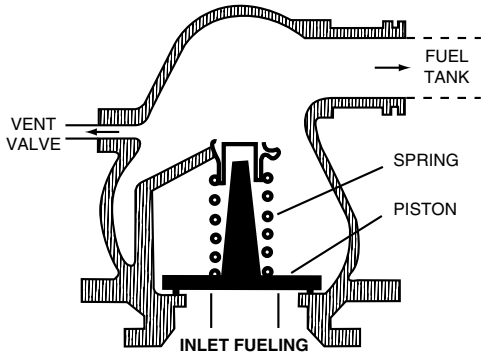


Figure 5-14. Refueling Adapter

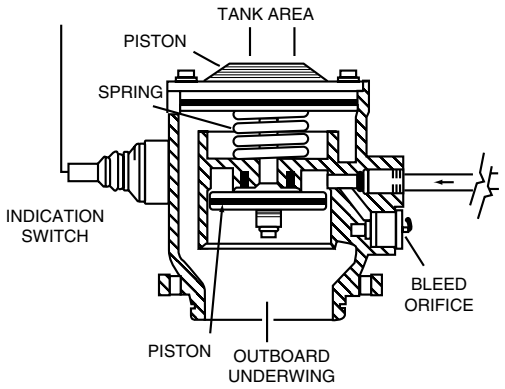


Figure 5-15. Refueling Vent Valve

Refueling Vent Valves

A refueling vent valve (Figure 5-15), is located in each outboard tank to prevent structural damage to the wing in the event of overpressurization during refueling. Because pressure is vented to the atmosphere, the valve incorporates a screen flame arrestor.

The valve has a poppet outlet, a position indicating switch, and a bleed port.

The vent valve is opened by fuel pressure from the refueling adapter. When the valve reaches the open position, the position indicating switch deenergizes the vent valve CLOSED light on the refueling panel.

The position indicating switch is also electrically connected to the respective pilot valve solenoid to prevent fuel from entering the tank in the event the refueling vent valve does not open.

When refueling is complete, the fuel pressure from the refueling adapter is bled off through the vent valve bleed port. The refueling vent valve closes, and the position indicating switch energizes the CLOSED light.

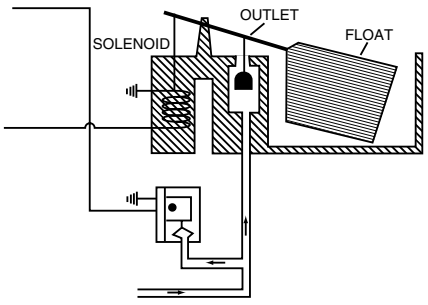


Figure 5-16. Pilot Valve

Pilot Valve

A pilot valve (Figure 5-16) is located in each outboard tank next to the refueling vent valve.

The pilot valve is a float-operated or solenoid-operated valve that controls the refueling shutoff valve. It closes the refueling shutoff valve by either:

1. Float position when the fuel reaches its maximum level;

or, during automatic operation:

2. Solenoid action at the level selected by the bugs on the fueling panel fuel quantity indicators. (The solenoid raises the float assembly when it is energized, causing the valve to simulate a full fuel tank.)

Refueling Shutoff Valve

The refueling shutoff valve (Figure 5-17) is an in-line, hydraulically controlled and actuated valve that controls the flow of fuel into the tank.

There is a refueling shutoff valve in each wing. The left wing shutoff valve is in the left inboard tank, and the right wing shutoff valve is in the right outboard tank.

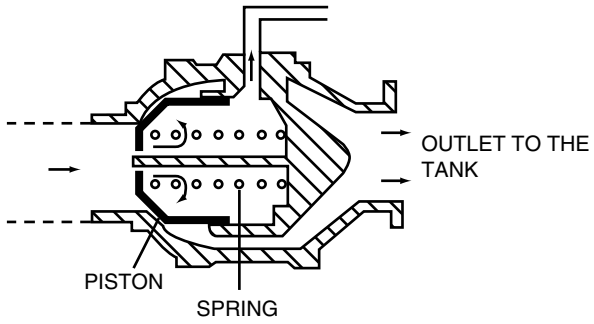


Figure 5-17. Refueling Shutoff Valve

The refueling shutoff valve consists of a piston operated by differential hydraulic pressure, normally held closed by a spring. The inside of the piston forms a chamber that is connected to the pilot valve.

A pressure switch is installed in the line between the piston chamber and the pilot valve. When pressure in the line reaches approximately 25 psi, the switch actuates, illuminating the appropriate CLOSED light on the refueling panel which indicates that the refueling shutoff valve is closed.

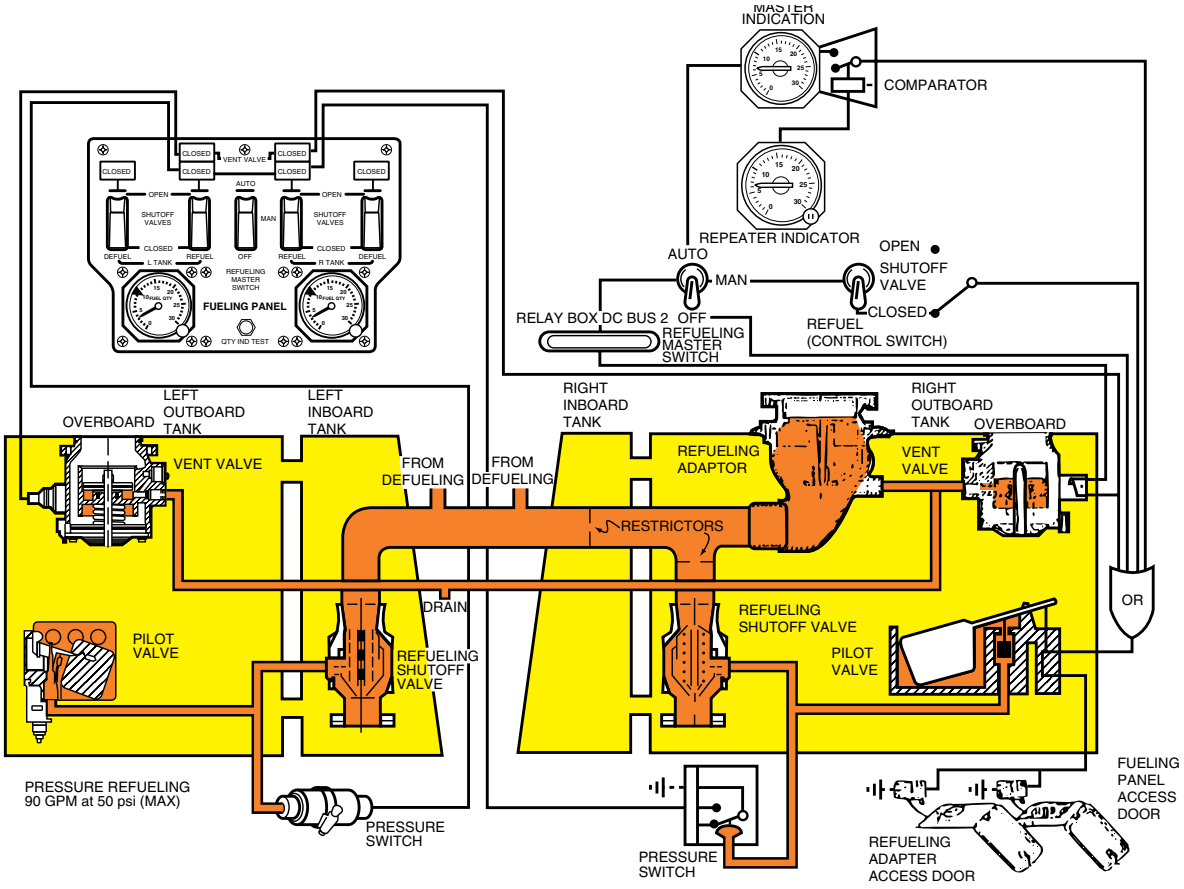


Figure 5-18. Refueling System Operation

Pressure Refueling Operation

The aircraft may be pressure refueled with or without electrical power. However, to prevent overpressurization and possible structural damage to the wing, pressure refueling should be conducted with electrical power on the aircraft. (When the refueling system is powered, a refueling valve is prevented from opening if the respective refueling vent valve does not open).

If unusual circumstances require refueling without electrical power, one of the following methods should be used:

- Overwing gravity refueling.
- Single point pressure refueling ***with the gravity refueling caps removed***.

When the refueling system is electrically powered, it may be operated in either the AUTO or MAN mode.

Opening either the refueling panel or refueling adapter access door energizes the solenoid in the pilot valve causing the float assembly to simulate a full tank.

At this time all indicating lights on the fueling panel, except the refueling shutoff valve lights, are illuminated.

Automatic (AUTO) Mode

To configure the refueling system for AUTO mode refueling:

- Connect fuel hose.
- Place refueling master switch in the AUTO position.
- Set bugs on the fuel quantity indicators to the desired fuel level.

When the refueling nozzle is opened, the refueling system functions as follows (Figure 5-18):

1. Fuel pressure from the refueling adapter is transmitted to the refueling shutoff valves, pilot valves, and refueling vent valves. No fuel flows into the tanks at this time.

2. In the shutoff valve, fuel pressure is transmitted through an orifice in the piston, to the pilot valve through a connecting line.

3. In the pilot valve, the float is being held in the "full" position by the solenoid, (energized when the access doors were opened).

4. A back-pressure is created in the line between the shutoff valve and the pilot valve.

   a. When this back-pressure reaches approximately 25 psi the pressure switch closes, illuminating the shutoff valve CLOSED light on the fueling panel.

   b. In the shutoff valve, this backpressure results in a force on the rear of the piston (larger area) greater than the force on the face of the piston (smaller area). This imbalance in forces, aided by a spring, holds the piston in the closed position.

5. Once the refueling vent valve is opened by fuel pressure from the refueling adapter, its position indicating switch deenergizes the vent valve CLOSED light and the pilot valve solenoid.

6. When the pilot valve solenoid deenergizes the float drops off of the "full" position and back-pressure in the pilot valve dissipates. As a result:

   a. The shutoff valve CLOSED light goes out.

   b. Fuel pressure overcomes spring pressure in the shutoff valve, displacing the piston thus opening the valve.

7. Fuel flows into the tank until the fuel quantity indicators reach the preselected (bug) quantity or the tanks are full. When either of these conditions are met, the refueling operation is automatically stopped.

Manual (MAN) Mode

In the manual mode the components of the refueling system have the same function as in the automatic mode. The only difference is that the refueling master switch is placed in the MAN position and the refueling shutoff valves have to be selected to the OPEN position.

In the MAN mode, the vent valve signal to the pilot valve solenoid (holding the float in the "full" position) will not be deenergized until the shutoff valve switch is placed to OPEN, regardless of vent valve position.

During manual mode refueling, fuel flow stops when any of the following occurs:

- The tanks are full.
- The refueling shutoff valves are selected CLOSED.
- The refueling master switch is selected OFF.

DEFUELING

The defueling system (Figure 5-19) provides a means to pressure or gravity defuel the airplane if required. Normally, the wing tanks are pressure defueled first and then gravity defueled to remove the last of the fuel.

Gravity Defueling

Gravity defueling is accomplished by connecting a special defueling adapter, with a manual shutoff valve, to the manual defueling valves and draining the fuel into a suitable container. The manual defueling valves are located on the underside of each wing in the inboard tanks.

Pressure Defueling System

Pressure defueling is controlled from the fueling panel through the defueling shutoff valves. To defuel the airplane, a standard fuel hose is attached to the fueling adapter and the aircraft's electric boost pumps are used to provide pressure. Suction (maximum 0.5 psi) from ground equipment may also be used for defueling.

Defueling Shutoff Valves

The defueling shutoff valves connect the crossfeed line to the refueling line. They are identical to the engine fuel shutoff valves.

The defueling shutoff valves are powered by DC bus 2 and are controlled by the corresponding defueling switch on the fueling panel. It is not necessary to have the refueling panel master switch on to operate the defueling valves.

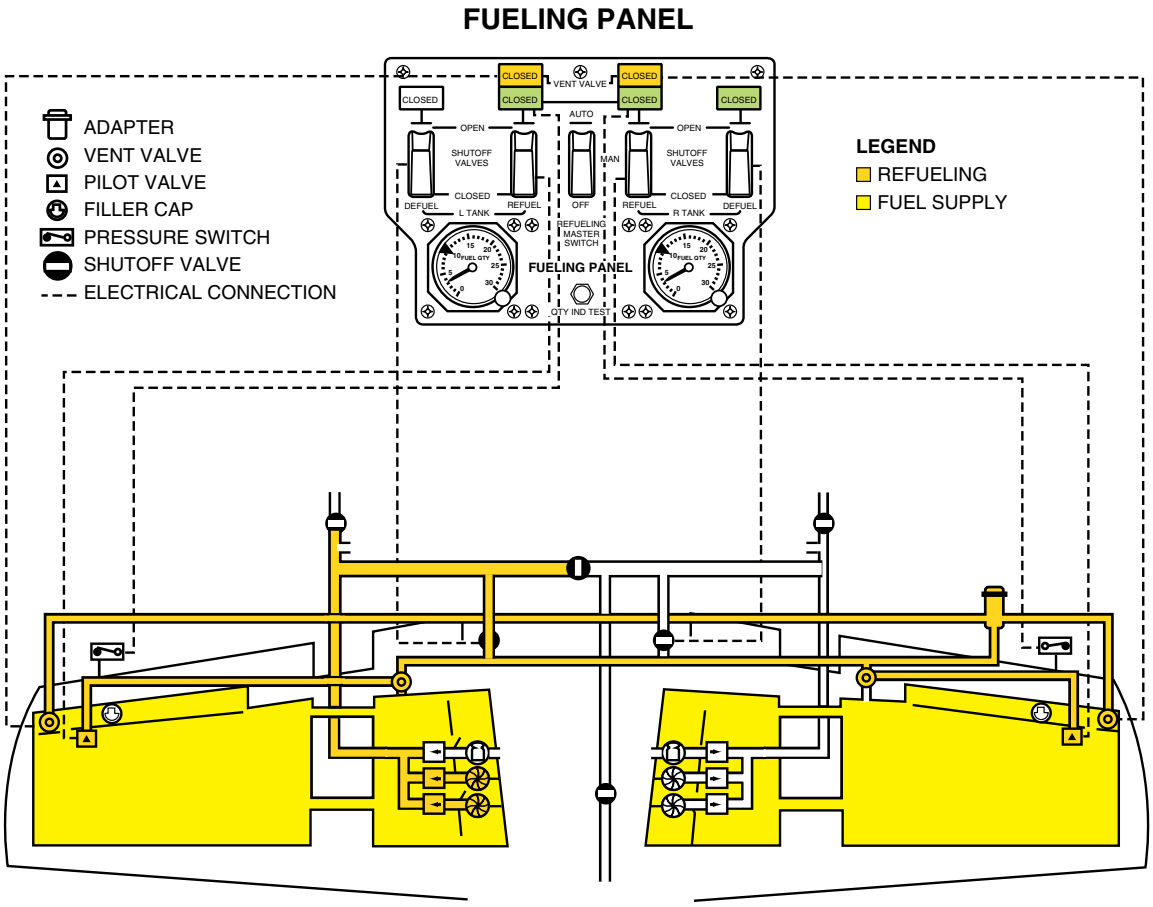


Figure 5-19. Defueling System

The defueling shutoff valve opens when the respective defueling switch is moved to the OPEN position. The green CLOSED defueling shutoff valve indicating light remains illuminated until the valve has reached the fully open position.

Pressure Defueling System Operation

Electrical power, provided by the airplane electrical system or a ground power unit, is required to pressure defuel the airplane. A hose with a standard refueling nozzle is attached to the airplane refueling adapter.

The defueling shutoff valves are opened by the defueling switches on the refueling panel while the boost pumps are actuated by their respective switches in the cockpit. The fuel flow starts at the collector tank, goes through the selected boost pump, through the check valves, into the main fuel lines. From the main fuel lines, the fuel travels through the crossfeed line to the corresponding defueling shutoff valve. From the shutoff valves, the fuel travels through the refueling shutoff valve line to the refueling adapter.

The master fuel quantity indicators in the cockpit or the repeater fuel quantity indicators at the fueling panel can be used to determine the fuel quantity in the fuel tanks. When the desired fuel quantity is reached, the boost pumps are switched OFF. The defuel shutoff valves are switched CLOSED and the fuel line is disconnected.

NOTE

The refueling vent valves are not open during the defueling operation.

In the event of accidental asymmetrical defueling or refueling, it is possible, on the ground, to transfer fuel from one tank ***into*** the other tank by defueling the high side and refueling the low side. No refueling hose is required.

The aircraft boost pumps are used to transfer the fuel.

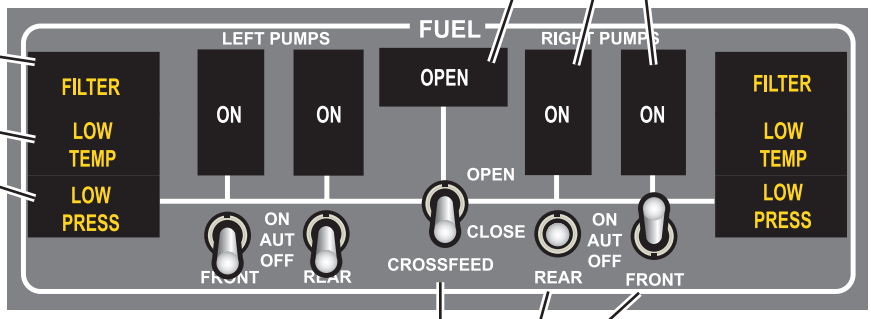
ENG NO.1 FILTER LIGHT (AMBER)
ILLUMINATED—PRESSURE DIFFERENTIAL HAS REACHED 1.5 PSID ACROSS THE LOW-PRESSURE FILTER, OR 50 PSID ACROSS THE HIGH-PRESSURE FILTER. THE FILTERS BYPASS AT 3 PSID AND 50 PSID, RESPECTIVELY. TRIGGERS AMBER FUEL LIGHT ON MAP.

ENG NO.1 LOW TEMP LIGHT (AMBER)
ILLUMINATED—INDICATES LOW FUEL TEMPERATURE. TRIGGERS AMBER FUEL LIGHT ON MAP.

ENG NO.1 LOW PRESS LIGHT (AMBER)
ILLUMINATED—INDICATES LOW FUEL PRESSURE. TRIGGERS AMBER FUEL LIGHT ON MAP.

CROSSFEED OPEN LIGHT (WHITE)
ILLUMINATED—WHEN CROSSFEED VALVE REACHES FULLY OPEN POSITION. REDUNDANT WITH WHITE CROSSFEED OPEN LIGHT ON FUEL MANAGEMENT PANEL.

ENG NO.2 PUMP ON LIGHTS (WHITE)
ILLUMINATED—RELEVANT PUMP IS ON.



NOTE:
THERE ARE NO LOW FUEL LIGHTS OR WARNINGS

ENG NO. 2 FRONT AND REAR ELECTRIC FUEL BOOST PUMP SWITCHES
ON—TURNS ON RELEVANT ELECTRIC PUMP.
AUT—PUMP AUTOMATICALLY TURNS ON IF FUEL PRESSURE IS LOW. PUMP AND RELEVANT LIGHT WILL CYCLE ON AND OFF. POSITION SWITCH TO ON.
OFF—TURNS OFF RELEVANT ELECTRIC PUMP.

CROSSFEED SWITCH
OPEN—OPENS CROSSFEED VALVE AND CLOSES MOTIVE FLOW VALVES.
CLOSE—CLOSES CROSSFEED VALVE AND OPENS MOTIVE FLOW VALVES.

Figure 5-20. Overhead Fuel Panel

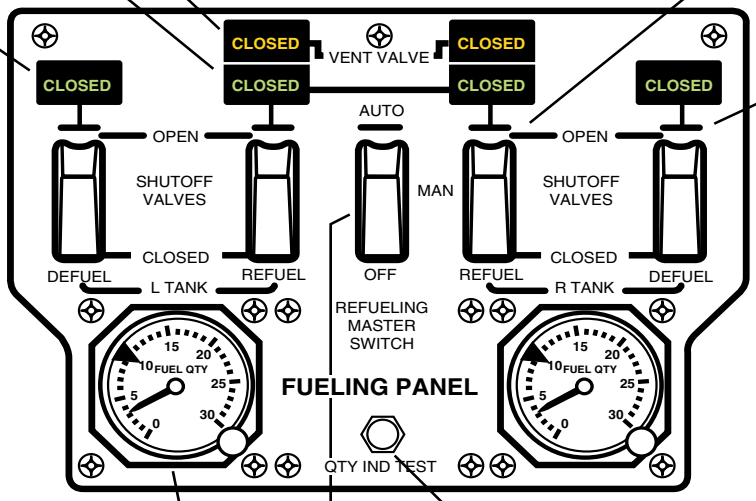
LEFT TANK DEFUELING CLOSED LIGHT (GREEN)
ILLUMINATED – INDICATES DEFUELING SHUTOFF VALVE IS CLOSED.

LEFT TANK REFUELING CLOSED LIGHT (GREEN)
ILLUMINATED – INDICATES REFUELING SHUTOFF VALVE IS CLOSED.

RIGHT TANK DEFUELING SWITCH
OPEN – OPENS RIGHT TANK DEFUELING SHUTOFF VALVE, ALLOWING RIGHT TANK DEFUELING.
CLOSED – CLOSES RIGHT TANK DEFUELING VALVE.

RIGHT TANK REFUELING SWITCH
OPEN – OPENS RIGHT TANK REFUELING SHUTOFF VALVE, ALLOWING RIGHT TANK REFUELING.
CLOSED – CLOSES RIGHT TANK REFUELING SHUTOFF VALVE.

LEFT TANK VENT VALVE CLOSED LIGHT (AMBER)
ILLUMINATED – INDICATES VENT VALVE IS CLOSED, INHIBITING REFUELING OF LEFT WING.



LEFT TANK FUEL QUANTITY INDICATOR
INDICATES FUEL QUANTITY IN TANK. THE DESIRED FUEL QUANTITY FOR AUTOMATIC REFUELING IS SELECTED THROUGH A BUG MOVED BY THE SET KNOB.

QUANTITY INDICATOR TEST BUTTON
PRESSED – RIGHT AND LEFT QUANTITY INDICATOR (AND COCKPIT QUANTITY INDICATORS) POINTERS ARE DRIVEN TO THE END OF THE SCALE.

REFUELING MASTER SWITCH
AUTO – AUTOMATIC OPERATION. BUG ON THE FUEL QUANTITY INDICATOR MUST BE SET TO THE DESIRED QUANTITY. UPON REACHING BUG SETTING, REFUELING OPERATION IS AUTOMATICALLY SHUT OFF.
MAN – MANUAL OPERATION. PLACE REFUELING SHUTOFF SWITCHES TO OPEN. FUEL QUANTITY IS MONITORED ON QUANTITY INDICATORS. WHEN DESIRED QUANTITY IS REACHED, REFUELING IS STOPPED BY RETURNING SWITCHES TO THE CLOSED POSITION.

Figure 5-21. Underwing Fueling Panel

QUESTIONS

1. The total usable fuel capacity of the EMB-120 is:

A. 5,525 pounds
B. 5,732 pounds
C. 5,777 pounds
D. 6,223 pounds

2. The EMB-120 fuel system consists of:

A. Four electric boost pumps
B. Four scavenger jet pumps
C. Two main jet pumps
D. All of the above

3. The firewall shutoff valve is operated through the:

A. Firewall shutoff valve limit switch
B. Fuel control unit
C. Fire extinguisher T-handle
D. Copilot's fuel monitoring switch

4. Crossfeed operation is indicated by:

A. A red light
B. A green light
C. An amber light
D. A white light

5. When pressure refueling, the airplane must be electrically powered.

A. True
B. False

6. APU fuel used goes through the fuel totalizer.

A. True
B. False

7. In case of a power loss to the fuel flow indicators:

A. The digital indication goes blank.
B. The digital indication reads LOW.
C. The digital indication reads HIGH.
D. The digital indication reads OFF.

CHAPTER 6
AUXILIARY POWER UNIT

CONTENTS

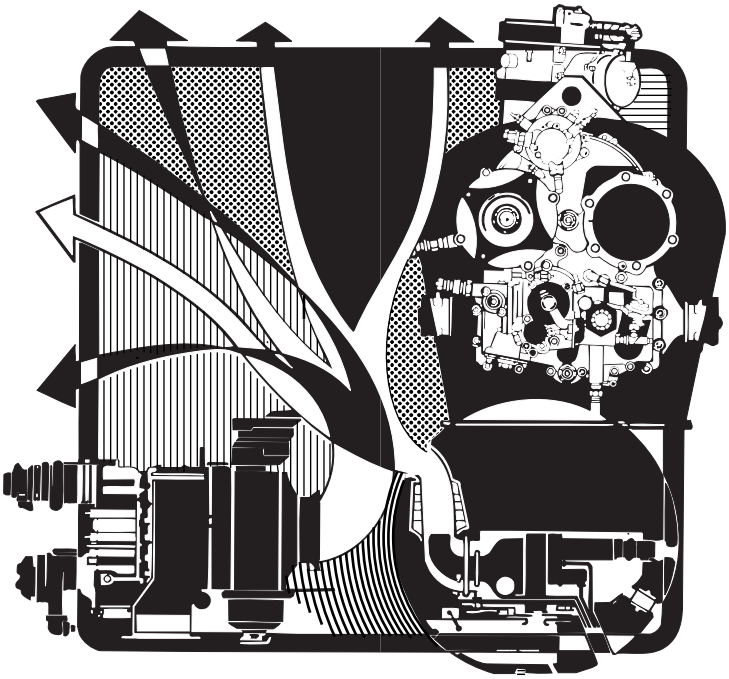
Page

INTRODUCTION 6-1
GENERAL 6-1
MAJOR SECTIONS 6-2
Compressor Section 6-2
Power Section 6-3
Accessory Section 6-3
APU SYSTEMS 6-4
Electrical System 6-4
Fuel System 6-4
Lubrication System 6-5
Pneumatic System 6-5
Automatic Shutdown 6-6
APU CONTROL AND MONITORING 6-6
APU Panel 6-6
Fuel Totalizer 6-9
OPERATION 6-9
Operation Notes 6-10
QUESTIONS 6-12

ILLUSTRATIONS

Figure	Title	Page
6-1	APU Layout	6-2
6-2	Major Sections	6-3
6-3	APU Installation (External)	6-5
6-4	APU Panel	6-6
6-5	Fuel Totalizer	6-9
6-6	APU Panel	6-11

CHAPTER 6
AUXILIARY POWER UNIT



INTRODUCTION

The EMB-120 Brasilia has an APU manufactured by either Garrett Turbine Engine Company (AiResearch) or Hamilton Sundstrand (Sundstrand Turbomach). The APU drives a DC generator, capable of powering the airplane's electrical system, and provides bleed air for air conditioning and pressurization.

GENERAL

The auxiliary power unit (APU) models GTCP 36-150(A) and (AA) are 25 SHP gas turbine engines consisting of a single-stage centrifugal compressor, a reverse flow annular combustor, and a single-stage radial turbine. This chapter refers to the Garrett GTCP 36-150(AA) APU.

The APU is a source of both electric and pneumatic power and may be operated in conjunction with, or independent from, the engines. It may be started and operated both on the ground and in flight.

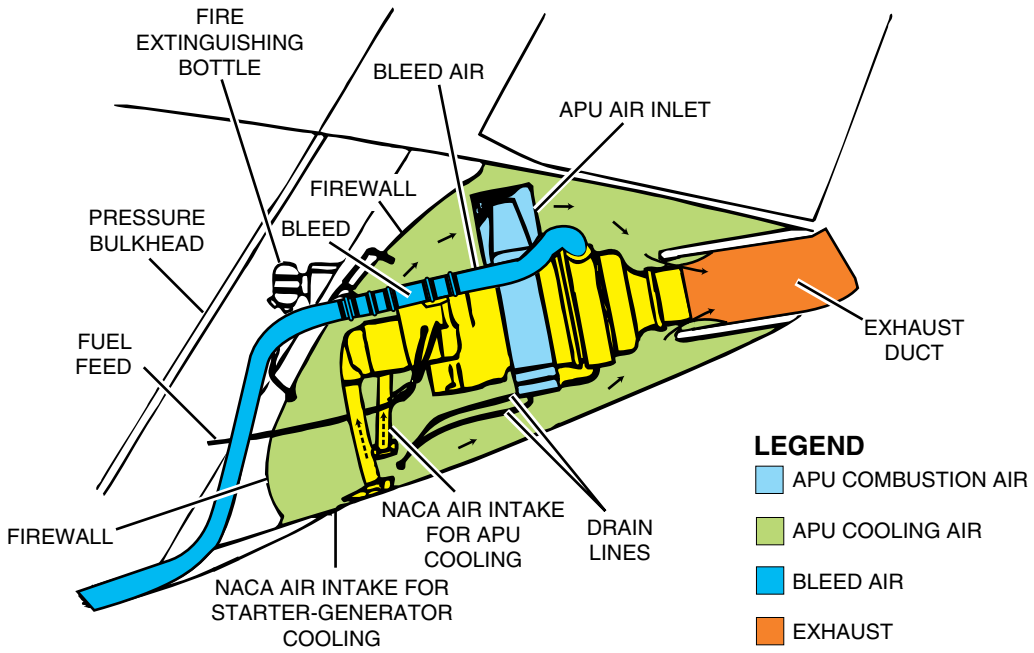


Figure 6-1. APU Layout

The APU compartment is in the aircraft tail cone, isolated by a stainless steel firewall (Figure 6-1). An inspection door on the left side of the compartment provides access to the APU components.

In flight, the APU bay is cooled by airflow entering a NACA inlet on the lower right side of the tail cone. On the ground, a fan mounted on the APU starter-generator induces the required cooling airflow. The flush-mounted APU combustion air intake is located on the upper right side of the tail cone.

An hourmeter on the APU crankcase registers total running time; and a maintenance panel, in the compartment between the aft pressure bulkhead and the APU compartment, indicates the cause of an automatic shutdown.

Controls and monitoring indicators for APU operation, fire detection, and extinguishing are on an APU control panel on the cockpit overhead panel.

MAJOR SECTIONS

For descriptive purposes, the APU is divided into three major sections (Figure 6-2):

- Compressor section
- Power section
- Accessory section

COMPRESSOR SECTION

The compressor section includes an air intake assembly, a single-stage centrifugal compressor rotor, and a two-stage diffuser.

The compressor section compresses and directs ambient air to the power section for combustion. It also provides bleed air for operating aircraft pneumatic systems.

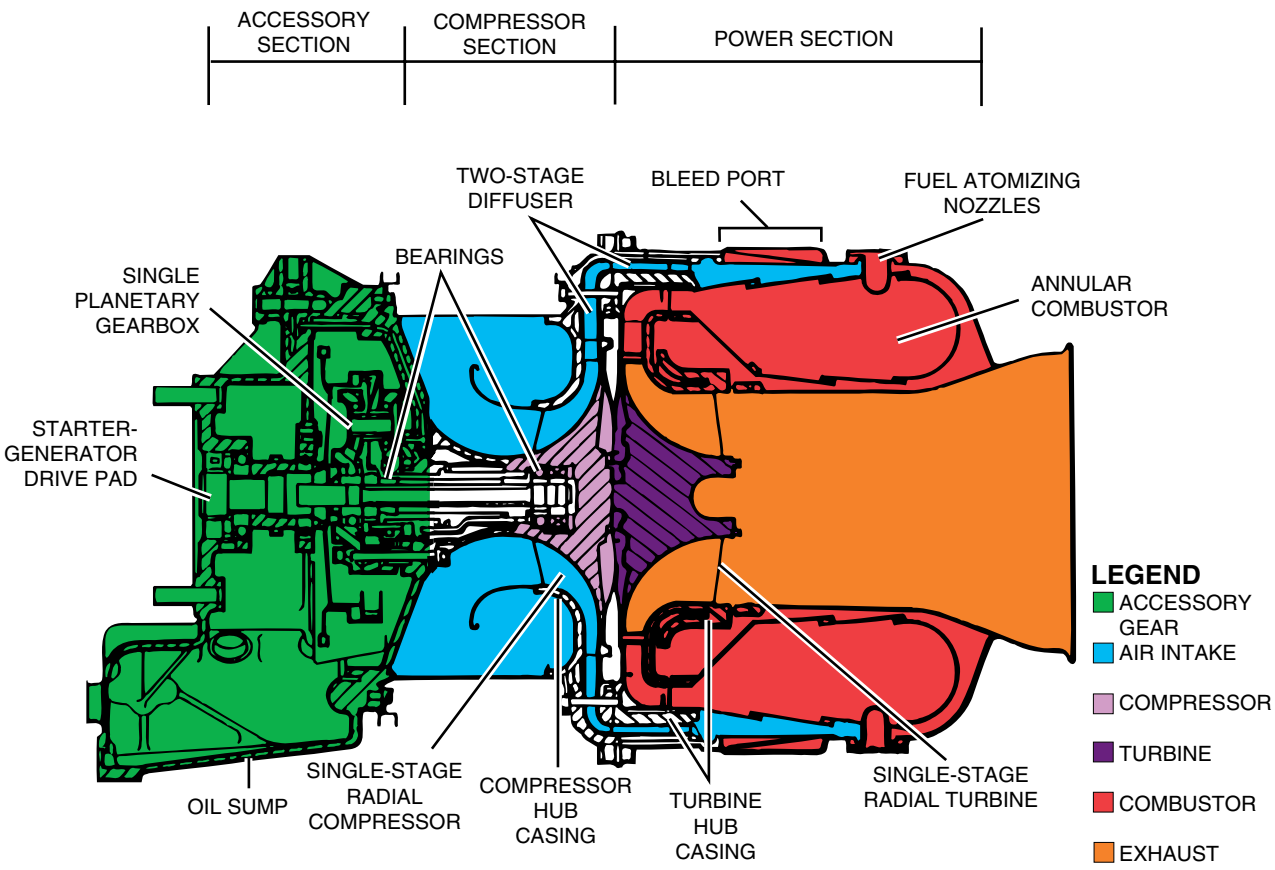


Figure 6-2. Major Sections

POWER SECTION

The power section combines the compressed air with a fuel mixture and converts them into shaft power.

The power section contains a deswirl assembly (stators), a turbine containment assembly, a reverse flow annular-type combustion chamber, and a single-stage radial turbine. The combustion chamber includes the spark igniter plug, fuel nozzle assembly, and EGT sensor.

Compressed air coming from the compressor section passes through the deswirl deflector and enters the turbine containment assembly. The air then flows through the combustion chamber where fuel is injected and combustion occurs. The hot gases produced in the combustion chamber pass from the turbine nozzle to the single-stage radial turbine impelling rotation and driving the main drive shaft. The main shaft provides power for the accessory gearbox.

ACCESSORY SECTION

The accessory section contains the accessory gearbox, fuel control unit (FCU), overspeed sensor, and the low oil pressure and high oil temperature switches.

The gearbox reduces the APU's main shaft speed to drive the starter-generator and the oil pump.

The APU oil system is self-contained and integrated with the gearbox.

APU SYSTEMS

The APU systems consist of the following:

- Electrical
- Fuel
- Lubrication
- Pneumatic
- Automatic shutdown

ELECTRICAL SYSTEM

The APU's electrical system includes the following subassemblies:

- Electronic control unit (ECU)
- Starter-generator
- Ignition unit

Electronic Control Unit (ECU)

The ECU governs the APU. It controls the starting sequence, acceleration, governed speed operation, temperature limits and shutdown. It incorporates the fuel management control logic and the load control valve logic.

The ECU receives input signals from:

- High oil pressure switch
- High oil temperature switch
- Starter-generator
- Master control switch
- Auxiliary shutdown switch
- Overspeed sensor
- Exhaust gas temperature (EGT) sensor
- Load control valve actuating switch

The ECU provides output signals to:

- Fuel control unit (FCU)
- Fuel solenoid valve
- Ignition unit
- Starting relay
- Hourmeter
- Maintenance panel

Starter-Generator

The APU starter-generator is physically and functionally identical to the engine starter-generator.

During the APU start sequence, the starter-generator functions as a starter and drives the compressor/turbine. When the starting cycle is over, the starter-generator operates as an APU-driven generator.

The APU starter-generator is attached to the accessory gearbox on the forward section of the APU.

Ignition Unit

The ignition unit is used to develop the output voltage to energize the igniter plug. The unit is a capacitive discharge, energy storage system. It is powered by 28 VDC from the airplane's electrical system and emits pulsing high voltage (18 to 24 KV) to the igniter.

FUEL SYSTEM

The APU fuel system provides pressurized, metered fuel to the combustion chamber. The system includes a fuel control unit (FCU), fuel flow dividers, and fuel nozzles. It provides fuel in accordance with the preprogrammed schedule in the FCU.

The FCU pressurizes and meters fuel going to the fuel flow dividers. The fuel flow dividers distribute the fuel from the FCU to six fuel nozzles; the nozzles atomize and inject the fuel into the combustion chamber in a swirl pattern.

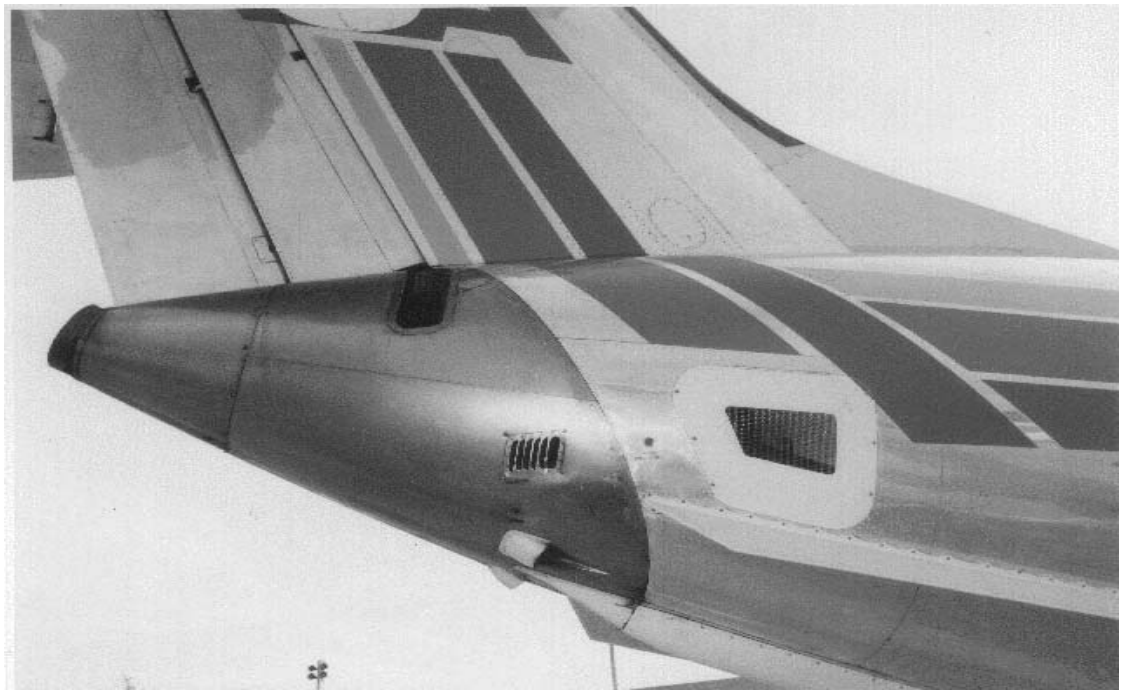


Figure 6-3. APU Installation (External)

The APU fuel is normally supplied by the right aircraft fuel tank. Fuel in the left tank may be used when the crossfeed valve is open.

With the right engine running, motive flow provides pressurized fuel to the APU for starting and operation. With only the left engine operating, an electric boost pump is required for fuel pressure.

LUBRICATION SYSTEM

The APU has a self-contained lubrication system totally integrated with the accessory gearbox. The system is designed to function without the need for an external heat exchanger. If the oil temperature exceeds 325°F (163°C), a thermostat installed on the oil tank sends a signal to illuminate the amber HIGH TEMP light on the APU CONTROL panel.

The APU utilizes an oil sump and an oil pump. The 1.89L (2 U.S. quarts) oil sump stores the oil required for the APU. It is integral with the bottom portion of the accessory gearbox and has a filler cap and dipstick on the left side. A draining point is assembled with a magnetic drain plug for chip detection.

The oil pump, driven by the accessory gearbox, distributes oil through the APU.

PNEUMATIC SYSTEM

The APU pneumatic system is subdivided into cooling and bleed-air systems. The cooling subsystem provides cooling for the APU, and the bleed-air subsystem provides compressed air to the aircraft bleed-air systems.

The APU compartment is cooled by air entering through an air scoop located on the lower right side of the tail cone (Figure 6-3). Air for engine operation enters through a screened intake on the upper right side of the tail cone (Figure 6-3). Additionally, the APU starter-generator is cooled by a fan, which rotates with the starter-generator, and is ducted to an air scoop on the lower left side of the tail cone (Figure 6-3).

The APU bleed-air system provides compressed air for the aircraft pneumatic systems. The bleed-air system operates automatically, controlling bleed air from the APU compressor section through a load control valve.

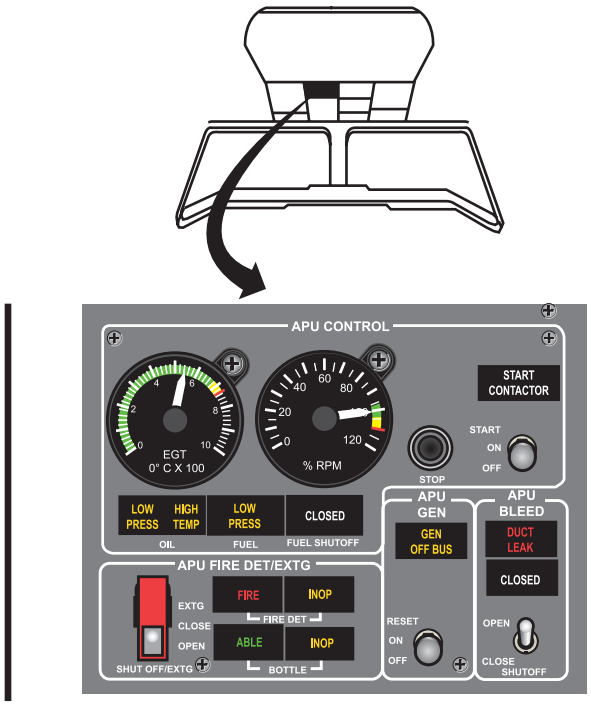


Figure 6-4. APU Panel

AUTOMATIC SHUTDOWN

The APU automatic shutdown is controlled by the ECU on the ground or in flight. On the ground, the APU is automatically shut down for any of the following:

- APU fire—a function of the fire bell ringing for 10 seconds or longer. If alarm is canceled before 10 seconds elapses, auto shutdown will not occur.
- Low oil pressure—below 31 psi, above 95% rpm for 10 seconds
- High oil temperature—from 141° to 147°C
- Overcurrent of components for 3 seconds
- Overvoltage—32 volts
- Overspeed—110% rpm for 1/2 second
- Loss of speed sensor—loss of rpm signal
- Over temperature during start—EGT exceeds 732°C for 1/2 second above 90% rpm.
- Loss of EGT sensor
- Loss of ECU 28-VDC power

In flight automatic shutdown occurs only for:

- Overspeed
- Overvoltage

APU CONTROL AND MONITORING

APU PANEL

The APU controls and indicators are located on an APU panel on the cockpit overhead panel (Figure 6-4). The panel, which provides for monitoring and control of the auxiliary power unit's operation, is divided into four sections:

- APU control
- APU bleed
- APU generator
- APU fire detection/extinguishing

APU CONTROL Section

The APU CONTROL section (Figure 6-4) incorporates the following switches and indicators:

- APU master switch
- APU stop button
- Start contactor light
- Low oil pressure light
- High oil temperature light

- Low fuel pressure light
- APU fuel closed shutoff valve lithe
- APU rpm indicator
- Exhaust gas temperature (EGT) indicator

Master Switch—The APU master switch is a three-position (START/ON/OFF) switch. Switch position functions are as follows:

- START— Initiates start cycle, (momentary position).
- ON—Applies power to the ECU, opens the fuel shutoff valve, and keeps the APU running during normal operation.
- OFF—Commands the APU shutdown. Deenergizes the ECU, closes the fuel shutoff valve, and deenergizes rpm and EGT indicators and indication lights.

  The white bleed-air and fuel shutoff CLOSED lights remain illuminated on the APU control panel for ten seconds after the APU master switch is selected to off.

STOP Button—The APU stop button shuts down the APU by sending an electronic 110% rpm overspeed signal which simultaneously checks the ECU fault protection circuit. It is normally used to shut down the APU so that rpm and EGT indications may be monitored during spool down.

START CONTACTOR Light—When the starting cycle is initiated, the white START CONTACTOR light illuminates indicating the start relay is energized. At 50% rpm, the ECU disengages the starter and extinguishes the light.

Oil LOW PRESS Light—If oil pressure drops below 31 psi for over 10 seconds with the APU operating above 95% rpm, the amber LOW PRESS light illuminates. Also, the amber APU light on the MAP illuminates, triggering a single chime aural alert, and the master CAUTION lights flash.

Fuel LOW PRESS Light—When the fuel pressure in the APU falls below 9 psi, a pressure switch installed in the fuel supply line illuminates the amber fuel LOW PRESS light. Also, the amber APU light on the MAP illuminates, triggering a single chime aural alert, and the master CAUTION lights flash.

The APU will not start with the APU fuel LOW PRESS light illuminated.

Fuel Shutoff CLOSED Light—The white fuel shutoff CLOSED light illuminates when the APU fuel shutoff valve, located in the right wing root area, is fully closed.

rpm Indicator—The APU utilizes a magnetic sensor in the reduction gearbox to sense APU rpm. The sensor provides a signal to the ECU where it is processed and sent to the rpm indicator.

The indicator is powered by DC bus 2. When a power or signal loss occurs, the pointer indicates below zero on the scale.

The rpm indicator is marked as follows:

- GREEN arc, from 96 to 104% rpm, indicates the normal operating range.
- YELLOW arc, from 104 to 110% rpm, indicates the caution range of operation.
- RED radial, at 110% rpm, indicates the maximum permitted rpm.

Exhaust Gas Temperature (EGT) Indicator—The APU utilizes a thermocouple in the exhaust duct to sense the exhaust temperature. The thermocouple sends a signal to the ECU where it is processed and sent to the EGT indicator.

The indicator is powered by DC bus 2. When a power or signal loss occurs, the pointer indicates below zero on the scale.

The EGT indicator is marked as follows:

- GREEN arc, from 0 to 680°C, indicates the normal operating range.
- YELLOW arc, from 680 to 732°C, indicates caution range of operation.

- RED radial, at 732° C, indicates the maximum permitted EGT .

APU BLEED Section

The APU BLEED section (Figure 6-4) has the following control switch and indicators:

- APU bleed shutoff switch
- APU duct leak light
- APU bleed closed light

SHUTOFF Switch—The APU bleed SHUTOFF switch, through a torque motor and the ECU, controls the bleed shutoff valve. Switch position functions are as follows:

- OPEN—Four seconds after the APU reaches 95% rpm, the bleed valve opens allowing bleed air to enter the aircraft pneumatic system.
- CLOSE—The bleed shutoff valve closes and the white APU bleed CLOSED light illuminates.

  The bleed shutoff valve also closes if the APU rpm drops below 95% for an EGT overtemperature, or when the APU is supplying the same pack as an engine.

Bleed CLOSED Light—The white APU bleed CLOSED light illuminates when the APU bleed shutoff valve is closed.

DUCT LEAK Light—When a bleed-air leak occurs and the temperature exceeds 71°C (160°F), temperature sensors located along the lines send a signal illuminating the red DUCT LEAK light. Also, the red DUCT LEAK light on the MAP flashes, the master WARNING lights flash, and a voice message "DUCT LEAK" is heard.

APU GEN Section

The APU is equipped with a starter-generator identical to the one used for the engines. Generator operation is covered in Chapter 2, Electrical Power Systems.

The APU GEN section (Figure 6-4) consists of the following:

- APU generator switch
- GEN OFF BUS light.

Generator Switch—The APU generator switch is a three-position switch with the following functions:

- RESET—Electrically resets the generator, (momentarily position).
- ON—Connects the APU generator to the central DC bus.
- OFF—Disconnects the APU generator from the central DC bus.

GEN OFF BUS Light—The amber GEN OFF BUS light illuminates when the APU generator is disconnected from the central DC bus.

APU FIRE DET/EXTG Section

The APU FIRE DET/EXTG section (Figure 6-4) contains the following switches and indicators:

- APU shutoff/extinguishing switch
- APU fire warning light
- APU fire detector inoperative light
- APU fire extinguishing bottle integrity light
- APU fire extinguishing bottle pressure condition light

SHUTOFF/EXTG Switch—The SHUTOFF/EXTG switch is a guarded, three-position switch, with the following positions:

- EXTNG—Discharges the fire extinguishing agent into the APU compartment.
- CLOSE—Closes the APU fuel shutoff and APU bleed valves, disconnects the APU

generator from the central DC bus, and deenergizes the APU control panel. The white APU bleed CLOSED light, fuel shutoff CLOSED light, and amber GEN OFF BUS light remain illuminated.

- OPEN—Normal operating position.

FIRE Warning Light—When a fire or overheat condition is sensed by the fire detector, the red FIRE light illuminates and remains illuminated until the fire or overheat condition is eliminated. Also, the red FIRE APU light will flash on the MAP, the master WARNING lights will flash, and a bell sound will be heard.

Fire Detector INOP Light—If at any time the gas is lost from the detector loop, the amber Fire Det INOP light illuminates. Also, the amber APU light on the MAP illuminates, triggering a single-chime aural alert, and the master CAUTION lights flash. The INOP light also illuminates during the main fire system test.

Bottle ABLE Light—The green Bottle ABLE light illuminates during the fire detection extinguishing test, when the fire extinguishing bottle explosive firing circuits are intact. This light illuminate only during the test sequence.

Bottle INOP Light — The amber Bottle INOP light illuminates when the APU fire extinguishing bottle is empty or inadequately pressurized. Also, the amber APU light on the MAP illuminates, triggering a single-chime aural alert, and the master CAUTION lights flash. The light also illuminates during the main fire system test.

The lamp TEST button on the engine fire control panel, when depressed, also illuminates all four lights in the APU FIRE DET/EXTG section.

FUEL TOTALIZER

APU fuel consumption is monitored by the main fuel system fuel totalizer (Figure 6-5). Because the APU does not utilize a fuel flow transmitter, its fuel consumption is computed from signals sent to the fuel totalizer by the ECU, bleed control switch, and generator control switch. The APU's fuel consumption is computed by the fuel totalizer using the following predetermined rates:

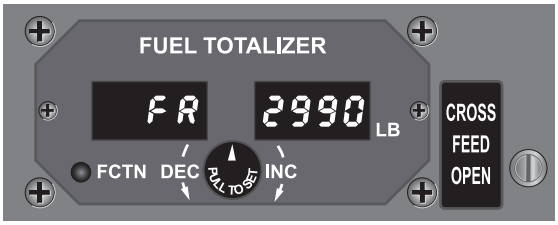


Figure 6-5. Fuel Totalizer

- APU only: 45 lb/hr
- APU and GEN: 53 lb/hr
- APU and BLEED: 90 lb/hr
- APU, GEN and BLEED: 98 lb/hr

NOTE

Fuel consumption for "A" APU's post-mod SB 120-049-0007 and "AA" APU is approximately 30% higher than the figures above.

OPERATION

The APU may be started using main battery power or external power. Recommended main battery voltage for starting is 24 volts, 22 volts is minimum. At least one electric boost pump must be turned on to pressurize the fuel line and turn out the fuel LOW PRESS light on the overhead APU panel.

The APU starting cycle is initiated when the master switch on the APU panel is placed in the START position and power from the central DC bus is applied to the starter.

The white START CONTACTOR light on the APU panel illuminates, indicating that the starter is driving the rotating components of the APU.

When the APU reaches 10% of its operating speed, the ECU energizes the ignition unit and opens the fuel solenoid shutoff valve allowing fuel to flow to the fuel nozzles in the combustion chamber.

The starter continues to accelerate the APU until the unit reaches 50% of its operating speed. The ECU then disengages the starter and extinguishes the white START CONTACTOR light.

The APU continues to accelerate until it reaches 95% rpm where the ignition circuit is deenergized. After the ignition circuit deenergizes, the ECU permits shaft power extraction through the starter-generator in the generator mode and, after 4 seconds, pneumatic power extraction through the load control valve.

Once the APU is started, the electrical power to keep it running is supplied by the APU generator regardless of the position of the electrical panel PWR SELECT switch or the APU generator switch.

The APU may be started on the ground or in flight. In flight, it may be started at altitudes up to 20,000 feet and at airspeeds between 135 to 240 KIAS.

Refer to the *Aircraft Flight Manual* limitations section for specific limitations.

OPERATION NOTES

- During the engine/APU fire detection/extinguishing test, if the TEST button is held more than 10 seconds without canceling the fire bell, the APU will automatically shut down.

- Due to its intake location on the upper right side of the tail cone, the APU has a tendency to ingest right engine exhaust fumes, especially with the propeller feathered. Therefore, after right engine start, recommend using engine bleed instead of APU bleed.

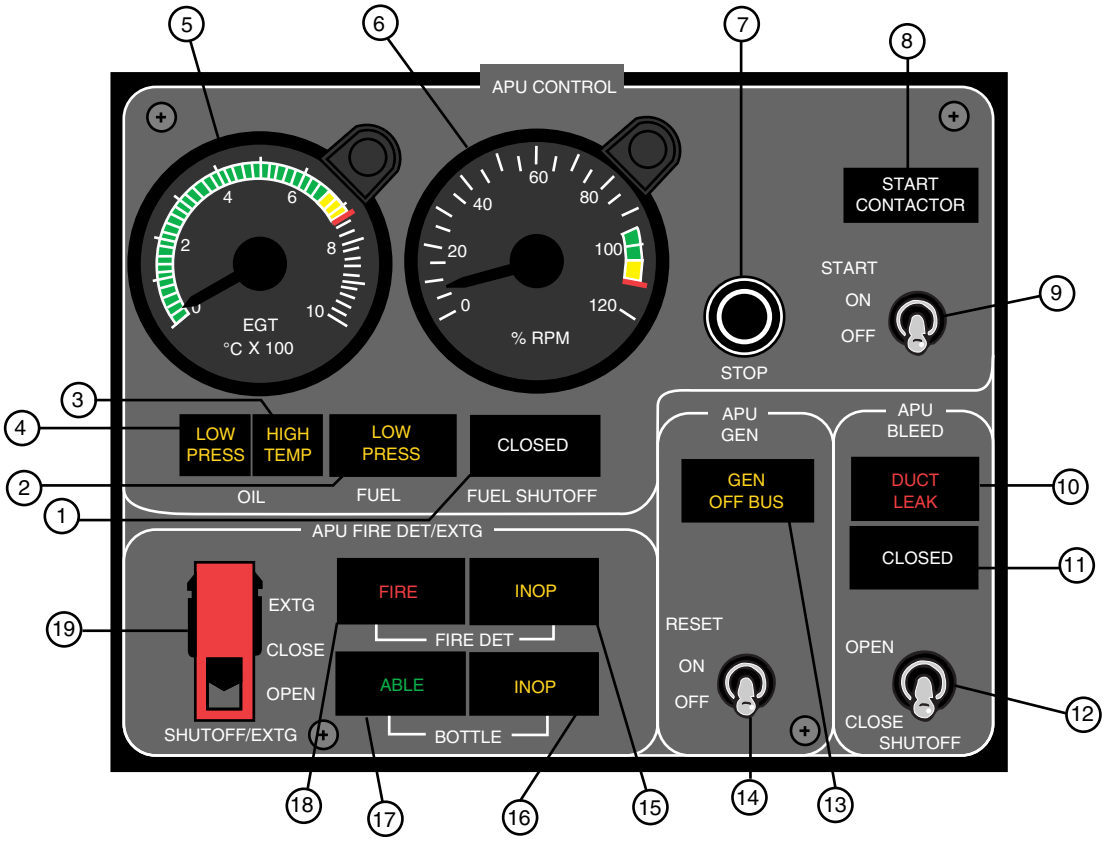
- The APU bleed logic is set up so that the APU cannot supply bleed air to the same air conditioning pack that is being supplied by engine bleed. In this case, priority is given to the engine and the APU bleed closes automatically.

- Do not spray water, cleaning, or deicing fluids into the tail cone APU maintenance door located on the upper right tail cone forward of the APU firewall. These fluids may penetrate the APU ECU and cause the APU to overspeed.

- Once the APU has been shut down and the APU master switch has been turned off, ***do not*** turn the switch back on for at least one minute. If the APU has not spooled down completely and the master switch is turned on, the fuel may be turned on again with a resulting stack fire.

- APU bleed may be utilized to supply air conditioning and pressurization during takeoff to reduce engine temperatures and to increase aircraft performance.

- To prevent electrical transients from affecting sensitive electronic equipment, place the backup battery switch in the ARM position prior to APU start.



1. APU FUEL SHUTOFF valve CLOSED light (white)
   Illuminated—Shutoff valve is closed
2. FUEL LOW PRESS light (amber)
   Illuminated—Fuel pressure is below 9 psi
3. Oil HIGH TEMP light (amber)
   Illuminated—Oil temperature is above 163°C
4. Oil LOW PRESS light (amber)
   Illuminated - Oil pressure is below 31 psi
5. Exhaust gas temperature (EGT) indicator
   Green band: 0–680°C
   Yellow band: 680–732°C
   Red radial: 732°C
6. APU rpm indicator
   Green band: 96–104%
   Yellow band: 104–110%
   Red radial: 110%
7. APU STOP button
   Pressed—Shuts the APU down by inputting an electronic overspeed signal and checking simultaneously the ECU fault protection circuit
8. START CONTACTOR light (white)
   Illuminated—Starter is operating, i.e., from starter actuation until APU reaches 50% rpm
9. APU master switch
   START—Initiates start cycle (momentary position)
   ON—Energizes the ECU, commands the fuel shutoff valve to open and allows the APU to keep running after starting
   OFF—Commands the APU shutdown. ECU is deenergized, fuel shutoff valve is closed, and control panel is deenergized. Fuel shutoff CLOSED and bleed CLOSED lights remain on 10 seconds after shutoff valve is closed.
10. APU DUCT LEAK light (red)
    Illuminated—Bleed-air leak with temperature exceeding 71°
    Illuminates in conjunction with red DUCT LEAK light on MAP.
11. APU BLEED CLOSED light (white)
    Illuminated - APU bleed shutoff valve closed
12. APU BLEED SHUTOFF valve switch
    OPEN—Opens APU bleed shutoff valve 4 seconds after 95% rpm, allowing APU to provide bleed air
    CLOSE—Closes APU bleed shutoff valve
13. APU GEN OFF BUS light (amber)
    Illuminated—APU generator is disconnected from the central DC bus
14. APU generator switch
    RESET—Electrically resets the generator (momentary position)
    ON—Connects the APU generator to the central DC bus
    OFF—Disconnects the APU generator from the central DC bus
15. APU fire detector INOP light (amber)
    Illuminated—Failure of any fire detector
16. APU fire extinguisher bottle pressure condition INOP light (amber)
    Illuminated—Bottle empty or inadequately pressurized
17. APU fire extinguisher bottle ABLE light (green)
    Illuminated - Bottle explosive firing circuits are intact. Illumi nates only during fire detection extinguishing test sequence.
18. APU FIRE warning light (red)
    Illuminated—Fire/overheat condition is sensed by the detec tor. It remains on until the fire/overheat condition disappears. Illuminates in conjunction with red FIRE APU light on the MAP
19. APU SHUTOFF/EXTG switch
    OPEN—APU normal operation
    CLOSE—Causes automatically and simultaneously:
    - APU fuel and bleed valves close
    - APU fuel shutoff CLOSED light and APU bleed CLOSED light illuminates
    - APU generator disconnected from central DC bus
    - APU GEN OFF BUS light illuminates

    EXTG—Discharges fire extinguishing agent

Figure 6-6. APU Panel

QUESTIONS

1. The APU provides a source of:
A. Pneumatic power
B. Electrical power
C. Fuel flow pressure
D. Both A and B

2. The APU can receive fuel from the left fuel tank:
A. True
B. False

3. The APU will automatically shutdown if rotating parts exceed:
A. 102%
B. 105%
C. 110%
D. 115%

4. During the APU start cycle, the:
A. GCU commands starter disengagement
B. FCU commands starter disengagement
C. ECU commands starter disengagement
D. EEC commands starter disengagement

5. The APU can be operated in flight:
A. True
B. False

CHAPTER 7
POWERPLANT

CONTENTS

Page

INTRODUCTION 7-1

GENERAL 7-1

ENGINE 7-2

- General 7-2
- Major Sections 7-4
- Engine Systems 7-7
- Engine Control 7-16
- Engine Monitoring 7-24

PROPELLER 7-29

- General 7-29
- Propeller Assembly 7-29
- Propeller Control Components 7-31
- Propeller Safety Features 7-38
- Propeller Operation 7-43
- Propeller Synchronization System 7-44

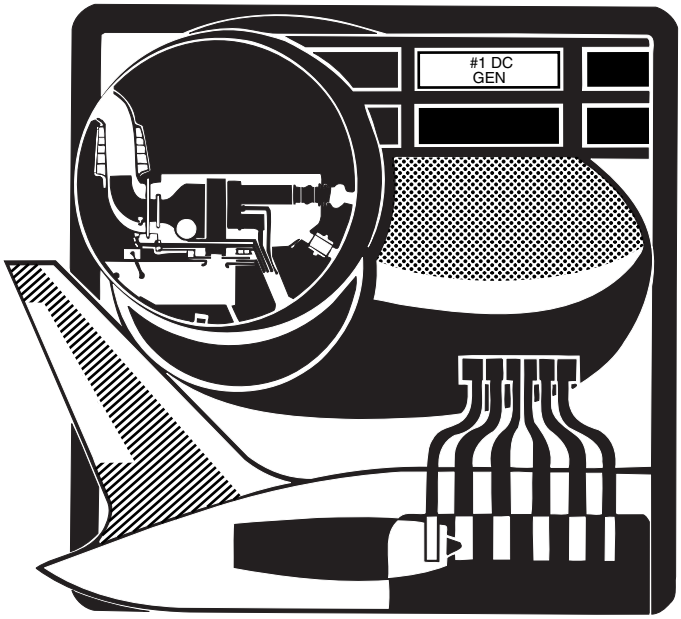
QUESTIONS 7-47

ILLUSTRATIONS

Figure	Title	Page
7-1	Engine Installation	7-2
7-2	Engine Layout	7-3
7-3	Engine Cross Section (Typical)	7-4
7-4	Stations	7-6
7-5	Engine Cowlings	7-7
7-6	Engine Oil Tank/Sight Glass	7-8
7-7	Engine Oil System	7-9
7-8	Hydromechanical Metering Unit (HMU)	7-12
7-9	Fuel Flow Indicators	7-13
7-10	Start/Ignition Panel	7-14
7-11	EEC Input/Output Schematic	7-16
7-12	EEC Controls and Indicators	7-17
7-13	Center Pedestal Console	7-20
7-14	Engine Control System Schematic	7-21
7-15	IDLE 1 (2) UNLK Light	7-22
7-16	Engine Instrument Location	7-24
7-17	Torque Indicating System	7-25
7-18	Engine Instruments	7-27
7-19	Powerplant Control Panels	7-28
7-20	Hamilton Sundstrand 14RF-9 “Commuter” Propeller	7-29
7-21	Blade Construction	7-30
7-22	Propeller Control Components	7-31
7-23	Propeller Oil Supply	7-32

7-24 Propeller Control System 7-34
7-25 Propeller Servomechanism during Pitch Change Operations 7-37
7-26 Powerplant Control Panel 7-39
7-27 AUTO FEATHER Control Panel 7-42
7-28 Synchronization Control Panel..... 7-44
7-29 Propeller System Controls and Indicators 7-46

CHAPTER 7
POWERPLANT



INTRODUCTION

This chapter describes the powerplant and propeller installed on the EMB-120 Brasilia. In addition, discussion is provided regarding related engine systems such as oil, fuel, and ignition; powerplant and propeller controls and instrumentation; as well as engine starting and propeller synchronization.

The information presented in this chapter must not be construed as being equal to or superseding any information from the manufacturers or the FAA. The values used for pressures, temperatures, rpm, power, etc., are used for training purposes only. While most values are accurate, actual operating values must be determined from the appropriate sections of the *Approved Flight Manual.*

GENERAL

The aircraft is equipped with two Pratt & Whitney PW 118, PW 118A, or 118B turboprop engines, each flat rated at 1,800 SHP. The PW 118A and B provide better performance when operating in high ambient temperatures.

Through reduction gearing, each engine drives a Hamilton Sundstrand 14RF-9 four-bladed "commuter" propeller. The propellers are constant- speed, full-feathering reversible units that feature a composite blade design.

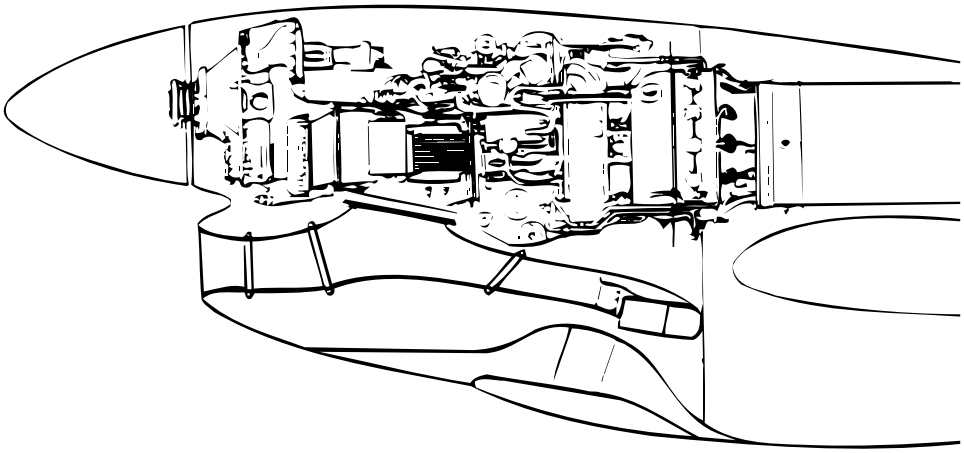


Figure 7-1. Engine Installation

ENGINE

GENERAL

The engines are mounted in the upper portion of the close-cowled nacelle structures (Figure 7-1). The exhaust is ducted to the atmosphere through the rear portion of the nacelle, directing the gas flow slightly downward in relation to the engine longitudinal centerline.

The engine is composed of two modules, the turbomachinery module (TMM) and the reduction gearbox (RGB), joined together by an interconnecting case (Figure 7-2).

The TMM, or main engine, is made up of three independent spool assemblies mounted on coaxial shafts and a reverse flow annular combustion chamber.

The rotating components of the TMM are two centrifugal compressors, each connected to/driven by a single-stage axial turbine, and a two-stage axial power turbine which drives the inner third shaft and consequently the RGB.

The compressors and their turbines form the low- and high-pressure spool assemblies.

The RGB, through two reduction gear stages, drives the propeller.

Each engine also includes two accessory drive sections. One is driven by the TMM and the other by the RGB.

Engine Operation Overview

During the starting cycle, the engine internal airflow is initiated by the starter-generator rotating the high-pressure (HP) spool.

Suction flow from the HP compressor draws air from the air inlet system to the low-pressure (LP) compressor.

This air is compressed by the LP compressor and then routed through the first-stage diffuser ducts to the HP compressor, where it is compressed further. It is then routed through the second stage diffuser ducts to the combustion chamber, where it is mixed with fuel being sprayed into the combustion chamber by the fuel nozzles.

During the start cycle, this fuel/air mixture is ignited by the ignition system igniter plugs. The expanding gases flow rearward and drive the HP and LP turbine stages which, in turn, drive their respective compressors to draw in more air for combustion.

As the engine speed increases, the fuel/air mixture is increased by the hydromechanical metering unit (HMU) fuel schedule until the engine has achieved a self-sustaining speed with continuous combustion.

The flow of expanding gases from the HP and LP turbines then continues on rearward to the two-stage power turbine which, through the torque shaft, drives the RGB, accessories, and the propeller.

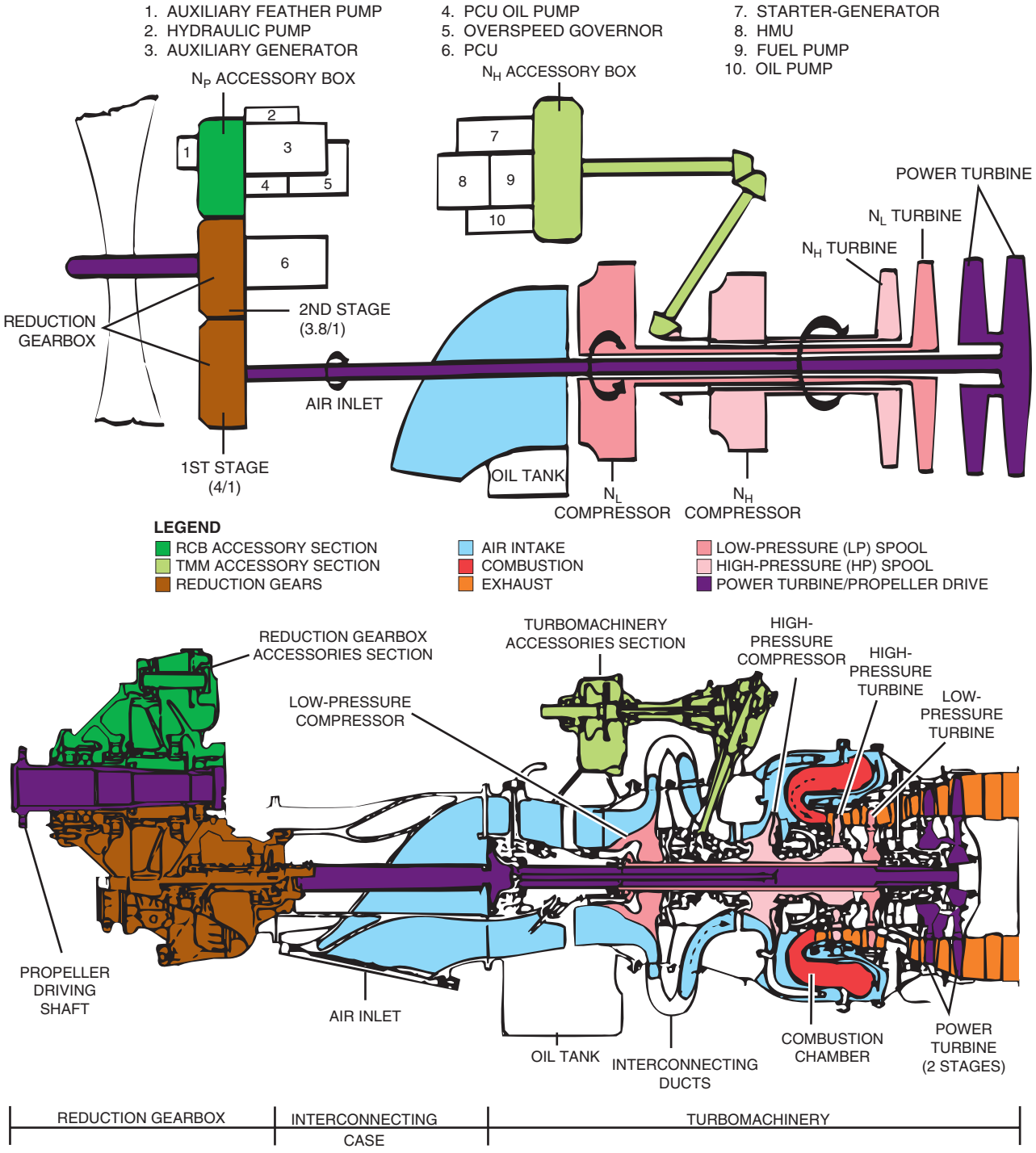


Figure 7-2. Engine Layout

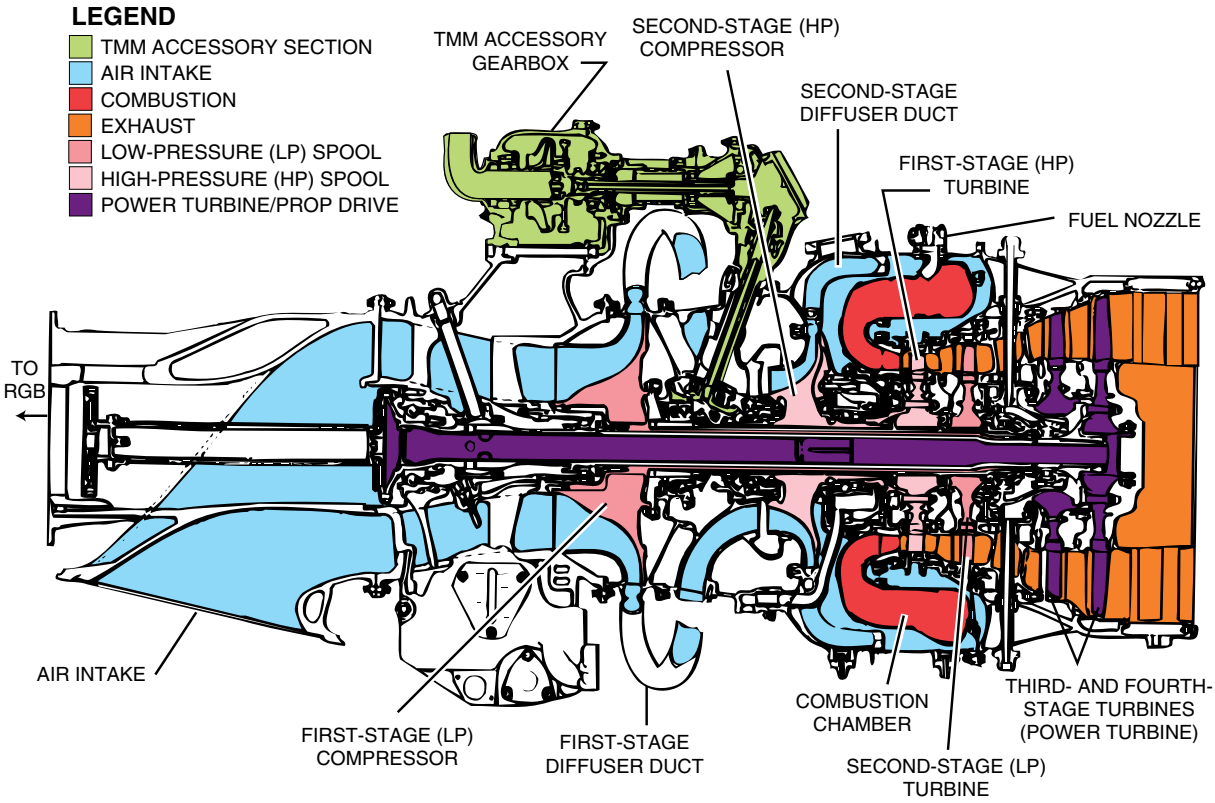


Figure 7-3. Engine Cross Section (Typical)

MAJOR SECTIONS

- Air inlet section
- Compressor section
- Combustion section
- Turbine section
- Reduction gearbox section
- Accessory drive section
- Cowling, drains, and vents

Air Inlet Section

The air inlet section routes air from the air inlet scoop to the engine compressor section through an S-shaped duct. The S-shape of the duct creates a deviation in the airflow from the air inlet to the engine. The result is that denser particles, by their inertia, are discharged overboard through a bypass duct. This process constitutes the continuous flow inertial separation system.

Compressor Section

The two-stage, centrifugal compressor section (Figure 7-3) provides the combustion section with enough airflow to sustain continuous combustion under all operating conditions.

The section consists of the following:

- First-stage compressor
- First-stage diffuser ducts
- Second-stage compressor
- Second-stage diffuser ducts

The ***first-stage compressor*** is a low-pressure (LP) centrifugal compressor driven by the second-stage

turbine. It rotates counterclockwise (as viewed from the rear).

The LP compressor receives airflow from the air inlet section into the center of the rotating LP impeller. Through centrifugal force, the air is figuratively "thrown" outward and away from center into a decreased space, where it is compressed and routed through the external ***first-stage diffuser ducts*** to the next compression stage.

The ***second-stage compressor*** is a high-pressure (HP) centrifugal compressor driven by the first stage turbine. Its clockwise rotation is opposite to the first stage (LP) compressor.

The HP compressor receives airflow from the first stage diffuser ducts into the center of the rotating HP impeller. It is compressed and routed, through the internal ***second-stage diffuser ducts***, to the combustion chamber.

Combustion Section

The combustion section is made up of a case which includes:

- Combustion chamber
- Fuel nozzles
- Igniter plugs

Air enters the annular, reverse-flow ***combustion chamber*** via the second-stage diffuser ducts. The direction of the entering airflow is changed 180° to allow for an overall shorter engine length.

Fuel for combustion, supplied by the engine fuel system, is injected into the annular chamber properly atomized by 14 ***fuel nozzles***. The fuel/air mixture is then ignited by two spark ***igniter plugs***.

Turbine Section

The four-stage, axial flow turbine section converts the energy of the expanding gases exiting the combustion chamber into rotational force used to drive the compressors, propeller, and accessories.

The following components make up the turbine section:

- First-stage turbine
- Second-stage turbine
- Third- and fourth-stage turbines
- Thermocouples

The ***first-stage turbine*** drives the HP compressor and TMM accessory gearbox. The turbine and compressor together make up the HP spool.

The ***second-stage turbine*** drives the LP compressor. The turbine and compressor together make up the LP spool.

The interconnected ***third- and fourth-stage turbines,*** or power turbine, drive the RGB, RGB accessories, and propeller. The power turbine rotates in the same direction, clockwise, as the high-pressure spool.

Nine bimetallic ***thermocouple*** probes project into the expanding gases between the second-stage turbine and the power turbine to monitor inter-stage turbine (T_6) temperature.

The thermocouples are connected in parallel so that output voltage, sent to the respective T_6 indicator, is proportional to the average turbine temperature.

Reduction Gearbox Section

The RGB is located in front of the TMM above the engine air inlet. It reduces the rpm from the power turbine shaft by a ratio of approximately 15:1, and increases the torque delivered to the propeller.

Accessory Drive Sections

The engine has two accessory drive sections, one on the TMM and the other on the RGB.

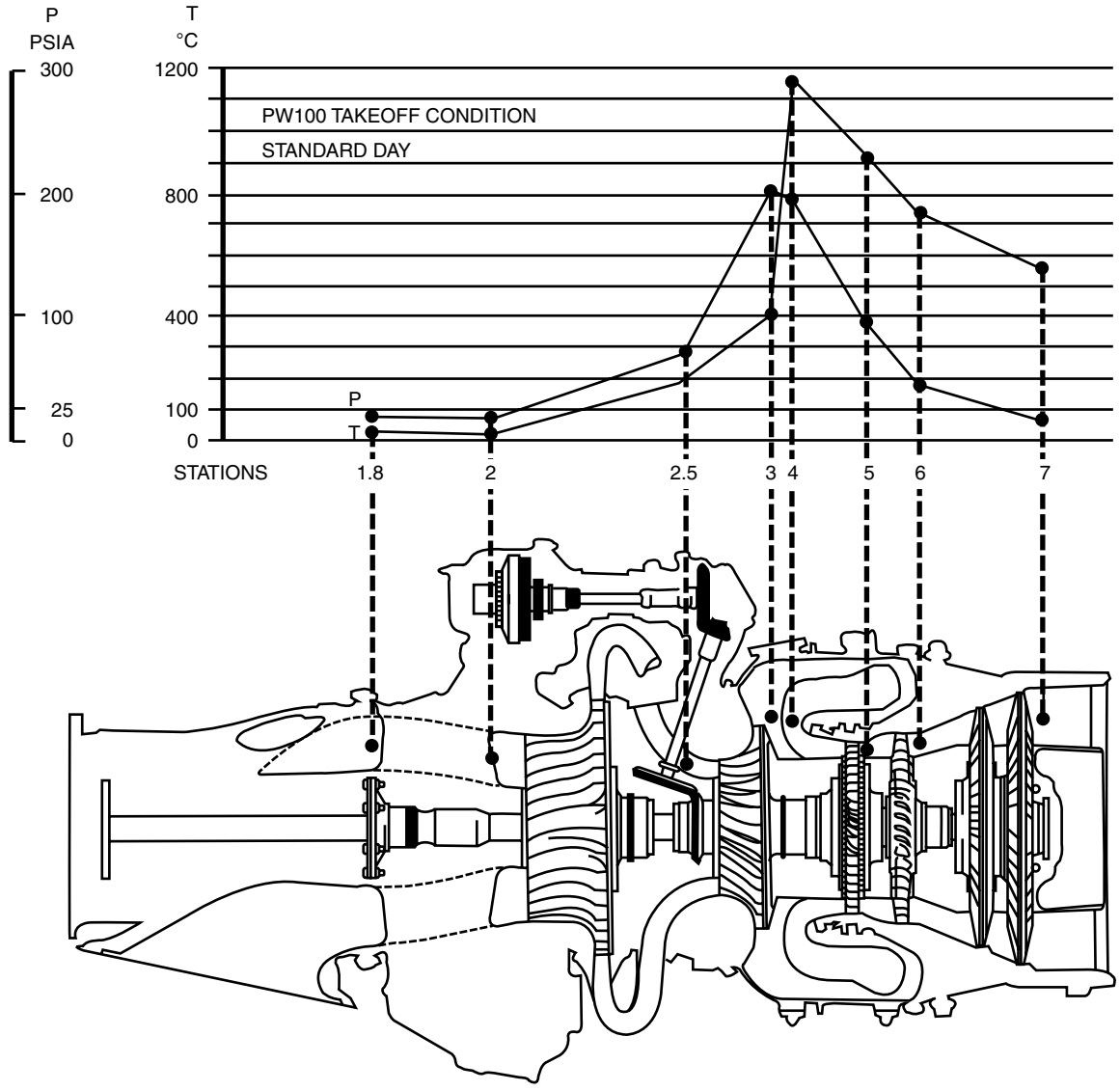


Figure 7-4. Stations

The TMM accessory gearbox (driven by the HP spool) drives the following:

- Starter-generator
- High-pressure fuel pump
- Oil (and scavenge) pump assembly
- Air/oil separator

Hydromechanical metering unit is located on, but not driven by the TMM accessory gearbox.

The RGB accessory gearbox (driven by the power turbine) drives the accessories:

- Hydraulic pump
- Overspeed governor
- Propeller oil pump assembly
- Propeller control unit (PCU)
- Auxiliary generator

Cowling, Drains, and Vents

The engine cowlings (Figure 7-5) make up a streamlined surface covering the engine and exhaust duct.



Figure 7-5. Engine Cowlings

The forward cowling houses the engine accessory section. A quick access door on the left side of the forward cowling provides access to the engine oil tank, sight gage, and filler cap. The aft cowling encloses the TMM and engine tailpipe.

The cowlings incorporate numerous NACA air inlets and outlets to provide cooling air for various engine components. Internal engine air is vented through a centrifugal breather air/oil separator and discharged into the engine exhaust duct. This venting operation is aided by the venturi effect of the engine exhaust flow. Oil recovered by the separator is returned to the engine oil storage tank.

Drain lines collect fuel, oil, and water from various points on the engine. These drain lines connect to a common drain mast on the outboard side of each nacelle and are discharged into the air stream.

ENGINE SYSTEMS

Bleed-Air System

Engine bleed air is used in the engine for sealing the turbomachinery bearing seals and cooling internal engine components. It is also used for the aircraft pneumatic system. Low-pressure bleed air is tapped from the LP compressor discharge (engine station 2.5) at the 12- and 2-o'clock positions. High-pressure bleed air is tapped from the HP compressor discharge (engine station 3.0) at the 10-o'clock position.

Engine Bleed Air—The valve at the 2-o'clock position is the $P_{2.5}/P_3$ switching valve. Its purpose is to select the bleed pressure used to seal the turbomachinery bearing oil seal and the air used internally in the engine. This valve is fully automatic and should not be confused with the $P_{2.5}$ and P_3 air used by the aircraft pneumatic system.

Pneumatic System Bleed Air—The 12-o'clock and 10-o'clock bleed ports, designated $P_{2.5}$ and P_3, are used to supply the aircraft's pneumatic system with low- and high-pressure bleed air.

The $P_{2.5}$ bleed port (LP bleed valve) is located at 12-o'clock position of the intercompressor case and bleeds air from the low-pressure stage through a flow limiting check valve. The check valve prevents the airflow from reversing into the engine when the P_3 bleed is open.

The P_3 bleed port (HP bleed valve) is located at the 10-o'clock position of the front part of the high-pressure diffuser combustion chamber case and bleeds air from the high-pressure stage through a venturi.

Refer to Chapter 9, "Pneumatics," for additional information.

Engine Oil System

The engine oil system provides a constant flow of filtered oil under controlled pressure and temperature for lubricating and cooling the engine bearings, reduction gears, and the gears of the RGB and TMM accessory sections. The oil system also actuates as well as lubricates the propeller servomechanism.

The engine oil system consists of the following subsystems:

- Storage system
- Pressure system
- Scavenge system
- Venting system
- Oil indicating system

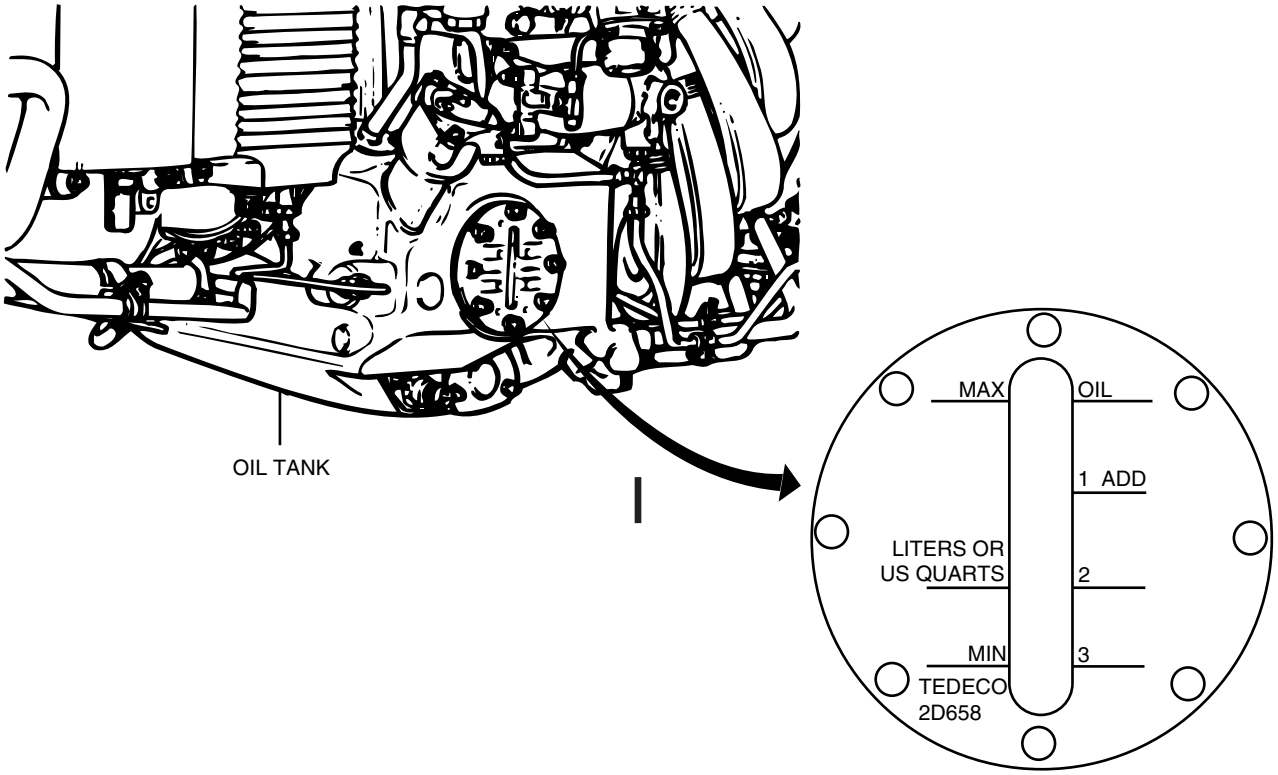


Figure 7-6. Engine Oil Tank/Sight Glass

Storage System—A 2.4 gallon (19.3 pound) engine oil tank forms an integral part of the engine air inlet. The tank (Figure 7-6), accessible through the oil access and inspection door on the left cowling, has a filler neck, a cap with a quantity indicating dipstick, and a sight-glass quantity gage (graduated in quarts low).

The maximum oil usage is one quart per four hours or $2^1/_2$ quarts per 10-hour period. For the most accurate reading of oil quantity, Pratt and Whitney recommends it be checked within 30 minutes of shutdown.

A 0.3 gallon (2.5 pound) auxiliary oil tank, or electric feather pump reservoir (EFP), is located inside the RGB. It is utilized for the storage of oil used by the auxiliary oil pump to feather the propeller should an engine oil system failure occur.

The tank is automatically filled by the engine oil system and when the engine is running or dry motored by the starter-generator.

Pressure System—The pressure system, as shown in Figure 7-7, delivers filtered oil under pressure to the RGB and TMM lubricating networks. It consists of the following components:

- Oil pump pack
- Oil cooler
- Pressure filter
- Pressure regulating valve

The ***oil pump pack***, mounted on the TMM accessory gearbox, is driven by the engine N_H spool. It provides mounting for the pressure pump, scavenge pumps, and the low-temperature relief valve.

The pressure pump is a high-capacity, positive-displacement, mechanical gear pump. Output volume and pressure are directly proportional to engine speed. The scavenge pumps are also positive-displacement, mechanical gear pumps.

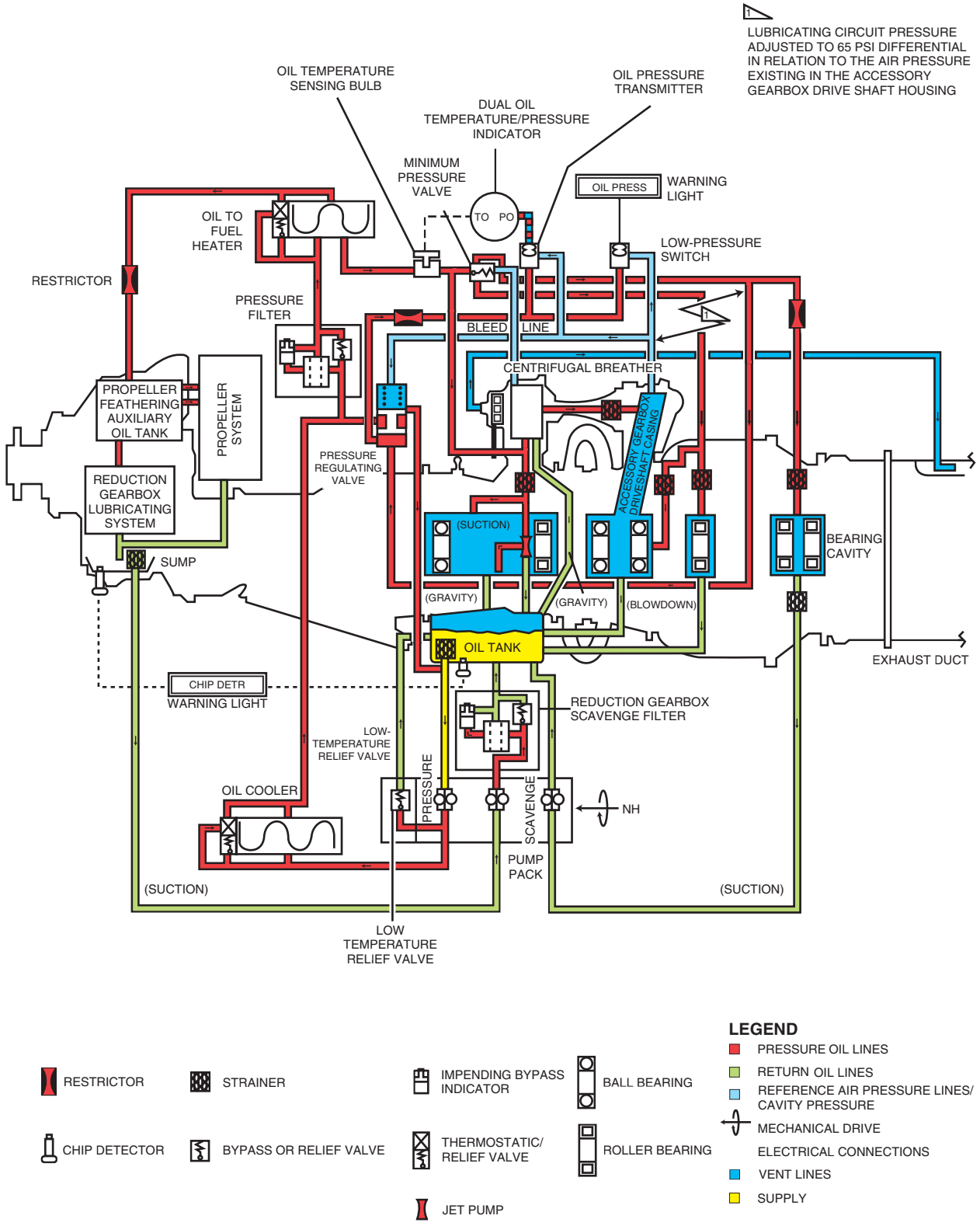


Figure 7-7. Engine Oil System

The oil pressure system components are protected during engine start by the low-temperature relief valve. The valve prevents the oil pump discharge pressure from exceeding 210 psi by returning part of the oil to the engine oil tank.

The ***oil cooler*** is an oil-to-air heat exchanger located in a separate duct below the engine intake duct.

The flow of oil through the oil cooler is controlled by a thermostatic/bypass valve. The valve maintains oil temperature within the normal operating range, and also opens to bypass oil when the pressure differential between the oil cooler inlet and outlet exceeds a preset limit.

NOTE

Extended ground operation in feather or ground idle reduces airflow to the oil cooler, resulting in high oil temperatures.

The ***pressure filter*** is located on the left side of the engine above the oil sight glass. It includes a bypass valve that opens when pressure differential across the filter exceeds 20 psid.

The pressure regulating valve maintains a constant differential between the oil pressure in the number 6 and 7 bearing cavities and the air pressure in the TMM accessory section gear cavity. The excess oil pressure is returned to the oil pump.

Scavenge System—There are two oil scavenge systems: one for the TMM lubricating network, and one for the RGB lubricating network.

The ***TMM scavenge system*** returns oil to the engine oil tank through internal passages and external tubing.

Oil from the number 1 and 2 bearings is returned by both gravity draining and an ejector jet pump.

Oil from the number 3, 4, and 5 bearing cavities is returned using the air pressure from labyrinth seal leakage.

Oil from the number 6 and 7 bearing cavities is returned by a scavenge pump mounted on the oil pump pack.

The ***RGB scavenge system*** includes a positive-displacement, mechanical gear pump mounted on the oil pump pack, and a scavenge filter. The filter incorporates a bypass valve in the event of clogging.

Venting System—The engine oil system uses a centrifugal breather to separate oil from the internal engine air before venting overboard.

The centrifugal breather is inside the TMM accessory gearbox. Oil separated by centrifugal force is returned to the engine oil storage tank while the air is vented overboard through external tubes along the right side of the engine into the engine exhaust duct. The venturi effect of the engine exhaust gas flow assists the venting.

Oil Indicating System—The oil indicating system provides a visual indication of the oil pressure and temperature measured in the TMM, and magnetic chip detection in the oil tank and RGB sump. It includes the following components and subsystems:

- Temperature/pressure indicator
- Temperature sensing bulb
- Pressure transmitter
- Low-pressure warning system
- Magnetic chip detection warning system

A combined oil ***temperature/pressure indicator*** for each engine is installed on the cockpit main instrument panel. They are back-lighted, dual analog pointer displays with the oil temperature on the left scale and oil pressure on the right.

The indicators are powered by 28 VDC, and 5 VDC for lighting. They include a circuit that drives the indicators off scale in the event of a power loss.

An oil ***temperature bulb*** senses the temperature of the oil downstream of the fuel heater and sends it to the temperature/pressure indicator.

An oil ***pressure transmitter*** provides input to the pressure side of the oil temperature/pressure indicator.

The transmitter is connected to the oil system by two lines. One line provides total oil pressure input and the other provides TMM accessory gear cavity air pressure.

The transmitter converts the reference oil and air pressures into differential oil pressure and sends it to the temperature/pressure indicator.

A ***low oil pressure warning system*** for each engine provides a visual and aural warning when the oil pressure falls below 40 psid.

When a differential pressure switch connected to the oil system pressure regulating valve is activated at 40 psid, an aural alarm of three chimes sounds, and the red OIL PRESS 1 or OIL PRESS 2 light on the MAP flashes. The voice warning "OIL" is given and the master WARNING lights flash. The low oil pressure warning is inhibited during engine start and intentional engine shutdown.

NOTE

The engine low oil pressure switch is also utilized to prevent overheat damage to the propeller blades by the electrical deicing circuit. The propeller blades may be heated only when the oil pressure is above 40 psid.

Engine operation with oil pressure between 40 and 55 psid is permitted only if the N_H is below 75%.

Oil pressure below 40 psid requires engine shutdown.

The ***magnetic chip detection system*** provides an indication of the presence of metal particles in the oil. The magnetic chip detectors are installed in the reduction gearbox and the engine oil tank. If a chip is detected, the system provides a visual alert to maintenance personnel by illumination of an amber CHIP DET 1 or CHIP DET 2 light located in the battery compartment.

NOTE

Chip detector lights on some unmodified aircraft are installed on the MAP and sound an aural alert.

Engine Fuel System

The engine fuel system receives fuel from the airplane's fuel tanks and delivers it to the combustion section with enough fuel flow to sustain continuous combustion under all operating conditions. For description of components prior to the HMU, see Chapter 5, "Fuel System".

Hydromechanical Metering Unit (HMU)—The HMU (Figure 7-8) is the engine fuel control unit. It is mounted on the forward end of the engine fuel pump (which is located on, and driven by, the TMM accessories section). The HMU controls the minimum and maximum limits of fuel flow to the engine as a function of the power lever angle (PLA) and high-pressure compressor discharge pressure (P_3). It also provides the motive fuel flow to operate the fuel tank ejector jet pumps.

The ***motive flow valve*** allows fuel in excess of the engine requirements to return to the fuel tank, supplying motive-flow pressure to operate the tank ejector jet pumps.

Pressure relief and ***regulating valves*** maintain the necessary pressure differential within the HMU by diverting part of the fuel flow to the ejector jet pump in the engine fuel pump.

A ***power lever valve***, actuated by the power lever, determines the rate of fuel flow. A potentiometer, located at the bottom of the power lever valve shaft, provides the engine electronic control (EEC) with an electrical signal proportional to the power lever angle. This signal is used by the EEC in modifying the HMU fuel schedule supplied to the engine.

A ***P_3 sensor and servo***, utilizing engine bleed air, works in conjunction with the power lever valve to determine the basic fuel flow schedule to the engine. It varies the fuel flow to the engine based on the HP compressor discharge pressure.

Signals from the EEC to a ***torque motor*** operate a valve that modifies the basic fuel scheduled by the power lever valve and P_3 servo position.

A ***deenrichment solenoid valve***, electrically controlled by the EEC, prevents engine compressor stalling associated with high-altitude power lever transients. The solenoid is energized above 14,000

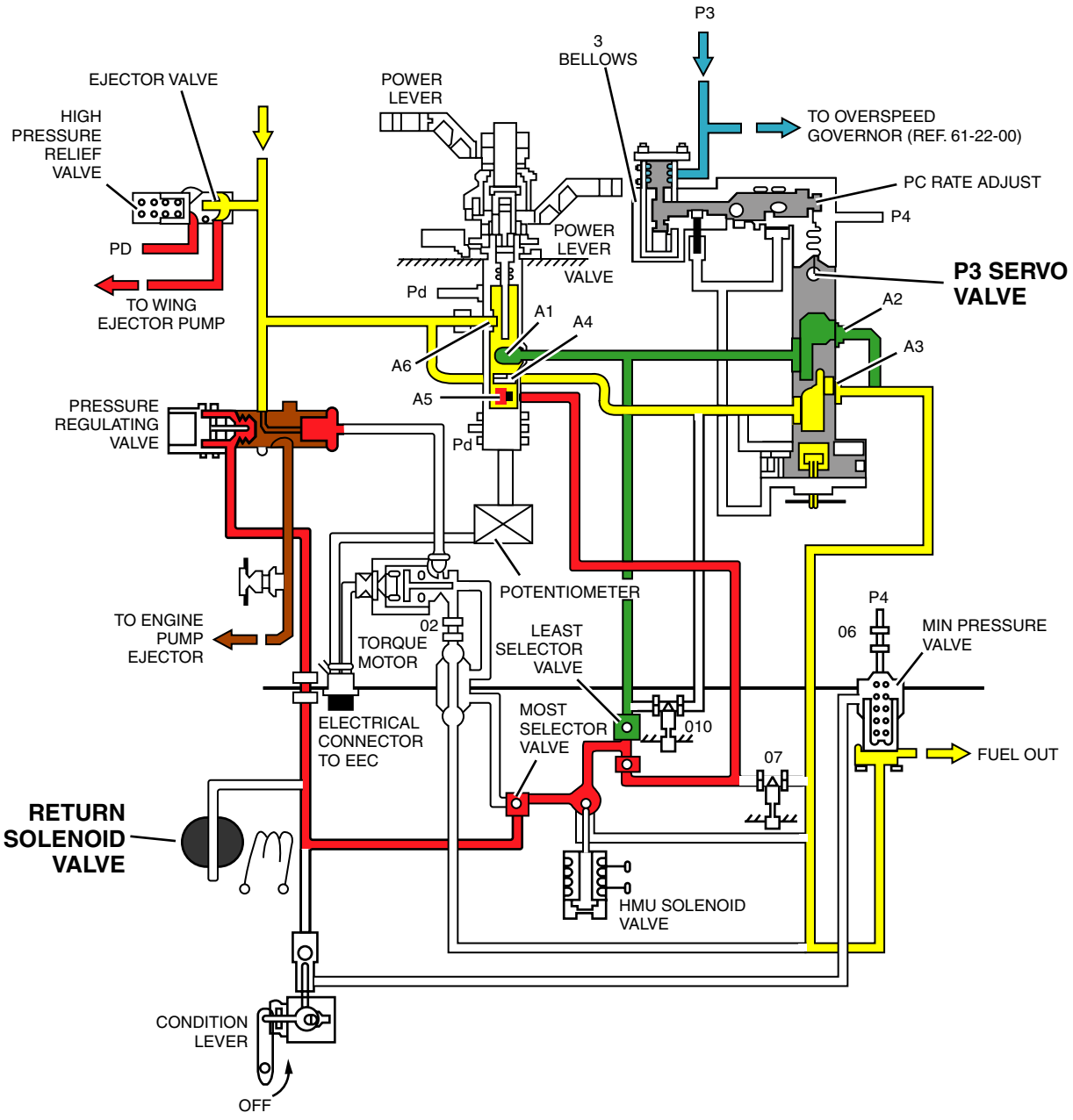


Figure 7-8. Hydromechanical Metering Unit (HMU)

feet MSL to deenrich the HMU basic fuel schedule. Below 14,000 feet MSL, the solenoid is deenergized.

The ***HMU return solenoid valve*** is energized only during the start cycle. It purges air from the HMU and provides fuel pressure to operate an ejector jet pump within the EPA drain tank. The jet pump returns fuel collected from the last shutdown to the fuel tank.

A spring-loaded ***minimum pressure valve*** is located at the HMU outlet. During engine start, the valve ensures adequate fuel pressure within the HMU to operate the control valves and supply pressure to the EPA drain tank ejector jet pump.

At about 6% N_H, the HMU discharge pressure overcomes the spring pressure and allows fuel to pass to the flow divider and on to the fuel nozzles.

Flow Divide—The fuel flow divider, located on the lower portion of the combustion section, distributes high-pressure fuel to the primary and secondary fuel nozzles.

During engine start, when the condition lever is moved out of the cutoff position, fuel pressure from the engine fuel pump lifts the primary spool of the flow divider and allows fuel to flow to the primary fuel nozzles. As the engine speed increases, fuel pump pressure lifts the secondary spool allowing fuel to flow to the secondary nozzles.

During engine shutdown, as fuel pressure decreases, spring pressure closes the flow divider spools. The primary and secondary lines of the fuel nozzles are then drained to the EPA drain collector tank.

Fuel Lines—The primary and secondary fuel lines supply the fuel nozzles. A third line provides fuel drainage in the event of a leakage in the nozzle assembly "O" rings.

Fuel Nozzles—Fourteen air-blast fuel nozzles supply the properly atomized fuel to be burned in the combustion chamber. Each nozzle has provisions for primary and secondary ports, however, only 10 nozzles have primary ports in use.

Fuel Flow Indicating—Each engine fuel flow indicating system consists of a flow transmitter and a cockpit analog/digital indicator (Figure 7-9). The transmitter measures fuel flow between the HMU and the fuel nozzles It compensates for temperature and density to provide "true" fuel flow, in pounds per hour (lb/hr), to the cockpit indicator. For more information on fuel indications, see Chapter 5, "Fuel System".

Engine Ignition System

Each engine ignition system provides high-voltage electrical energy to ignite the fuel/air mixture in the combustion chamber, and includes the following components:

- Ignition control switch
- Ignition lights
- Exciter unit
- Igniter plugs

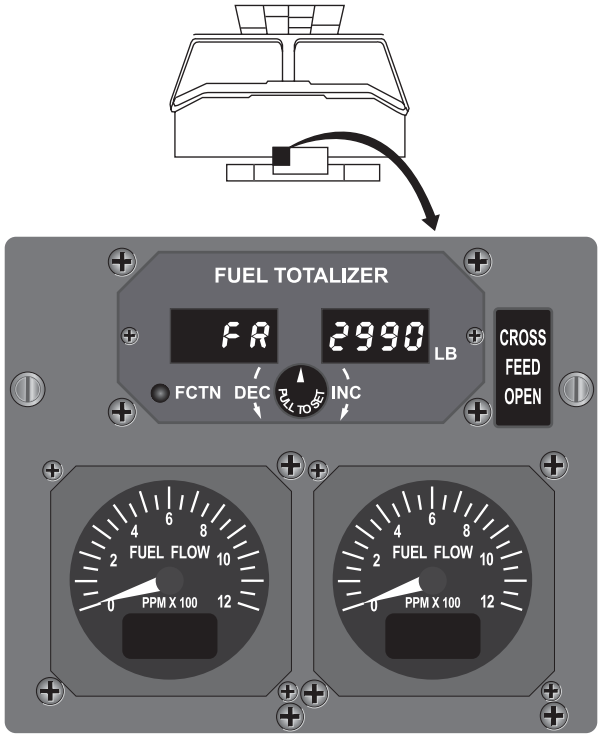


Figure 7-9. Fuel Flow Indicators

Ignition Control Switches—The ignition control switches are located on the START/IGNITION panel on the cockpit overhead panel (Figure 7-10). They are powered by the respective emergency DC bus. The ignition control switch positions are as follows:

ON—The ignition circuit is continuously energized.

AUTO—The ignition circuit is automatically energized whenever the starting cycle is initiated by selecting the engine start switch to START. The ignition circuit is automatically deenergized at 50% N_H by the GCU. The ignition circuit may be manually deenergized by moving the engine start switch to the ABORT position.

OFF—The ignition circuit is deenergized, (even if the start cycle is initiated).

The OFF position of the ignition control switch is used for the following:

- Dry motoring the engine to replenish the RGB auxiliary oil tank

- Clearing the engine of fuel after a wet start
- Lowering the T_6 temperature after a hot start
- Compressor and/or turbine washes

Ignition Light—The ignition light, next to the ignition switch, illuminates when the ignition circuit is energized.

Exciter Unit—The electronic exciter unit, mounted on the engine, contains solid-state circuitry, transformers, and capacitors used to generate the electrical energy required by the engine ignition system.

When the unit is energized, the capacitor is charged until the energy stored reaches approximately 4 joules. The capacitor then discharges through high-voltage leads to the igniter plugs.

Igniter Plugs—In the combustion chamber of each engine are two identical, independent igniter plugs. The plugs are primarily used to ignite the fuel/air mixture during engine start. Continuous ignition may be used in adverse weather conditions to provide protection from engine flame outs .

Engine Starting System

Start System Components—Each engine starting system includes the following components:

- Start switch
- Engine start contactor
- Starter-generator
- Generator control unit (GCU)
- Start relays

The engine ***start switches***, on the overhead START/IGNITION control panel (see Figure 7-10), have the following two momentary positions:

ON—Signals the GCU to begin the starting cycle. The GCU closes the start contactor and activates the ignition circuit. The starting cycle ends when the GCU receives a signal from the N_H sensor at 50% N_H.

ABORT— Interrupts the engine start cycle by sending an artificial 50% N_H signal to the GCU.

The engine ***start contactor*** connects the central DC bus to the starter-generator during the start cycle.

The ***starter-generator***, located on the right side of the TMM accessory gearbox, drives the N_H spool during the start cycle.

The starter-generators are individually controlled by ***generator control units (GCUs)*** located inside the right and left cockpit consoles. Each GCU controls all functions and associated components for starting and electrical generation.

There are multiple ***relays*** that control the ignition system, start contactors, and starter-generator

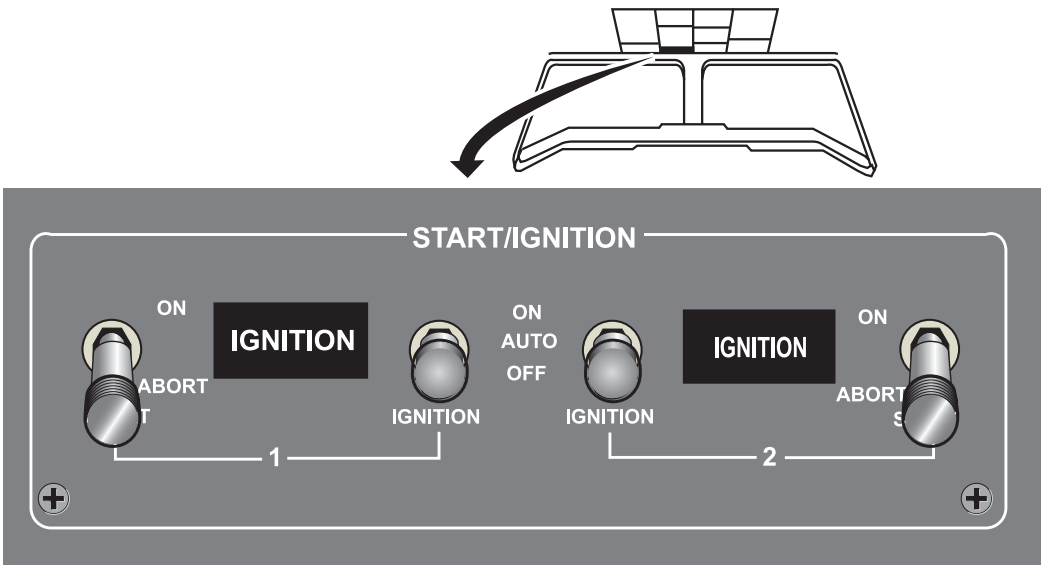


Figure 7-10. Start/Ignition Panel

operation, as well as protect against starter-generator short circuits, and prevent simultaneous starting of both engines.

The relays are in the respective DC relay box and operated by the GCUs.

Starting System Operation—The engine starting cycle is an automatic sequence in which the ignition and starting systems are simultaneously activated and deactivated.

The electrical power for engine starting may be from an external power source or the airplane battery.

The APU generator, or the generator of the other engine, may be used to assist the battery during an engine start. The output of a generator alone is not sufficient to provide the power required for an engine start.

During this description of starting system operation, the following prestart configuration is assumed:

- Power select switch —EXTernal
- Generator switches—OFF
- Bus tie switches—ON
- Fuel boost pump —ON
- Power levers—GROUND IDLE
- Condition levers—FUEL CUTOFF
- Ignition switches—AUTO

Momentarily placing the starting switch to the ON position sends a signal to the GCU that initiates the following start sequence:

- The ignition circuit is energized (white IGNITION light illuminates).
- The start contactor connects the available power on the central DC bus to the starter-generator, which drives the N_H spool assembly.
- When N_H reaches 10%, the pilot moves the condition lever from the CUT-OFF to the FEATHER position, introducing fuel into the engine.
- The HMU return solenoid valve, energized by the GCU, purges air from the HMU and the empties the EPA drain collector tank into the respective wing fuel tank.
- The HMU provides fuel through the flow-divider primary fuel nozzles to the combustion chamber. The secondary fuel nozzles activate later in the start cycle when fuel pressure is higher.
- The igniter plugs ignite the fuel/air mixture, as indicated by a T_6 rise in approximately five seconds (maximum 10 seconds).

NOTE

T_6 should be monitored for a hot start as the HP spool continues to accelerate.

- When N_H reaches 50%, the starter-generator signals the GCU to terminate the start cycle (white IGNITION light goes out).
- The GCU opens the start contactor, turns off the ignition circuit, deenergizes the HMU solenoid, and enables the generator function of the starter-generator. At this point, the engine is self-sustaining and will continue to accelerate to approximately 62% N_H and stabilize.

Hot Start Indication—If the T_6 is rising rapidly with a relatively slow N_H acceleration, an imminent hot start is indicated. The condition lever should be moved to FUEL CUTOFF and, at the end of the 30-second starter limit, the START switch should be moved to ABORT. The cause of the hot start should be investigated before another start is attempted.

Wet Start Indication—If no T_6 rise is observed, and N_H stagnates or stabilizes below 50% N_H, it may be assumed that the engine has failed to light-off and a probable wet start has occurred.

In this case, the condition lever should be moved to FUEL CUTOFF and the IGNITION switch moved to OFF. At the end of the 30-second starter limit, the START switch should be moved to ABORT.

The cause of the wet start should be investigated before another start is attempted.

Possible reasons for the no-start condition are:

- Ignition switch in the off position
- Ignition switch circuit breaker out
- Igniters inoperative
- Out of gas

ENGINE CONTROL

Electronic Engine Control (EEC)

The electronic engine control (EEC) is a solid-state, engine control device that works in conjunction with the HMU. The microprocessor-based EEC executes a program defined by its programmable read-only memory (PROM). It is mounted on the engine air inlet front case, at the 9-o'clock position.

The purpose of the EEC is to modify the HMU fuel schedule to improve engine performance. It has the authority to increase the fuel flow over and above what the HMU would schedule for the same power lever position, but only within the maximum and minimum fuel flow limits established by the HMU.

The EEC utilizes several external inputs and provides output to the torque motor in the HMU (Figure 7-11).

The EEC is powered from its respective emergency DC bus. When emergency bus voltage is interrupted, power is supplied by the backup emergency DC bus (except during an electrical emergency condition).

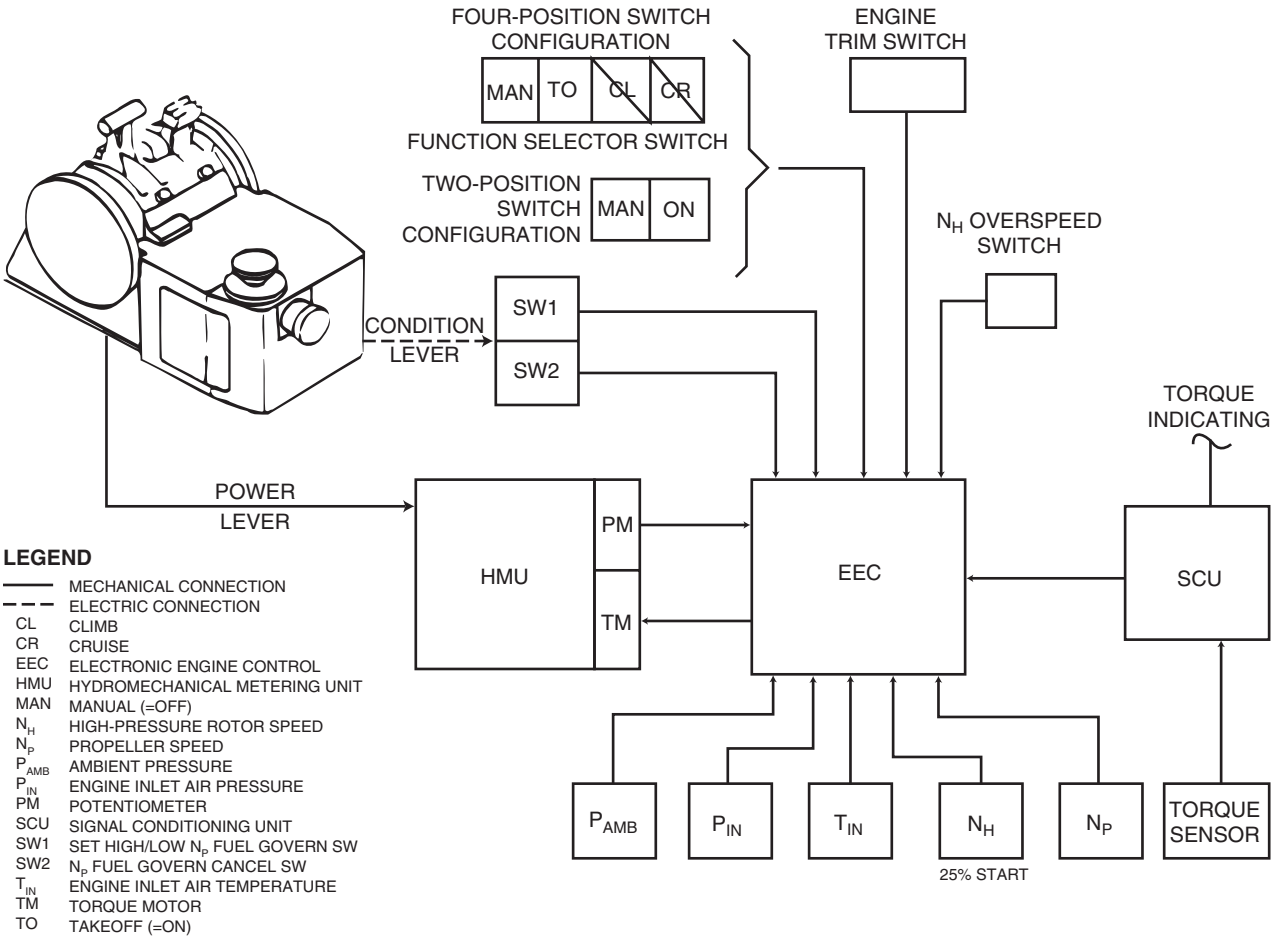


Figure 7-11. EEC Input/Output Schematic

The engine may be started with the EEC on or off. With the EEC on, starting will be characterized by two temperature peaks; with the EEC off, start is characterized by slower starting and only one temperature peak.

Power Rating Selector—The EEC uses the power rating selector on the overhead EEC panel (Figure 7-12) to program the EEC control function. The power rating selector has the following positions:

- MAN (EEC off)—Selects the manual, HMU only, mode.
- TO/CL/CR (EEC on)—Selects the automatic, control functions, mode.

The EEC operates the same in each position. Originally, the EEC system was designed to operate takeoff, climb, and cruise maximum torque indicator bugs. As these functions have not yet received regulatory approval, the torque indicator bugs are inactive.

The EEC panel also has two white MANUAL lights to monitor the EECs. They illuminate when the respective EEC power rating selector is placed in the MAN position, or when the EEC automatically reverts to the manual mode.

In addition, there are two red warning lights on the glareshield panel labeled EEC 1 and EEC 2 (Figure 7-12). They illuminate when a failure is detected in the respective EEC.

EEC Control Functions—The following control functions are performed by the EEC any time in the TO or ON position and engine speed is above 25% N_H:

- Idle speed governing
- N_P fuel governing
- Acceleration/deceleration control
- Fixed torque climb
- HMU deenrichment

N_H idle speed governing occurs with the condition lever in feather and the power lever in ground or flight idle. The EEC performs the following function:

- In ground idle, the EEC maintains an idle speed of 62% N_H.
- In flight idle, the EEC maintains an idle speed of 72% N_H.

N_P fuel governing is the process of controlling propeller speed during taxi and reverse by varying the fuel flow to the engine. In a free turbine engine, propeller speed is a function of blade angle and exhaust gas flow over the power turbine.

During taxi, the crew uses the power levers in the Beta range to change blade angle. The EEC maintains specific minimum N_P by increasing fuel flow as the power lever is moved forward and decreasing fuel flow as the power lever is moved aft.

During reverse, the EEC increases fuel flow to the engine to assist in decelerating the aircraft.

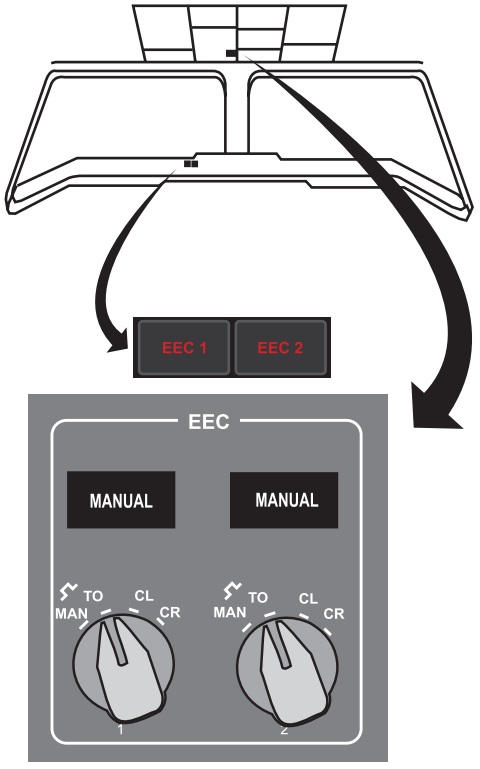


Figure 7-12. EEC Controls and Indicators

With N_P fuel governing, the EEC maintains the following conditions:

- 50% N_P with the condition lever at MIN rpm
- 65% N_P with the condition levers at MAX rpm
- 80% N_P with the power levers in reverse

Without N_P fuel governing, the following conditions exist:

- When the power levers are moved forward, blade angle increases and N_P decreases.
- When the power levers are moved aft, blade angle decreases and N_P increases.
- When the power levers are moved into reverse, no engine spool-up occurs.

N_P fuel governing is canceled by the following:

- Condition lever in feather
- Operation of the autofeather system
- Pulling the engine fire handle

The ***acceleration/deceleration control*** function of the EEC controls the acceleration of the engine to improve spool-up time and eliminate compressor stalls, and controls the deceleration of the engine, thereby preventing flameouts.

Fixed Torque Climb—In most aircraft to maintain the same power output during the climb, the crew must continuously move the power levers forward.

In this aircraft, the EEC fixed-torque climb feature maintains the selected torque during the climb by increasing fuel flow to the engine. As a result, the T_6 temperature increases and small, occasional power lever reductions are necessary to keep the T_6 temperature within climb limits.

The ***HMU deenrichment*** solenoid valve control system is activated depending upon the altitude of the aircraft. The valve is either open or closed.

Under standard sea-level conditions (except for N_P and N_H speed governing functions), the EEC and the HMU schedule approximately the same fuel flow to the engine at idle.

During climb, the EEC maintains the selected fixed torque by constantly increasing the fuel flow over and above what the HMU would schedule for the same power lever position.

At approximately 14,000 feet, the HMU fuel schedule has become too rich for the EEC to optimize by the addition of fuel. Also, at high altitudes with only the HMU controlling fuel flow, the engine is subject to compressor stalls on slam accelerations.

The deenrichment solenoid valve control system includes the following two altitude switches:

- 14K switch
- 10K switch

The 14K switch:

- Energizes the solenoid when climbing through 14,000 feet, closing the valve
  - This leans the HMUs fuel schedule to allow the EEC to control engine acceleration and deceleration at high altitudes, and prevent power-lever-induced compressor stalls.
- Deenergizes the solenoid when descending through 14,000 feet, opening the valve
  - If the HMU deenrichment solenoid failed to deenergize on descent, the HMU fuel schedule would remain lean. Reducing the power lever to idle could result in engine flame out due to insufficient fuel flow.

The 10K switch: triggers an EEC warning descending through approximately 10,000 feet if the HMU deenrichment solenoid fails to deenergize

A three-chime aural alert sounds, the red EEC 1 or EEC 2 light on the glareshield panel illuminates, a voice warning “ENGINE CONTROL” is given, and the master WARNING lights flash.

When this occurs, crew action should be performed as follows:

- Power rating selector MAN
- Refer to checklist
- Maintain torque above 20% with the power lever
- Make slow power lever movements to minimize flame-out possibility (HMU solenoid may be stuck in the high-altitude, lean position).

HMU Solenoid Test—After engine start or during taxi out, the EECs are alternately switched off. Under most conditions, this should cause a slight drop in fuel flow and, consequently, N_H. However, if the N_H drops below 50% or the engine flames out, the HMU solenoid may be stuck in the energized position. If the problem is verified (by duplicating the N_H drop when repeating the check with the bleed OFF), maintenance action is required prior to flight.

EEC Failures—When an electrical, sensor, or software malfunction occurs, the system reverts to manual control (HMU only). Reversion to manual control is identified by the following:

- Power loss, recoverable by advancing the power lever
- Power lever stagger for equivalent torque
- Slower than normal acceleration
- Faster than normal deceleration
- Loss of the fixed torque climb feature

When the EEC provides a signal to revert to the manual mode, the following occurs:

- Torque motor is inhibited
- White MANUAL light on the EEC panel illuminates
- Relevant red EEC 1 or EEC 2 light on the glareshield panel illuminates
- Three-chime aural alert sounds, and "ENGINE CONTROL" voice warning is given
- Master WARNING lights flash

Fail Fixed—If a peripheral EEC sensor should fail the EEC 1 or EEC 2 warnings are generated, but the EEC will not revert to manual. The EEC torque motor current is frozen at its present value to prevent engine spool backs during takeoff. This is called a fail-fixed malfunction and occurs at altitudes less than 14,000 feet.

On aborted takeoffs, this function enhances reverse thrust asymmetry. The fail-fixed function is canceled when the EEC is reset or turned off.

With the EEC inoperative, consult the approved *Airplane Flight Manual,* "Limitations" section, for the maximum aircraft operating altitude.

EEC Safety Features

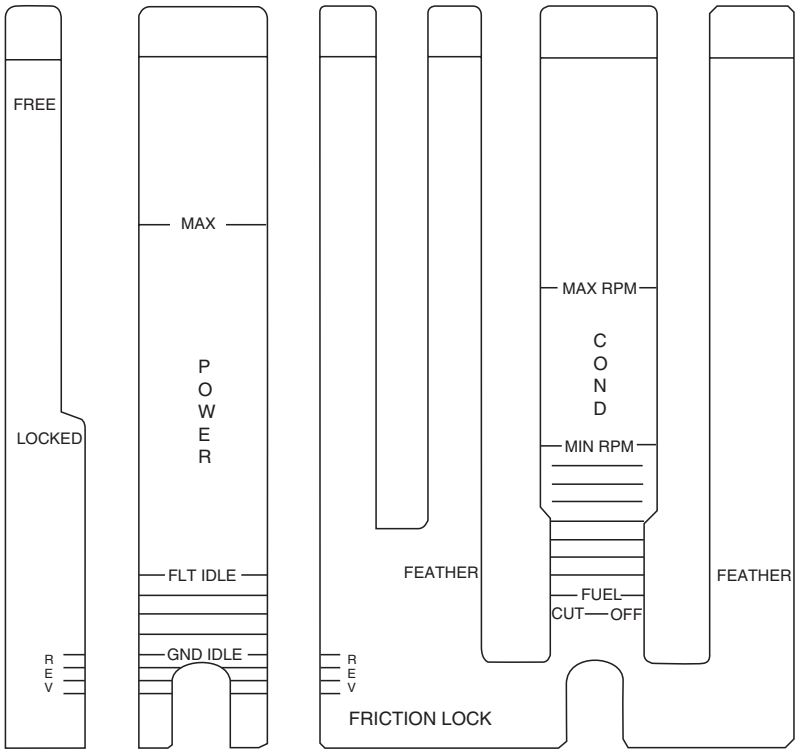
- N_H overspeed protection at 103% (EEC switches off)
- N_H underspeed protection below 60% N_H (EEC switches off)
- N_P >100% or N_P accelerating faster than a predetermined rate, EEC reduces fuel flow to control N_P within 106% (feature enabled when N_P >80%)

Engine Power Controls

Engine power control is accomplished by two pairs of levers that control power output, fuel shutoff, propeller pitch and speed, and propeller feathering by mechanical means.

The power levers and condition levers are located on the center pedestal console (Figure 7-13).

Power Levers—The power levers are located on the left side of the console. They are mechanically connected to, and the primary input for, the HMU (Figure 7-14). The power levers are also interconnected with the propeller control unit (PCU) for direct blade angle control (BETA mode) and electrically connected to the EEC.



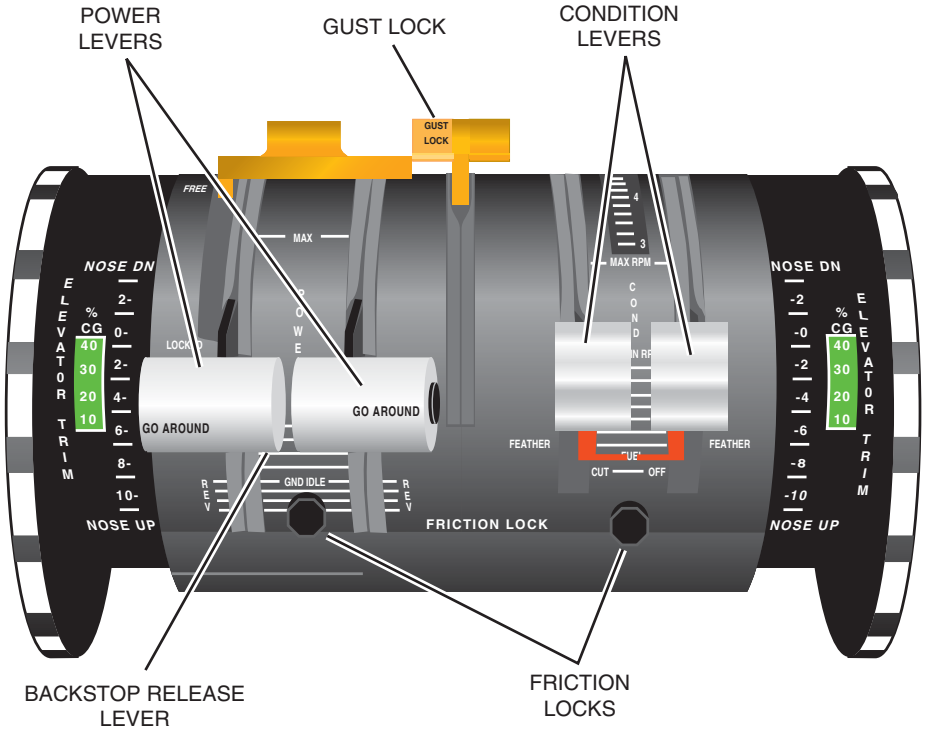


Figure 7-13. Center Pedestal Console

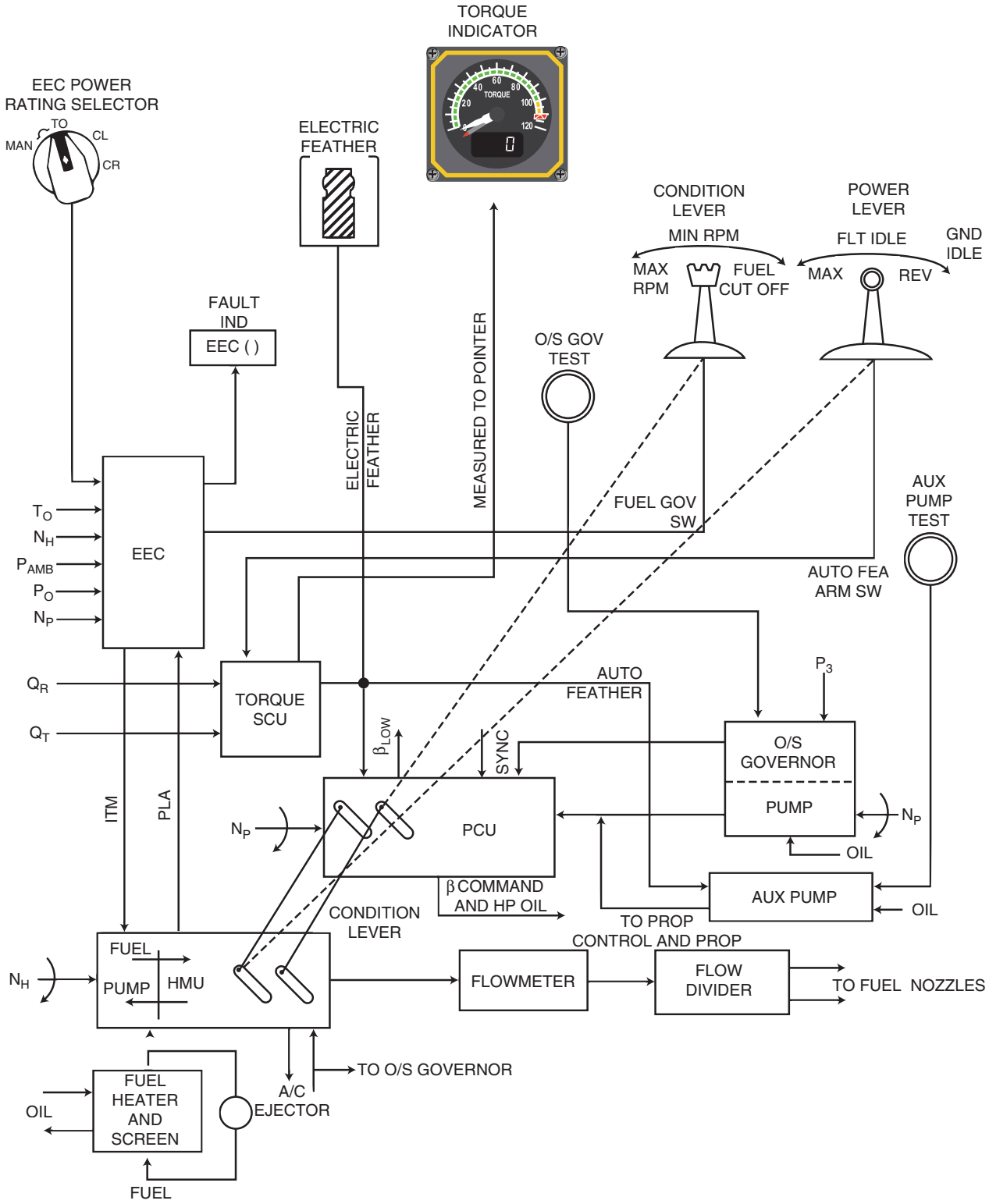


Figure 7-14. Engine Control System Schematic

Control Ranges—Each power lever has two distinct control ranges separated by a mechanical stop: the flight operation range and the ground operation range (BETA range)

The ***flight range*** is from the flight idle (FLT IDLE) position to maximum power (MAX). Flight idle stops prevent the power levers from inadvertently being moved back into the ground operation range.

In the flight range, the power lever controls the power output or spool-up (N_H) of the engine. Propeller rpm (N_P) remains constant at the value selected by the control lever . (The PCU compensates for the increase or decrease in engine power by varying the blade angle to maintain the selected rpm.)

The power levers, through the HMU, increase or decrease the basic fuel flow schedule to the engine. The power lever position angle is transmitted to the EEC through the potentiometer on the HMU power lever valve shaft.

To enter the ***ground (beta) range***, the flight idle stops must be overridden by pulling up on the stop release levers installed between the power levers.

The ground range is from just below the flight idle stop (FLT IDLE), through the ground idle detent (GND IDLE), to reverse (REV).

In the beta range, the power lever controls propeller pitch for taxiing and reverse during ground operations. The power levers are mechanically connected to the PCU to allow beta range propeller blade angle scheduling.

There is no N_H spool-up in the ground idle range. There is spool-up in the reverse range to maintain the propeller speed as the blade angle continues toward reverse. (Max N_P in reverse is 80%.)

The flight idle stop should only be actuated on the ground after the main landing gear and nosewheel are in contact with the runway.

The crew must be sure that the flight idle stop levers are not accidentally actuated in flight.

WARNING

Selecting the power control levers below flight idle in flight is known to cause catastrophic propeller overspeeds and is expressly forbidden by the *AFM* and *Airworthiness Directive*.

Aircraft Post-Mod. SB 120-076-0009 or SN 120-178 and subsequent are equipped with an electrical flight idle stop device that prevents power lever movement below flight idle in flight. The system utilizes solenoid locks located in the engine nacelles that are activated when the main gear proximity sensors indicate a weight-off-wheels condition. A time delay of 10 seconds is incorporated into the system.

Airplanes Post-Mod. SB 120-76-0018 or SN 120.345 and subsequent are equipped with an IDLE 1 (2) UNLK light that illuminates when the flight idle stop device is not operational in flight (Figure 7-15).

This stop is electrically actuated by means of a solenoid installed in each of the nacelles; when the airplane is in flight, the solenoid is energized and the power lever cannot be set below flight idle; when the airplane is on the ground, the solenoid is deenergized and the power lever may be moved below flight idle, allowing ground idle and reverse settings.

The stop activation (in flight) and deactivation (on the ground) is automatically commanded by the weight-on-wheels switches installed on the main landing gear shock absorbers (air/ground system).

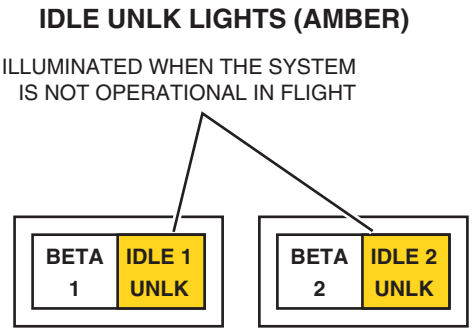


Figure 15. IDLE 1 (2) UNLK Light



CAUTION

Never set power lever below flight idle in flight. Apply reverse only after the nosewheel is on the ground.

The power levers activate the following switches installed in the control pedestal inner structure:

- Takeoff warning switches
- Autofeather system switches
- Secondary low-pitch stop switch
- Landing gear warning switches
- Prepressurization switch (left power lever only)

A gust lock system is installed to prevent takeoffs with the controls locked. The power levers are blocked from being moved forward of flight idle when the gust lock is in the LOCKED position. They have full range of movement in the unlocked, FREE, position.

Friction locks adjusts both the power and condition levers resistance to motion. The amount of resistance is controlled by knobs located between each set of levers on the lower end of the pedestal.

Condition Levers—The condition levers are located on the right side of the console. They are mechanically connected to the PCU to control propeller pitch for constant speed and feathering operations (see Figure 7-14). The condition levers are also connected to the HMU to control engine shutdown.

Each control lever performs the following operations:

- Propeller constant-speed control
- Propeller mechanical feathering control
- Engine fuel shutoff control

Control Range/Positions—Each condition lever has the following control range and positions separated by mechanical stops:

- RPM select
- FEATHER
- FUEL CUTOFF

In the ***RPM select*** range, an infinite number of rpm selections are available between MAX RPM (100% N_P) and MIN RPM (80% N_P):

- Normal climb is conducted at 90 or 100% N_P
- Normal cruise is conducted at 85% N_P
- Descents at 85%
- Landings at 100%
- MAX RPM—In flight, selects 100% NP constant speed operation

  During taxiing, selects 65% N_P speed governing as provided by the EEC when the power lever is scheduling engine power at less than required for constant-speed operation.

- MIN RPM—In flight, 80% is the minimum governing rpm. During taxiing, selects 50% N_P speed governing as provided by the EEC when the power lever is scheduling engine power at less than required for constant-speed operation.

  The minimum rpm stop prevents the condition lever from being moved into the feather position inadvertently when selecting MIN RPM.

  The minimum rpm stop may be overridden by lifting the condition lever up and aft into the feather position.

The FEATHER position mechanically feathers the propeller and cancels N_P fuel governing.

A feather stop prevents the condition lever from inadvertently being moved into the fuel cutoff position when selecting feather.

The feather stop may be removed by pushing forward on the stop release lever and moving the condition lever into the fuel cutoff position.

The FUEL CUTOFF position shuts down the engine by mechanically shutting off the fuel to the engine at the flow divider.

The condition levers activate the following switches installed in the control pedestal inner structure:

- High/low N_P fuel governing switches
- N_P fuel governing cancel switches

Emergency Shutdown Control—The emergency shutdown control for each engine is a fire handle, located on the fire protection system panel. Pulling the handle electrically commands simultaneous engine shutdown and propeller feathering.

When a fire handle is pulled, the following occurs:

- Firewall fuel shutoff valve is closed
- Firewall hydraulic shutoff valve is closed
- Engine bleed-air valve is closed
- Deicing flow control valve is closed
- Electrical feathering system is actuated
- Fire extinguishing system is armed

ENGINE MONITORING

Engine Indicating System

Each engine indicating system includes one or more sensors. The value measured by the sensors is converted into electrical signals and sent to the respective indicators on the center instrument panel (Figure 7-16). These instruments provide a digital indication on a liquid crystal display (LCD), and/or an analog indication by moving a pointer over a graduated scale.

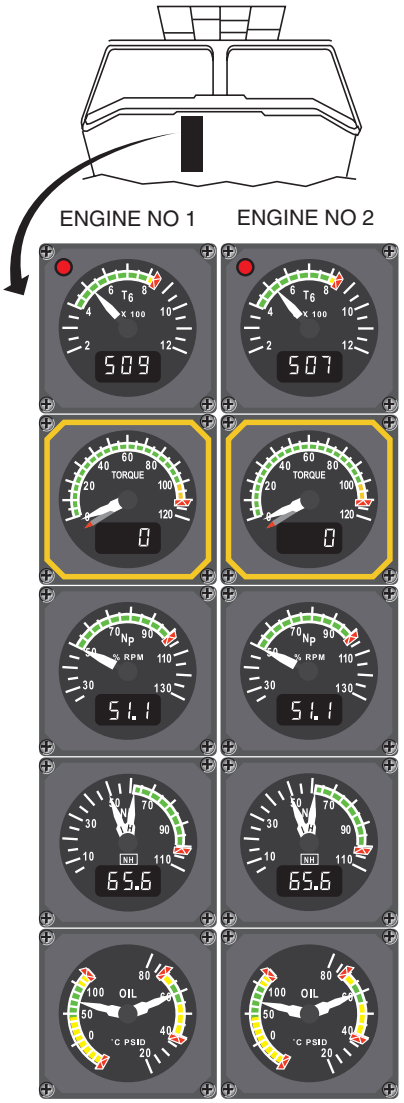


Figure 7-16. Engine Instrument Location

All engine indicating systems have the following operating characteristics:

- Engine instruments are 28-VDC powered
- Engine instrument lighting is 5-VDC powered

When a sensor signal loss occurs:

- The LCD indicates zero
- The pointer moves to the first low mark on the scale

When a power supply failure occurs:

- The LCD blanks
- The pointer moves off the low end of the indicating scale

T6 Indicators—The T_6 interturbine temperature indicator amplifies the signal voltage from the thermocouples and changes it into presentable analog and digital indications.

The T_6 indicator incorporates a red over-temperature warning light at the upper left corner of the dial. When the temperature exceeds 816°C, this warning light illuminates, a three-chime aural alert sounds, a voice warning of "T_6" is given, and the master WARNING lights flash.

The warnings are canceled when the T_6 temperature drops below 816°C.

The indicators are powered from their respective emergency DC bus.

Torque Indicating System—The engine torque indicating system (Figure 7-17) is the primary reference in the aircraft for setting engine power.

Each torque system consists of the following components:

- Torque shaft
- Reference shaft
- Torque pick-up sensor
- Signal conditioning unit
- Torque indicator

The ***torque shaft***, located in the engine interconnecting case, is the shaft that physically connects the TMM to the RGB. It is the drive shaft that transmits power from the engine to the propeller.

The torque shaft fits inside a sleeve, or ***reference shaft***, that is connected only at the TMM end.

At the RGB end of both shafts (the unattached end of the reference shaft) are toothed reference rings mounted with the teeth of each interposed.

The variation of the space between the teeth is proportional to the engine torque applied to the torque shaft. (The torque shafts twists while the reference shaft, unattached at one end, does not.)

The ***torque pick-up sensor*** detects the difference in spacing between the teeth and sends the information to the SCU.

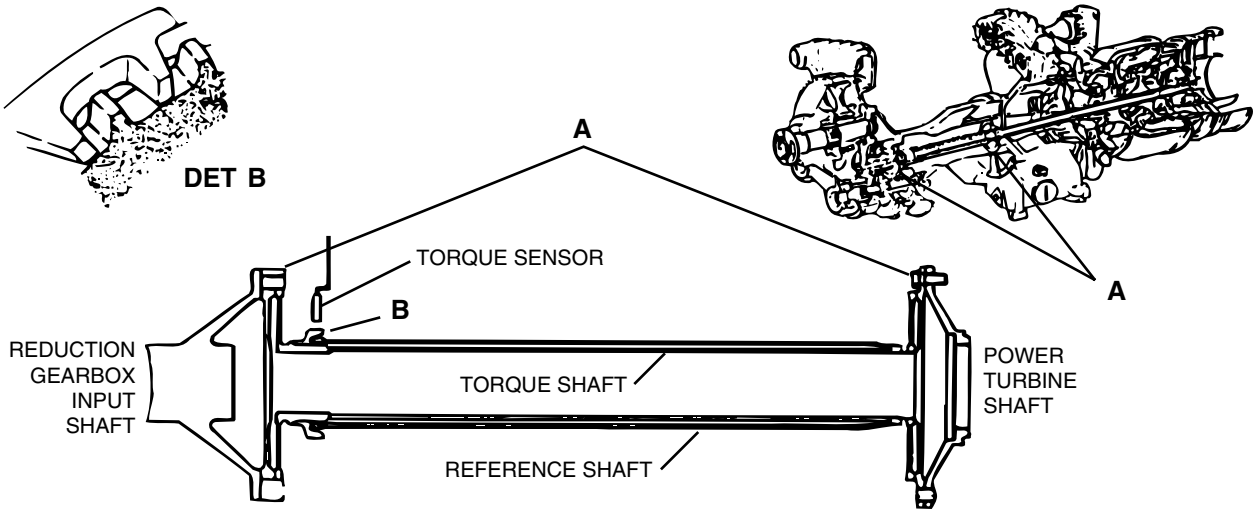


Figure 7-17. Torque Indicating System

The ***signal conditioning unit*** (SCU) generates a calibrated torque signal, which drives the analog pointer and LCD of the ***torque indicator*** on the main instrument panel. It also supplies information to the autofeather system for arming and activation.

The torque indicator of each engine is powered by the respective emergency DC bus.

Speed Indicating System—The following speed sensors transmit proportional train of pulses to the digital and/or analog indicator. In addition, they also send signals to the EEC which uses them for engine fuel control scheduling.

The propeller speed (N_P) sensor is located on the RGB.

The high-pressure spool speed (N_H) sensor, is located on the upper portion of the TMM accessories gearbox.

The low-pressure spool speed (N_L) sensor is located on the left side of the engine near the compressor case.

NOTE

The LCD display on the N_H /N_L indicator displays N_H.

The indicators are powered from the respective DC buses.

Oil Indicating System—The engine oil indicating system is discussed in the Engine Oil System section of this chapter.

A summary of engine indicators and their parameters for the PW 118 and PW 118A engines is presented in Figure 7-18.

Figure 7-19 illustrates the powerplant control panels and indicates the functions of each.

ENGINE NO. 1 **ENGINE NO. 2**





INTERTURBINE TEMPERATURE INDICATOR

INCORPORATES A RED WARNING LIGHT
RED LIGHT ON: ABOVE 816° C
RED LIGHT OFF: BELOW 816° C
GREEN ARC: 400 TO 800° C
YELLOW ARC: 800 TO 816° C
RED RADIAL. 816° C





TORQUE INDICATOR

INCORPORATES A BUG, WHICH RECEIVES SIGNAL FROM EEC
GREEN ARC: 0 TO 100%
YELLOW ARC: 100 TO 110%
RED RADIAL. 110%

NOTE:
TORQUE BUGS ARE NOT ACTIVATED





PROPELLER SPEED INDICATOR (N_P)

GREEN ARC: 50 TO 100%
RED RADIAL: 100%





HIGH-PRESSURE SPOOL SPEED/LOW PRESSURE SPOOL SPEED INDICATOR (N_H/N_L)

GREEN ARC: 62 TO 100% (N_H ONLY)
PW 118A: 62 TO 102% (N_H ONLY)
RED RADIAL: 100%
PW 118A: 102%





OIL TEMPERATURE/PRESSURE INDICATOR

OIL TEMPERATURE:

GREEN ARC:	45 TO 100° C
PW 118A:	45 TO 115° C
YELLOW ARC:	–40 TO 45° C
AND	110 TO 115° C
PW 118A:	–40 TO 45° C
AND	115 TO 125° C
RED RADIAL.	–40 AND 115° C
PW 118A:	–40 AND 125° C

OIL PRESSURE:

GREEEN ARC: 55 TO 65 PSID

YELLOW ARC: 40 TO 55 PSID
AND 65 TO 70 PSID

RED RADIAL: 40 AND 70 PSID

Figure 7-18. Engine Instruments

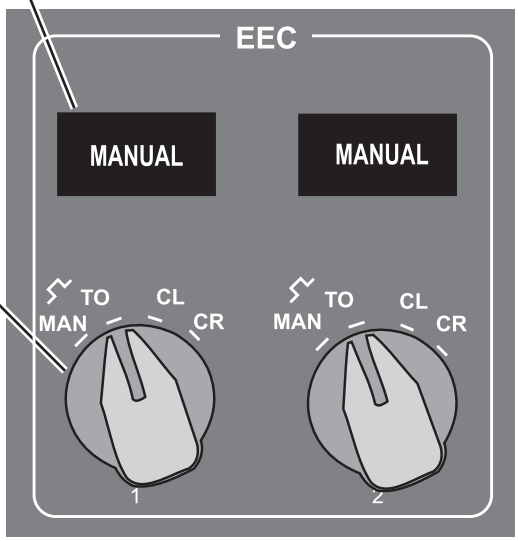
ENGINE NO. 1 MANUAL LIGHT (WHITE)

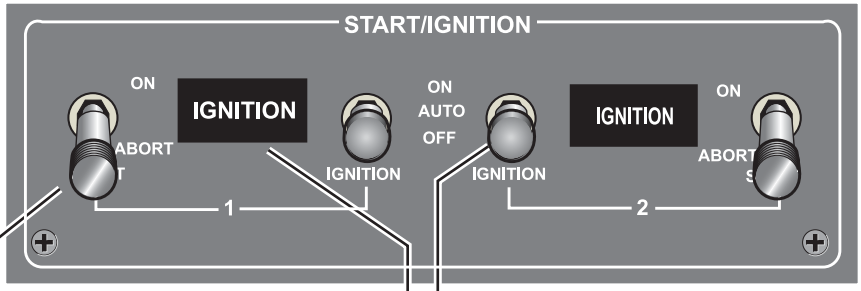
ILLUMINATED—WHEN POSITIONING THE EEC POWER RATING SELECTOR ON MAN POSITION OR WHEN EEC REVERTS AUTOMATICALLY TO MANUAL MODE

POWER RATING SELECTOR

MAN—SELECTS THE MANUAL MODE
TO—SELECTS THE TAKEOFF CONDITION
CL—SELECTS THE CLIMB CONDITION
CR—SELECTS THE CRUISE CONDITION

NOTE:
PULL KNOB TO CHANGE FROM MAN TO TO POSITIONS.
SINCE BUGS ARE DISABLED, THERE IS NO DIFFERENCE BETWEEN TO, CL, AND CR POSITIONS.





START SWITCH

ON—SIGNALS THE GCU TO BEGIN THE STARTING CYCLE. THE GCU CLOSES THE START CONTACTOR AND ACTIVATES THE IGNITION CIRCUIT. THE STARTING CYCLE ENDS WHEN THE GCU RECEIVES A SIGNAL FROM THE N_H SENSOR AT 50% N_H.

ABORT—INTERRUPTS THE ENGINE START CYCLE BY SENDING AN ARTIFICIAL 50% N_H SIGNAL TO THE GCU

IGNITION CONTROL SWITCH

ON—THE IGNITION CIRCUIT IS CONTINUOUSLY ENERGIZED.

AUTO—THE IGNITION CIRCUIT IS AUTOMATICALLY ENERGIZED WHENEVER THE STARTING CYCLE IS INITIATED BY SELECTING THE ENGINE START SWITCH TO START. THE IGNITION CIRCUIT IS AUTOMATICALLY DEENERGIZED AT 50% N_H BY THE GCU. THE IGNITION CIRCUIT MAY BE MANUALLY DEENERGIZED BY MOVING THE ENGINE START SWITCH TO THE ABORT POSITION.

OFF—THE IGNITION CIRCUIT IS DEENERGIZED, (EVEN IF THE CYCLE IS INITIATED).

IGNITION LIGHT (WHITE)

ILLUMINATED—WHEN THE IGNITION CIRCUIT IS ENERGIZED



GLARESHIELD PANEL

EEC LIGHTS (RED)

ILLUMINATED—WHEN ANY FAILURE IS DETECTED ON RESPECTIVE EEC OR ON HMU ENRICH SOLENOID VALVE

Figure 7-19. Powerplant Control Panels

PROPELLER

GENERAL

PW-118 and PW-118A engines are equipped with Hamilton Standard 14RF-9, four-bladed, "commuter" propellers (Figure 7-20). The propeller is driven by the engine power turbine assembly through the reduction gearbox.

The reduction ratio between the power turbine output shaft and the propeller is approximately 15:1 (20,000 rpm of the power turbine corresponds to approximately 1,300 rpm of the propeller).

Propeller maximum governed speed varies from 1,300 to 1,309 rpm (100.0-100.7% N_P). The pitch adjustment range varies from +79.2° (feather) to –15.0° (reverse), measured at the 42-inch blade station.

The oil used for propeller control is supplied by the engine lubrication system.

The following subjects are covered in this section:

- Propeller assembly
- Propeller control components
- Propeller safety features
- Propeller operation
- Propeller synchronization system

PROPELLER ASSEMBLY

The propellers are constant-speed, full feathering reversible units that feature a composite blade design. The four blades are attached to a one-piece aluminum barrel. Inside the hub is a dome assembly that houses a double-acting hydraulic pitch change servomechanism. The hub and dome are enclosed within an aerodynamic spinner.



Figure 7-20. Hamilton Standard 14RF-9 "Commuter" Propeller

Propeller Blades

The primary structural member of the propeller blade is a solid aluminum spar. The spar is covered with a fiberglass shell and filled with low-density polyurethane foam to form the blade (Figure 7-21).

To provide leading edge erosion protection, the blade incorporates a nickel sheath over the outboard section (station 42 to the tip) and an abrasion resistant polyurethane sheath over the inboard section (station 42 to the blade root).

Each blade is also fitted with imbedded leading edge electrical deicers, lightning damage protection features, and an overall erosion resistant coating.

Propeller Specifications

The 14RF-9 propeller specifications are as follows:

- Diameter..... 10 ft 6 in.
- Flight idle
  blade angle..... 17.6° at station 42
- Ground idle
  blade angle..... –4.5° at station 42
- Reverse
  blade angle..... –15.0° at station 42
- Feather
  blade angle..... 79.2° at station 42
- Governing speed
  MAX (100% N_P) ..... 1,300 rpm
- Governing speed
  MIN (80% N_P) ..... 1,040 rpm
- Tip speed
  at 100% N_P ..... 715 fps (0.71 M)
- Tip speed
  at 80% N_P ..... 572 fps (0.57 M)

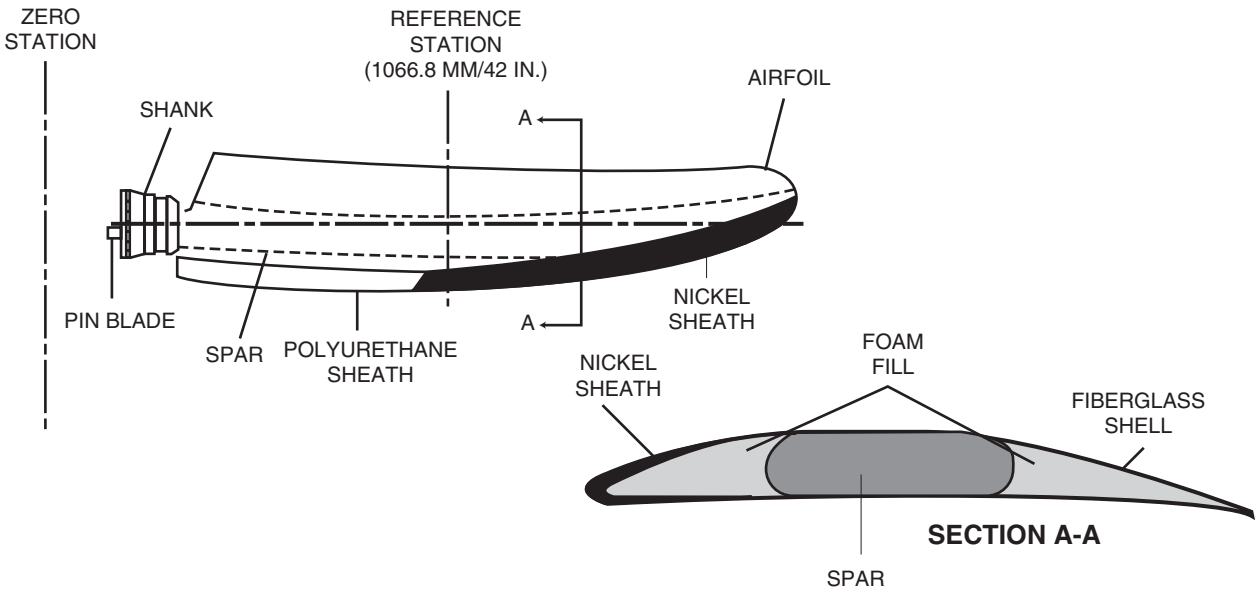


Figure 7-21. Blade Construction

PROPELLER CONTROL COMPONENTS

The propeller control system is comprised of the following components (Figure 7-22):

- Auxiliary oil tank
- Auxiliary oil pump
- Propeller oil pump
- Propeller control unit
- Propeller servomechanism

Auxiliary Oil Tank

The auxiliary oil tank is the propeller system oil tank. It is a pressurized, 0.3 U.S. gallon tank, that is integral to the reduction gearbox. The tank is continuously supplied by the engine oil system at approximately 100 psi and feeds both the electric auxiliary oil pump and the mechanical main propeller oil pump.

The auxiliary oil tank always keeps a minimum oil level for feathering the propeller in an emergency. It has sufficient quantity for one full-blade actuation from flight idle to full feather. This reserve oil supply is not available to the main propeller oil pump.

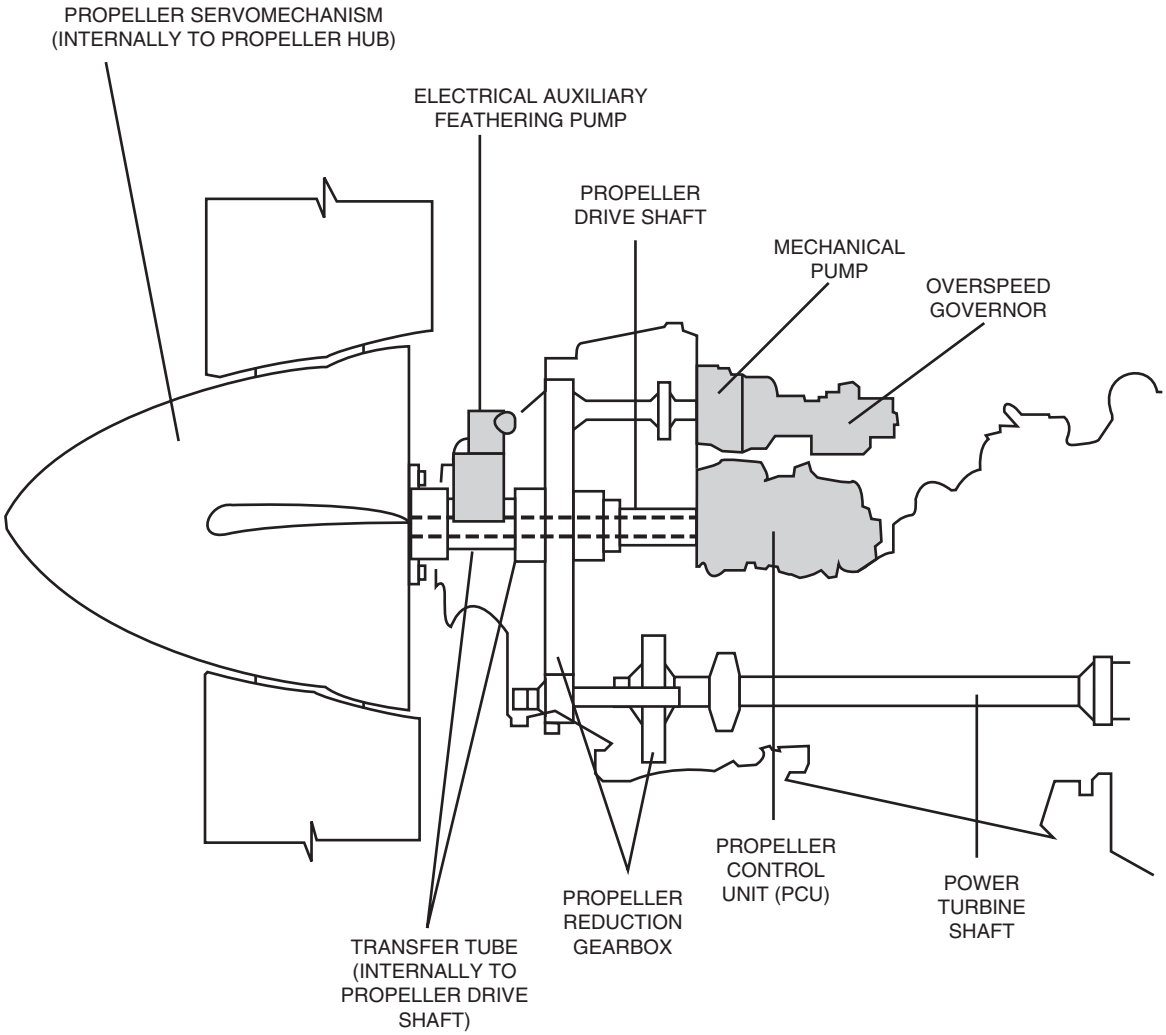


Figure 7-22. Propeller Control Components

Auxiliary Oil Pump

The auxiliary oil pump, mounted on the front of the RGB, gets its oil supply from the auxiliary oil tank (Figure 7-23). The auxiliary oil pump is an electric motor-driven, positive displacement pump. It provides approximately 820 psi pressure for the propeller feathering system when the mechanical prop oil pump is inoperative. If the auxiliary oil tank is depleted the auxiliary oil pump has no effect on blade angle.

The electrical auxiliary oil pump operates when the following actions are conducted:

- Auxiliary pump test button is pressed
- Autofeather system is triggered
- Electrical feather switch is operated
- Fire handle is pulled

Auxiliary Oil Pump Test—There are two auxiliary oil pump test buttons located on the overhead POWER PLANT control panel. Each auxiliary oil pump is to be tested before engine start (MIN oil temperature to feather the propeller is 0°C).

The proper test sequence is as follows:

1. Power levers ..... REVERSE
2. Control lever ..... MIN
3. TEST button ..... PRESS
4. Observe blade angle ..... DECREASE
5. Observe Beta light..... ON
6. TEST button ..... RELEASE

NOTE

This test is normally followed by the electric feather switch test. The auxiliary oil pump motor duty cycle limit is 20 seconds, which is more than the time needed to move the blade from flight idle to full feather blade angle. The auxiliary pump is powered by emergency 28-VDC buses and is provided with a timer to limit its operation time to 20 seconds to protect the motor from burning out and/or the pump from being run dry when the auxiliary oil tank is empty.

The timer may be reset by the following:

- Releasing the test switch
- Turning off the autofeather system
- Turning off the electrical feather switch
- Pushing in the fire handle

If the auxiliary oil tank is empty, it will be necessary to carry out a dry motoring of the engine to replenish it. The auxiliary oil pump outlet incorporates a check valve to prevent reverse oil flow when the propeller oil pump is operating and the auxiliary oil pump is not (Figure 7-23).

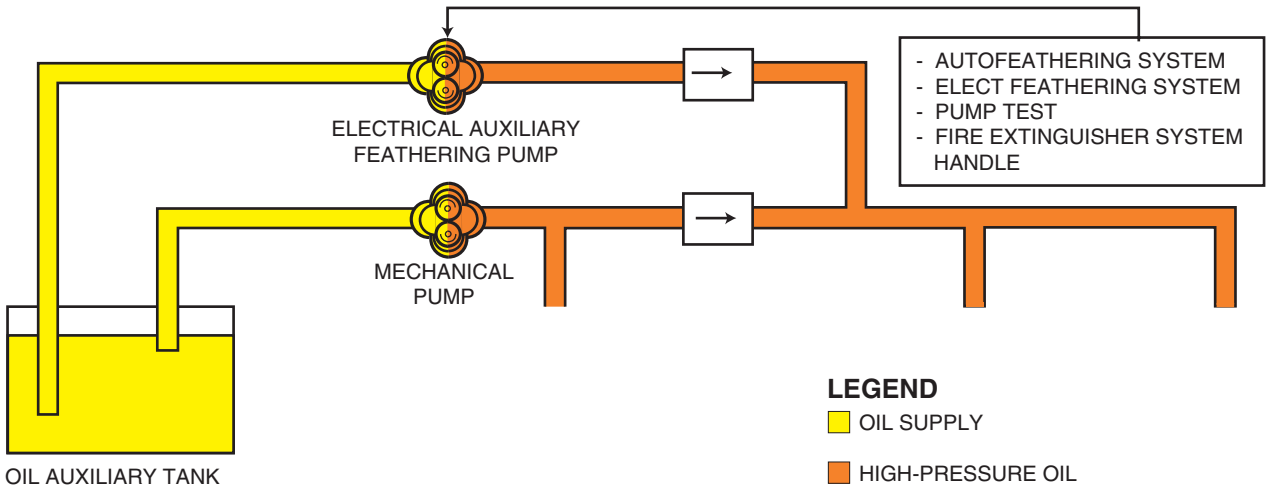


Figure 7-23. Propeller Oil Supply

Propeller Oil Pump

The propeller oil pump, driven by the RGB, boosts the engine oil pressure from 100 psi to 780 psi. It supplies the propeller control unit, the overspeed governor, and the propeller servomechanism. The propeller oil pump outlet incorporates a check valve to prevent reverse oil flow when the propeller oil pump is inoperative and the auxiliary oil pump is actuated (see Figure 7-23).

Propeller Control Unit (PCU)

The PCU (Figure 7-24) is the main control component in the propeller system. It is responsible for controlling propeller speed and selecting propeller pitch.

The PCU is commanded by the condition and power levers in the cockpit. The condition lever controls propeller speed (N_P) in flight and mechanical feathering. The power lever controls the flight low-pitch lock schedule and propeller pitch during taxi and reverse operations.

The PCU servo piston and pitch change screw are connected to the transfer tube end. The opposite end is connected to the selector valve in the propeller servomechanism.

The propeller is normally controlled by the PCU. The PCU is mounted on the RGB, directly behind the propeller hub. It contains the following:

- Primary governor
- Least selector valve
- Reverse valve
- Beta valve
- Mechanical feathering valve
- Electrical feathering solenoid
- Low blade angle switch
- Synchrophaser torque motor

Primary Governor—The primary governor, also known as the propeller speed governing section, is supplied high-pressure oil (780 ±30 psi) from the propeller oil pump. Commanded by the condition lever, it controls propeller pitch and speed in flight. The speed governing section is operational between 80 and 100% N_P.

The governor reduces and meters high-pressure oil to control the propeller pitch for constant-speed governing in response to flyweight force versus the speed selected by the condition lever. The metered, or control pressure (P_C), is one half of the supply pressure (P_S) in an onspeed condition.

The speed governor section increases or decreases propeller pitch until the propeller speed (N_P), selected by the condition lever, is reached.

During engine operations with a governed propeller (pitch control by means of the speed governor section)—for every condition lever position between MIN and MAX RPM there is a specific propeller speed, and for every power lever position from FLT IDLE to MAX PWR there is a predetermined minimum blade angle (flight low pitch).

The flight low pitch is controlled by the beta valve. It limits the speed governor action, not allowing a propeller pitch below minimum established values.

Least Selector Valve—The least selector valve acts as a hydraulic discriminator between the speed governor section and the overspeed governor (Figure 7-24). As metered pressure (P_C) from the primary governor is one half of the supply pressure through the overspeed governor, the least selector remains shifted to the primary governor. The metered pressure is routed through the reverse valve to the servo piston to control blade angle during constant-speed governing operation.

If an overspeed occurs, supply pressure from the overspeed governor is dumped and becomes the lesser pressure. The least selector valve shifts to drain metered pressure as necessary to limit the overspeed.

Reverse Valve—The reverse valve, controlled by the power lever, selects either the speed governor section or the beta valve to vary the control pressure (P_C) and, consequently, propeller pitch.

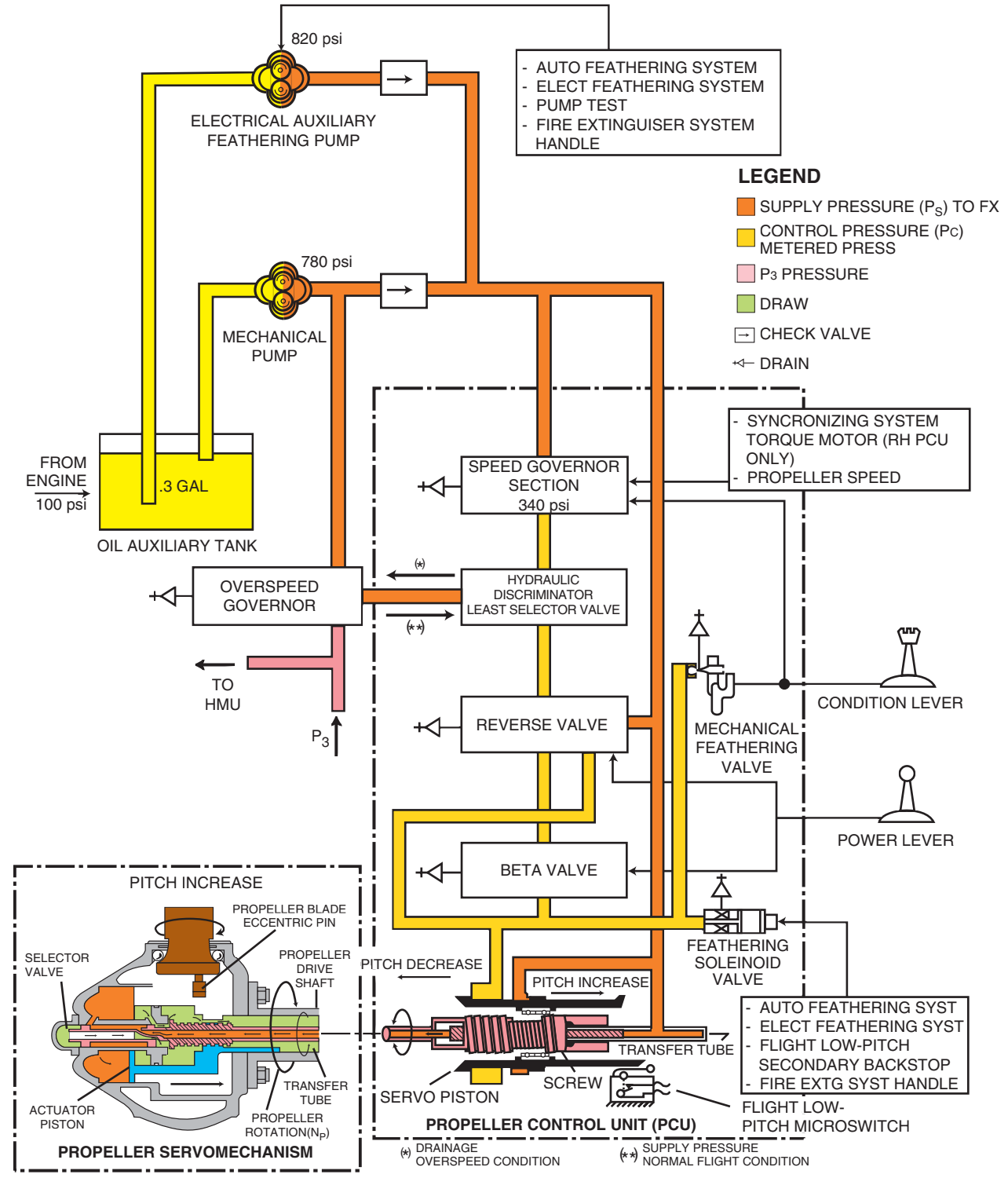


Figure 7-24. Propeller Control System

For power lever positions from FLT IDLE to MAX PWR, (typical in-flight positions), the reverse valve uses the speed governor section to control propeller pitch, down to the flight low pitch backstop limit.

For power lever positions from FLT IDLE down to maximum reverse (typical on ground positions), the reverse valve hydraulically blocks the speed governor section and propeller pitch control is by the Beta valve.

WARNING

Never select the power levers below FLT IDLE in flight.

In flight, the power lever operating range must be limited to positions equal to or above FLT IDLE. It ensures governed blade angles (speed governor section controlling the propeller pitch) that are far above those commanded by the Beta valve.

The actuation of the reverse valve in flight (power lever below FLT IDLE) disables both the speed governor section and the overspeed governor, and controls the propeller pitch angles using the Beta valve schedule (minimum blade angle schedule). The Beta schedule angles are much lower, towards flat pitch, than the angles scheduled by the speed governor section. This reduced blade angle in flight causes the propeller to extract energy from the airstream, driving the power turbine shaft to very high overspeed. Serious damage to the engine and excessive propeller drag may result.

Beta Valve—During ground operation, as the power lever is moved aft of flight idle into the ground range, the Beta cam schedules the Beta valve to command blade angle.

The Beta valve is commanded by the power lever. In flight it controls the primary low pitch backstop, on the ground it controls propeller pitch for taxi and reverse operations.

On the ground, with the propeller operating in the Beta mode (pitch control by means of the Beta valve)—for every power lever position from maximum reverse to just below FLT IDLE there is a corresponding predetermined blade angle.

During taxi and reverse operations, propeller speed (N_P) is controlled by the EEC:

- With the condition lever at MAX RPM and the power lever between FLT IDLE and 10° PLA (just below GND IDLE), the EEC ensures a minimum speed of 65% N_P.
- With the condition lever between MAX and MIN RPM, the EEC assures a minimum speed of 50% N_P for every power lever position
- With EEC OFF, the propeller speed depends upon pressure altitude and air density.

In flight, and with the power lever between FLT IDLE and MAX PWR, the Beta valve acts as a propeller pitch backstop. This ensures that blade angles will not be lower than the pre-established minimums.

At FLT IDLE, the low pitch is 17.6° (at blade station 42). This low-pitch value increases as the power lever is moved above FLT IDLE.

With the power lever below FLT IDLE, the propeller blades are positioned below the flight low pitch. This condition is indicated by the illumination of the Beta light on the glareshield. The Beta light illuminates anytime blade angle is 12.6° or less, (5° below the flight low pitch corresponding to FLT IDLE position). The Beta lights are normally illuminated during taxi and reverse operations.

NOTE

Beta light illumination in flight is abnormal. It indicates either Beta valve failure in limiting the flight low pitch, or a failure in the Beta indication circuit.

Mechanical Feathering Valve—The mechanical feathering valve is actuated by the feather position of the condition lever. It dumps metered pressure and allows supply pressure to feather the propeller.

Electrical Feathering Solenoid —The electrical feathering solenoid valve is activated by the following:

- Autofeather system
- Electric feathering system
- Flight low pitch secondary backstop
- Fire handle

When activated, the solenoid valve dumps metered pressure (P_C) and allows supply pressure (P_S) to feather the propeller.

The secondary low pitch stop system dumps metered pressure (P_C) to allow supply pressure (P_S) to increase blade angle above 12.6° without fully feathering the prop.

Low Blade Angle Switch—The low blade angle switch, also called the Beta switch, is activated by the servo piston at 12.6° blade angle. It illuminates the Beta light and, if the power lever is in the flight range, activates the secondary low pitch backstop system.

Synchrophaser Torque Motor—The torque motor provides a corrective signal from the synchrophaser system to the right PCU primary governor section to adjust the speed and phase angle of the right propeller in relation to the left propeller.

PCU Operation

Following is a summary of PCU operation:

- The PCU controls the propeller pitch through the relative rotation of the oil transfer tube housed in the propeller shaft. (The transfer tube normally rotates in the same direction, and at the same speed, as the propeller.)
- Relative rotation of the oil transfer tube imparts an axial (forward and aft) movement to the selector valve in the propeller hub. The valve controls the routing of oil to one side or the other of the actuator piston.
- Relative rotation of the oil transfer tube is caused by a frictionless ballscrew mechanism in the PCU at the aft end of the transfer tube. The screw is actuated by a servo piston.
- The servo piston is a dual acting piston:
  - The aft side is continuously connected to supply pressure, P_S, oil from the propeller oil pump (780 ±30 psi), or the electrical auxiliary feathering pump (820 ±30 psi).
  - The forward side receives metered-control oil pressure, P_C, from the governor and control mechanisms within the PCU.
- The surface areas of the two sides of the piston are unbalanced. The aft (supply pressure) side is one half the area of the forward (control pressure) side. In a steady state, on speed condition, the control pressure is exactly one half the supply pressure.
- The servo piston is controlled by varying the control pressure on the forward side of the piston.
- Control pressure (P_C) (metered) drives propeller toward reverse, decreasing pitch, and therefore increasing propeller speed.
- Supply pressure (P_S) (constant) drives propeller toward feather, increasing pitch, and therefore decreasing propeller speed.

Propeller Servomechanism

The propeller servomechanism is totally responsible for propeller pitch change operation. It is a hydromechanical assembly located inside the propeller hub (Figure 7-25) and incorporates the following components:

- Oil transfer tube
- Pitch change valve
- Actuator piston
- Pitch lock gap

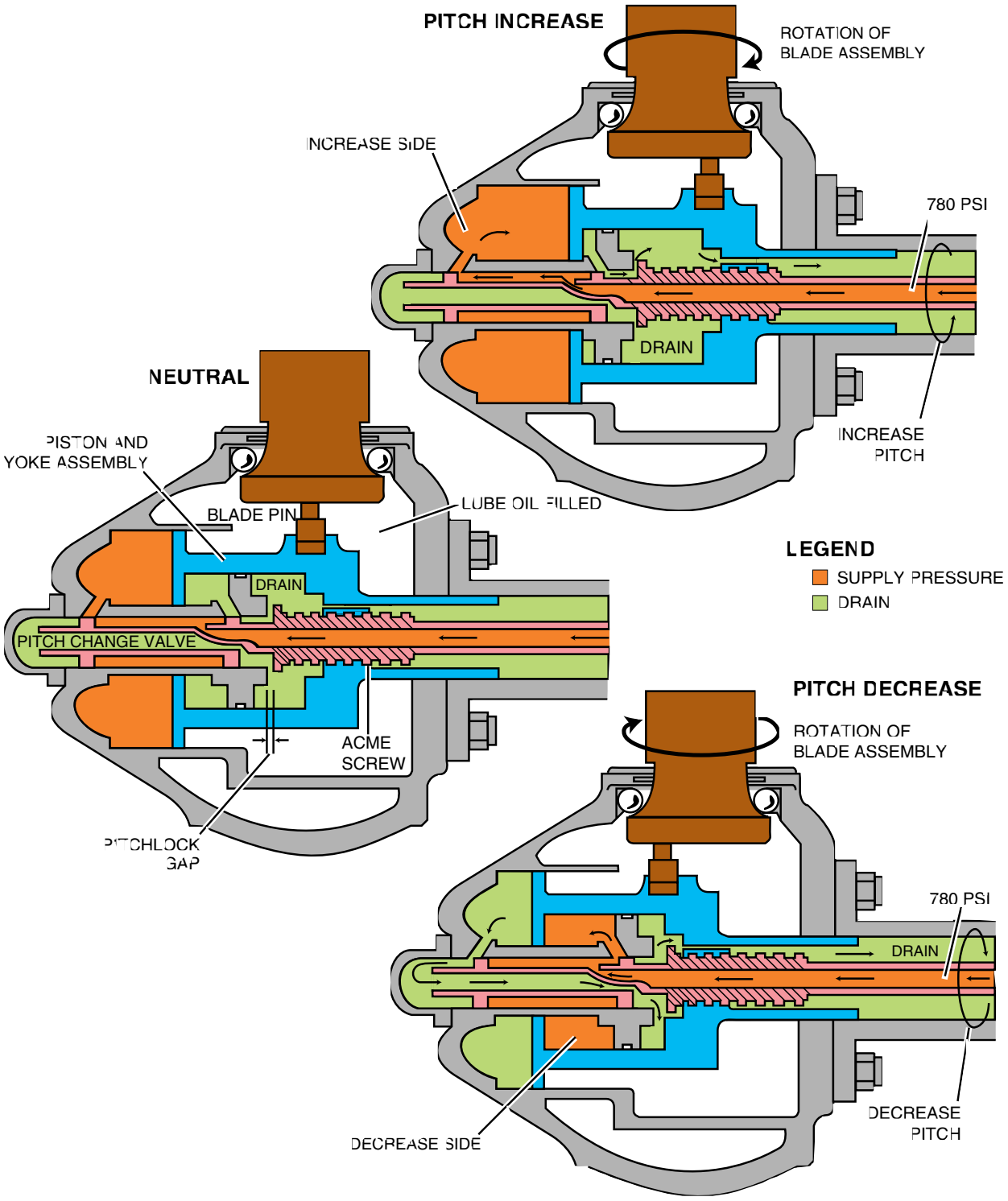


Figure 7-25. Propeller Servomechanism during Pitch Change Operations

The oil transfer tube, pitch change valve, and actuator piston all rotate with the propeller.

Oil Transfer Tube—The oil transfer tube carries supply pressure oil to the pitch change valve for operating the actuator position. The tube is rotated (relative to propeller rotation) by the frictionless ballscrew mechanism in the PCU (in response to fore and aft movement of the PCU servo piston). The tube rotation repositions the pitch change valve.

Pitch Change Valve—The pitch change valve assembly within the propeller servomechanism directs supply pressure oil from the transfer tube to the pitch increase or pitch decrease side of the actuator piston.

Part of the selector valve assembly is the reverse-threaded acme screw, which changes the transfer tube's relative rotation into axial movement.

Actuator Piston—The actuator piston is a dual-acting assembly that repositions the propeller blade's offcenter yoke assembly to change blade angle.

Supply oil is directed to the front side of the piston to increase pitch and to the aft side of the piston to decrease pitch.

When either side of the piston is receiving supply oil, the pitch change valve directs the opposite side of the piston to drain.

Pitch Lock Gap—The pitch lock gap is a small (.030–inch) space between the acme screw portion of the selector valve assembly and the propeller hub assembly. Oil from either side of the actuator piston passes through this gap to drain. Should the supply of oil to the PCU fail, the aerodynamic forces on the propeller (which tend to drive it toward flat pitch) close the pitch lock gap. This shuts off the drain to both sides of the actuator piston, creating a hydraulic lock.

Propeller Servomechanism Operation

- Unbalanced P_S/P_C pressure on the PCU servo piston causes a rotation of the oil transfer tube relative to the propeller shaft.
- In the servomechanism, the reverse threaded acme screw changes the tube's relative rotation to fore or aft selector valve displacement.
- The valve is positioned such that one of the actuator piston chambers is connected to the supply pressure, and the other is connected to drain.
  - Pressure to the forward side of the piston increases the blade angle toward feather.
  - Pressure to the aft side of the piston decreases the blade angle toward reverse.
- The actuator piston moves, repositioning both the selector valve and the propeller blade eccentric pin. (The selector valve and actuator piston continue to move as long as there is relative rotation of the transfer tube.)
- When the selected pitch is reached, the PCU servo piston returns to equilibrium, relative rotation of the transfer tube stops, the selector valve returns to its balanced (actuator piston chambers closed) position, and the actuator piston stops.

PROPELLER SAFETY FEATURES

The "commuter" propeller is equipped with the following automatic and manual safety features:

- Overspeed governor
- Pitch lock
- Primary low pitch stop
- Secondary low pitch stop
- Mechanical feathering
- Electrical feathering
- Emergency feathering and shutdown
- Autofeather system

Overspeed Governor

The overspeed governor is mounted on and driven by the propeller mechanical oil pump. Using oil pressure supplied by the pump, it limits propeller overspeed in case of a primary governor malfunction.

During normal onspeed operation, the overspeed governor routes high-pressure oil to the least selector valve in the PCU, keeping it shifted to the primary governor (see Figure 7-24).

When an overspeed occurs, the overspeed governor flyweights open and dump the high-pressure oil to drain. This shifts the least selector valve, dumping the metered pressure oil to drain. Supply pressure oil moves the servo piston to increase pitch. As the blade angle increases, propeller rpm decreases to maintain a steady state speed of 103% N_P.

In the event of hydraulic circuit failure, as indicated by an overspeed in excess of 103% N_P, the overspeed governor bleeds P_3 air from the HMU P_3 sensor and servo to reduce fuel flow to the engine. This causes the engine and propeller speed to decrease. The P_3 bleed is fully open at approximately 109%.

N_P Overspeed Governor Test—Two overspeed governor test buttons are located on the overhead panel (Figure 7-26), and are for maintenance use only.

Pitch Lock

Pitch lock protection from propeller overspeeds is provided in the servomechanism by the pitch lock gap feature.

If supply oil pressure were lost, the centrifugal twisting moment and airloads acting on the propeller would cause an uncommanded pitch decrease resulting in a propeller overspeed.

When the propeller decreases pitch approximately 1° (2% NP) below commanded pitch, the pitch lock gap closes. This precludes the escape of oil from either side of the actuator piston, creating a hydraulic lock, which prevents further blade angle decrease.

A pitch locked propeller acts just like a fixed-pitch propeller where rpm follows power lever position and airspeed changes. As soon as supply pressure oil is restored to the servomechanism, the propeller returns to normal operation.

Primary Low Pitch Stop

With the power lever at FLT IDLE, the primary low pitch stop prevents the propeller blade angle from decreasing below 17.6°. This prevents uncommanded reverse blade angles in flight, which would cause a major propeller overspeed.

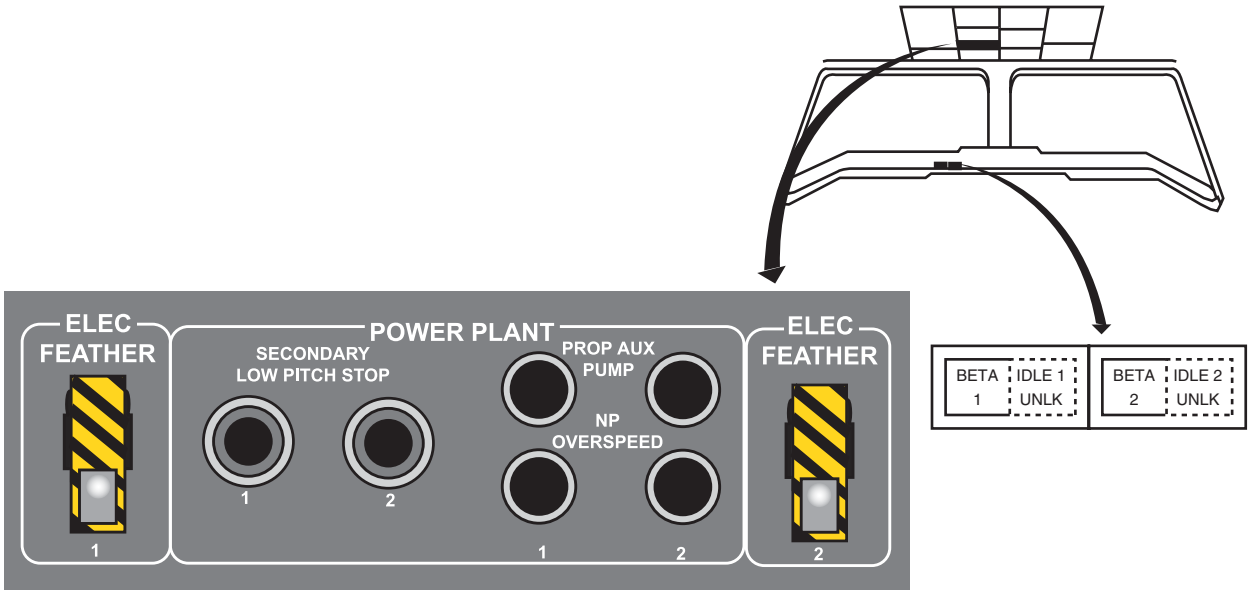


Figure 7-26. Powerplant Control Panel

When the power lever is moved below flight idle into the ground (Beta) range, the primary low pitch stop is removed by the reverse valve to permit Beta range and reverse for ground operation.

Secondary Low Pitch Stop

With the power lever at FLT IDLE or above, the secondary low pitch stop prevents a propeller blade angle less than the flight low pitch in the event of a primary low pitch stop failure. The system is automatically actuated when the propeller pitch decreases below 12.6°.

The secondary low pitch stop system has two microswitches:

- Flight low pitch microswitch
- Secondary low pitch microswitch

The ***flight low pitch microswitch*** is actuated by the PCU piston. It closes whenever the propeller blade angle is below 12.6° (5° below the flight low pitch corresponding to FLT IDLE position).

The ***secondary low pitch microswitch*** is located within the control pedestal. It is closed whenever the power lever is positioned at FLT IDLE or above (normal flight position).

When actuated, the secondary low pitch system causes the BETA light to illuminate and energizes the feathering solenoid valve in the PCU.

When energized, the solenoid valve drains the control pressure (P_C) line, causing the propeller pitch to increase above 12.6°. The system is then deactivated by the opening of the low pitch microswitch, and the BETA light goes out.

If propeller pitch decreases again, the process is repeated and propeller pitch will cycle around 12.6°.

NOTE

An in-flight failure of the low pitch microswitch (locking in the closed position) will suddenly feather the propeller. This condition is indicated by an NP decrease and a torque increase. Reduce the power lever to FLT IDLE and open the affected engine Beta circuit breaker, located on the circuit breaker panel (see checklist).

Secondary Low Pitch Stop Test—Two secondary low pitch stop test buttons are located on the overhead panel (see Figure 7-26).

BETA Lights

Two amber BETA lights on the left glareshield panel (see Figure 7-26) are illuminated whenever the propeller blade angle reaches 12.6°.

A flashing BETA light in flight indicates the primary low pitch stop has failed and the secondary low pitch stop system is operating.

A BETA light during ground operations is normal. (Primary low pitch stop is removed and secondary is inhibited).

Mechanical Feathering

Mechanical feathering is accomplished by positioning the condition lever to feather. This opens the mechanical feather valve, which dumps the servo piston metered pressure P_C to drain. The remaining supply pressure shifts the servo piston, feathering the propeller.

With a propeller oil pump failure or reduction of oil supply, mechanical feathering may not be possible. In this event, the electrical feathering system must be used to feather the propeller.

Electrical Feathering

The propeller is electrically feathered by actuation of the feathering solenoid valve in the PCU, and the auxiliary electrical feathering pump.

The ***auxiliary electrical feathering pump*** supplies oil pressure for propeller feathering independent of the engine lubricating circuit and mechanical oil pump.

When actuated, the pump operates for 20 seconds and turns off automatically. This is more than enough time to fully feather the propeller, even from reverse pitch.

The ***auxiliary electrical feathering pump test*** is conducted prior to engine start, as follows:

1. Power lever ..... MAX REVERSE
2. Condition lever ..... MIN RPM
3. PROP AUX PUMP button ..... PRESS
4. BETA light ..... ON
5. PROP AUX PUMP button..... RELEASE

Electrical propeller feathering is controlled by the following:

- Electrical feathering system which may receive inputs from either of the following:
  - Automatic feathering system
  - Emergency feathering system

Electrical Feathering System—The electrical feathering system provides a means for propeller feathering in the following conditions:

- Engine oil pressure loss
- Engine inoperative
- Engine shutdown due to fire

The electrical feathering system is actuated by the guarded ELEC FEATHER switch on the overhead panel (see Figure 7-26) or by input from either the automatic or emergency feathering systems. The system is normally off, with the ELEC FEATHER switch guard lowered.

When the system is actuated the following occurs:

- PCU feathering solenoid valve is energized
- Electrical auxiliary feathering pump is energized
- Automatic feathering system control circuit for the opposite propeller is interrupted

Once turned off the electrical feathering system is available for immediate reuse, provided there is sufficient oil quantity in the oil auxiliary tank.

NOTE

The oil auxiliary tank may be replenished by dry motoring the engine.

Electric Feathering System Test

NOTE

The electric feather system test is conducted following, and in conjunction with, the auxiliary oil pump test. Therefore, the propeller blade angle is in the Beta range with the engines not running.

The test is conducted as follows:

1. Power levers..... GROUND IDLE
2. Electric feather switch..... FEATHER
3. BETA light..... OUT
4. Observe propeller blade angle ..... FEATHER
5. Electric feather switch..... NORMAL

Automatic Feathering System— Electrically controlled automatic feathering is accomplished by the autofeather system. The system improves aircraft performance by quickly reducing the asymmetrical drag of a windmilling propeller following a power loss on takeoff or go around.

The autofeather system actuates the feathering solenoid valve and the auxiliary electrical feathering pump through the electrical feathering system.

The autofeather system is armed for takeoff and two-engine approaches. It is turned off for single-engine approaches.

The autofeather system only feathers one propeller, the other is automatically locked out.

Autofeather system components are:

- The AUTO FEATHER control panel
- Torque signal conditioning units (SCUs) on each engine
- Control pedestal microswitches

The AUTO FEATHER ***control panel*** (Figure 7-27) directly beneath the POWER PLANT control panel, has the following controls:

- ON/OFF switch—Arms the system
- Green ARMED light—Illuminates when arming requirements are met
- TEST switches—Simulate high torque on respective engine

The **SCUs** provide the autofeathering system with the engine torque input.

The control pedestal ***microswitches*** are energized when the respective power lever angle (PLA) is greater than 62°.

Autofeather system arming requirements (green ARM light on) are:

- Autofeather control switch—ON
- PLA—Both greater than 62°
- Torque—Both engines above 62 ±1.4% (takeoff power)

When the autofeather system is triggered (torque on either engine drops below 23.6 ±2.5%), the following occurs:

- Green ARMED light goes out

Then, after a 0.5 second delay,

- Interlock relay locks out the other propeller
- Failed engine propeller feathers
- Failed engine N_P fuel governing is canceled

NOTE

The engine does not shut down.

After automatic feathering system actuation, propeller unfeathering is possible only after AUTO FEATHER switch is turned OFF.

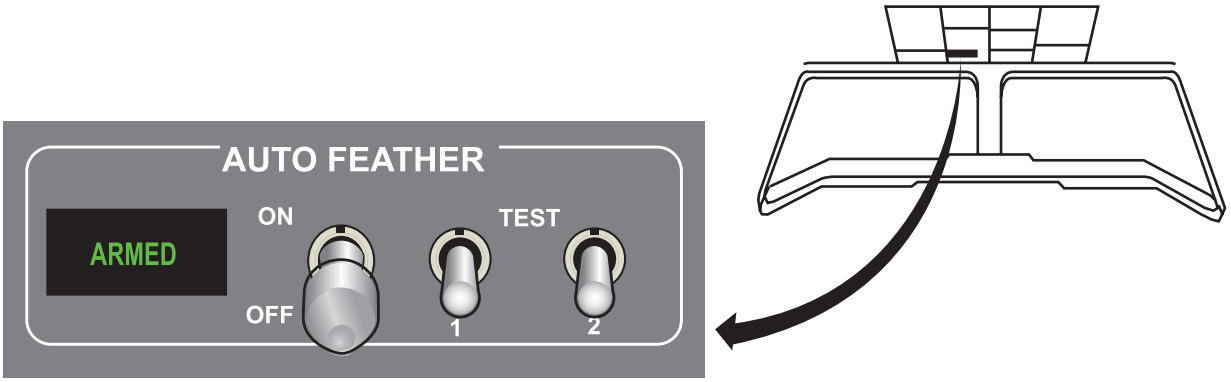


Figure 7-27. AUTO FEATHER Control Panel

When the power levers are advanced for takeoff and, after an eight-second delay, the autofeather system is not armed, a three-chime aural alert sounds, and a "TAKEOFF AUTOFEATHER" voice message warning is given.

Autofeather Test Sequence—After engine starting and before each takeoff, the autofeather system must be tested for each propeller.

The satisfactory test sequence is as follows:

1. Autofeather control switch..... ON
2. Both power levers ..... GROUND IDLE
3. Both TEST switches..... TEST
4. Both torque indications 75%
5. ARM light ..... ON
6. Left TEST switch ..... RELEASE
7. Left torque indication ..... DROP BELOW 22%
8. ARM light ..... OFF
9. Left N_P..... DROP TO 15–20%
10. Both TEST switches..... TEST
11. Both torque indications 75%
12. ARM light ..... ON
13. Right TEST switch..... RELEASE
14. Right torque indication ..... DROP BELOW 22%
15. ARM light ..... OFF
16. Right N_P ..... Drop to 15–20%
17. Both TEST switches..... RELEASE

Emergency Feathering and Shutdown—Emergency feathering and shutdown are accomplished by pulling the respective fire "T" handle.

Pulling the "T" handle results in the following:

- Electrically closes the firewall shutoff valves to shut down and isolate the engine
- Actuates the feathering solenoid valve and the auxiliary electrical feathering pump, through the electrical feathering system, to feather the propeller

PROPELLER OPERATION

WARNING

Never select the power levers below FLT IDLE in flight.

Following is a review of actions that take place in the PCU and in the hydraulic servomechanism when a pitch increase or decrease is commanded.

For pitch increase:

- The metered control pressure is reduced, allowing the supply pressure to move the PCU servo piston aft.
- The frictionless ball screw mechanism causes the transfer tube to rotate counterclockwise (relative to propeller rotation).
- The transfer tube rotation, via the pitch change valve's reverse-threaded acme screw, moves the valve forward.
- Supply pressure oil is routed to the increase pitch (forward) side of the piston, driving the blades to increase pitch via the blade shank off-centered pin.

For pitch decrease:

- The metered control pressure is increased, overriding the supply pressure and moving the PCU servo piston forward.
- The frictionless ballscrew mechanism causes the transfer tube to rotate clockwise (relative to propeller rotation).

- The transfer tube rotation, via the pitch change valve's reverse-threaded acme screw, moves the valve aft.

- Supply pressure oil is routed to the decrease pitch (aft) side of the piston, driving the blades to decrease pitch via the blade shank off-centered pin.

PROPELLER SYNCHRONIZATION SYSTEM

The synchronization system reduces high-frequency vibration caused by propellers operating at different speeds.

A synchrophaser feature adjusts phase angle relationship between propellers so that no two propeller blades cross the leading edges of the wing at the same time, thereby reducing low frequency noise.

The synchronization system is available under all flight conditions with the PCU operating in the constant speed mode.

Synchronization System Components

Components of the synchronization system are as follows:

- Synchronization control switch
- Pulse generator
- Synchrophaser
- Torque motor

Synchronization Control Switch—The propeller synchronization system is controlled by the two position ON/OFF switch on the overhead PROP SYNC panel (Figure 7-28).

The synchrophaser operates only when propeller rpm is in the 80 to 100% N_P range, and may be on during takeoff and landing.



Figure 7-28. Synchronization Control Panel

Pulse Generator—A pulse generator for each propeller produces electrical signals proportional to the speed and phase angle of the propeller. The signals are sent to the synchrophaser for processing and control.

Synchrophaser—The synchrophaser is a microprocessor. It takes the pulse generators' speed and phase inputs and sends a corrective signal output to the right PCU torque motor.

The left propeller is the master and the right is the slave. The synchrophaser compares speed and phase signals from the pulse generators, and transmits a corrective signal to the torque motor of the right slave PCU.

If the propeller speeds differ by more than 2.5% N_P, the system will not function. This prevents significant speed loss in the event of an engine failure.

The synchrophaser also keeps the phase angles between the left and right propeller within 5° of the preset value.

Torque Motor—The torque motor is a PCU component.

The right torque motor receives corrective signals from the synchrophaser. It adjusts the speed of the right propeller, to match the left, by varying the primary governor setting.

Synchronization System Operation

A difference between master and slave propeller N_P may be evidenced by an audible beat in propeller noise—the faster the beat, the greater the difference.

When audible beat or noise becomes noticeable, the best operation of the synchronization system is as follows:

1. Turn the synchrophaser OFF.
2. Set condition levers to the desired NP.
3. Match the NP as close as possible.
4. Turn the synchrophaser ON.

When speed sync is established, audible beats should subside.

After transients cease, the synchrophaser establishes the desired phase relationship within approximately 18 seconds, and general noise level should be reduced.

Turning the system off causes the slave propeller to return to its unbiased, mechanical rpm setting.

When the synchrophaser is not in use, a difference in governing rpm between propellers is normal. Condition lever positions may not match when N_P indications are equal, normally due to hysteresis or rigging.

Figure 7-29 illustrates the propeller system controls and indicators. Explanations of capabilities are also included.

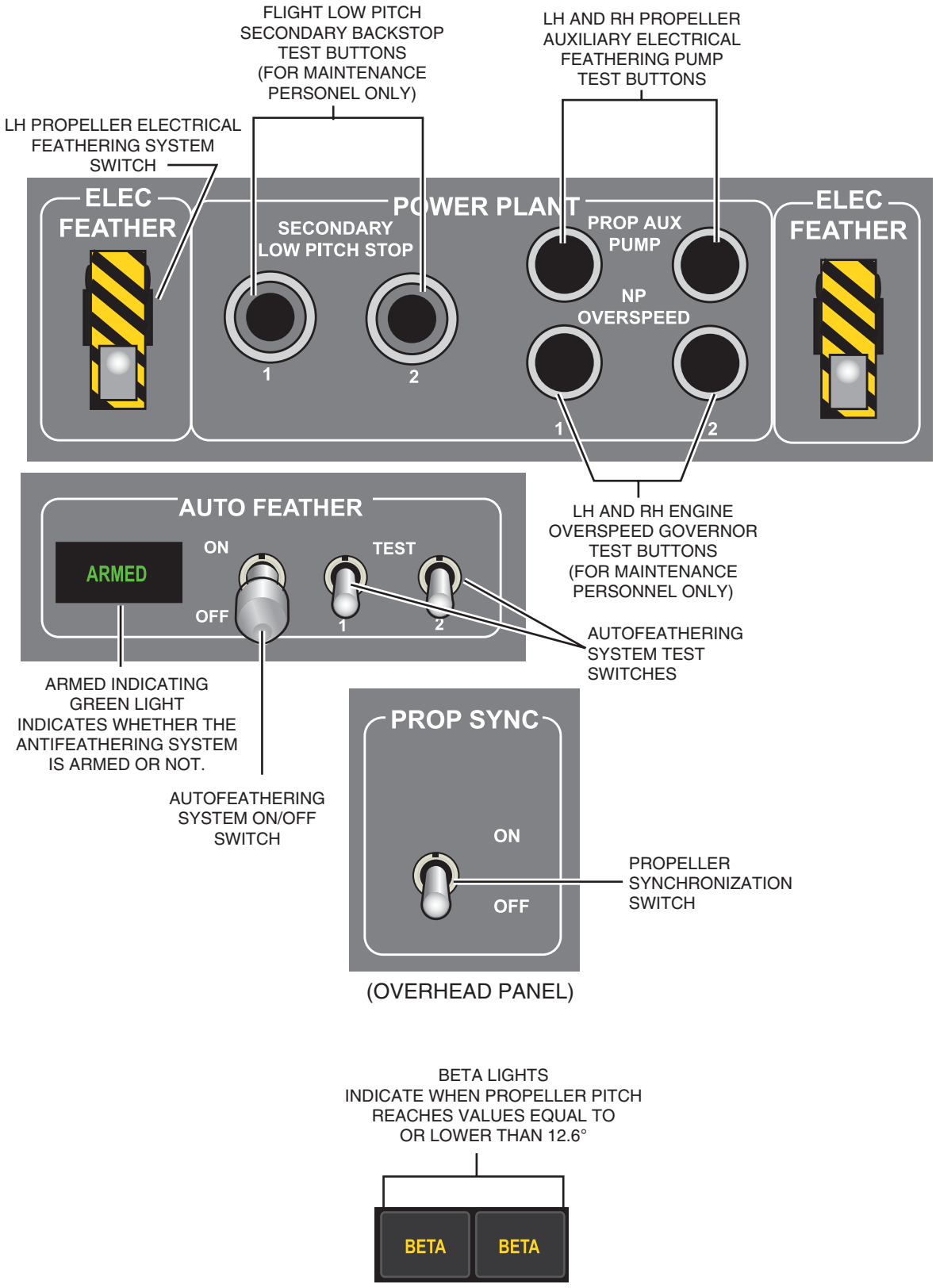


Figure 7-29. Propeller System Controls and Indicators

QUESTIONS

Engine

1. The engines of the EMB 120 are:
A. Garrett 1180 C
B. Pratt-Whitney PW 118
C. Air research 2131 EMB
D. Pratt-Whitney PT6 EMB

2. The engine has how many compressors?
A. 1
B. 2
C. 3
D. 4

3. The engine has how many turbines?
A. 1
B. 2
C. 3
D. 4

4. The engine horsepower rating is:
A. 1180
B. 1800
C. 1018
D. 2131

5. Engine power control is accomplished by operation of:
A. Fuel control unit (FCU)
B. Hydromechanical metering unit (HMU)
C. Turbine limiting device (TLD)
D. A combination of the electronic engine control (EEC) and the hydromechanical metering unit (HMU)

6. The device that provides essential fuel control functions in the absence of electrical power is:
A. Turbine limiting device (TLD)
B. Hydromechanical metering unit (HMU)
C. Electronic engine control (EEC)
D. Power surge control unit (PSCU)

7. The maximum torque allowed is:
A. 100% for 15 minutes
B. 110% for 5 minutes
C. 120% for 5 minutes
D. Maximum torque is limited by aircraft weight

8. The electronic engine control is normally used in the:
A. OFF position
B. TO or ON position
C. HI FLT position
D. MAX DIF position

9. If the EEC light on the glareshield illuminates, this indicates:
A. An electrical failure exists in the system
B. The power lever is set incorrectly
C. A mechanical failure exists in the system
D. Both A and C

10. Which turbine provides power to the propeller shaft?
A. High pressure turbine
B. Low pressure turbine
C. Power turbine
D. A combination of all three

11. Maximum continuous torque is:
A. 97%
B. 100%
C. 103%
D. 105%

12. Maximum continuous engine rpm is:
A. 95%
B. 100%
C. 103%
D. 104%

13. Minimum oil temperature for takeoff is:
A. 32°C
B. 45°C
C. 52°C
D. There is none

14. Oil pressure alarm light illuminates at:
A. 22 PSID
B. 40 PSID
C. 52 PSID
D. 55 PSID

15. Condition lever positions are:
A. MAX RPM, MIN RPM, FEATHER, FUEL CUT-OFF
B. MAX FLOW, MIN FLOW, GND IDLE, FUEL CUT-OFF
C. GND RPM, MAX PRESS, MIN PRESS, CUT-OFF
D. MAX FLOW, MIN RPM, FEATHER, GND IDLE

16. The PW 118 combustion section is defined as:
A. Can-type combustor
B. Reverse flow annular combustor
C. Can-annular combustor
D. Straight flow annular type

17. Power lever movement provides direct control to:
A. TSU
B. PCU
C. HMU
D. EEC

18. During engine start, with ignition system set to "AUTO":
A. The igniters come on automatically
B. The igniters go off automatically
C. The igniters come on if engine fails
D. Both A and B

19. What if any indication does the pilot have to indicate that the engine igniters are energized?
A. Chime will sound, ignition light on MAP will illuminate
B. Chime will sound, ignition light on start/ignition panel will illuminate
C. A white ignition light on the start/ignition panel will illuminate
D. All of the above

20. After initiation, the start cycle is automatically interrupted at approximately:
A. 30% N_H
B. 50% N_H
C. 60% N_H
D. By position of power levers

21. Engine oil is also used for:
A. Nosewheel steering
B. Fuel heating
C. APU dampening
D. Flap retraction

22. Each engine has how many compressor bleed ports?

A. 1
B. 2
C. 3
D. None

23. Fuel flow to the engine is metered by the:

A. FFU
B. HMU/FFU
C. HMU/EEC
D. FMU/EEC

24. The EEC has a function called:

A. Fail-store
B. Fail-proof
C. Fail-fixed
D. Fail-save

25. EEC control function is activated when:

A. 25% N_P
B. 25% N_H
C. 25% N_L
D. 25% N_C

26. If gust lock is set, power levers are restricted in movement.

A. True
B. False

27. The condition lever has free movement from feather to fuel cut-off.

A. True
B. False

28. Engine compressors are:

A. Axial (2)
B. Centrifugal (2)
C. 1 axial—1 centrifugal
D. Annular (2)

29. The free power turbine is:

A. A single-stage axial flow
B. A two-stage axial flow
C. A single-stage centrifugal flow
D. A two-stage centrifugal flow

30. The engine bleed ports are called:

A. N_1—N_3
B. $P_{2.5}$—P_3
C. P_2—P_3
D. $N_{2.5}$— P_3

QUESTIONS

Propeller

1. The EMB 120 propellers are made by:
A. Dowty-Royal
B. Hamilton-Sundstrand
C. Bendix-Keller
D. Pratt-Whitney

2. In normal operation, maximum governed propeller speed is:
A. 1,290–1,300 rpm
B. 1,300–1,312 rpm
C. 1,290–1,309 rpm
D. 1,300–1,322 rpm

3. Normal operational pitch range is:
A. 89.5° (feather) to –25° (reverse)
B. 79.2° (feather) to –10° (reverse)
C. 89.5° (feather) to –15° (reverse)
D. 79.2° (feather) to –15° (reverse)

4. In flight, N_P is controlled by:
A. PCU and condition levers
B. PCU and power levers
C. EEC and power levers
D. Both B and C

5. The main component responsible for selecting pitch is:
A. EEC
B. FCU
C. HMU
D. PCU

6. The main component for changing propeller pitch is:
A. Electronic engine control
B. Propeller servomechanism
C. Propeller pressure control
D. Both A and C

7. If through a malfunction the propeller becomes fixed pitch, what items will control N_P?
A. Power and speed changes
B. Power and EEC changes
C. Autopitch control
D. None of the above

8. During taxi and reverse, which unit controls N_P?
A. EEC
B. FCU
C. HMU
D. PCU

9. Pitch lock is a term used to describe:
A. A device which limits propeller lever movement at high speed
B. A device which allows power lever movement at high speed
C. A device which locks the propeller pitch in flight if propeller oil pressure is lost
D. A device which locks propeller pitch to prevent over travel in reverse

10. Which control input sets the propeller speed governor?
A. Power lever
B. Autofeather
C. Condition lever
D. Both B and C

11. When operating in the Beta range:

A. The speed governor system controls propeller pitch
B. The speed governor system has no control of propeller pitch
C. The speed governor acts in concert with condition lever
D. None of the above

12. Beta valve position is adjusted by the:

A. Power lever
B. Condition lever
C. Autofeather
D. Both B and C

13. Propeller blade angle in flight idle is:

A. 22.6°
B. 17.6°
C. 12.6°
D. 7.6°

14. In flight, if the engine has failed and the propeller has not been feathered, it will then feather automatically.

A. True
B. False

15. If on approach with autofeather armed and engine fails, will its propeller autofeather?

A. Yes
B. No

16. Propeller sync is activated by:

A. System selected ON and $N_P > 50\%$
B. System selected ON and $N_H > 50\%$
C. System selected ON and $N_P > 80\%$
D. System automatically on when $N_P > 90\%$

17. The main purpose of the least selector valve is to:

A. Monitor fuel flow to the HMU.
B. Select the proper prop speed governor (normal or overspeed)
C. Provide additional oil pressure to prop dome
D. All of the above

CHAPTER 8
FIRE PROTECTION

CONTENTS

Page

INTRODUCTION 8-1

GENERAL 8-1

DETECTION SYSTEMS 8-2

Engine/Wheel Well Fire Detection System 8-2

APU Fire Detection System 8-4

Fire Detection Systems Test 8-5

Smoke Detection System 8-6

FIRE EXTINGUISHING SYSTEMS 8-7

Engine/Wheel Well Fire Extinguishing System 8-7

APU Fire Extinguishing System 8-10

Portable Fire Extinguishers 8-11

ELECTRICAL POWER SOURCES 8-11

Smoke Detection System (Optional) 8-13

Class-C Baggage Compartment Smoke Detection and Fire-Extinguishing System (Optional) 8-14

QUESTIONS 8-16

ILLUSTRATIONS

Figure	Title	Page
8-1	Engine/Wheel Well Fire Detector Location	8-2
8-2	Fire Detector	8-2
8-3	Engine Fire Control Panel	8-3
8-4	APU Fire Detection	8-4
8-5	APU Warning and Indicator Lights	8-5
8-6	Smoke Detector Location	8-6
8-7	Selector Switch	8-6
8-8	Fire Extinguisher Bottle (Left)	8-7
8-9	Engine Fire Extinguishing Controls and Indicators	8-8
8-10	Engine/Wheel Well Fire Extinguishing System	8-9
8-11	APU Fire Extinguishing System and Controls	8-10
8-12	Fire Protection Controls and Indicators	8-12
8-13	Smoke Detector Locations	8-13
8-14	Smoke Detector Panel (Right Closet—Typical)	8-14
8-15	Baggage Detection/Extinguishing Panel	8-15

CHAPTER 8
FIRE PROTECTION



INTRODUCTION

The EMB 120 Brasilia fire protection system provides for the detection, warning and extinguishing of fire in each engine/main wheel well compartment, and within the auxiliary power unit (APU) compartment. An optional smoke detection system is available for the passenger cabin and cargo compartment. Portable hand-held extinguishers are also provided.

GENERAL

Each nacelle is equipped with three sensing elements. One is installed in the landing gear wheel well, one in the engine accessory section and the other in the exhaust area. A control module relays the signals to indication and warning devices on the glareshield engine fire control panel, the multiple alarm panel, and the audio warning unit.

The engine fire extinguishing system is a two-shot system. If an engine fire is not extinguished with actuation of the first bottle, the second bottle is available for discharge into the same engine.

The APU fire protection system includes a sensing element in the APU compartment, a control module, and indications on the APU fire control panel and multiple alarm panel.

The optional smoke detection system consists of four photo-electric cells, a selector switch, and a control amplifier which relays signals to the multiple alarm panel and the aural warning unit.

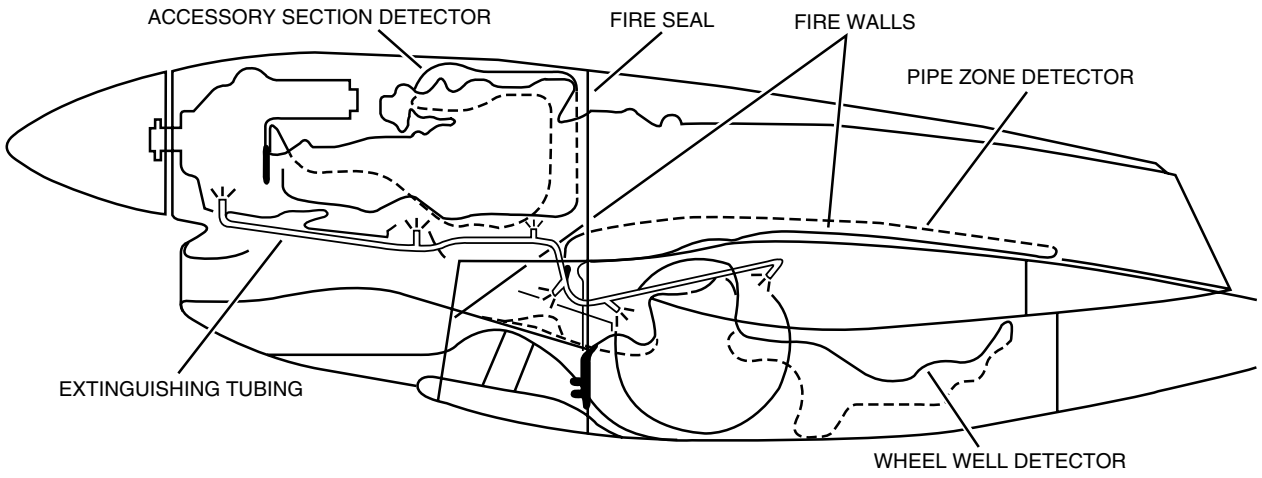


Figure 8-1. Engine/Wheel Well Fire Detector Location

DETECTION SYSTEMS

ENGINE/WHEEL WELL FIRE DETECTION SYSTEM

General

The engine/wheel well fire detection system provides an immediate warning in the event of a fire or general overheat condition in the engine accessory compartment, the tailpipe compartment, and the wheel well area (Figure 8-1).

The detection system consists of three detectors and two control modules per nacelle. Additionally, there are four discrete fire indication lights for each nacelle, located on the engine fire control panel in the cockpit.

Fire Detectors

The fire/overheat detectors (Figure 8-2), sense a temperature increase above normal. There is a detector located in each of the three main regions of the engine nacelle.

The detectors are designed to be virtually free of any false alarms. Cuts, bends, twists, abrasions, or even excessive deforming of the sensor should not cause false warnings.

The sensor tube contains a fixed volume of helium gas. As the tube senses an overall temperature increase its internal gas pressure increases proportionally. When the force of the gas pressure overcomes the reference force in the dual responder alarm switch, an electrical signal activates the warning devices.

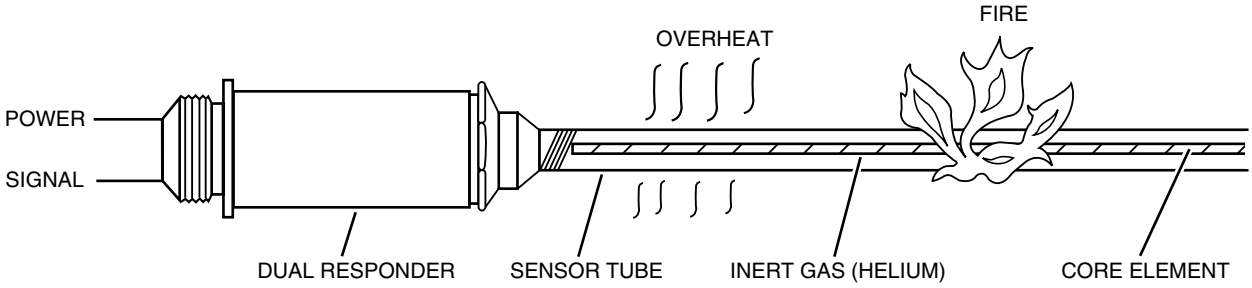


Figure 8-2. Fire Detector

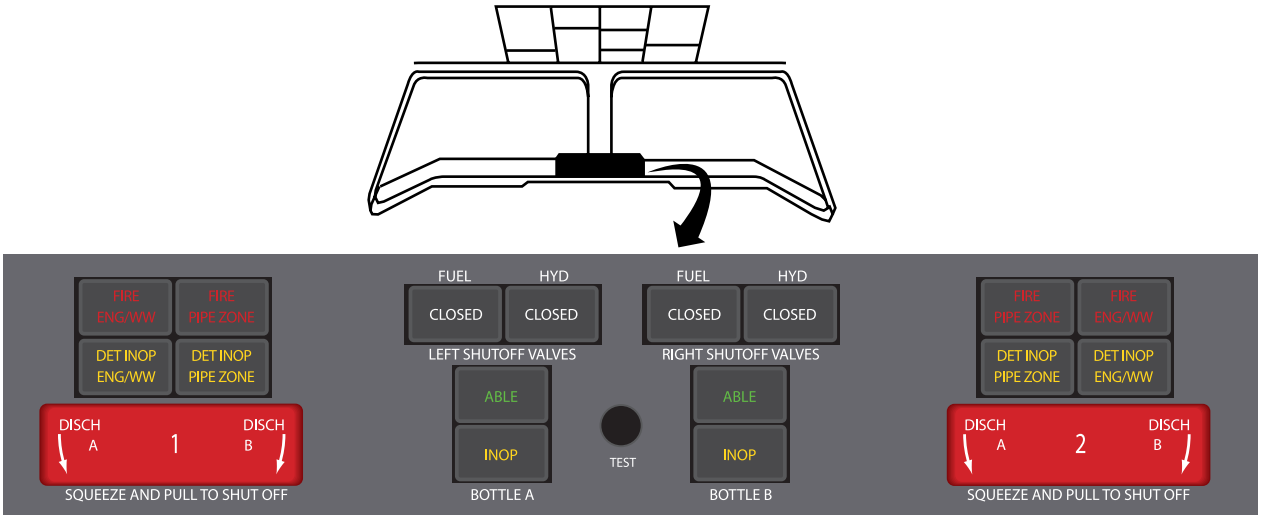


Figure 8-3. Engine Fire Control Panel

The sensor tube's capillary central core contains a fixed volume of helium. This central core element provides sensing for localized fire or high intensity heat. The core releases an extremely large gas volume quickly when sensing this heat and activates the alarm switch.

When either the general overheat or localized fire condition disappears, the detection system returns to its standby condition without requiring manual resetting.

If the gas leaks out of the tube, an INOP caption on the fire control panel illuminates indicating the respective fire detector is inoperative.

Control Modules

Each engine nacelle has two control modules. They provide the electrical interface between the fire detectors, the test button, and the fire indication and warning lights.

One control module is for the engine compartment and wheel well area, and the second is for the tail pipe area.

Indication and Warning Lights

Each nacelle is provided with four discrete fire indication and warning lights, located on the engine fire control panel (Figure 8-3).

The four lights are:

- FIRE ENG/WW
- FIRE PIPE ZONE
- DET INOP ENG/WW
- DET INOP PIPE ZONE

Engine/Wheel Well Fire Detection Operation

A fire or overheat in the engine, wheel well, or tail pipe area causes the detector in that area to send a signal to its control module. The control module then sends signals to the appropriate indicator light (FIRE ENG/WW or FIRE PIPE ZONE), to the master WARNING lights, and to the aural warning unit which actuates a continuous bell.

If one of the detectors fails, the relevant amber DET INOP light illuminates and the master CAUTION lights will flash to warn the pilots.

The flashing master WARNING/ CAUTION lights and the bell may be canceled by pressing the alarm cancel button next to each master WARNING/ CAUTION light.

APU FIRE DETECTION SYSTEM

The APU fire detection system (Figure 8-4), consists of one fire and overheat detector, similar to the detectors in the engines and wheel wells, and one control module. In addition, the APU has warning and indicator lights on both the overhead APU fire control panel and the multiple alarm panel (MAP).

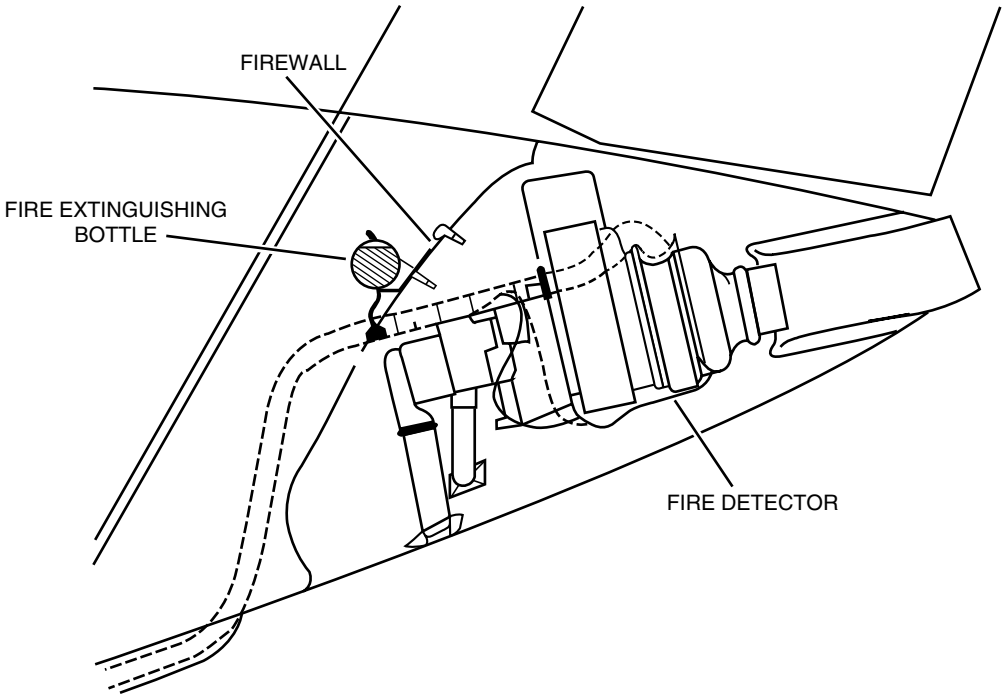


Figure 8-4. APU Fire Detection

A TEST button, located on the engine fire control panel, is used to test the fire detection system integrity. When the TEST button is pressed all indication and warning lights on the engine fire control panel and the APU fire control panel illuminate, the master WARNING and CAUTION lights flash, and the continuous bell sounds.

Indication and Warning Lights

The APU fire control panel (Figure 8-5), has two indicator lights associated with the fire detection system. They are the red FIRE light and the amber INOP light.

The two lights on the MAP (Figure 8-5), activated by the APU fire detection system are the red FIRE APU warning light and the amber APU caution light.

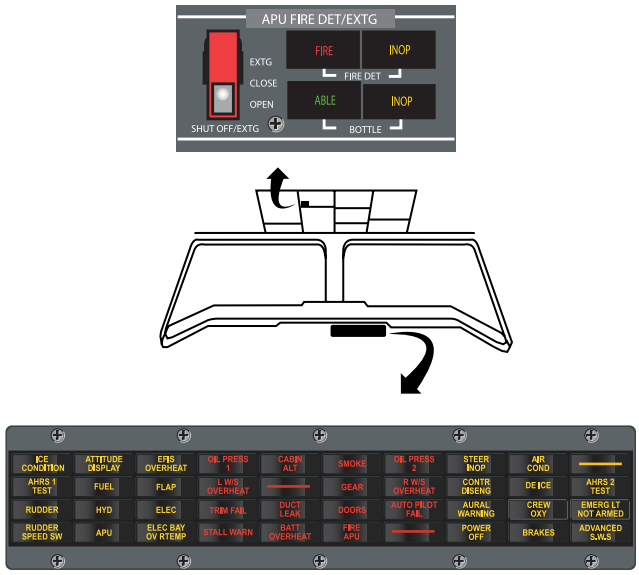


Figure 8-5. APU Warning and Indicator Lights

APU Fire Detection Operation

A fire or overheat in the APU compartment causes the detector to send a signal to the control module. The control module then sends a signal to the FIRE light on the APU fire control panel, to the multiple alarm panel FIRE APU light, to the master WARNING lights on the glareshield panel, and to the aural warning unit which actuates a continuous bell.

The flashing master WARNING lights and the bell may be canceled by pressing the alarm cancel button next to each master WARNING light.

The FIRE light and FIRE APU light remain on as long as the fire or overheat condition exists.

If the APU fire detector fails, the INOP light on the APU fire control panel illuminates. The APU light on the MAP illuminates, and the master CAUTION lights on the glareshield panel flash. The master CAUTION lights may be canceled by pressing the alarm cancel button beside each master CAUTION light.

The INOP light and APU light cannot be canceled.

FIRE DETECTION SYSTEMS TEST

The TEST button on the engine fire control panel is used to test both the engine/wheel well and APU fire detection system integrity.

When the TEST button is pressed:

- All lights on the engine fire control panel, the APU fire control panel, and the FIRE APU and APU lights on the MAP illuminate.
- The master WARNING and CAUTION lights flash.
- The continuous bell is sounded

The flashing master WARNING and CAUTION lights, and the bell, may be canceled by pressing the alarm cancel button located next to each master WARNING/CAUTION light.

If the TEST button is held for more than ten seconds during ground operations, the APU will automatically shutdown. (It is a function of the fire bell, not the test button, that automatically shuts down the APU).

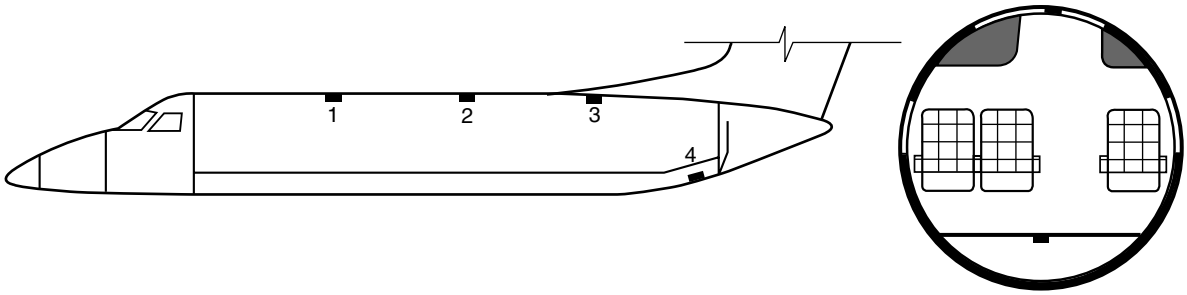


Figure 8-6. Smoke Detector Location

SMOKE DETECTION SYSTEM

The smoke detection system provides an immediate and positive alarm to fires which may occur in the passenger cabin or in the unsupervised baggage compartment. It is an essential component during all cargo operations.

The smoke detection system consists of four smoke detectors, one control amplifier, and one selector switch.

Smoke Detectors

There are four photo-electric smoke detectors located in the cabin and cargo area (Figure 8-6).

Two detectors are located in the passenger cabin ceiling, and two are located in the baggage compartment, one in the ceiling and one under the floor. The detector under the floor of the baggage compartment senses cabin exhaust airflow.

Smoke Detection System Selector Switch

The smoke detection system selector switch is located on the right side of the cockpit overhead panel (Figure 8-7).

The six position switch has four smoke detector test positions in addition to the ON and OFF positions.

Smoke Detection System Operation and Test

When the photoelectric detectors sense the presence of smoke in the passenger cabin or cargo compartment, the three-chime aural alert sounds, the red SMOKE light on the MAP illuminates, the master WARNING lights flash, and the “SMOKE” voice warning is given.

When the smoke density drops to a safe level, the system automatically resets.

To test the smoke detectors, place the selector switch to positions 1, 2, 3 then 4. In each of the positions, the 3 chime aural alert, the red SMOKE light on the MAP, the flashing master WARNING lights, and the “SMOKE” voice warning are all activated.

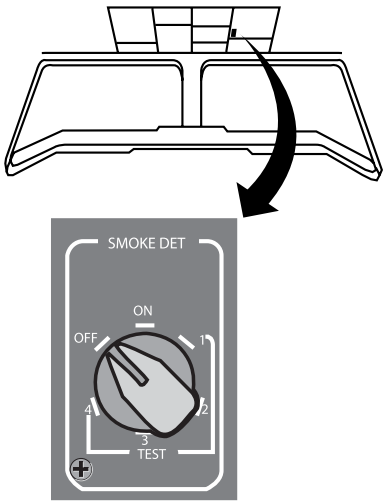


Figure 8-7. Selector Switch

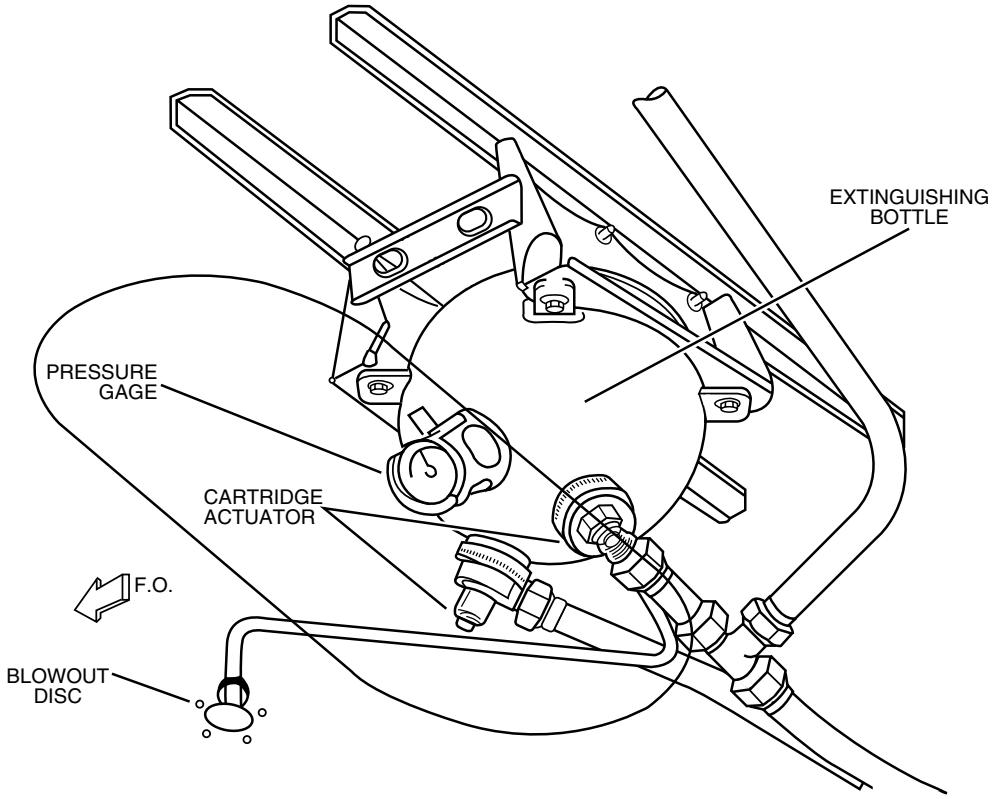


Figure 8-8. Fire Extinguisher Bottle (Left)

FIRE EXTINGUISHING SYSTEMS

The aircraft fire extinguishing system consists of two independent subsystems. One subsystem is used to extinguish fires in the engines and main gear wheel wells, and the other is used for extinguishing fires in the APU compartment.

ENGINE/WHEEL WELL FIRE EXTINGUISHING SYSTEM

The engine/wheel well fire extinguishing system consists of two bottles containing the extinguishing agent, and associated controls and indicators.

Extinguisher Bottles

The two extinguisher bottles are located on each side of the fuselage center wing section under the passenger cabin floor (Figure 8-8).

Each bottle is charged with 6.5 lb of CBrF3 (Bromotrifluoromethane), and pressurized to 600 psi with nitrogen.

A pressure indicator is provided for maintenance personnel to visually check bottle pressure. The pressure indicator also initiates a signal to the engine fire control panel in the cockpit if bottle pressure drops below 275 psi, lighting the amber INOP light for Bottle A or Bottle B.

Extinguisher Bottles Thermal Discharge Indicator - Two discharge indicators (blowout disks) are located in the fuselage skin adjacent to the bottles (Figure 8-8). These red plastic discs eject from their seats if the extinguisher bottle discharges on its own due to thermal expansion.

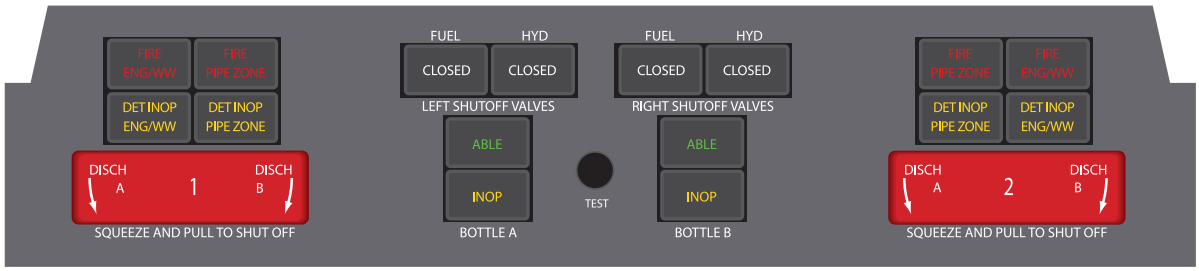


Figure 8-9. Engine Fire Extinguishing Controls and Indicators

Controls And Indicators

Fire Handles - The engine/wheel well fire extinguishing system is controlled by the two fire handles, labeled 1 and 2, on the engine fire control panel (Figure 8-9).

Panel Indicator Lights - The status of each engine/wheel well fire extinguisher bottle is indicated by the ABLE and INOP lights on the engine fire control panel (Figure 8-9).

The green ABLE lights indicate the integrity of the extinguisher bottles' firing elements (squibs). The ABLE light illuminates only during the fire system test.

The amber INOP light indicates that the bottle is empty or is in an abnormal condition, (i.e. low pressure). The INOP lights illuminate any time an abnormal condition exists, and during the fire system test.

The four other lights are associated with the engine/wheel well fire extinguishing system are the white Fuel CLOSED and white Hyd CLOSED light for each engine (Figure 8-9).

The Fuel and Hyd CLOSED lights, when illuminated, indicate that the respective shutoff valve is fully closed.

Handle No.1 controls the valves and extinguishing agent for the left nacelle while handle No. 2 is for the right.

Engine/Wheel Well Fire-Extinguishing System Operation

Operation of the engine/wheel well fire extinguishing system is as follows:

- Squeeze the fire handle to release the safety lock.
- Pull the handle out to close the respective valves and arm both fire bottles.
- Rotate the handle either left or right to actuate either bottle A, or bottle B.

When the handle is rotated the cartridge is fired, driving a slug through a segmented closure disc. The slug penetration provides full flow passage for agent discharge.

The handle pulled, left or right, determines which engine/wheel well, left or right, receives the extinguishing agent.

The direction of rotation of the handle determines which bottle is used.

The agent is discharged to the engine and wheel well zones only. No agent is discharged into the pipe zone.

When a fire handle is pulled, the following functions take place on that engine (Figure 8-10):

- Fuel shutoff valve closes (Fuel CLOSED light illuminates).
- Hydraulic shutoff valve closes (Hyd CLOSED light illuminates).
- Engine bleed valve closes (Bleed CLOSED light illuminates on the cockpit overhead panel).
- Flow control valve for engine air inlet deicing system closes.
- Autofeather system activated
- Both fire bottles are armed.

The TEST button is used to test the fire extinguishing systems integrity. The test is identical to the test performed on the fire detection system.

When the TEST button is pressed:

- All lights on the engine fire control panel, the APU fire control panel, and the FIRE APU and APU lights on the MAP illuminate.
- The master WARNING and CAUTION lights flash.
- The continuous bell is sounded.

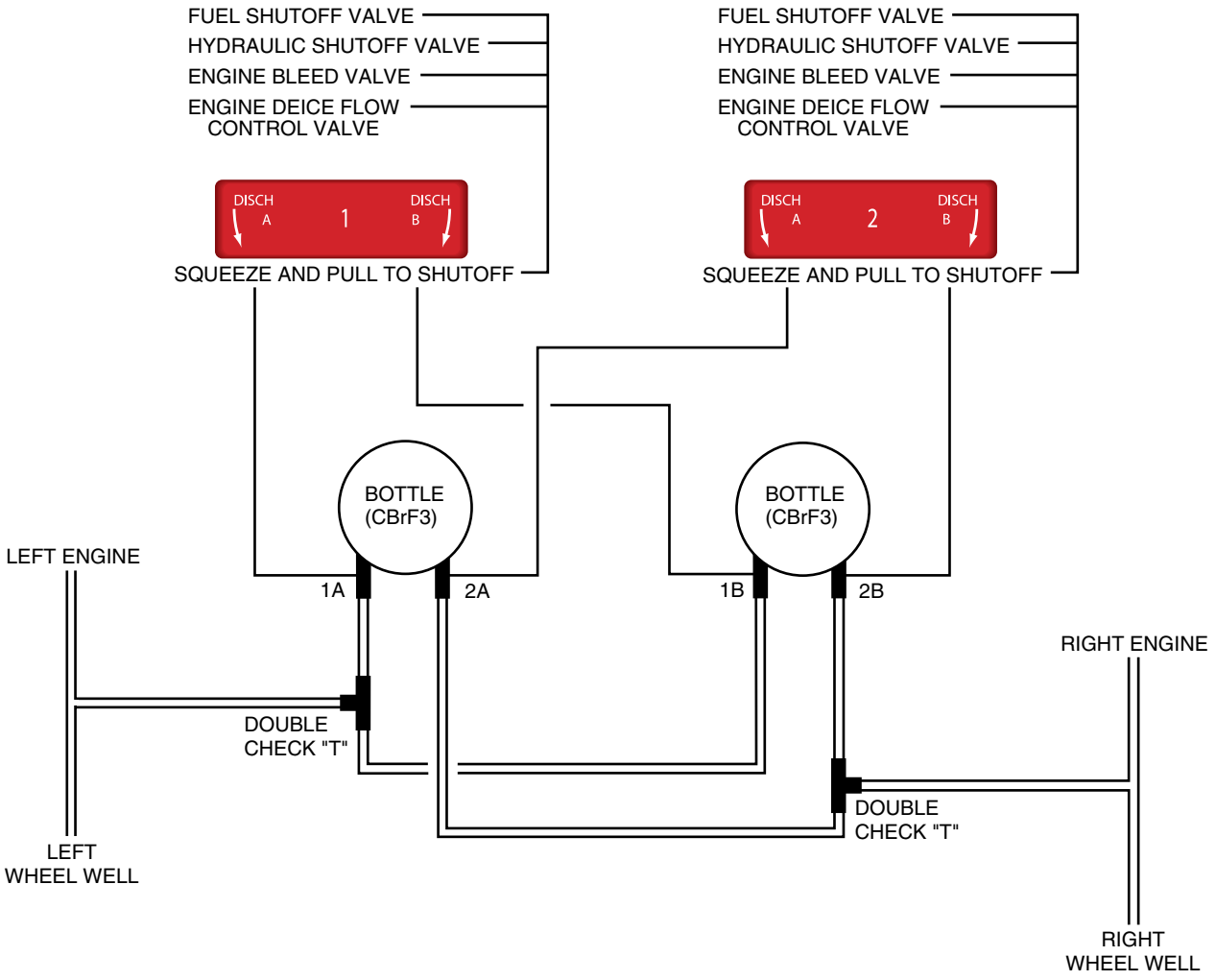


Figure 8-10. Engine/Wheel Well Fire Extinguishing System

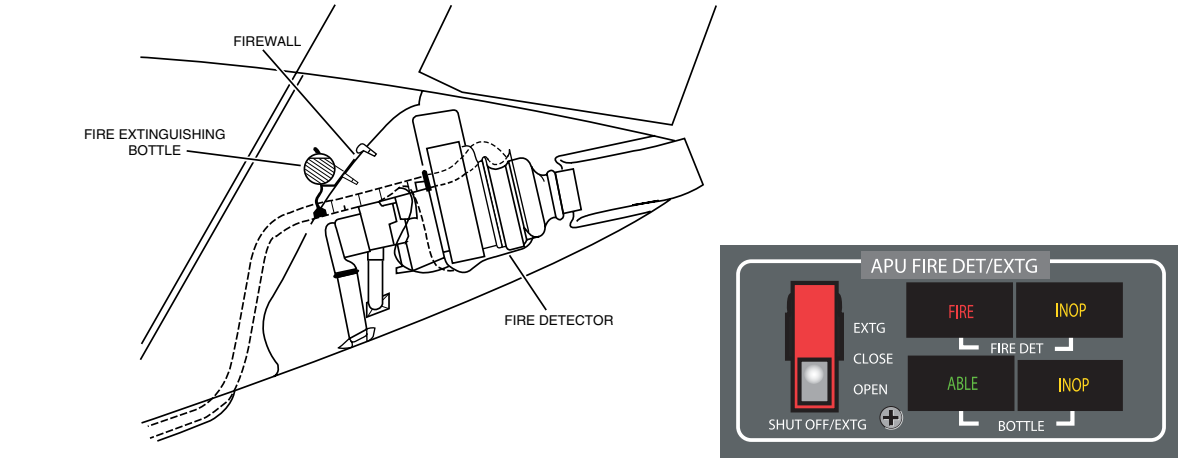


Figure 8-11. APU Fire Extinguishing System and Controls

APU FIRE EXTINGUISHING SYSTEM

The APU fire extinguishing system consists of one bottle containing the extinguishing agent, and the associated controls and indicators on the APU fire control panel.

APU Extinguisher Bottle

The APU extinguisher bottle is located in the tail cone section of the aircraft forward of the stainless steel firewall (Figure 8-11). It is charged with 1 lb of CBrF3, and pressurized to 600 psi with nitrogen.

A pressure indicator is provided for maintenance personnel to visually check bottle pressure. The pressure indicator also initiates a signal to the APU fire control panel if bottle pressure drop below 275 psi, lighting the amber Bottle INOP light.

Extinguisher Bottle Discharge Indicator - A thermal discharge indicator (blowout disc) is located in the fuselage skin in the lower right tail cone area. This red plastic disc is ejected from its seat if the extinguisher bottle discharges on its own due to thermal expansion.

Controls and Indicators

APU Fire Extinguishing Switch - The APU fire extinguishing system is controlled by a red guarded switch, labeled SHUTOFF/EXTG, on the APU fire control panel (Figure 8-11).

Panel Indicator Lights - The status of the APU extinguishing bottle is indicated by the Bottle ABLE and INOP lights on the APU fire control panel (Figure 8-11).

The green ABLE light indicates the integrity of the extinguisher bottle's firing element (squib). The ABLE light illuminates only during the fire system test.

The amber INOP light indicates that the bottle is empty or is in an abnormal condition, (i.e. low pressure). The INOP light illuminates any time an abnormal condition exists, and during the fire system test.

When the Bottle INOP light is triggered, the APU light on the MAP illuminates and the master CAUTION lights on the glareshield flash. The master CAUTION lights may by canceled by pressing the alarm cancel button, but INOP and APU lights remain on as long as the bottle is empty.

APU Fire-Extinguishing System Operation

The APU fire-extinguishing system is operated by the red guarded switch on the APU fire control panel.

Functions of the switch positions are:

- OPEN—Normal operating position
- CLOSE—Initiates the following:
  - Closes the APU fuel and bleed-air valves
  - Disconnects the APU generator from the central bus
  - Deenergizes the APU control panel, causing the APU to shutdown in preparation for extinguisher discharge
- EXTG—Discharges extinguishing agent into the APU compartment.

After the agent has been discharged, the bottle INOP light on the APU control panel, the APU light on the MAP illuminate, and the master CAUTION lights flash. The master CAUTION lights may by canceled by pressing the alarm cancel button, but INOP and APU lights remain on as long as the bottle is empty.

PORTABLE FIRE EXTINGUISHERS

The aircraft is equipped with two portable fire extinguisher bottles. Each bottle contains 2.5 pounds of HALON and may be used on all classes of fires.

One extinguisher is attached to the rear of the pilot's seat, the other below the flight attendant's seat. Some operators install a third optional extinguisher in the rear of the passenger cabin.

The extinguishers are operated by pulling out the locking pin and depressing the operating lever.

A gage on each bottle indicates its condition or state of charge.

ELECTRICAL POWER SOURCES

Hot battery bus:

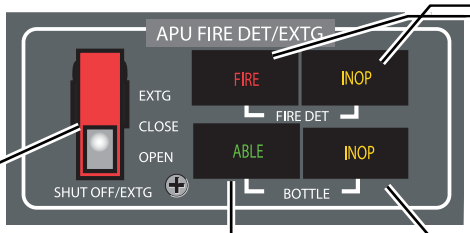
- A and B squibs (both bottles)

28-VDC emergency bus 1:

- Left nacelle detection
- Bottle "A" INOP light

28-VDC emergency bus 2:

- Bottle "B" INOP light
- APU detection and extinguishing
- Right nacelle detection
- Smoke detection



APU FIRE WARNING LIGHT (RED)
ILLUMINATED - Fire/overheat condition sensed by the detector. Illuminates in conjunction with red FIRE APU light on multiple alarm panel. Remains illuminated until the fire overheat condition disappears.

APU FIRE DETECTOR INOPERATIVE LIGHT (AMBER)
ILLUMINATED Failure of any fire detector.

APU SHUTOFF/EXTINGUISHER SWITCH
OPEN - APU system in normal operation
CLOSE Following occurs automatically and simultaneously:
- APU fuel and bleed valves close.
- APU bleed CLOSED light and fuel shutoff CLOSED light illuminate.
- APU generator disconnected from central bus.
- APU GEN OFF BUS light illuminates.

EXTG Fire extinguisher agent discharged.

APU FIRE EXT BOTTLE ABLE LIGHT (GREEN)
ILLUMINATED (Fire system test only) Extinguisher bottle in proper condition for discharge.

APU FIRE EXT BOTTLE PRESSURE CONDITION LIGHT (AMBER)
ILLUMINATED Bottle empty, or inoperative due to pressure/charge problem.

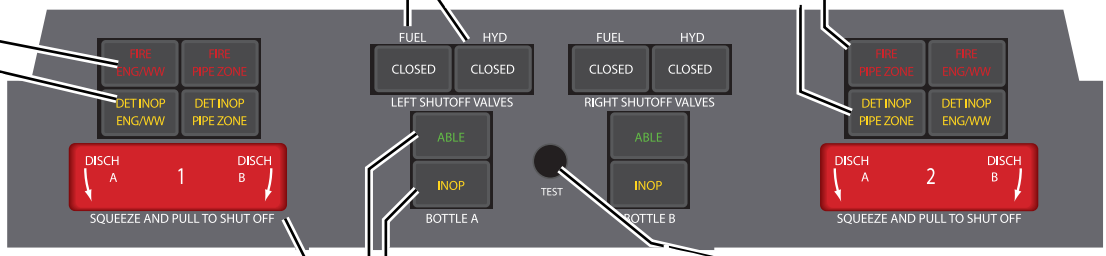
ENGINE/WHEEL WELL No. 1 FIRE WARNING LIGHT (RED)
ILLUMINATED - Fire/overheat condition sensed by engine or wheel well detector. Remains illuminated until fire/overheat condition disappears.

ENGINE/WHEEL WELL No. 1 FIRE DETECTOR INOPERATIVE LIGHT (AMBER)
ILLUMINATED - Failure of engine or wheel well fire detector.

ENGINE No. 1 FUEL AND HYDRAULIC SHUTOFF VALVES POSITION INDICATION LIGHTS (WHITE)
ILLUMINATED - Valve is fully closed. (Actuated by pulling fire handle).

PIPE ZONE No. 2 FIRE WARNING LIGHT (RED)
ILLUMINATED Fire/overheat condition sensed by pipe zone detector. Remains illuminated until fire/overheat condition disappears.

PIPE ZONE No. 2 FIRE DETECTOR INOPERATIVE LIGHT (AMBER)
ILLUMINATED Failure of pipe zone fire detector.



ENGINE/WHEEL WELL No. 1 FIRE EXTINGUISHING HANDLE
Squeezing and pulling handle results in following occuring automatically and simultaneously:
- Engine fuel, hydraulic and bleed air shutoff valves close.
- Air inlet de-icer ejector flow control valve closes.
- Actuation of propeller electrical feathering system.

Rotating handle discharges fire extinguishing agent
- Counter clockwise rotation fires left bottle.
- Clockwise rotation fires right bottle.
- Both bottles discharge into same region.

FIRE EXTINGUISHING BOTTLE INTEGRITY LIGHT (GREEN)
ILLUMINATED (Fire system test only) Extinguisher bottle in proper condition for discharge.

FIRE EXTINGUISHING BOTTLE PRESSURE CONDITION LIGHT (AMBER)
ILLUMINATED Bottle empty, or inoperative due to pressure/charge problem.

ENGINE, WHEEL WELL, PIPE ZONE AND APU FIRE PANELS TEST BUTTON
PRESSED Checks fire detection and extinguishing circuits.
Following lights must illuminate:
- All lights in engine/wheel well, pipe zone and APU fire control panels.
- CLOSED fuel shutoff valve light on APU panel.

Following lights must flash:
- Master WARNING and CAUT ON lights.
- FIRE APU and APU lights on the multiple alarm panel.

Aural warning (bell) must sound continuously.

Figure 8-12. Fire Protection Controls and Indicators

SMOKE DETECTION SYSTEM (OPTIONAL)

NOTE

The smoke detection system is applicable to airplanes operating in the Combi or Full Cargo Version.

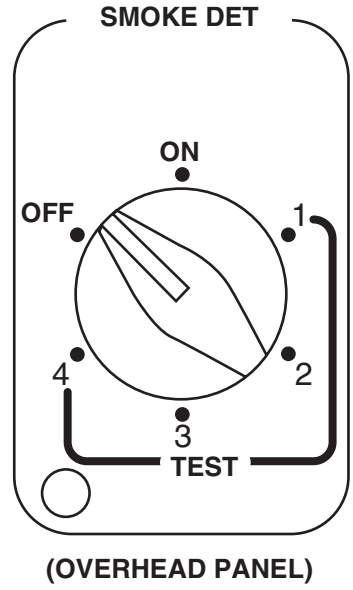
The smoke detection system is activated by a rotary switch on the overhead panel, which also checks the detectors individually for correct operation.

Three smoke detectors are installed in the ceiling, and another one is installed near the pressurization control valves (Figure 8-13).

In the presence of smoke, the system activates the aural alarm and the red SMOKE light illuminates, on the multiple alarm panel (Figure 8-14). As the smoke density drops to a level associated with a safe condition, the system is automatically reset.

CAUTION

The recirculation fans, if left installed on cargo aircraft, must be selected off before flight because they may interfere with the operation of the smoke detectors.



SMOKE DETECTION SYSTEM CONTROL SWITCH

OFF—System deactivated (for passenger configuration).
ON—System armed (when operating in the combi for full cargo version).
1/2/3/4—Operationally checks the respective detector: The aural/visual alarm is activated, indicating correct operation of the system.

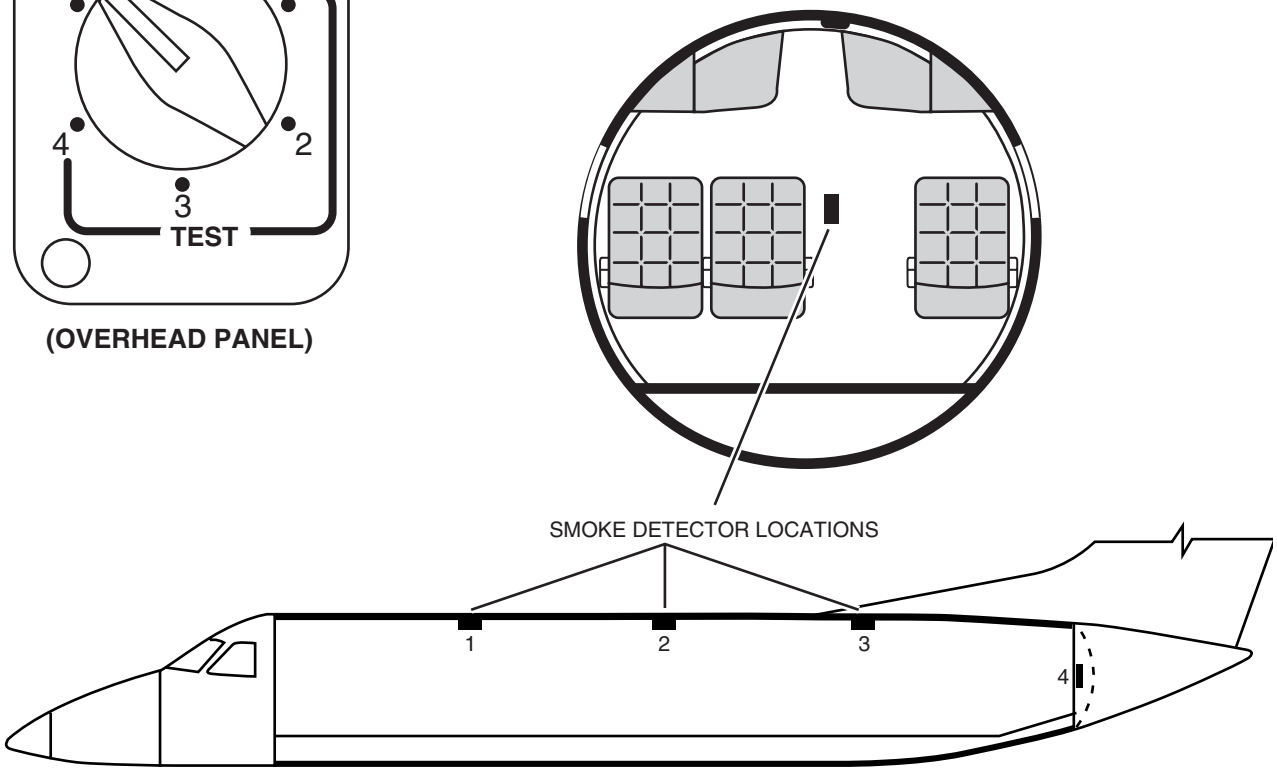


Figure 8-13. Smoke Detector Locations

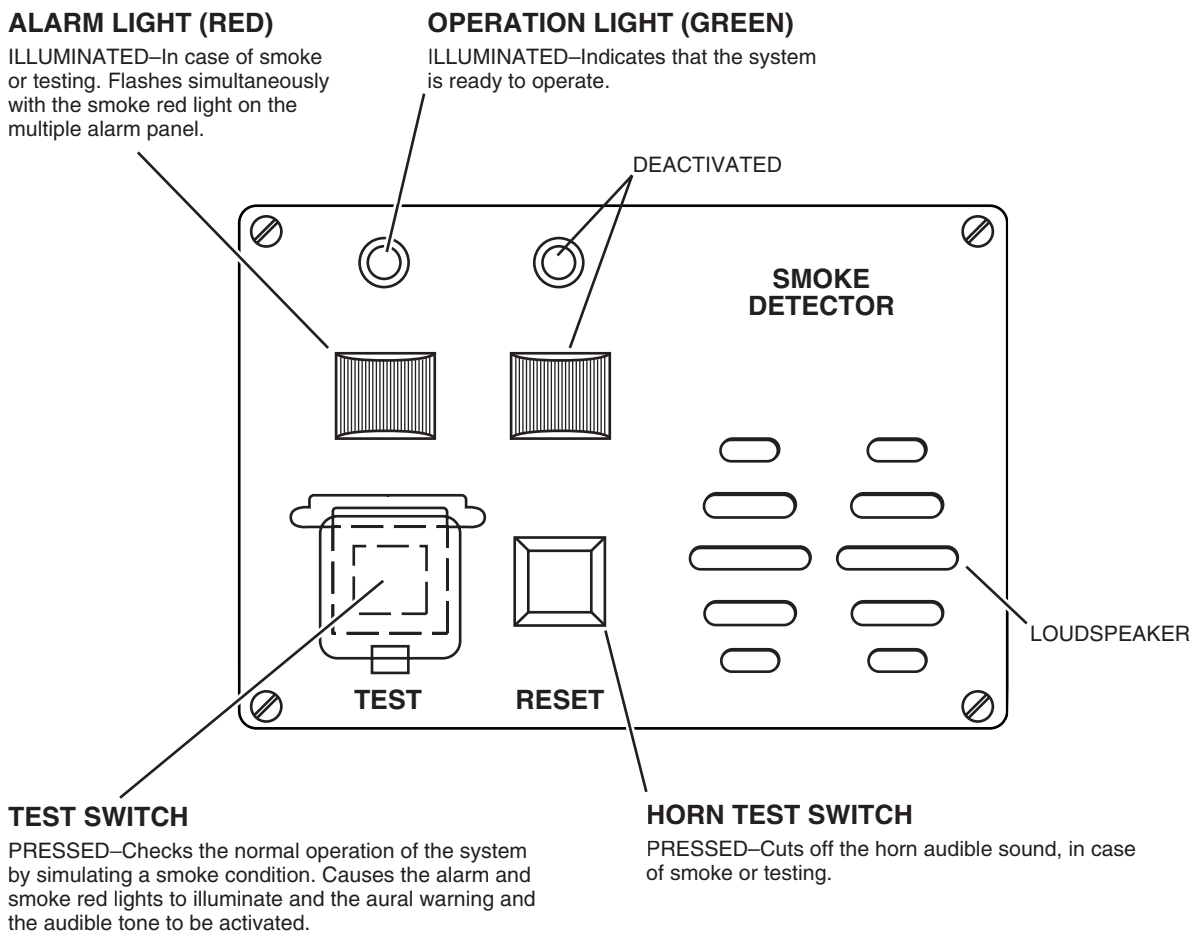


Figure 8-14. Smoke Detector Panel (Right Closet—Typical)

CLASS-C BAGGAGE COMPARTMENT SMOKE DETECTION AND FIRE-EXTINGUISHING SYSTEM (OPTIONAL)

The airplanes Post Mod. SB 120-26-0025 or SN 120.350 and subsequent are equipped with a Class-C baggage compartment that contains a smoke detection and fire-extinguishing system. The system comprises of the following:

- Two smoke detectors
- High-rate discharge bottle, intended to fill the baggage compartment instantaneously
- Metering discharge bottle intended to provide the concentration-fire-level extinguishing agent for at least 60 minutes
- Dedicated smoke detection and fire-extinguishing control panel located on the glareshield panel (Figure 8-15)

In case smoke is detected inside the baggage compartment, a red WARN light illuminates the baggage fire-extinguishing button located on the control panel, the red SMOKE light illuminates on the multiple alarm panel, and the SMOKE aural warning sounds.

TEST SWITCH

The smoke detection and fire-extinguising system is tested by means of a three-position switch. The switch has the TEST 1, TEST 2, and center position. The TEST 1 and TEST 2 positions are spring-loaded to the center OFF position.

When the switch is positioned in TEST 1:

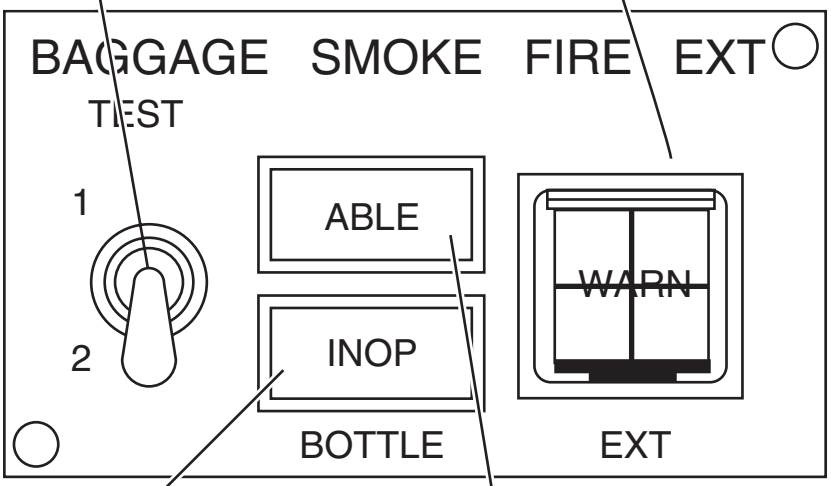
- Both fire extinguisher bottles and one smoke detector are tested.
- ABLE, INOP, and WARN lights illuminated on the BAGGAGE SMOKE/FIRE EXT panel.
- SMOKE light illuminated on the multiple alarm panel.

When the switch is positioned in TEST 2:

- The remaining smoke detector is tested.
- WARN light illuminated on the BAGGAGE SMOKE/FIRE EXT panel.
- SMOKE light illuminated on the multiple alarm panel.

BAGGAGE FIRE-EXTINGUISHING PUSHBUTTON (RED—GUARDED)

- When pressed, discharges both baggage fire extinguisher bottles.
- Illuminates when smoke is detected in the baggage compartment.
- Remains lit as long as there is smoke inside the baggage compartment.



INOP LIGHT (AMBER)

THE INOP light illuminates when:

- TEST 1 is performed
- The fire extinguisher bottles are empty or present a low-pressure state

ABLE LIGHT (GREEN)

- Illuminates when TEST 1 is performed. It indicates normal operation of the fire extinguisher bottles.

Figure 8-15. Baggage Detection/Extinguishing Panel

When the baggage fire-extinguishing button is pressed, both bottles are discharged and an amber INOP light illuminates on the control panel to indicate that the bottles are empty. The control panel also has a switch that allows the flight crew to test the system.

The bottles are installed in a nonpressurized compartment at the tail cone and are filled with Halon 1301 gas.

The electric power for the smoke detection/extinguishing system is supplied by 28-VDC EMERG DC bus 1.

QUESTIONS

1. If the smoke detection system is activated and the alarm sounds, how is it reset?

A. Turn switch to set and then back to ON
B. As smoke level drops to a safe level, it resets automatically
C. Turn switch to OFF, then back to ON
D. Any one of the above

2. The sensor tube detects an overheat condition by means of:

A. The expansion copper wire core
B. A sensitive titanium core
C. Expansion of internal gasses in its core
D. A sensitive infrared receiver

3. The fire detection system provides warning in:

A. Engine area
B. Wheel well area
C. Tailpipe
D. All of the above

4. Fire suppression is available in the:

A. Engine area
B. Wheel well area
C. Tailpipe area
D. A and B only

5. In case of fire, both fire-extinguishing bottles can be discharged to either engine.

A. True
B. False

6. The APU has both a fire detector and a fire extinguisher in its compartment.

A. True
B. False

CHAPTER 9
PNEUMATICS

CONTENTS

Page

INTRODUCTION 9-1

GENERAL 9-1

BLEED-AIR SYSTEM 9-3

Engine Bleed-Air System 9-3

APU Bleed-Air System 9-6

BLEED-AIR SYSTEM OPERATION 9-7

Engine Bleed Air 9-7

APU Bleed Air 9-7

WARNING SYSTEMS 9-8

Duct Leak Warning 9-8

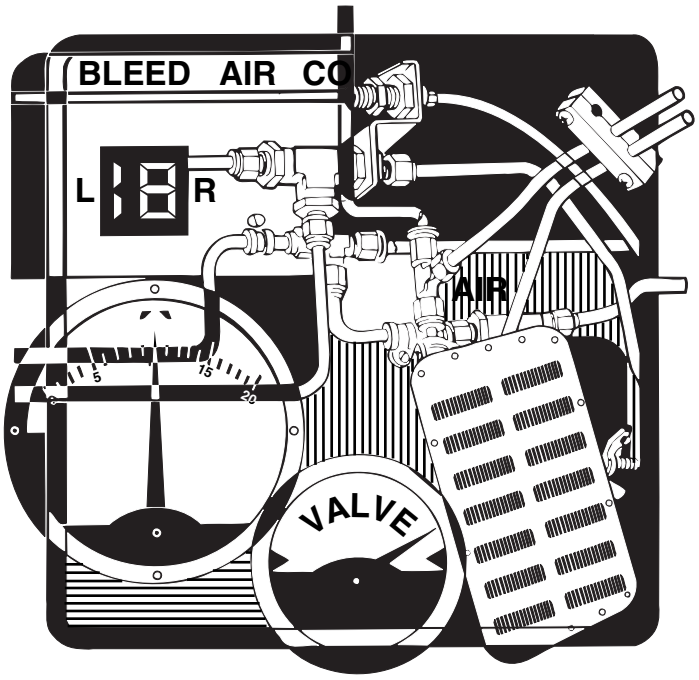
Engine Bleed Overheat 9-8

QUESTIONS 9-10

ILLUSTRATIONS

Figure	Title	Page
9-1	Pneumatic System	9-2
9-2	Pneumatic Valve and Schematic (Typical)	9-4
9-3	PAC Panel	9-5
9-4	APU Bleed-Air Controls	9-6
9-5	Pneumatic System Controls and Indicators	9-9

CHAPTER 9
PNEUMATICS



INTRODUCTION

The EMB-120 pneumatic system supplies engine bleed air for air conditioning/pressurization, pneumatic deicing, and pressurization of the hydraulic reservoirs.

GENERAL

There are two identical bleed-air systems, one from each engine. Each engine has two sources of bleed air; a low pressure bleed-air port ($P_{2.5}$) from the LP compressor, and a high pressure bleed-air port (P_3) from the HP compressor.

This bleed air is used to supply the air-conditioning/ pressurization system, the pneumatic deice system, and to pressurize the hydraulic system reservoirs.

A cross bleed line interconnects the left and right systems and permits either bleed system to supply the air-conditioning/ pressurization system.

APU bleed air may also be used to supply the air-conditioning/pressurization system.

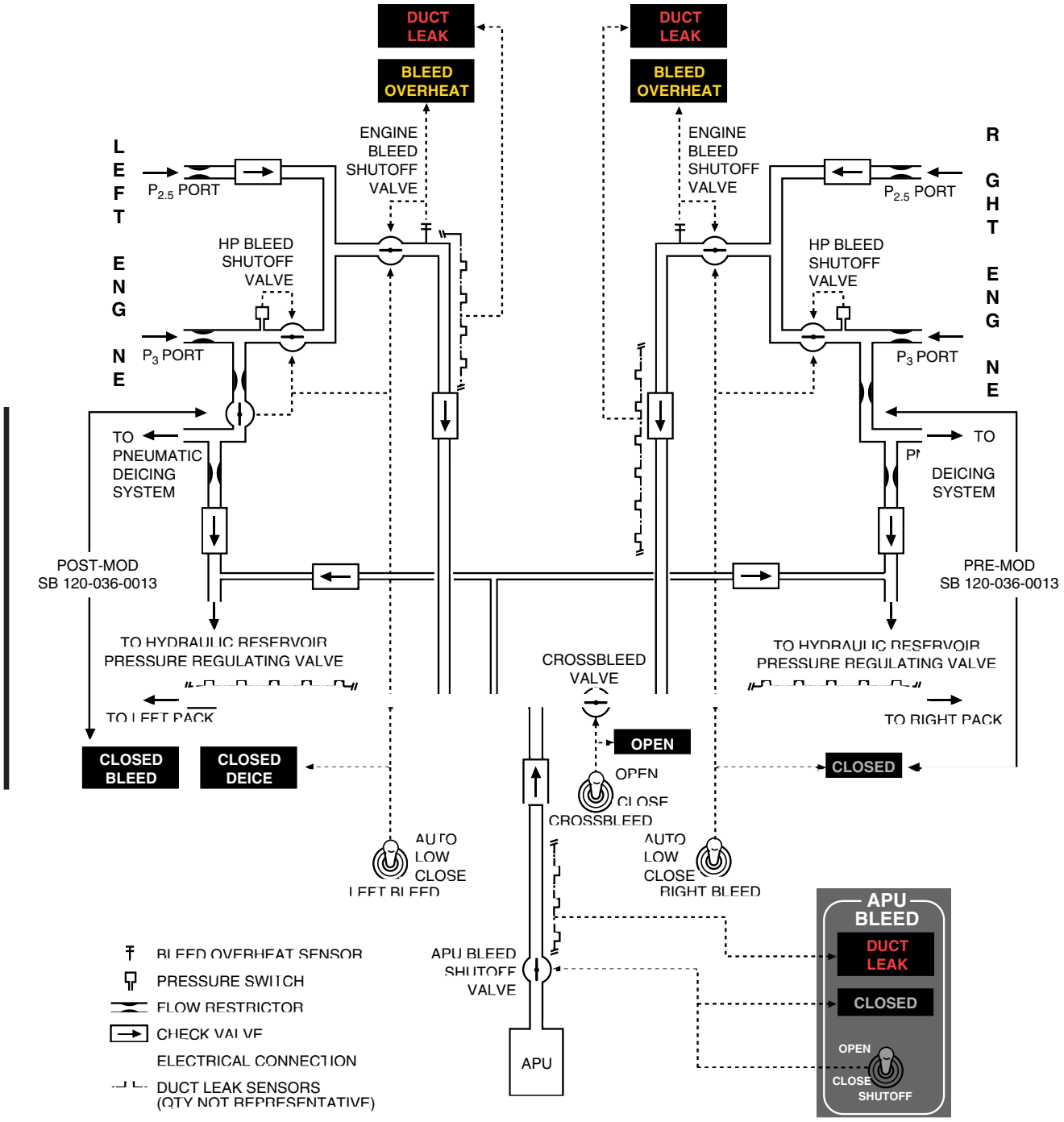


Figure 9-1. Pneumatic System

BLEED-AIR SYSTEM

ENGINE BLEED-AIR SYSTEM

Components

Each engine bleed-air system includes the following components, (Figure 9-1):

- LP bleed check valve
- HP bleed valve
- HP pressure switch
- Engine bleed shutoff valve
- Engine bleed check valve

Both systems are interconnected by a crossbleed valve.

Airplanes post-mod SB 120-036-0013 and subsequent are equipped with an additional bleed shutoff valve to the pneumatic deicing system that opens and closes with the engine bleed switch. The white CLOSED light is modified to display CLOSED BLEED and CLOSED DEICE.

LP Bleed Check Valve

Low pressure, $P_{2.5}$, bleed air is obtained from the LP compressor discharge air through a port in the engine case at engine station 2.5.

Each LP bleed port is provided with a flow restrictor and a check valve. The restrictor limits the volume of air bled from the LP compressor. The check valve keeps the LP port closed when the HP bleed valve is open. It also prevents reverse flow of air from the HP stage to the LP stage.

HP Bleed Valve

High pressure, P_3, bleed air is obtained from the HP compressor discharge air through a port in the engine case at engine station 3.0.

Each HP bleed is provided with a flow restrictor and an HP bleed shutoff valve. The restrictor limits the volume of air bled from the HP compressor. The HP bleed valve is an electrically controlled, pneumatically actuated butterfly valve.

The HP bleed valve is controlled by the HP pressure switch and the respective LEFT or RIGHT BLEED switch on the overhead PNEUMATIC/AIR CONDITIONING (PAC) panel.

HP Pressure Switch

The HP pressure switch controls the HP bleed valve when the respective engine bleed switch is in AUTO.

The HP pressure switch senses HP compressor discharge pressure and functions as follows:

CLOSE—When the HP compressor discharge pressure exceeds 58 psi, the pressure switch electrically commands the HP shutoff valve to close.

OPEN—When the HP compressor discharge pressure is below 45 psi, the pressure switch electrically commands the HP shutoff valve to open.

Engine Bleed Shutoff Valve

Each engine bleed-air system is provided with an engine bleed shutoff valve between the engine bleeds and the pneumatic crossbleed manifold.

The valve is physically located in the inboard leading edge of each wing next to the engine nacelle.

This engine bleed valve is used to isolate the engine bleed-air supply in the event of a malfunction in the system or to prevent the engine bleeds from causing any power loss during maximum performance or single engine operation. The engine bleed shutoff valve is controlled by the respective engine bleed switch on the overhead PAC panel.

The engine bleed shutoff valve also closes when the engine fire handle is actuated or when the bleed overheat switch, located downstream of the bleed shutoff valve, senses an overheat condition.

When the bleed shutoff valve is closed, the white CLOSED light on the PAC panel illuminates.

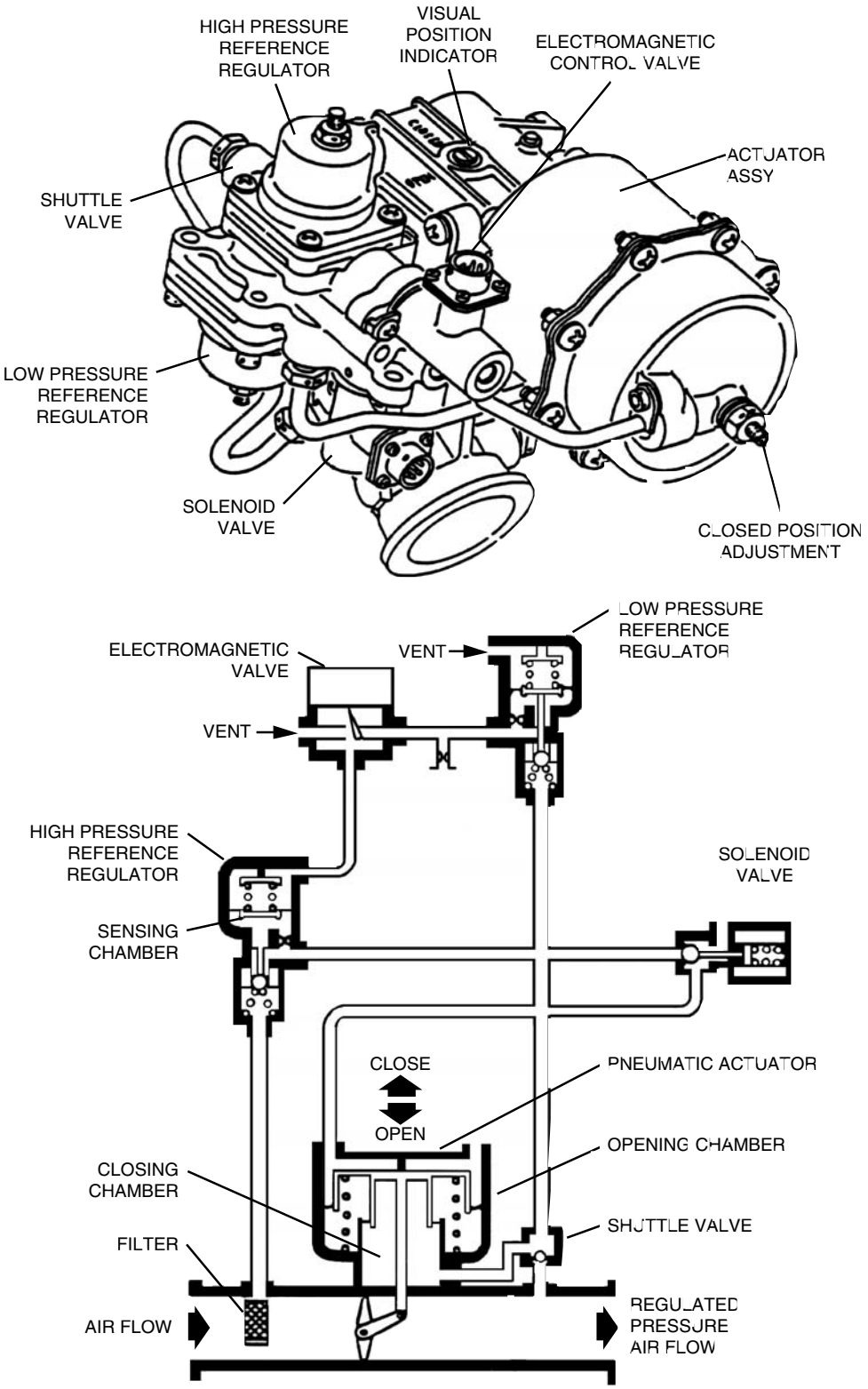


Figure 9-2. Pneumatic Valve and Schematic (Typical)

Check Valve

A check valve is located downstream of the engine bleed shutoff valve to prevent a reverse flow of air into the engine from the other engine or the APU.

Crossbleed Valve

The crossbleed valve provides a connection between the right and left pneumatic systems. When the valve is open, either engine, or the APU, can provide bleed air for both air-conditioning packs.

The crossbleed valve is physically located next to the right wing root.

The valve is closed or opened by means of a CROSSBLEED switch on the PAC panel. When the valve is open, a white OPEN light on the PAC panel illuminates.

Pneumatic Valve

A typically pneumatic valve in this aircraft requires both electrical power, normally supplied by control panel switch, and bleed air from a portion of the pneumatic system. The loss of either electrical power or bleed air will cause the valve to close because of spring action. See Figure 9-2 for a picture and schematic of a typical pneumatic valve.

Control and Monitoring

Pneumatic/Air Conditioning (PAC) Panel

The PAC panel, (Figure 9-3), is located on the lower right side of the overhead panel and contains operating controls and indicator lights for the bleed air, and air-conditioning systems.

Each engine bleed valve is controlled by a three position switch labeled AUTO/LOW/CLOSE located on the PAC panel.

Functions of the switch positions are as follows:

AUTO—The HP bleed shutoff valve is controlled by the HP pressure switch. The pressure switch commands the valve to close when the high stage pressure reaches 58 psi.

CLOSE—The engine bleed valve and the HP bleed valve are both closed.

LOW—The HP bleed shutoff valve is kept closed and the bleed air is supplied through the low pressure bleed check valve.

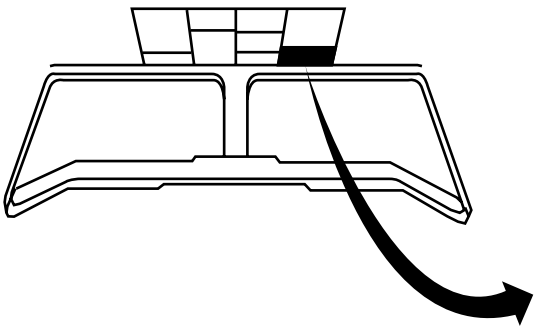




Figure 9-3. PAC Panel

Leading Edge, Empennage and Engine Air Inlet Deicing Controls

The operating controls and indicator lights for the pneumatic deicing boots on the leading edges of the wings, vertical and horizontal stabilizers, and engine air inlets are on the top and center sections of the overhead ICE/RAIN PROTECTION panel.

Refer to Chapter 10,"Ice and Rain Protection," for detailed information on deicing systems operation.

APU BLEED-AIR SYSTEM

The optional APU supplies bleed air on the ground or in flight for either or both air conditioning packs, and for hydraulic reservoir pressurization.

Refer to Chapter 6,"Auxiliary Power Unit," for more information on operation of the APU.

Components

APU Bleed Shutoff Valve

The APU bleed shutoff valve is installed in the APU bleed duct and, for normal operation, is controlled by the APU bleed SHUTOFF switch.

The valve also closes when the APU fire extinguishing switch is positioned to CLOSE.

In addition, the APU electronic control closes the APU bleed automatically should an APU abnormal operation condition exist.

APU Bleed Check Valve

A check valve is located downstream of the APU bleed shutoff valve to prevent reverse flow of air into the APU from engine bleed air.

Control and Monitoring

APU Bleed Section of APU Control Panel

The APU bleed-air SHUTOFF switch and indicator lights are in the APU BLEED section on the lower right side of the overhead APU CONTROL panel (Figure 9-4).

APU bleed air is controlled with both the SHUTOFF switch and the CROSSBLEED switch on the PAC panel.

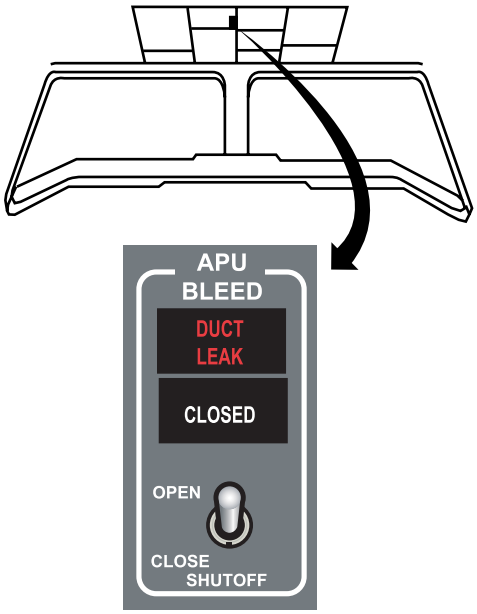


Figure 9-4. APU Bleed-Air Controls

BLEED-AIR SYSTEM OPERATION

ENGINE BLEED AIR

NOTE

The HP pressure is directly related to the LP pressure. When HP pressure is above 58 psi, the LP pressure is sufficient to operate the packs.

HP bleed air is used during lower power ground operation and low power descents due to the low bleed-air volume and pressure available from the LP bleed port.

As the engine power increases, air pressure from the LP bleed port is high enough to supply the pneumatic systems and the HP bleed valve is closed by the HP pressure switch.

Either the LP or HP bleed may be used. However, both bleeds cannot be used at the same time.

The HP bleed valve is enabled by the engine bleed valve switch in the AUTO position.

The HP pressure switch signals the HP bleed valve to close at 58 psi as the engine is accelerating (increasing P_3). The pressure switch re-opens the valve at 45 psi during engine deceleration (decreasing P_3).

During takeoff and landing the LP bleed is used to minimize power loss.

Takeoffs and landings may be made with all engine bleeds closed to completely eliminate power losses.

The packs may also be off (within certain operating limitations), or the APU may be used to operate the packs.

APU BLEED AIR

During ground, single engine, or maximum performance operation, the APU may be used as the source of bleed air to the air-conditioning packs.

The APU supplies the left pack through a duct connected to the left crossbleed manifold and pack. To supply the right pack, the crossbleed valve must be opened.

The system is designed so the APU and an engine can not simultaneously supply bleed air to the same pack. Priority is given to the engine.

An electrical circuit prevents the APU bleed from being opened if the left engine bleed is open and the crossbleed is closed. If the crossbleed is open, the APU bleed can only be opened if the right engine bleed is closed also.

The following example demonstrates bleed valve operation:

- Engines—Before start
- APU—Operating
- APU bleed switch—OPEN
- Engine bleed switches—CLOSE
- Crossbleed switch —OPEN
  - When the right engine is started and the right engine bleed switch is selected to LOW or AUTO, the APU bleed valve closes. The right engine supplies the right pack directly and the left pack through the crossbleed valve.
  - If the crossbleed valve is closed, the APU bleed valve opens and supplies the left pack. The right engine continues to supply the right pack.
  - When the left engine is started and the left engine bleed switch is set to LOW or AUTO, the APU bleed valve closes. The left engine supplies the left pack and the right engine supplies the right pack.

WARNING SYSTEMS

Each bleed-air system is protected by a duct leak warning system. The engine bleed-air system also includes an engine bleed overheat warning.

DUCT LEAK WARNING

The duct leak warning has sensors installed close to each duct interconnection. Besides monitoring the air conditioning bleed ducts, the duct leak warning also monitors the pneumatic lines of the deicing system and the air conditioning packs. The sensors are activated if the temperature in the bleed duct region reaches 91° C (195° F).

If a duct bleed-air leak is sensed in either engine bleed-air system the following occurs:

- A red DUCT LEAK light illuminates on the PAC panel.
- A three-chime aural alert sounds.
- The voice warning “DUCT LEAK” is given.
- A red DUCT LEAK light illuminates on the MAP.
- The master WARNING lights flash.

ENGINE BLEED OVERHEAT

The engine bleed overheat warning system is activated by an overheat switch installed downstream of the engine bleed-air shutoff valve.

If the overheat switch senses a temperature in excess of 315° C (600° F), the following occurs:

- The engine bleed shutoff valve closes.
- The amber BLEED OVERHEAT light on the PAC panel illuminates.
- A single-chime aural alert sounds.
- The amber AIR COND light on the MAP illuminates.
- The master CAUTION lights flash.

LEFT **DUCT LEAKAGE** LIGHT (RED)
ILLUMINATED—Bleed-air leak in the bleed ducts with temperature exceeding 91°C (195°F)

Lights in conjunction with red DUCT LEAK light on MAP

LEFT **BLEED OVERHEAT** LIGHT (AMBER)
ILLUMINATED—Overheat condition in the left bleed duct with temperature exceeding 315°C (600°F)

Lights in conjunction with amber AIR COND light on MAP

The engine bleed shutoff and HP bleed shutoff valves close simultaneously.

LEFT BLEED **CLOSED** LIGHT (WHITE)
ILLUMINATED—Left engine bleed shutoff valve closed



CROSSBLEED **OPEN** LIGHT (WHITE)
ILLUMINATED—Crossbleed valve open

CROSSBLEED SWITCH
OPEN—Opens the crossbleed valve, connecting the left and right bleed systems

CLOSE—Closes the crossbleed valve, isolating the left and right bleed systems

RIGHT BLEED SWITCH
AUTO—The engine bleed shutoff valve is kept open. The HP bleed shutoff valve may be open or closed depending on the HP pressure as sensed by the HP pressure switch.

NOTE:
For airplanes Post Mod. SB 120-021-0015 or S/N 120.025 and subsequent, the HP bleed shutoff is kept open if the pack control selector is at HI position.

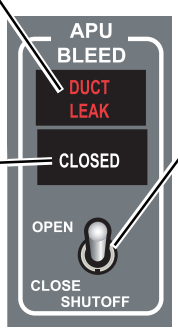
LOW—The engine bleed shutoff valve is kept open and the HP bleed shutoff valve is closed. Engine bleed is from LP port only.

CLOSE—Both the engine bleed shutoff valve and the HP bleed shutoff valve are closed.

APU **DUCT LEAKAGE** LIGHT (RED)
ILLUMINATED—Bleed-air leak in the APU duct with temperature exceeding 71°C (160°F).

Lights in conjunction with red DUCT LEAK light on MAP.

APU BLEED **CLOSED** LIGHT (WHITE)
ILLUMINATED—APU bleed shutoff valve closed



APU BLEED **SHUTOFF** VALVE SWITCH
OPEN—Opens APU bleed shutoff valve four seconds after 95% rpm, allowing APU to provide bleed air.

CLOSE—Close APU bleed shutoff valve

Figure 9-5. Pneumatic System Controls and Indicators

QUESTIONS

1. Engine bleed air is ducted from 3 separate ports.

A. True
B. False

2. The duct leak warning system is activated when temperature in the duct region reaches:

A. 99°F
B. 125°F
C. 160°C
D. 195°F

3. If a "duct leak" is detected, the bleed-air system will automatically shut off.

A. True
B. False

4. If a "bleed overheat" is detected, the bleed-air system will automatically shut off.

A. True
B. False



CHAPTER 10
ICE AND RAIN PROTECTION

CONTENTS

Page

INTRODUCTION 10-1
GENERAL 10-1
Deicing 10-1
Anti-Icing 10-2
Rain Removal 10-2
DEICING SYSTEM 10-3
Leading Edge Deicing 10-3
Engine Air Inlet Deicing 10-6
Propeller Deicing 10-7
Deicing System Monitor 10-9
ANTI-ICING SYSTEM 10-10
Components 10-10
Ice Detection and Speed Warning 10-14
RAIN REMOVAL 10-16
Windshield Wipers 10-16
QUESTIONS 10-16

ILLUSTRATIONS

Figure	Title	Page
10-1	EMB-120 Ice and Rain Protection	10-2
10-2	Pneumatic Deicing Systems Schematic	10-4
10-3	Leading Edge Deicing System Controls	10-5
10-4	Engine Deicing Controls	10-7
10-5	Propeller Deicing Controls	10-8
10-6	Propeller Deice Cycles	10-8
10-7	Deicing System Monitor Controls	10-10
10-8	Pitot-Static Anti-Ice Controls	10-11
10-9	Angle-of-Attack/Slip-Skid Anti-Ice Controls	10-12
10-10	Total Air Temperature Anti-Ice Controls	10-13
10-11	Windshield Anti-Ice Controls	10-13
10-12	Ice Detection Panel	10-14
10-13	Low Speed Alarm Panel	10-15
10-14	Windshield Wiper Controls	10-16
10-15	Ice and Rain Protection Systems Controls and Indicators	10-17

CHAPTER 10
ICE AND RAIN PROTECTION



INTRODUCTION

The EMB-120 Brasilia is approved for flight in known icing conditions when the required equipment is installed and operational.

System operation and tests must be in strict conformance with the FAA-approved *Airplane Flight Manual (AFM)*, the *Maintenance Manual*, or the operator's approved *Minimum Equipment List (MEL).*

GENERAL

The aircraft incorporates ice and rain protection systems designed to prevent or dispose of ice and rain formation on various parts of the aircraft. The ice and rain protection controls are all located on the upper right side of the cockpit overhead panel.

DEICING

Deicing systems are designed to remove ice after it has formed on an aircraft surface.

Deicing is provided for the following areas (Figure 10-1):

- Wing and empennage leading edges (pneumatic boots).
- Engine air inlet and bypass duct (pneumatic boots).
- Propeller blades (electric).

To reduce crew workload a full time automatic deicing systems monitor is provided to annunciate component or system malfunctions.

ANTI-ICING

Anti-icing systems are designed to prevent ice from forming on an aircraft surface.

Anti-icing is provided for the following areas (Figure 10-1):

- Pitot-static probes and static ports.
- Angle-of-attack sensors.
- Slip-skid sensor.
- Total air temperature probe.
- Forward windshields.

RAIN REMOVAL

Rain removal is provided to the forward windshields by electric motor-driven windshield wipers (Figure 10-1).

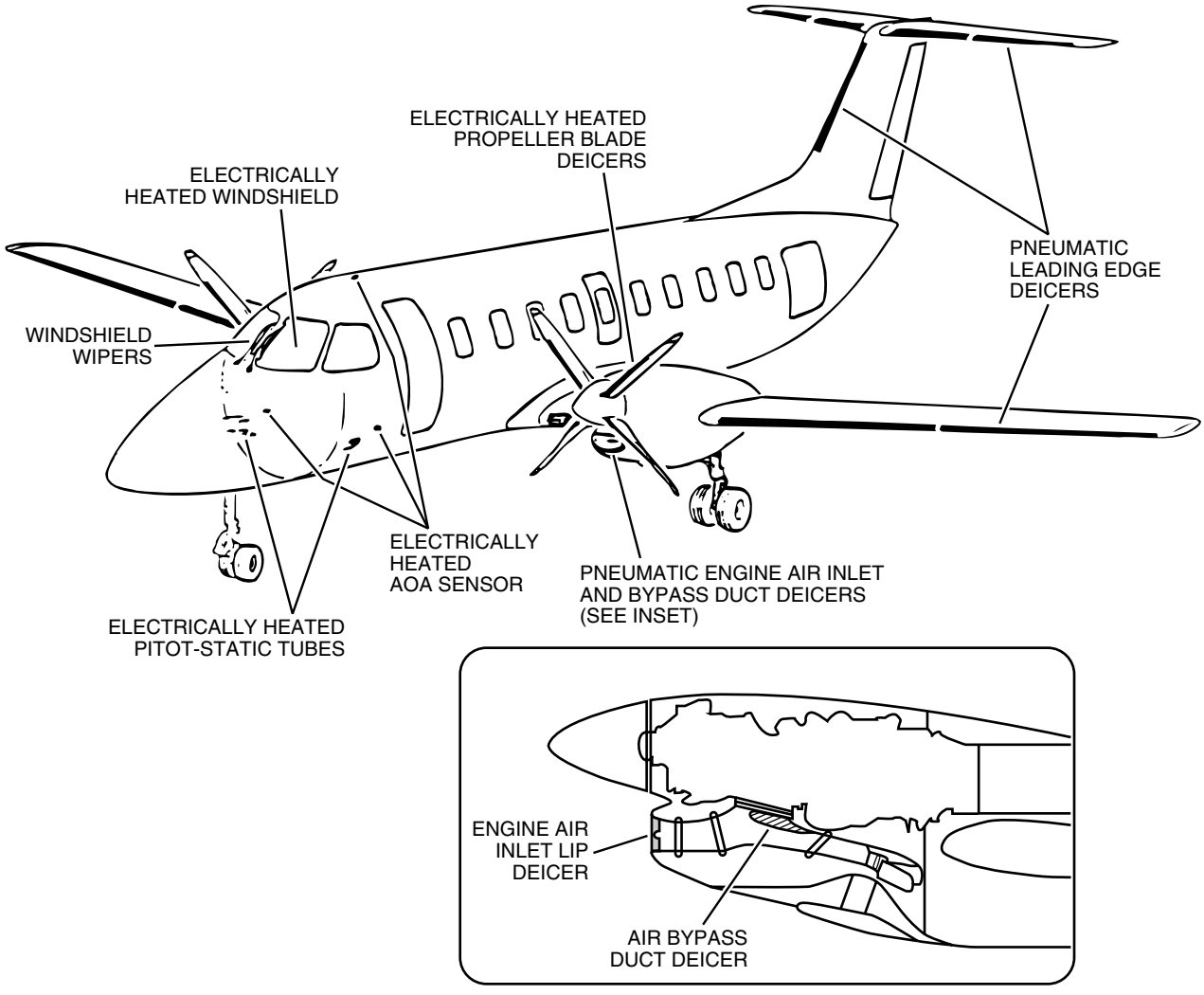


Figure 10-1. EMB-120 Ice and Rain Protection

DEICING SYSTEM

LEADING EDGE DEICING

The leading edges of the wing and the horizontal and vertical stabilizers are deiced by use of inflatable pneumatic boots (deicers), (Figure 10-1). The system deices the leading edges by pneumatically inflating the rubber deicers. As the deicers inflate they crack the ice, which is then removed by the airstream. A minimum of 80% NH on either engine is required for operation of this system.

Components

The wing and empennage leading edge deicing system consists of the following components (Figure 10-2):

- Deicers
- Pressure regulator valves
- Water separators
- Check valves
- Ejector flow control valves
- Timers
- Pressure switches

Deicers—The eleven deicers, constructed of rubber, are bonded to recessed areas of the leading edges to maintain the airfoil shape when the deicers are deflated.

Each deicer contains tubes that run the length of the deicers. These tubes are inflated simultaneously to crack the ice on the deicer surface.

Each wing has three deicers, designated outboard, middle, and inboard.

Each horizontal stabilizer has two deicers, an outboard and an inboard. A single deicer is installed on the vertical stabilizer.

Pressure Regulator Valves—The pressure regulator valves receive high-pressure bleed air from the engines and reduce it to a working pressure of 20 psi. If a regulator fails a pressure relief valve opens at 23 psi to protect the system.

Water Separators—The water separators remove water from the tubing and deicers. They use the cool surfaces of the separator and centrifugal force to condense the moisture in the air to water. The water is then drained overboard.

Check Valves—Check valves permit normal deicer system operation, in the event of single-engine operation, by preventing the loss of bleed air through the inoperative engine.

Ejector Flow Control Valves—The ejector flow control valves control the inflation and deflation of the deicers. They are controlled by the timers.

When the valves are electrically energized by the timer, air pressure enters the deicer tubes, inflating the deicer.

When the valves are deenergized, the air is routed through a venturi. This creates a vacuum in the deicer tubes, deflating the deicers.

Timers—For redundancy, there are two timers in the leading edge deicing system. Either may be selected to operate the system. Timer one is powered by Emergency DC Bus 1, and timer two by Emergency DC Bus 2.

The timer establishes the sequence and operating cycle of the ejector flow control valves.

The leading edge deicers are sequenced in pairs to maintain aerodynamic symmetry; and at a regular interval to minimize bleed-air losses. The deicer sequence is:

1. Outboard wings
2. Middle wings
3. Inboard wings and inboard horizontal stabilizers
4. Outboard horizontal stabilizers and vertical stabilizer

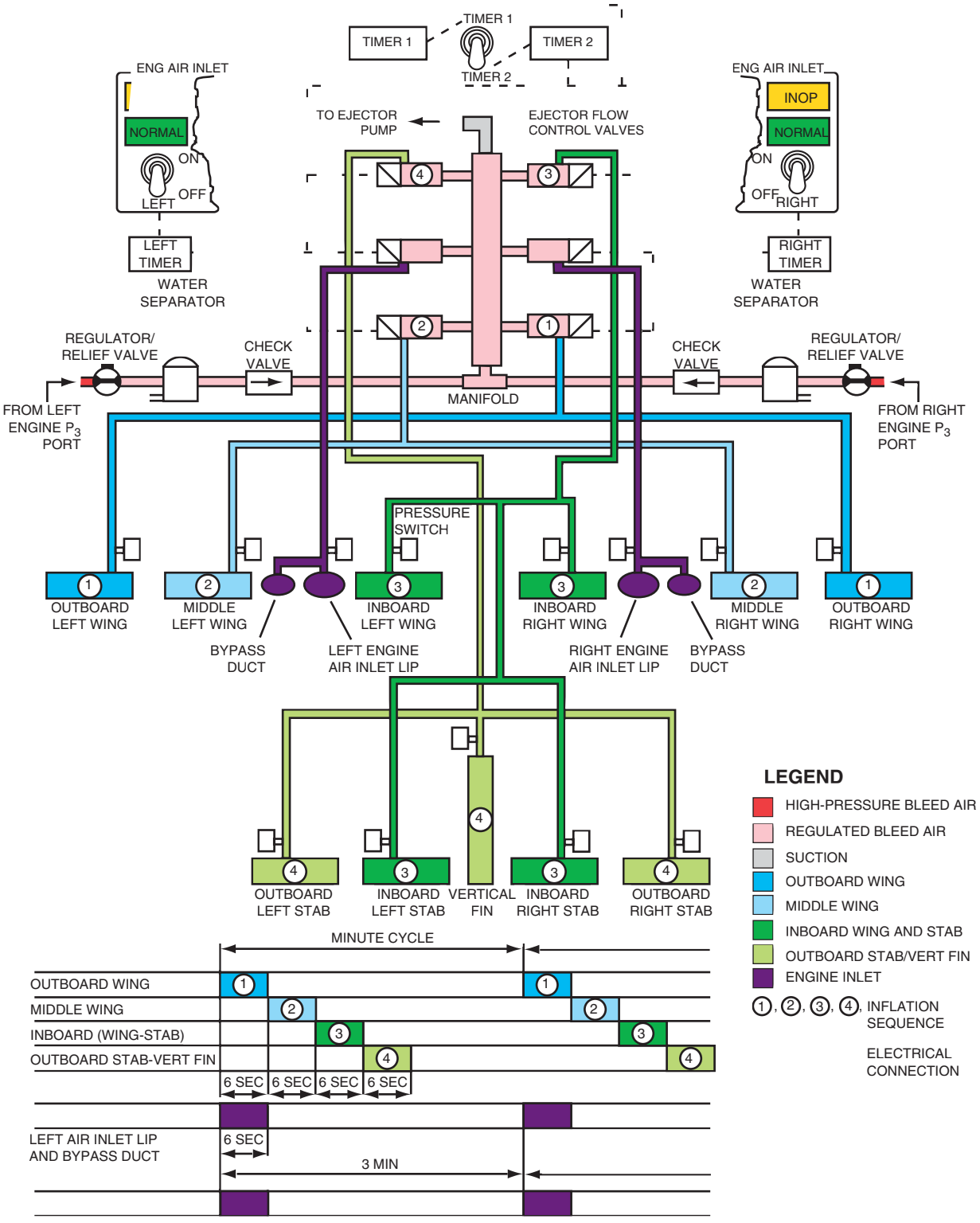


Figure 10-2. Pneumatic Deicing Systems Schematic

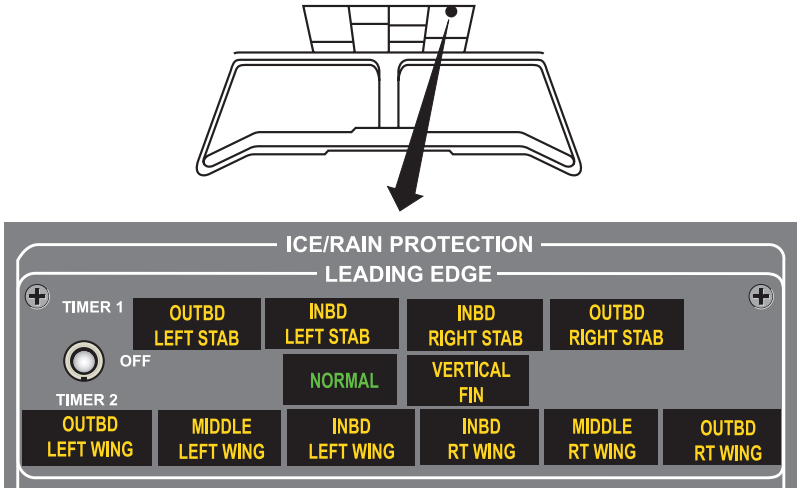


Figure 10-3. Leading Edge Deicing System Controls

The timer operation cycle is repeated every minute. Each deicer is inflated for six seconds and deflated for 54 seconds.

NOTE

Many aircraft have decals on the right side of this panel labeled HEAVY and LIGHT (FIGURE 10-3), but have a plug instead of a switch. In this case, the timer cycle has been hard-wired to the heavy position, per the pertinent AD.

Pressure Switches—The pressure switches provide the input used to monitor system operation.

As long as pressure in the deicer is above 16 psi, a green NORMAL light is illuminated on the control panel. If a deicer does not inflate, or remains inflated beyond the established time, the green light will extinguish and the applicable inoperative light illuminates.

Control and Monitoring

The wing and empennage leading edge deicing system is controlled and monitored by the flight crew using the switches and indicating lights in the LEADING EDGE section of the cockpit overhead ICE/RAIN PROTECTION panel (Figure 10-3).

The switch selects the timer to be used, and the operating cycle. The timer select switch, labeled TIMER 1/OFF/TIMER 2, selects the desired timer or turns the system off.

When a timer is selected, and the pressure in the deicer reaches 16 psi, the pressure switches send a signal to the deicing system monitor, which in turn sends a signal to illuminate the green NORMAL light on the overhead panel.

There are eleven amber indicator lights (one for each deicer) on the leading edge panel. If one of the deicers does not inflate, or remains inflated beyond the established time, the green NORMAL light extinguishes and the amber light corresponding to the malfunctioning deicer illuminates. These lights ***do not*** cycle with the deicer boots.

Operation

The timer select switch is used to turn the system on by selecting TIMER 1 or TIMER 2. Only one timer is in operation at a time, with the other available as a backup.

When the timer switch is placed to the TIMER 1 position, the system operates as follows (Figure 10-2):

- Timer 1 is energized and signals the ejector flow control valve for the outboard wings to energize.
- Bleed air from the engines passes through the regulators, separators, and check valves to the deicing system manifold where the appropriate ejector flow valve routes the regulated pressure to the outboard deicing boots.
- The boots inflate for 6 seconds.
- The timer then switches to the next ejector flow control valve in the sequence.
- The deenergized ejector flow control valve applies a vacuum to deflate the deicer boots.

The green NORMAL light illuminates when the leading edge deicing system is turned on (TIMER 1 or TIMER 2 selected), and the system is operating normally.

If a leading edge deicer malfunctions, the NORMAL light extinguishes and the appropriate amber indicator light illuminates.

ENGINE AIR INLET DEICING

The engine air inlet deicing system uses the same pneumatic control system as the wing and empennage leading edge deicing system. A minimum of 80% N_H from either engine is required to make this system operational.

Components

Both systems share the same pressure regulator valves, water separators, and check valves. They ***do not*** share ejector flow control valves, timers, or pressure switches.

Deicers—Each engine has two deicers. One of the deicers is in the air inlet lip and the other is in the upper forward region of the bypass duct (Figure 10-1 inset).

Ejector Flow Control Valves—The ejector flow control valves function the same as those in the leading edge deicing system.

Timers—Each engine has its own timer to provide deicer inflation and deflation. The left timer is powered by Emergency DC Bus 1, and the right by Emergency DC Bus 2.

When selected, the air inlet and bypass duct deicers for each engine are ***simultaneously*** inflated on a three-minute cycle. Each deicer is inflated for 6 seconds and deflated for 174 (Figure 10-2).

Each engine operates independently of the other engine and the leading edge deicing systems.

Pressure Switches—The pressure switches function the same as those in the leading edge deicing system.

Control and Monitoring

The engine deicing system is controlled and monitored by the flight crew using the switches and indicating lights in the ENG AIR INLET section of the ICE/ RAIN PROTECTION panel (Figure 10-4).

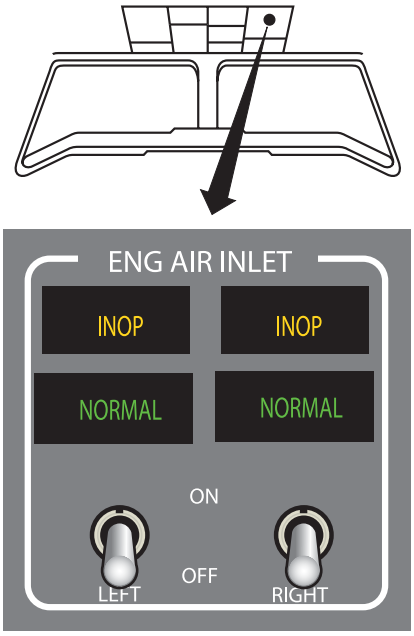


Figure 10-4. Engine Deicing Controls

The LEFT and RIGHT switches on the panel are used to select the timers of the left and right engine deicing system. The switches are labeled ON/OFF.

A green NORMAL light, above each switch, illuminates when that engine air inlet deicing system is selected and operating normally.

An amber INOP light, above each NORMAL light, indicates a failure of the engine deicers.

Operation

The engine deicing system is activated by placing the LEFT and RIGHT switches to the ON position.

When the timer switches are placed in the ON position, the system operates as follows (Figure 10-2):

- The timers are energized and signal the ejector flow control valves for the engine inlet and bypass ducts to energize.
- Bleed air from the engines passes through the regulators, separators, and check valves, to the deicing system manifold where the appropriate ejector flow valves route the regulated pressure to the engine deicing boots.
- The boots inflate for 6 seconds.
- The timers deenergize the ejector flow control valves, deflating the boots.
- The boots remain deflated for the remainder of the 3 minute cycle.

PROPELLER DEICING

The aircraft uses electrical power for deicing the propellers.

Heating elements, embedded in the propeller blades, melt the ice, which is then ejected from the blades by centrifugal force.

Components

The propeller deicing system consists of the following components:

- Timers
- Brush blocks
- Slip rings
- Heaters

Timers—The timers establish the operation cycle of the propeller deicing heaters.

There are two timers in the propeller deicing system. Either timer may be used for deicing. Timer one is powered by DC Bus 1, and timer 2 by DC Bus 2.

Brush Blocks—The brush blocks provide the means of transferring electrical power from the front of the engine to the slip rings on the propeller. Each engine contains a brush block with six brushes. The brushes contact the slip rings on the propeller.

Slip Rings—The slip rings receive the electrical power from the brush blocks and transfer this power to the propeller heater elements.

Heaters—The propellers are heated by electric elements integrally molded into the leading edge of each propeller blade, and electrically connected to the slip ring.

The left prop heater is powered by Relay Box DC Bus 1, and the right prop heater by Relay Box DC Bus 2.

Control and Monitoring

The controls and indicators for the propeller deicing system are located in the PROPELLER section of the overhead ICE/RAIN PROTECTION panel (Figure 10-5).

The system is controlled by two switches:

- TIMER SELECT switch—labeled TIMER 1/OFF/TIMER 2. Turns on the system with the selection of either timer. Only one timer is in operation at a time, the other is a backup.

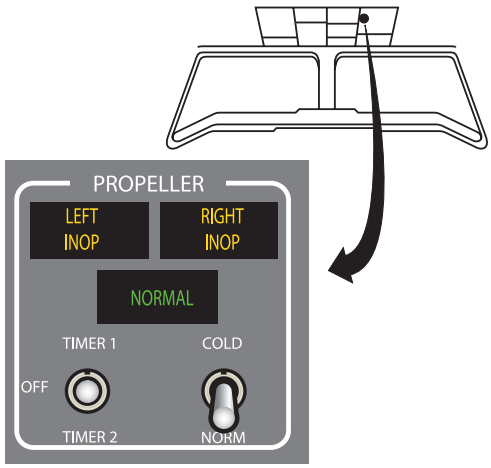


Figure 10-5. Propeller Deicing Controls

- CYCLE SELECT switch—labeled COLD/NORM. Selects one of two cycle times (Figure 10-6):
- COLD—20 seconds on, 60 seconds off
- NORM—10 seconds on, 60 seconds off

The propeller deicing system is monitored with a green NORMAL light and two amber lights (LEFT INOP and RIGHT INOP).

When the propeller deicing system is turned on by placing the timer switch to either the TIMER 1 or TIMER 2, and the system is operating normally, the green NORMAL light illuminates.

The amber LEFT INOP and RIGHT INOP lights indicate a deicer failure.

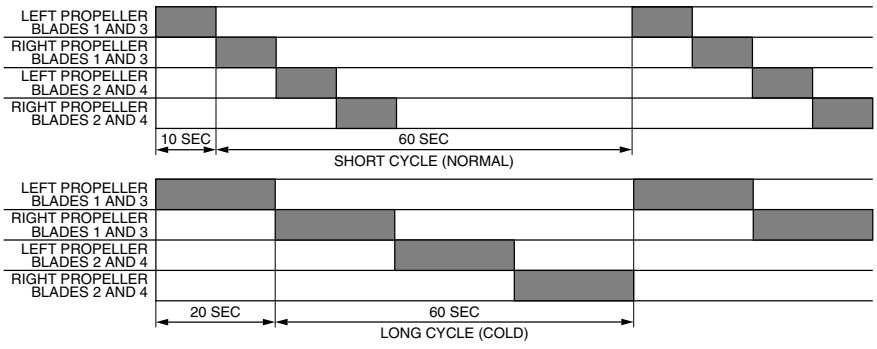


Figure 10-6. Propeller Deice Cycles

Operation

The propeller deicing system is turned on by selecting TIMER 1 or TIMER 2. A COLD or NORM cycle is selected depending upon the icing conditions.

The recommended temperatures for selecting COLD or NORM are:

- Temperature colder than minus 10°C use COLD.
- Temperature warmer than minus 10°C use NORM.

In the TIMER 1 position, electrical power is supplied from DC Bus 1 to both brush blocks.

The brush blocks transfer the electrical power to the slip rings which are connected to the propeller blade heating elements.

Electrical power is supplied to opposite propeller blades on the same propeller to ensure dynamic balance under icing conditions and to keep a proper sequence.

The heating sequence is:

- Left propeller/blades 1 and 3
- Right propeller/blades 1 and 3
- Left propeller/blades 2 and 4
- Right propeller/blades 2 and 4

Protection against overheat damage to the propeller during ground operations is provided through the engine low oil pressure switch. The switch prevents propeller deicing system operation when oil pressure is below 40 psid.

If the propeller deicing system cycling is interrupted, or the electrical current is below 35 amps or above 50 amps, the following occurs:

- The green NORMAL light extinguishes.
- The LEFT INOP or RIGHT INOP light illuminates.
- The DEICE light on the MAP illuminates.
- A single-chime aural alert sounds.
- The master CAUTION lights flash.

DEICING SYSTEM MONITOR

The deicing system monitor provides a constant operational check of all leading edge deicers, engine deicers, and the propeller deicing system; as well as the indicator/alarm lights associated with these systems. It also self-tests its own operation.

The monitor is automatically powered and operating when the aircraft is powered. It is powered by two separate and isolated power supplies: DC Bus 1 and DC Bus 2.

When a deicing system is turned on and functioning properly, the monitor illuminates the green NORMAL light for that system.

If the system is not functioning properly, the deicing system monitor extinguishes the green NORMAL light and illuminates the amber INOP light for the failed system. As a result:

- The DEICE light on the MAP illuminates.
- A single-chime aural alert sounds.
- The master CAUTION lights flash.

If a fault exists in a monitored system that is ***not*** energized, with the exception of extinguishing the NORMAL light (which was not on), the deicing system monitor generates the same indications and warnings as with an energized system.

The controls for the deicing system monitor are located in the MONITOR section of the overhead ICE/RAIN PROTECTION panel (Figure 10-7).

The MONITOR section contains a TEST/ON switch and an amber INOP light.

The TEST/ON switch is spring-loaded to the ON position. The TEST position checks all deicing system lights and the monitor.

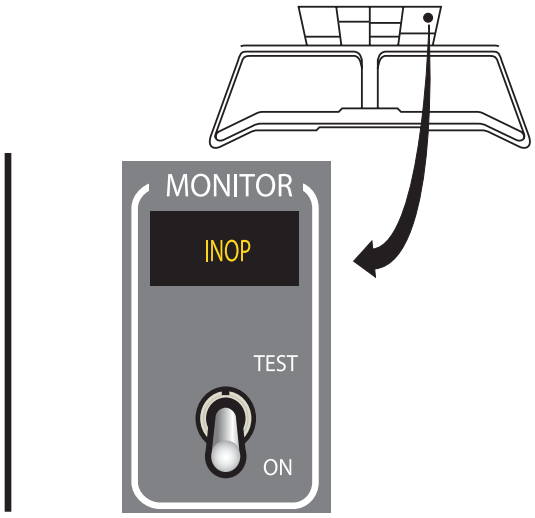


Figure 10-7. Deicing System Monitor Controls

The INOP light illuminates when a fault occurs in the monitor.

The deicing system monitor ***does not*** check the full operation of the deicing systems, only their abnormal or inoperative status.

The deicing system monitor ***does*** check its own internal circuitry through self-test.

Monitor Test

When the TEST/ON switch is placed in the TEST position and no failure is detected, the following occurs:

- Any illuminated light on the deicing panel extinguishes.
- A single-chime aural alert sounds.
- The INOP light on the MONITOR panel illuminates.
- The DEICE light on the MAP illuminates
- The master CAUTION lights flash.
- All lights on the deicing panels illuminate.
- All lights on the deicing panels extinguish.

After the 7-second test time, the equipment revalidates all deicing systems.

If a failure is detected in the deicing system monitor itself, the monitor INOP light and the DEICE light on the MAP illuminate and remain on until the fault is rectified.

Failure of the monitor will not affect the operation of the monitored systems. However, the green NORMAL lights and amber INOP lights associated with the systems will not illuminate. If a failure occurs in one of the monitored systems, that failure would not be annunciated.

ANTI-ICING SYSTEM

Anti-icing protection is provided for several components by electrically heated anti-ice elements.

COMPONENTS

Electrical anti-icing protection is provided for the following components:

- Pitot-static probes and ports
- Angle-of-attack sensors
- Slip-skid sensor
- Total air temperature probe
- Windshields

If any electrical anti-icing system should fail the indications to the crew are identical. Those indications are:

- The related INOP light illuminates.
- The DEICE light on the MAP illuminates.
- A single-chime aural alert sounds.
- The master CAUTION lights flash.

Pitot-Static Probes and Ports

The aircraft incorporates three combined pitot-static probes and two related outflow valve ports which are anti-iced by the use of electrical heating elements.

The three pitot-static probes (and power supply) are:

- The pilot's pitot-static probe (Emergency DC Bus 2).
- The copilot's pitot-static probe (DC Bus 2).
- The auxiliary pitot-static probe (DC Bus 3).

The two related outflow valve static ports are:

- The pneumatic outflow valve static port (DC Bus 2).
- The electropneumatic outflow valve static port (DC Bus 3).

The pitot-static anti-icing system is controlled from the PITOT/STATIC section of the overhead ICE/RAIN PROTECTION panel (Figure 10-8).

There are three ON/OFF switches labeled:

1—Controls the pilot's pitot-static heater.

2—Controls the copilot's pitot-static heater and the pneumatic outflow valve static heater.

AUX—Controls the auxiliary pitot-static heater and the electropneumatic outflow valve static heater.

There are also three amber INOP lights above each switch. The lights illuminate when:

- The pitot-static switches are OFF
- The pitot-static switches are ON and the respective system fails.

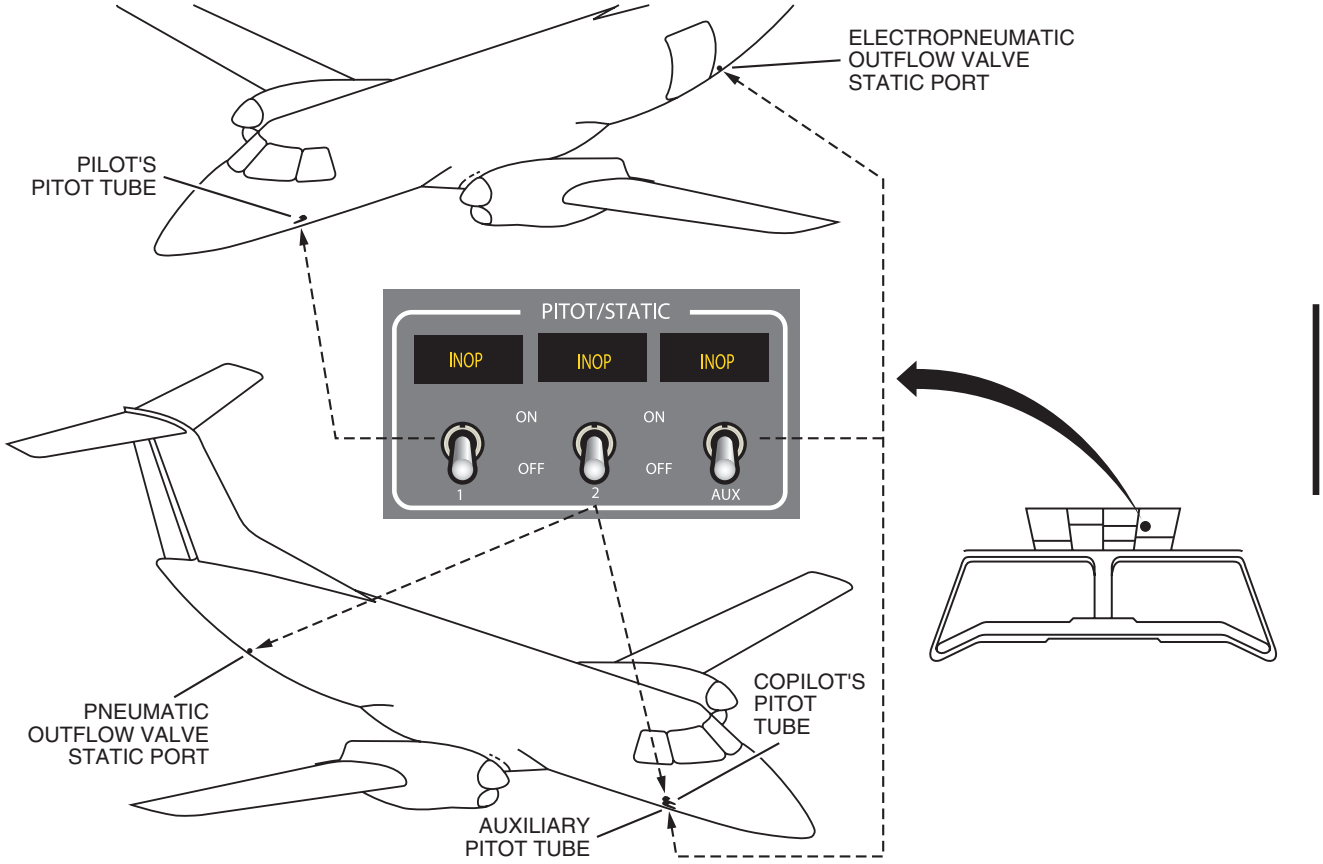


Figure 10-8. Pitot-Static Anti-Ice Controls

Angle-Of-Attack Sensors

The aircraft incorporates two angle-of-attack sensors that are anti-iced by the use of electrical heating elements.

The left AOA sensor anti-icing heater is powered by DC Bus 1, while the right AOA sensor anti-icing heater is powered by DC Bus 2.

The angle-of-attack sensor anti-icing is controlled by the ON/OFF switches, labeled LEFT and RIGHT, in the AOA SENSOR section of the ICE/RAIN PROTECTION panel (Figure 10-9).

An amber INOP alarm light is located above each switch. Should the LEFT or RIGHT system fail, the respective INOP light illuminates accompanied by the associated indications and warnings.

Slip-Skid Sensor

The aircraft incorporates one slip-skid sensor. It is anti-iced by the use of an electrical heating element powered by DC Bus 2.

The slip-skid sensor anti-icing heater is controlled by the use of an ON/OFF switch located on the SLIP section of the ICE/RAIN PROTECTION panel (Figure 10-9).

An amber INOP light is located above the switch.

If the slip-skid sensor anti-icing system should fail, the INOP light illuminates accompanied by the associated indications and warnings.

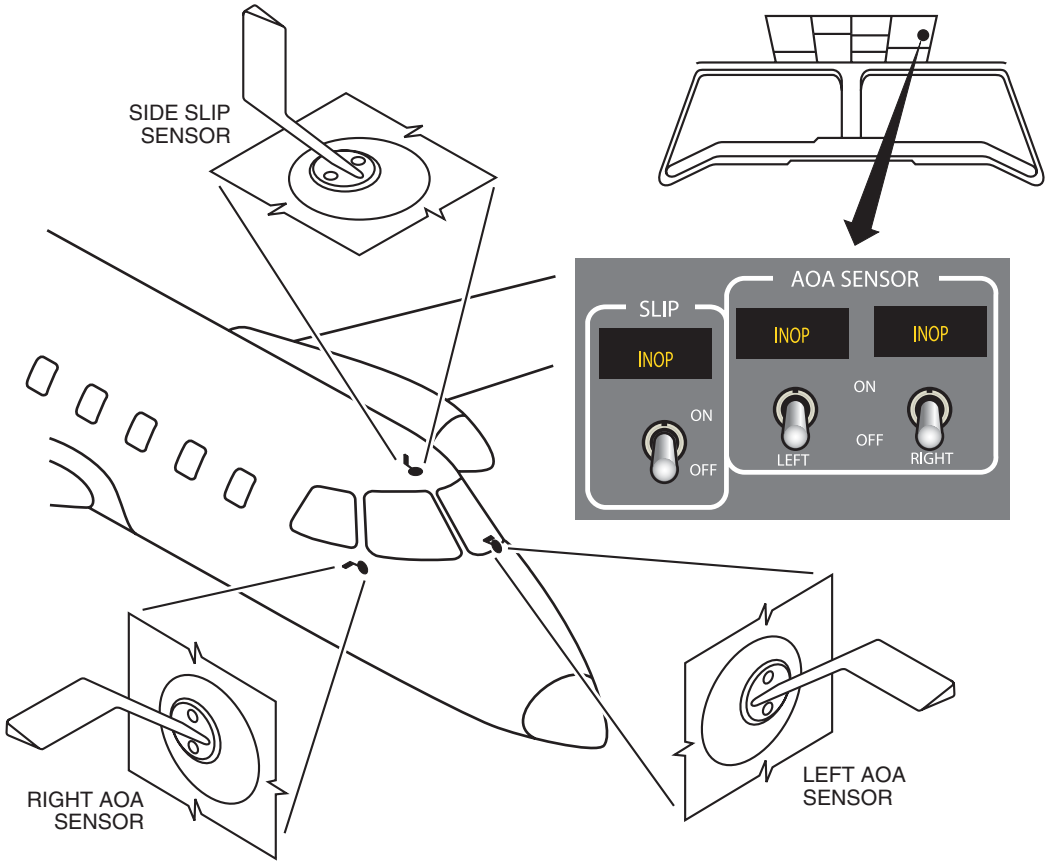


Figure 10-9. Angle-of-Attack/Slip-skid Anti-Ice Controls

Total Air Temperature Probe

The total air temperature probe is anti-iced by an electrical heating element powered by DC Bus 1.

The heater is controlled by the ON/OFF switch in the TAT section of the ICE/RAIN PROTECTION panel (Figure 10-10).

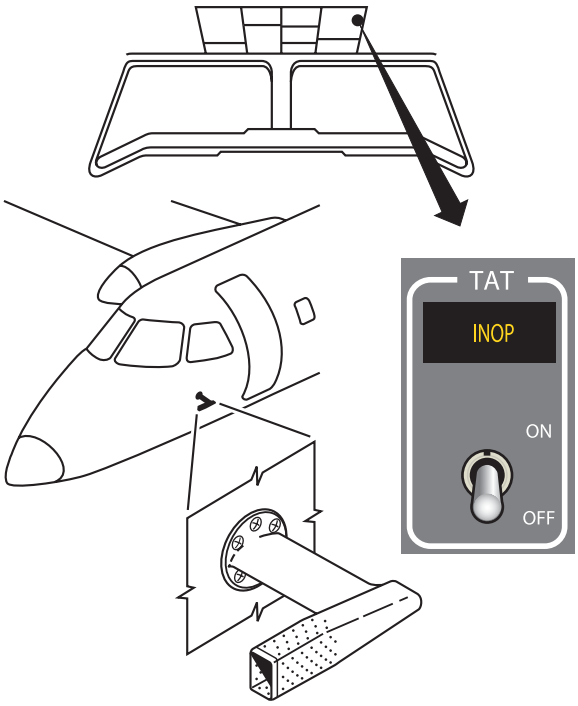


Figure 10-10. Total Air Temperature Anti-Ice Controls

An amber INOP light is located above the switch. If a total air temperature anti-icing system should fail, the related INOP light illuminates accompanied by the associated indications and warnings.

Windshields

The aircraft incorporates heater elements in the interlayer of the pilot's and copilot's windshields.

The left windshield heater is powered by Relay Box DC Bus 3, and the right by Relay Box DC Bus 2.

The windshield anti-icing system is controlled from the WINDSHIELD section of the ICE/RAIN PROTECTION PANEL (Figure 10-11). There are two ON/OFF switches, each labeled LEFT and RIGHT, and a TEST switch labeled LEFT and RIGHT.

Two amber INOP lights and two green ON lights are located above the ON/OFF switches.

Windshield Anti-Icing System Operation—The windshield controller maintains the windshield temperature between 40°C and 45°C with an overheat limit of 55°C. When a windshield is being heated, the related green ON light is illuminated on the panel.

The rate at which the ON lights cycle on and off is dependent upon the ambient temperature. In extremely cold temperature, the ON lights may be on continuously as the windshield may never reach 45°C.

If a system failure occurs, the relevant green ON light extinguishes and its amber INOP light illuminates, accompanied by the associated indications and warnings.

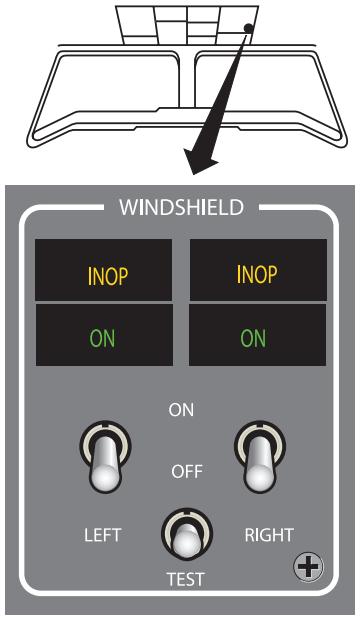


Figure 10-11. Windshield Anti-Ice Controls

If a windshield temperature exceeds 55°C, an overheat warning is received. The crew should switch off the related windshield anti-icing system and refer to the checklist.

Windshield Overheat System

Test—The windshield overheat circuit may be checked using the TEST switch. The switch must be held to the LEFT and RIGHT positions.

If the overheat detection system is working properly the following occurs:

- A 3-chime alert sounds.
- The voice warning "WINDSHIELD" is given.
- The red LW/S OVERHEAT or RW/S OVERHEAT light, and the amber DEICE light on the MAP illuminate.
- The master WARNING and CAUTION lights flash.

ICE DETECTION AND SPEED WARNING

Ice Detection System

The ice detection system consists of an ICE DETECTION panel and an ice detector on the aircraft nose (Figure 10-12).

The ice detector is vibrating probe. When the aircraft enters an icing environment, the ice collected on the probe causes the frequency of the probe to decrease, generating a signal that activates the ICE CONDITION amber light on the multiple alarm panel. The ice detects ice accretion as little as 0.5mm (0.20 inch) of ice.

The icing signal remains activated for 60 seconds. After 60 seconds, the ice detector heater is activated to deice the probe. When the probe's natural frequency is recovered, the heater is deenergized and a new detection cycle is initiated. When operating in icing conditions, the probe signal is continuously activated and the ICE CONDITION light remains illuminated as long as icing conditions persist.

The ice detection system is powered by DC bus 2. The system is automatically turned on when the aircraft is energized and remains activated during the entire flight.

NOTE

The system performs a self-test when the aircraft is energized.

The DEICE amber light illuminates on the multiple alarm panel simultaneously with ICE CONDITION.

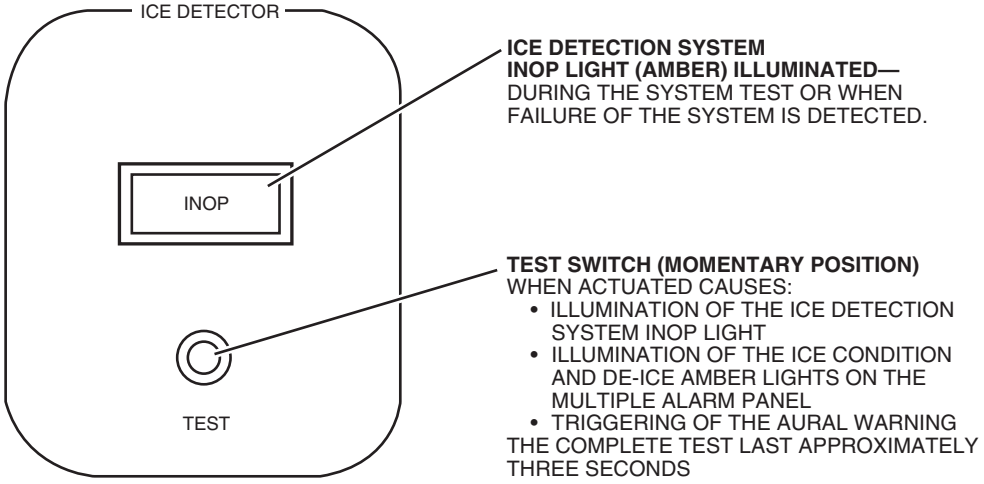


Figure 10-12. Ice Detection Panel

Low Speed Alarm System

The low speed alarm system is designed to warn the pilot that airspeed is less than 160 KIAS in icing conditions.

Components

The low speed alarm panel is a box on top of the glareshield panel. It consists of a buzzer, amber indicator labeled LOW SPEED, and a test switch to test both the lighted indicator and the buzzer (Figure 10-13).

Test

The low speed alarm should be tested as a first flight of the day item. When the test switch is activated, the buzzer should sound and be cancelled by the ALARM CANCEL on the caution/warning panel. The LOW SPEED light should illuminate and remain until the test switch is released.

Operation

The low speed alarm is activated when the conditions below occur simultaneously when aircraft is in flight:

- Airspeed drops below 160 knots
- Flaps are set to UP/CRUISE position (zero position)
- Either the ice detector has been activated or the leading-edge deice boots have been turned on

NOTE

The aircraft being in flight is based on the right landing gear retracted for aircraft compliant with SB 120-30-0033.

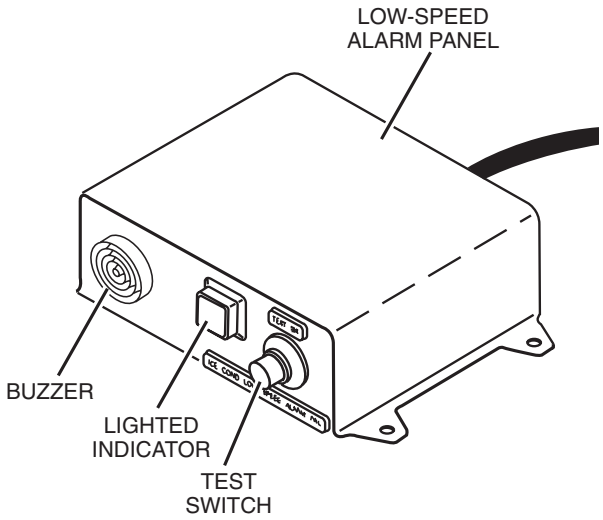


Figure 10-13. Low Speed Alarm Panel

Activation

If all conditions exist, the system causes the lighted LOW SPEED indicator to illuminate and the aural warning buzzer to sound.

Once the alarm condition is established, the buzzer can be cancelled by pressing ALARM CANCEL on the caution/warning panel. The LOW SPEED light remains illuminated until the condition is corrected. The buzzer remains disabled as long as the alarm conditions do not reoccur.

NOTE

Aircraft not compliant with SB 120-30-0033 cannot cancel the alarm buzzer. Both the light and buzzer will continue until the condition is corrected.

RAIN REMOVAL

WINDSHIELD WIPERS

The aircraft's rain removal system consists of two windshield wipers mounted at the base of the windshields. Each windshield wiper is driven by its own electric motor. The left windshield wiper motor is powered by DC Bus 1 and the right by DC Bus 2.

The windshield wipers are individually controlled by two rotary switches, labeled LEFT and RIGHT, in the WINDSHIELD WIPER section of the ICE/RAIN PROTECTION panel (Figure 10-14).

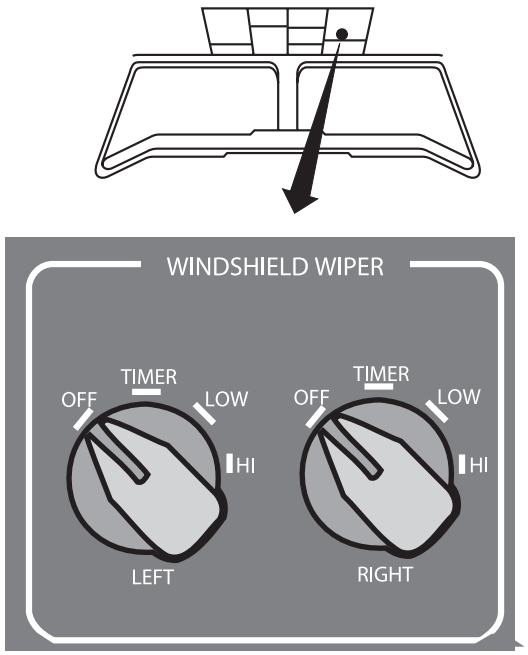


Figure 10-14. Windshield Wiper Controls

Each rotary switch has four positions:

OFF—The wiper blade is placed in the parked position, parallel with the center windshield vertical post.

TIMER—The wiper blade operates intermittently with a time interval of eight seconds between wiper cycles. (A wiper cycle is two strokes of a wiper blade beginning and ending at the parked position.)

LOW—The wiper blade operates at approximately 80 strokes per minute.

HIGH—The wiper blade operates at approximately 140 strokes per minute.

When both wiper switches are selected to the same speed, the windshield wipers operate synchronously. A failure in one wiper has no effect on the operation of the other.

CAUTION

Never operate the windshield wipers on a dry windshield or at a speed in excess of 160 KIAS

Never clean the windshield with anything but soap, water, and a clean soft cloth.

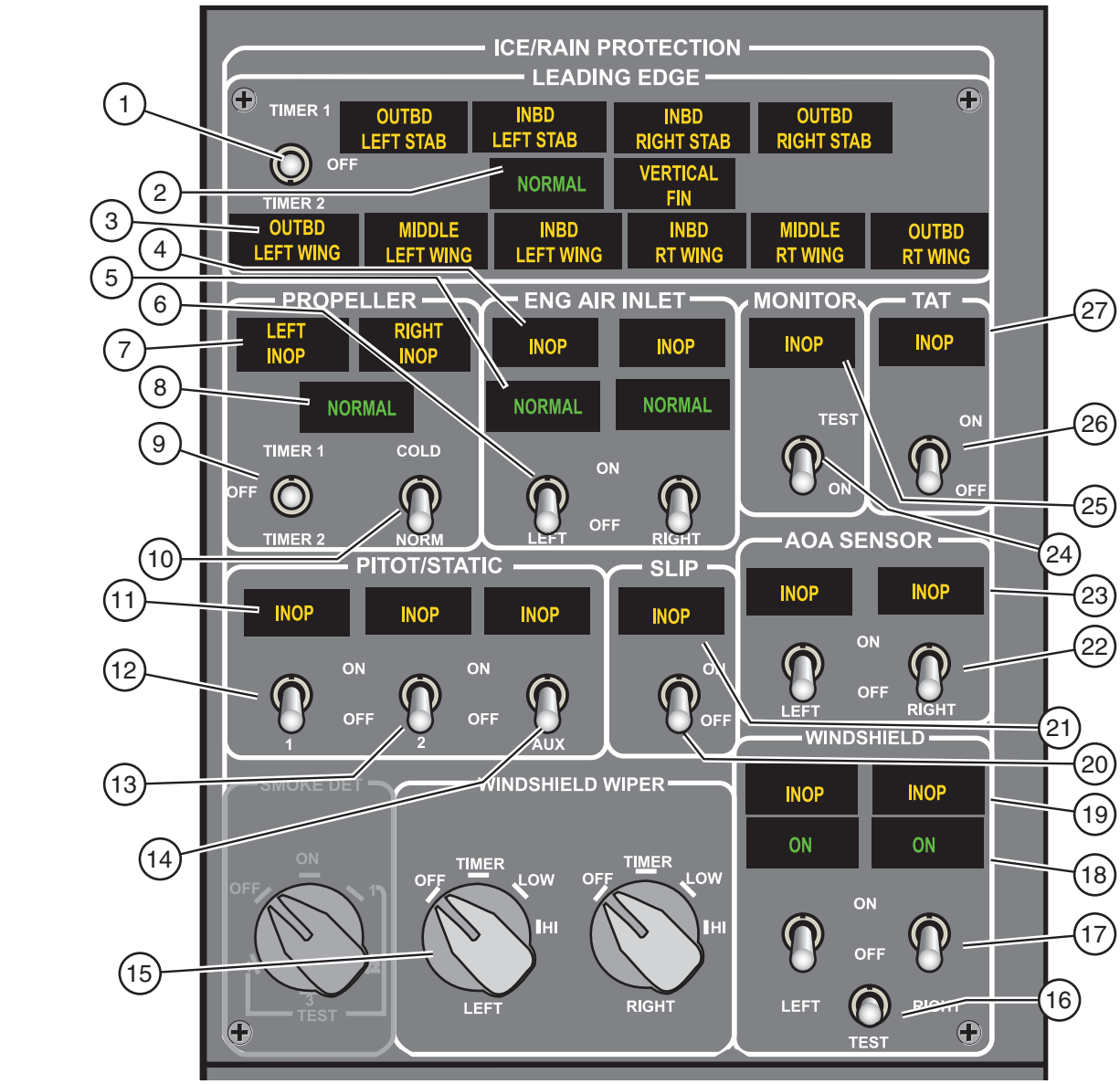


Figure 10-15. Ice and Rain Protection Systems Controls and Indicators

Leading edge system operates at one-minute cycle duration.

1. Leading edge deicers switch

   TIMER 1—The system is energized and is controlled by timer 1.

   OFF—The system is unpowered.

   TIMER 2—The system is energized and is controlled by timer 2.

2. Leading edge deicers NORMAL light (green)

   Illuminated—Leading edge deicing system is turned on (timer 1 or timer 2), and operating normally.

3. OUTBD LEFT WING deicer light (amber)

   Illuminated—Outboard left wing deicer malfunction (green NORMAL light extinguishes).

4. Left engine air inlet deicer INOP light (amber)

   Illuminated—Left engine air inlet deicer malfunction (green NORMAL light extinguishes).

5. Left engine air inlet deicer NORMAL light (green)

   Illuminated—Left engine air inlet deicing system is on and operating normally.

6. Left engine air inlet deicer switch

   ON—The left engine air inlet deicer system and timer are energized. Deicers operate on a three-minute cycle; inflated for six seconds, and deflated for remainder of cycle.

   OFF—The left engine air inlet deicer system is unpowered.

7. Propeller deicing system LEFT INOP light (amber)

   Illuminated—Fault detected in the left propeller deicing circuit (green NORMAL light extinguishes).

8. Propeller deicing system NORMAL light (green)

   Illuminated—Propeller deicing system is on and operating normally.

9. Propeller deicing system switch

   TIMER 1—The system is energized and is controlled by timer 1.

   OFF—The system is deenergized.

   TIMER 2—The system is energized and is controlled by timer 2.

10. Cycles selection switch

    COLD—Selects 20 seconds on 60 second off cycle (temperature less than –10° C).

    NORM—Selects 10 seconds on 60 second off cycle (Temperature above –10° C).

11. Captain's pitot-static heater INOP light (amber)

    Illuminated—Captain's pitot-static heater is not powered (i.e., switch is OFF, or switch is ON and heater has failed).

12. Captain's pitot-static heater switch

    ON—Energizes the captain's pitot-static heater.

    OFF—Heater deenergized.

13. Copilot's pitot-static heater/pneumatic outflow valve static heater switch

    ON—Energizes the copilot's pitot-static heater and pneumatic outflow valve static heater.

    OFF—Heaters deenergized.

14. AUX pitot-static heater/electropneumatic outflow valve static heater switch

    ON—Energizes the auxiliary pitot-static heater and electropneumatic outflow valve static heater.

    OFF—Heaters deenergized.

15. LEFT windshield wiper control switch

    OFF—Blade is parked parallel to the windshield vertical post.

    TIMER—Intermittent operation in single cycles (two strokes), with an eight seconds time interval between cycles.

    LOW— Wiper operates at approximately 80 strokes/minutes.

    HIGH— Wiper operates at approximately 140 strokes/minutes.

16. Windshield overheat test switch

    • Must be held in LEFT or RIGHT position

    LEFT—Checks the left windshield overheat circuitry, alarms, and indicators.

    RIGHT—Checks the right windshield overheat circuitry, alarms, and indicators.

17. RIGHT windshield heater switch

    ON—Right windshield heater powered. Controller maintains windshield temperature between 40° and 45° C.

    OFF—Right windshield heater unpowered.

18. Right windshield ON light (green)

    Illuminated—Right windshield is being heated

19. Right windshield INOP light (amber)

    Illuminated—Right windshield heater system failure (green ON light extinguishes).

20. Slip-skid sensor heater switch

    ON—Power applied to slip-skid sensor heater.

    OFF—Heater unpowered.

21. Slip-skid sensor heater INOP light (amber)

    Illuminated—Slip-skid sensor heater failed.

22. RIGHT AOA sensor heater switch

    ON—Power applied to right AOA sensor heater.

    OFF—Heater unpowered.

23. Right AOA sensor heater INOP light (amber)

    Illuminated—Right AOA sensor heater failed.

24. Deicing system monitor switch

    ON—Deicing system monitor is powered (spring-loaded position).

    TEST—Checks all deicing system lights and the monitor.

25. Deicing system monitor INOP light (amber)

    Illuminated—Deicing system monitor fault.

26. Total air temperature probe heater switch

    ON—Power applied to TAT probe heater.

    OFF—Heater unpowered.

27. Total air temperature probe heater INOP light (amber)

    Illuminated—TAT probe heater failed.

QUESTIONS

1. What prevents moisture from entering tubing or deicers?

A. A heater system
B. A drain system
C. A water separator system
D. There is no way to prevent moisture from entering the tubes.

2. Deicer boots are operated by pneumatic air flowing through:

A. Two ejector flow control valves
B. Four ejector flow control valves
C. Six ejector flow control valves
D. Eight ejector flow control valves

3. Leading edge deicer boots are controlled by:

A. One timer
B. Two timers
C. One timer and heat sensor
D. Two timers and ice sensor

4. The modes of leading edge deicing are:

A. Light and heavy
B. Light and frost
C. Frost and heavy
D. On or off

5. Deice boots are inflated for a maximum of:

A. 2 seconds
B. 3 seconds
C. 6 seconds
D. 10 seconds

6. When the leading edge system is selected, the system cycles every:

A. 30 seconds
B. 60 seconds
C. 90 seconds
D. 180 seconds

7. Propeller deice is accomplished by:

A. Pneumatic pressure
B. Alcohol spray
C. Electrical heat
D. None of the above

8. The propeller deice system is protected from damage during ground operation by:

A. Weight-on-wheels switch
B. Pilot's safety switch
C. Low oil pressure switch
D. N_P switch

9. When the "cold" mode of propeller deice is used, the propeller blades are heated for:

A. 10 seconds
B. 20 seconds
C. 30 seconds
D. 60 seconds

10. The recommended temperature for selecting "cold" mode operation is:

A. Below 0°C
B. Below –10°C
C. Above –10°C
D. Above –10°F

11. The windshield heat system heats the windshield to approximately:

A. 20–25°C
B. 30–35°C
C. 40–45°C
D. 50–55°C

11. Should any segment of the anti-ice/deice system fail, the pilot would be alerted by a:

A. Red light
B. Green light
C. Amber light
D. White light

13. Angle-of-attack and side-slip sensors are deiced by:

A. Pneumatic pressure
B. Ram air
C. Electric heat
D. Bleed-air heat

14. Static ports are electrically anti-iced.

A. True
B. False

15. Windshield wipers move at a rate of _____ on high mode.

A. 80 strokes/minute
B. 100 strokes/minute
C. 120 strokes/minute
D. 140 strokes/minute

CHAPTER 11
AIR CONDITIONING

CONTENTS

Page

INTRODUCTION 11-1

GENERAL 11-1

AIR-CONDITIONING SYSTEM 11-3

Packs 11-3

Distribution System 11-5

Control and Monitoring 11-7

GASPER SYSTEM 11-11

Components 11-11

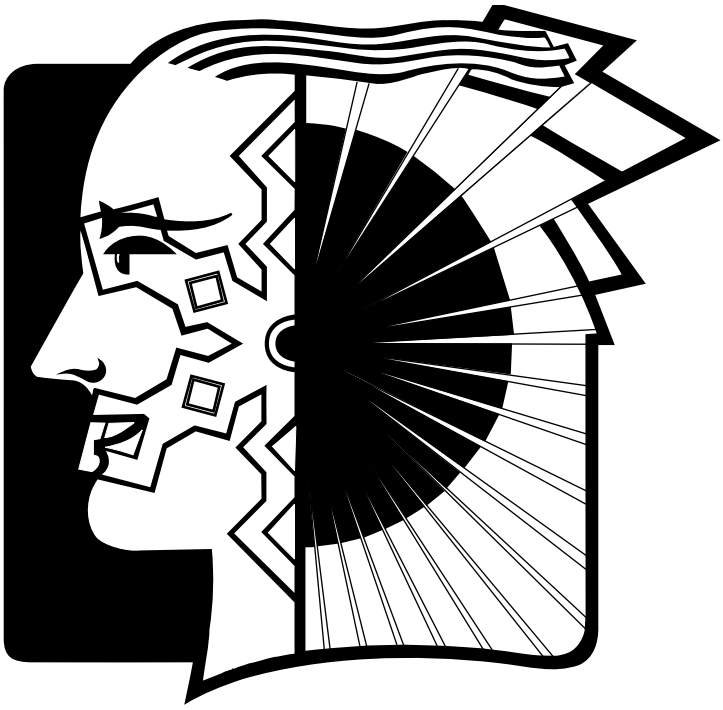
RAM-AIR SYSTEM 11-12

QUESTIONS 11-15

ILLUSTRATIONS

Figure	Title	Page
11-1	Air-Conditioning Pack Schematic	11-2
11-2	Air-Conditioning Distribution System	11-4
11-3	Air-Conditioning Distribution System— Full Cargo Configuration	11-5
11-4	Ground Air-Conditioning Connection	11-6
11-5	PAC Control Panel	11-7
11-6	Air-Conditioning System Controls and Indicators	11-13

CHAPTER 11
AIR CONDITIONING



INTRODUCTION

The EMB-120 air-conditioning system maintains a comfortable environment for crew and passengers. The engines or the auxiliary power unit (APU) may be used as the air source, either in flight or on the ground. Provisions have also been made for connecting an external conditioned air source on the ground. The aircraft is pressurized by controlling the exhausting of conditioned air from the cabin. Aircraft pressurization is covered in more detail in Chapter 12, "Pressurization."

GENERAL

The EMB-120 air-conditioning system provides cockpit and cabin cooling in the summer, and heating in the winter. It is also the source of aircraft pressurization.

Engine or APU bleed air is used to operate two pneumatically powered air conditioning packages, referred to as "packs," that supply the conditioned air.

The packs, installed in the wing-to-fuselage fairings at the aircraft wing roots, have independent controls, indicators and protection devices.

The air is delivered to the cockpit and cabin by a system of ducts and vents

Additional internal cabin air flow is provided by a gasper system that uses electric motor driven recirculation fans and separate gasper air outlets.

When the aircraft is unpressurized, a ram-air system may be used to supply outside air for ventilation.

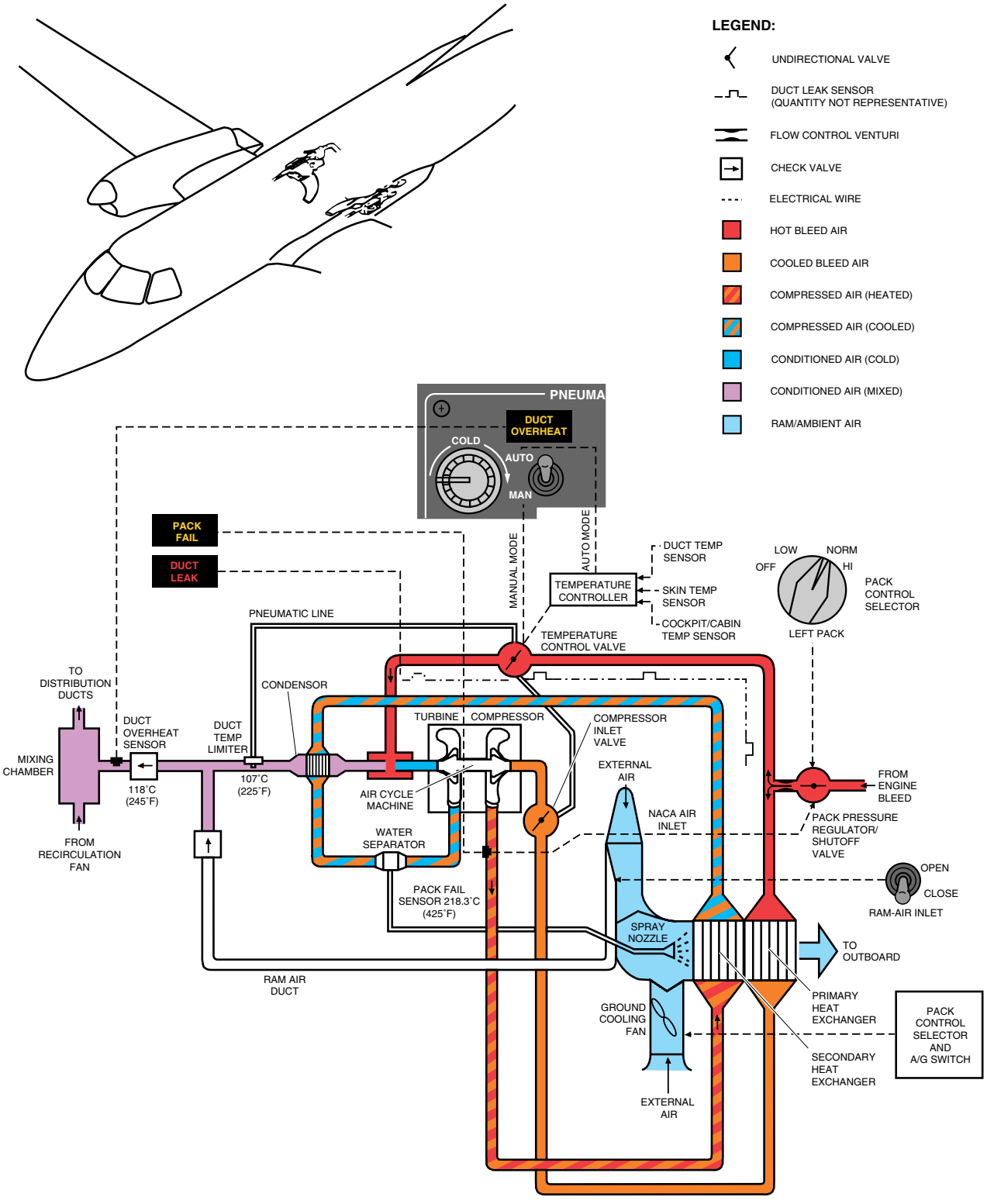


Figure 11-1. Air-Conditioning Pack Schematic

AIR-CONDITIONING SYSTEM

PACKS

The aircraft is equipped with two completely independent, pneumatically powered air conditioning packs (Figure 11-1). They provide cold, cool, warm or hot air to the cockpit and passenger cabin for comfort and pressurization.

The packs are operated by bleed air from one engine, both engines simultaneously, or from the APU.

Components

Each pack consists of the following major components:

- Primary heat exchanger
- Secondary heat exchanger
- Air cycle machine
- Condenser
- Water separator

Operation

The following discussion of pack operation is illustrated in Figure 11-1.

- Hot bleed air enters the pack through the pressure regulator/shutoff valve. This electropneumatically actuated valve regulates the pressure and flow of the air. (Loss of electric or pneumatic power closes the valve, regardless of the pack control selector position.)
- From the pressure regulator/shutoff valve the bleed air splits. One line goes through the conditioning cycle while the other bypasses the cycle and is routed, through the temperature control valve, to the air cycle machine output.
- The air going through the conditioning cycle first enters the primary heat exchanger where it is cooled by outside air. This outside air is from either a NACA inlet during flight, or induced by a cooling fan on the ground.
- After exiting the primary heat exchanger, the cooled bleed air passes through the compressor inlet valve and enters the air-cycle machine compressor, where its pressure is increased.
- Leaving the compressor, the air passes through the secondary heat exchanger, where the heat caused by compression is removed. (After the outside air has passed through both heat exchangers, it escapes overboard through grills in the sides of the fuselage aft of the wings.)
- The twice cooled high-pressure bleed air then passes through a condenser which is supercooled by the air exiting the cooling turbine. Any moisture remaining in the air at this point condenses into liquid and is collected by a water separator.

  The water from the water separator is routed through a water drain pipe and sprayed onto the secondary heat exchanger unit to improve heat exchanger efficiency.
- The air then enters the air-cycle machine turbine where, through expansion, it is cooled to temperatures near freezing or below. The increased velocity of the airflow, resulting from the expansion, is used to drive the turbine. The turbine, in turn, drives the compressor through an interconnecting shaft.
- The temperature of the air leaving the pack is monitored by the temperature controller. The temperature control valve, pneumatically linked to the compressor inlet valve, uses hot bypass air to control the output temperature. (When the temperature control valve is open, the compressor inlet valve is closed, and vice versa.).
- After the bypass air is added, the conditioned air goes through and cools the condenser. It then passes the temperature sensor that closes the temperature control valve at 225°F, and flows into the air-distribution system.

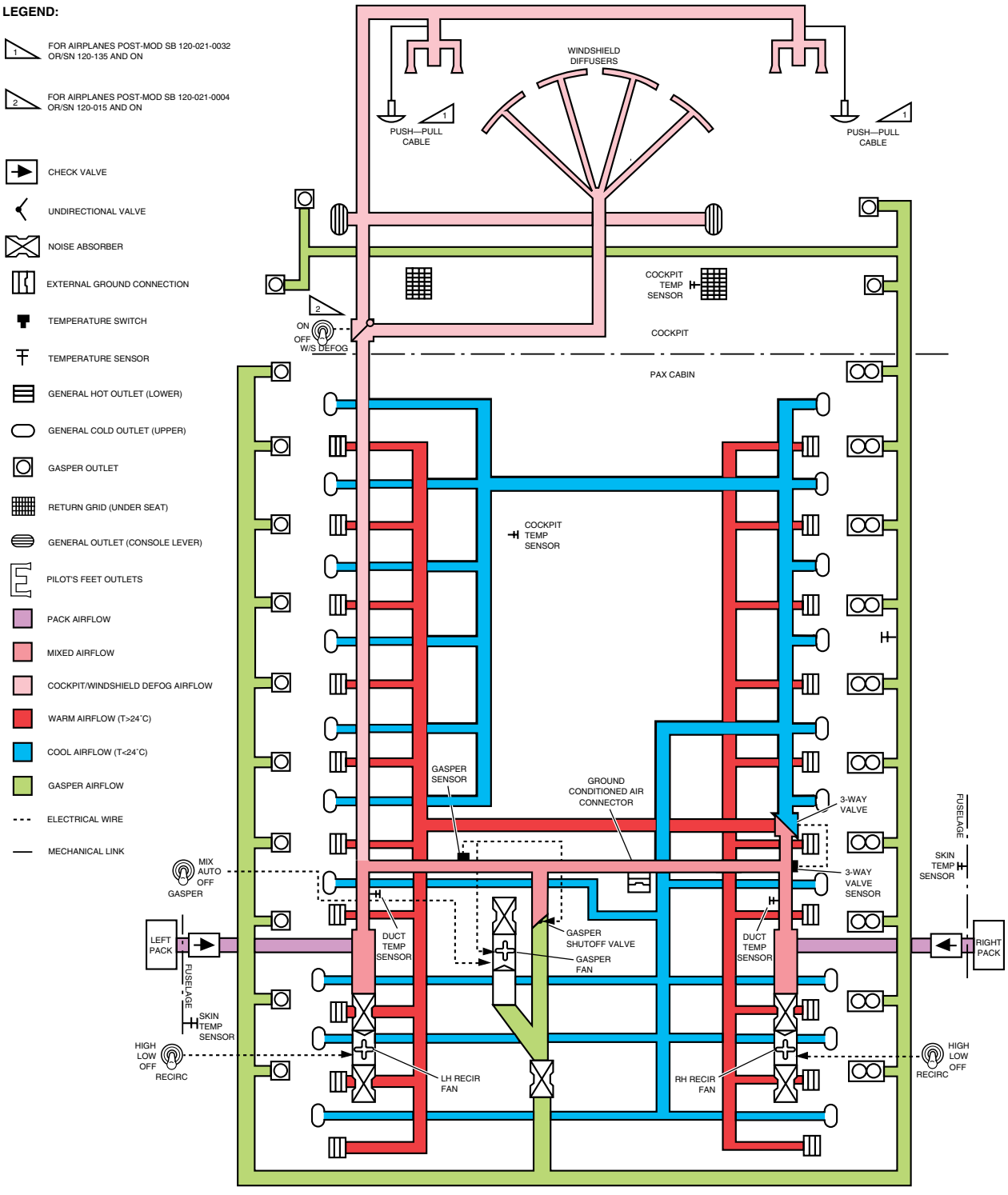


Figure 11-2. Air-Conditioning Distribution System

DISTRIBUTION SYSTEM

General

The air-conditioning distribution system, (Figures 11-2 and 11-3), uses a series of ducts and vents to provide the cockpit and passenger cabin with a temperature controlled environment. The ducts are shaped such that 60% of the airflow goes to the passenger cabin, and 40% to the cockpit, regardless of the arrangement of the packs. The system uses low-speed flow to attain maximum comfort at low noise levels.

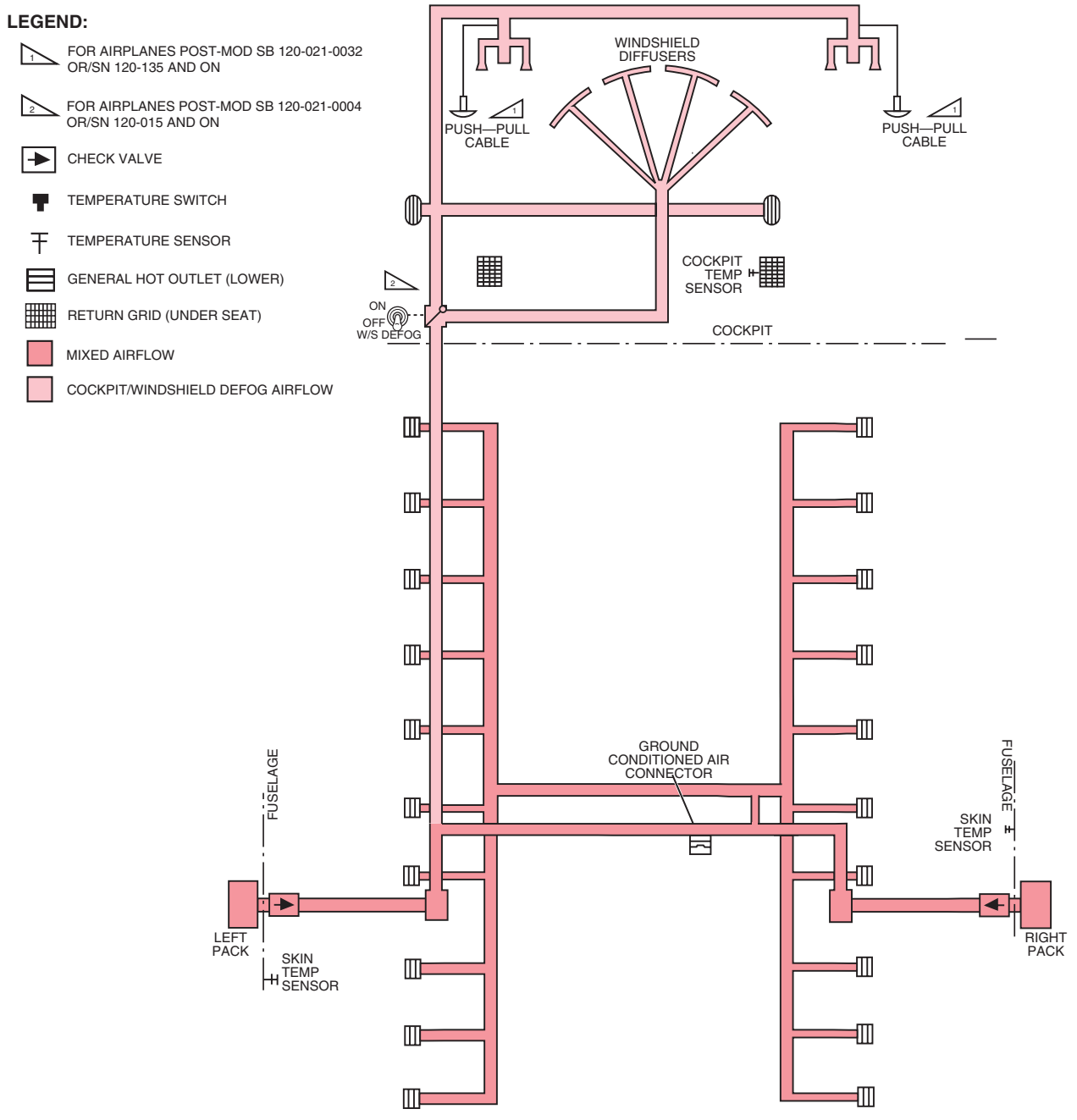


Figure 11-3. Air-Conditioning Distribution System—Full Cargo Configuration

Components

The distribution system includes the following components:

- Mixing chambers
- Recirculation fans
- Distribution ducts
- Ground Air-Conditioning Connection

Mixing Chambers

The mixing chambers combine air from the packs with recirculated air. The two mixing chambers are connected by the interconnection duct.

Recirculation Fans

A recirculation fan is installed in the ducting prior to each mixing chamber. The fans mix recirculated cabin air with conditioned air from the respective pack.

The two speed fans are controlled by a three position (OFF/LOW/HIGH) RECIRC switch on the overhead PAC panel. The fans may be operated with or without packs.

CAUTION

The recirculation fans, if left installed on cargo aircraft, must be selected off before flight because they may interfere with the operation of the smoke detectors.

Distribution Ducts

The distribution system includes two types of ducts/outlets:

- General
- Gasper

General—The cabin distribution system consists of a series of head and floor level outlets. The head level outlets are used if the air in the duct is cold; the floor level outlets are used if the air is hot.

The cockpit has four general outlets; two at floor level, and two at head level. There is also a foot air outlet for each crew member, and four windshield defogging diffusers.

Gasper—Individually adjustable eyeball type outlets, similar to those found on large airliners, are provided for each passenger and each pilot. The separate gasper system is discussed in more detail in the following section.

Ground Air-Conditioning Connection

A ground air-conditioning connection, (Figure 11-4), is in the lower right fuselage. It allows conditioned air to be supplied, through a check valve, directly into the interconnection duct.

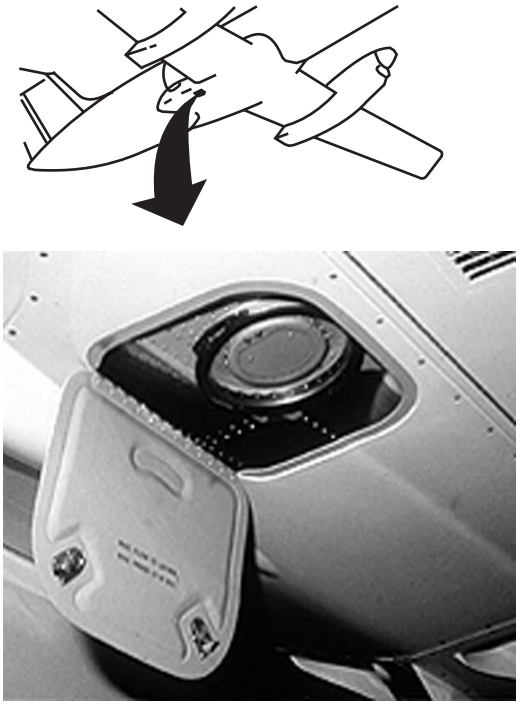


Figure 11-4. Ground Air-Conditioning System

Operation

Air delivered to the cabin is a combination of conditioned air from the packs and, when the recirculation fans are on, air drawn from the cabin under floor system.

The distribution system is designed using the principle that cold air descends and warm air rises. This sets up a circulation pattern and maintains a more even temperature.

A temperature sensor and a 3-way hot/cold selector valve are located just downstream of the right mixing chamber. The sensor determines duct temperature and commands the valve to direct cold air, (below 24°C/75°F), to the head level outlets; or hot air, (above 24°C/75°F), to the floor level outlets.

The air from the cabin and cockpit is exhausted through return air slots at floor level. The air travels aft under the floor where some is picked up by the recirculation fans. The remainder continues aft to the outflow valves and is exhausted overboard.

Instrument panel cooling is through natural cockpit exhaust in the floor. This also serves as a precaution against smoke.

CONTROL AND MONITORING

Controls

The pneumatic/air-conditioning (PAC) control panel, located in the lower right corner of the cockpit overhead panel (Figure 11-5), contains all controls and indicators for the pneumatic/air-conditioning systems.

In addition to the pneumatic system controls, discussed in the previous chapter, the PAC panel contains the following air-conditioning controls:

- Cockpit temperature mode selector switch
- Cabin temperature mode selector switch
- Temperature controls
- Windshield defogger switch
- Recirculation fan switches
- Pack control selector switches
- Ram-air inlet switch

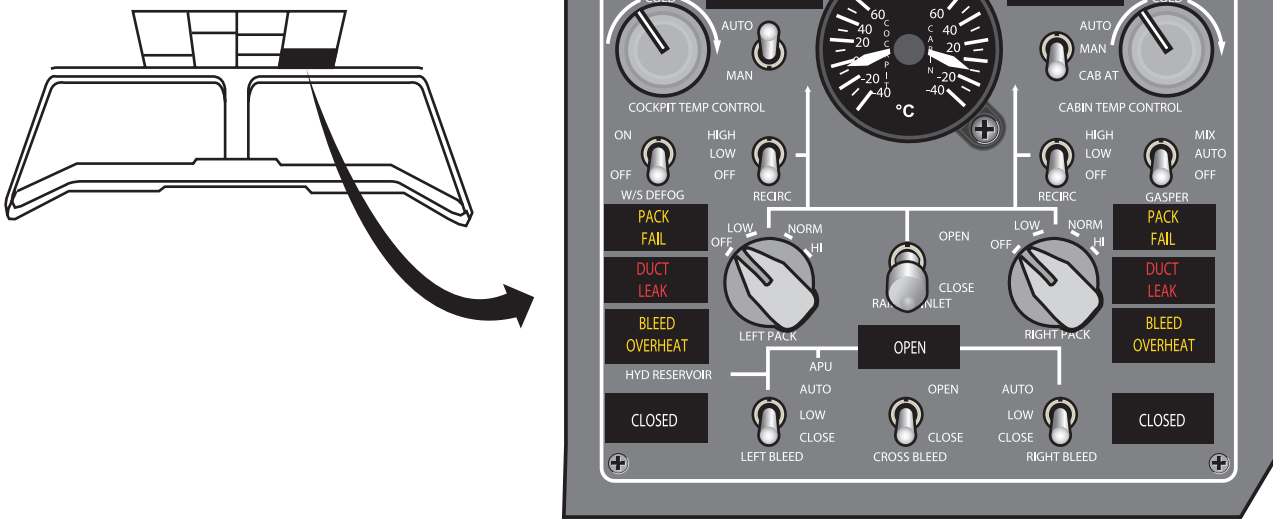


Figure 11-5. PAC Control Panel

Temperature Controls

There are two sets of temperature controls on the PAC panel, labeled COCKPIT TEMP CONTROL and CABIN TEMP CONTROL. Each set has a mode selector switch and a temperature adjust knob. The controls are used together to control cockpit or cabin temperature.

Cockpit Temperature Mode Selector Switch—The selector switch is a two position switch:

AUTO—Normal operating mode. Controls the left pack discharge temperature through the automatic temperature controller.

MAN—Used when automatic temperature controller fails. The temp control knob is directly connected to the pack temp control valve.

Cabin Temperature Mode Selector Switch—The selector switch is a three position switch:

AUTO—Normal operating mode. Controls the right pack discharge temperature through the automatic temperature controller.

MAN—Used when automatic temperature controller fails. The temp control knob is directly connected to the pack temp control valve.

CABIN AT—Cabin temperature control shifted to CABIN TEMP CONTROL knob on the attendant's panel.

Temperature Adjust Knobs—There are three temperature adjust knobs. Two are located on the PAC panel, one beside each mode selector switch, and are used to adjust cockpit and cabin temperature in the AUTO and MAN mode.

The third knob, the CABIN TEMP CONTROL knob on the attendant's panel, controls the right pack temperature controller (in an auto mode), when the CABIN TEMP CONTROL mode selector switch on the PAC panel is selected to CABIN AT.

Temperature Control Operation—The temperature controls use two valves to regulate the hot and cold airflow.

These valves and their functions are:

- Temperature control valve—A normally closed electropneumatic shutoff valve, located in the hot air bypass line. Receives information from either the electronic temperature controller (AUTO mode), or directly from the temperature adjusting knob (MAN mode).

- Compressor inlet valve—A normally open pneumatic valve, located upstream of the compressor inlet and slaved to the temperature control valve.

The valves are pneumatically connected and therefore operate together. Only the temperature control valve is electrically controlled. When the temperature control valve is closed, the compressor inlet valve is open; when the temperature control valve is open, the compressor inlet valve is closed (never fully).

AUTO MODE: With the temperature control mode selector in AUTO, the temperature in the cockpit/cabin is controlled by the relevant electronic temperature controller. The controller continuously compares temperature readings from skin, compartment, and supply duct sensors to the reference temperature set by the crew with the temperature knob.

The temperature range in the AUTO mode is between 60°F/15.5°C (full cold) and 90°F/32.2°C (full hot). The temperature may be varied within this range with the temperature adjust knob.

MANUAL MODE: With the temperature control mode selector in MAN, the temperature in the cockpit/cabin is controlled by the relevant temperature adjust knob. The temperature gage must be monitored to attain the desired temperature.

Use MAN only with an AUTO mode failure (temperature not maintained within the AUTO mode temperature limits).

In the MAN mode the temperature should be continuously monitored and adjusted. Selecting full hot or full cold could lead to a duct overheat or cause ice formation at the turbine outlet.

Windshield Defog Switch

The W/S DEFOG switch is a two position switch:

ON—Energizes a valve in the cockpit air conditioning line that diverts some (not all) air from the cockpit outlets to the windshield diffusers.

OFF—Diverter valve deenergized. Air from cockpit air conditioning line is routed to the cockpit outlets.

Recirculation Fan Switches

The right and left RECIRC switches are three position switches:

HIGH—Recirculation fan operates at maximum output.

LOW—Recirculation fan operates at minimum output.

OFF—Recirculation fan off.

Pack Control Selector Switches

The pack control selector switches, labeled LEFT PACK and RIGHT PACK, are four position switches with the following functions:

OFF—Pack pressure regulator/ shutoff valve fully closed.

NOTE

During ground operations, ground cooling fans supply airflow to the primary and secondary heat exchangers whenever the packs are on (LOW, NORM, HI). The fans are automatically switched off at weight off wheels.

LOW—Used for takeoff and approach. Minimizes bleed extraction to conserve engine power. Pack pressure regulator/shutoff valve regulates pack pressure to 15 psi.

NORM—Normal operating position. Pack pressure regulator/shutoff valve regulates the pressure between 15 and 35 psi, depending upon temperature demand.

HI—The pack pressure regulator/ shutoff valve regulates the pack operating pressure to 24 psi. If the engine bleed switch is in the AUTO position the engine P_3 bleed port opens, supplying high temperature bleed air to the pack.

For use in cold conditions when cabin temperature is below 19°C

NOTE

Use of the HI position in warm temperatures may cause bleed overheats or pack fails due to the high P_3 temperatures.



RAM AIR INLET Switch

The ram air inlet switch is a two position switch that operates two flapper valves, one in each NACA inlet:

OPEN—Outside air from NACA inlets directed into the distribution system (unpressurized flight only).

CLOSE—Air from NACA inlets directed to the pack heat exchangers. Must be closed to operate the packs.

Indications and Warnings

The air-conditioning system incorporates the following indicators and warnings:

- Temperature indicator
- DUCT LEAK light
- Duct overheat protection
- DUCT OVERHEAT light
- PACK FAIL light
- BLEED OVERHEAT light
- Bleed CLOSED light
- Crossbleed OPEN light
- System warnings

Temperature Indicator

The temperature indicator on the PAC panel provides an ambient temperature indication in °C for both the cockpit and passenger cabin.

DUCT LEAK Light

As previously discussed in Chapter 9 "Pneumatics," when sensors in the duct detect a temperature over 90° C/195° F the following occurs:

- The red DUCT LEAK light on the PAC panel illuminates
- A three-chime aural alert sounds
- The voice warning "Duct Leak" is given
- The red DUCT LEAK light on the MAP illuminates
- The master WARNING lights flash.

Duct Overheat Protection

The duct temperature limiter, a thermopneumatic device, closes the temperature control valve when the temperature in the conditioned air duct reaches 107°C/225°F. When the temperature control valve closes, the slaved compressor inlet valve opens). No cockpit indication is provided.

DUCT OVERHEAT Light

When the duct overheat thermostat, located just before the mixing chamber, senses duct temperature over 118°C/245°F, it illuminates the amber DUCT OVERHEAT light on the PAC panel.

PACK FAIL Light

The PACK FAIL light is activated by a 218.3°C/425°F set point thermostat installed at the compressor outlet.

Simultaneously with the light, the thermostat commands the pack pressure regulator/shutoff valve to close and turns off the ground cooling fan.

A PACK FAIL inflight may be caused by a bleed overheat condition, or failure of the primary heat exchanger.

A PACK FAIL on the ground indicates a probable ground cooling fan failure.

After a PACK FAIL, the system must be reset by turning the pack control selector OFF, then back on.

BLEED OVERHEAT Light

As previously discussed in Chapter 9, "Pneumatics," when the overheat switch (downstream of the engine bleed air shutoff valve) senses a temperature in excess of 315°C/600°F the amber BLEED OVERHEAT light on the PAC panel illuminates. Also, the engine bleed and HP bleed shutoff valves automatically close.

Bleed CLOSED Light

A white bleed CLOSED light on the PAC panel indicates the relevant engine bleed shutoff valve is closed.

Crossbleed OPEN Light

The white crossbleed OPEN light on the PAC panel illuminates when the crossbleed valve is open.

System Warnings

The illumination of the amber DUCT OVERHEAT, PACK FAIL or BLEED OVERHEAT light on the PAC panel is accompanied by the following additional warnings and indications:

- A single chime aural alert sounds
- The amber AIR COND light on the MAP illuminates
- The master CAUTION lights flash.

GASPER SYSTEM

The gasper system is provided to enhance comfort in the cockpit and cabin by providing an induced local air movement from individual outlets (Figure 11-2).

The individually adjustable eyeball type outlets, called gaspers, are similar to those found on large airliners.

One gasper is provided for each passenger, and two for each pilot.

The gasper air system supplies conditioned air in the cooling mode and recirculated air in the heating mode. There is also an option to combine conditioned air and recirculated air in extreme ambient conditions to augment gasper flow (double flow).

For ease of understanding of the gasper air system, the airflow from the air conditioning interconnection duct is referred to as "duct air," and the airflow from the gasper fan is referred to as "cabin air."

COMPONENTS

The components of the gasper air system are:

- Duct temperature sensor
- Gasper duct shutoff valve
- Gasper fan

Duct Temperature Sensor

The duct temperature sensor operates the gasper duct shutoff valve and the gasper fan. The duct temperature sensor is located in the air-conditioning interconnection duct between the mixing chambers.

Gasper Duct Shutoff Valve

A gasper duct shut-off valve is installed in the gasper duct between the interconnection duct and the gasper fan outlet.

When open, the valve permits the higher pressure duct air to flow into the gasper air system.

When closed, the valve prevents duct air from entering the gasper air system. The gasper fan, when operating, supplies cabin air to the gasper air outlets.

Gasper Fan

The gasper fan is a two speed fan installed in a duct under the cabin floor. It discharges through a check valve into the gasper air system downstream of the shutoff valve.

The fan is controlled by the three position (OFF/AUTO/MIX) GASPER switch located on the PAC panel.

The switch functions are as follows:

OFF—The shutoff valve opens and the gasper fan is off. Gasper outlet air is from the duct only.

AUTO:

- Duct temperature >75°F —The shutoff valve closes and the gasper fan operates on high. Gasper outlet air is from the cabin only.
- Duct temperature <75°F—The shutoff valve opens and the gasper fan is off. Gasper outlet air is from the duct only.

MIX—The shutoff valve opens and the fan operates at low speed. Gasper outlet air is from the duct and cabin regardless of duct temperature (double flow). Should not be used during flight as it may deliver warm air to the passengers' heads.

NOTE

In the event of smoke in the cabin, the gasper fan and recirculation fans should be turned OFF.

RAM-AIR SYSTEM

The ram-air system provides cabin air flow during unpressurized operation. It may be used during takeoff or when the air-conditioning packs are inoperative.

The system uses the same NACA air inlets that supply cooling air for the pack heat exchangers. Flapper valves re-direct the airflow from the heat exchangers to the distribution ducts just prior to the mixing chambers (see Figure 11-1).

To ventilate the cabin with ram air:

- Position the RAM AIR INLET switch on the PAC panel to OPEN
- Set the mode selector switch on the pressurization control panel to DUMP. This allows air to flow through the cabin and out the outflow valves.

NOTE

The RAM AIR INLET switch controls both NACA air inlet flapper valves.

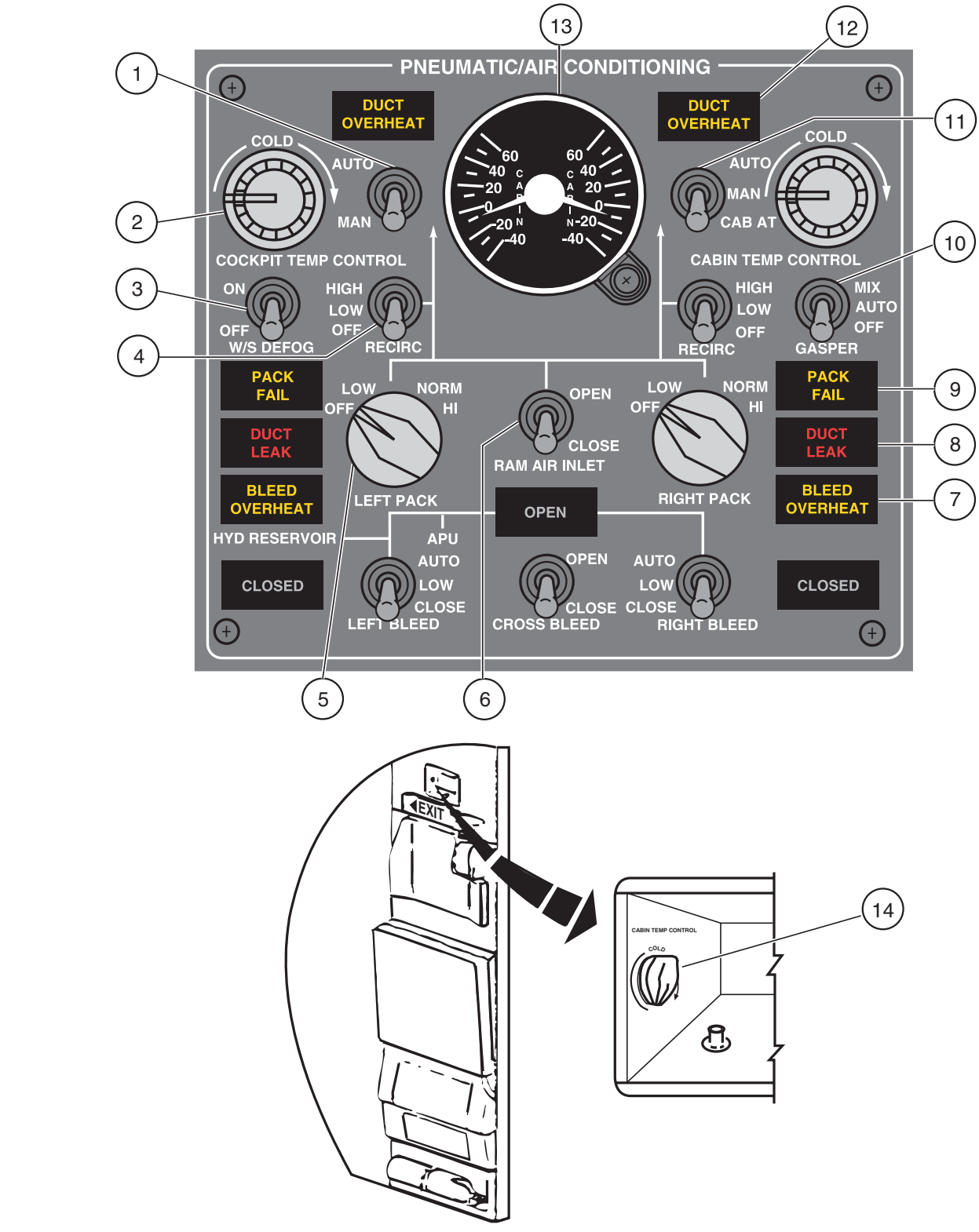


Figure 11-6. Air-Conditioning System Controls and Indicators

1. COCKPIT TEMP CONTROL mode selector switch
   AUTO—Normal operating mode. Controls the left pack discharge temperature through the automatic temperature controller.
   MAN—Used when automatic temperature controller fails. The temp control know is directly connected to the pack temp control valve.
2. Cockpit temperature adjusting knob
   Allows the pilot to set a desired temperature in the cockpit by actuating the left pack temperature control in either manual or automatic modes.
3. W/S DEFOG switch
   ON—Energizes a valve in the cockpit air conditioning line that diverts some (not all) air from the cockpit general outlets to the windshield diffusers.
   OFF—Valve deenergized. Airflow directed to cockpit general outlets only.
4. Left RECIRCulation fan switch
   OFF—Recirculation fan off.
   LOW—Recirculation fan operates at low rpm.
   HIGH—Recirculation fan operates at high rpm.
5. LEFT PACK control selector
   OFF—The pack pressure regulator/shutoff valve is fully closed.
   LOW—Used for takeoff and approach. The pack pressure regulator/shutoff valve delivers low bleed air pressure (15 psi) to the pack.
   NORM—Normal operating position. The pack pressure regulator/shutoff valve regulates pressure between 15 and 35 psi.
   HI—Used when cabin temperature is below 19°C. The pack pressure regulator/shutoff valve regulates the pressure to 24 psi.
6. RAM AIR INLET switch
   OPEN—Opens valves in both NACA inlets allowing outside air to be directed to distribution ducts.
   CLOSE—Airflow is directed to the pack heat exchangers.
7. Right BLEED OVERHEAT light (AMBER)
   Illuminated—Overheat condition in the right bleed duct with temperature exceeding 315° C (600°F).
   Illuminates in conjunction with amber AIR COND light on MAP.
   The engine bleed shutoff and HP bleed shutoff valves close simultaneously.
8. Right **DUCT LEAK** light (RED)
   Illuminated—Bleed-air leak in the right bleed ducts with temperature exceeding 90°C (195°F).
   Illuminates in conjunction with red DUCT LEAK light on MAP.
9. Right PACK FAIL light (AMBER)
   Illuminated—Indicates high temperature (218.3°C/425°F) at the compressor outlet. The pack pressure regulator/shutoff valve closes simultaneously.
10. GASPER switch
    OFF—Gasper fan is off and gasper shutoff valve is open.
    AUTO:
    Duct temp > 75°F—Shutoff valve closes and gasper fan operates on high.
    Duct temp < 75°F—Shutoff valve opens and gasper fan is off.
    MIX—Gasper fan operates on low and gasper shutoff valve is open.
11. CABIN TEMP CONTROL mode selector switch
    AUTO—Normal operating mode. Controls the left pack discharge temperature through the automatic temperature controller.
    MAN—Used when automatic temperature controller fails. The temp control knob is directly connected to the pack temp control valve.
    CABIN AT—Cabin temperature control shifted to CABIN TEMP CONTROL knob on attendant's panel.
12. Right DUCT OVERHEAT light (AMBER)
    Illuminated—Temperature in duct prior to mixing chamber exceeds 118°C (245°F).
13. Temperature Indicator
    Indicates cockpit and cabin temperatures in °C.
14. Attendant's CABIN TEMP CONTROL knob
    Allows the attendant to actuate the right pack temperature controller, controlling pax cabin temperature. The knob is effective in the automatic mode only. Enabled when the CABIN TEMP CONTROL mode selector switch is set to CAB AT.

QUESTIONS

1. The EMB 120 is air conditioned by:

A. A freon air-conditioning system
B. Cold air from an ambient induction system
C. Environmental control packs
D. Both A and B

2. On the ground an air conditioner malfunction (PACK FAIL) would probably be caused by:

A. Failure of the freon pump
B. Failure of the free air mixer unit
C. Failure of the ground cooling fan
D. Turbine temp control

3. Should there be a PACK FAIL malfunction, the system may be restarted by:

A. Turning the pack control selector OFF then on
B. Turning the pack control selector to RESET
C. Letting the duct cool down then go to RESET
D. The system will reset automatically when cool.

4. The conditioned air temperature is controlled by:

A. The speed of the compressor
B. The mixture of cold air from the turbine and bleed air
C. The speed of the hot air pump
D. All of the above

5. Temperature can be read for both the cockpit and the cabin.

A. True
B. False

6. With the air conditioning system in the AUTO mode, the temperature range can be controlled within:

A. 32°–60°F
B. 60°–90°F
C. 32°–100°F
D. The temperature range depends on the altitude of the airplane.

7. The manual mode of operation can be used at pilot's discretion.

A. True
B. False

8. Should the air-conditioning system malfunction and cause a DUCT OVERHEAT, the pilot will be warned by a:

A. Red light
B. Green light
C. Amber light
D. White light

9. The DUCT OVERHEAT light comes on when the internal temp reaches:

A. 150°F
B. 118°F
C. 200°F
D. 245°F

10. Air distribution ratio between cockpit and cabin is:

A. 50–50
B. 25–75
C. 40–60
D. 20–80

11. To increase air flow to the cabin, the system has:

A. One large squirrel cage recirculation fan
B. Two recirculation fans
C. A variable recirculation bypass fan
D. A duct distribution fan

12. Gasper fans should not be used in the MIX position during flight because:

A. A weight on wheels switch disables it.
B. It may blow warm air on the passengers heads.
C. It is sensitive to high altitudes.
D. It is not sensitive to high altitudes.

CHAPTER 12
PRESSURIZATION

CONTENTS

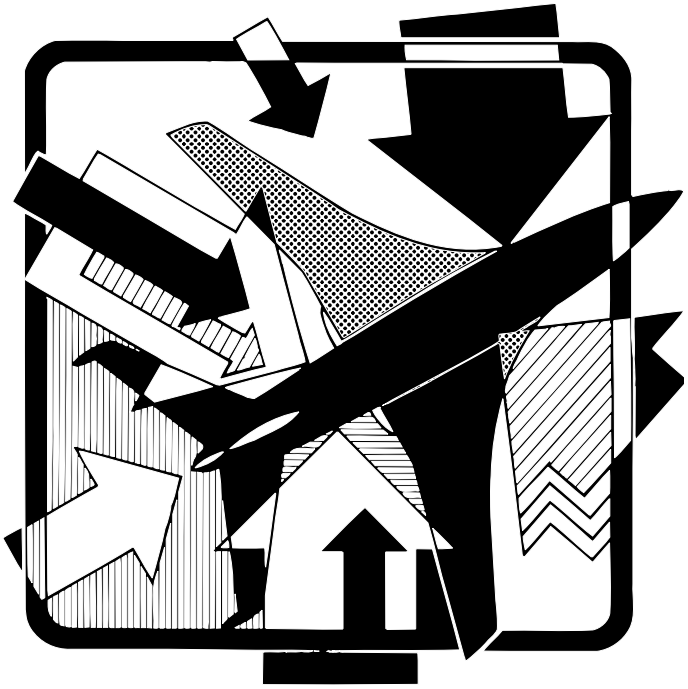
Page

INTRODUCTION 12-1
GENERAL 12-1
COMPONENTS 12-3
Electropneumatic Outflow Valve 12-3
Pneumatic Outflow Valve 12-4
Ejector Jet Pump 12-4
Blowout Panels 12-4
CONTROL AND MONITORING 12-5
Mode Selector Switch 12-5
Electronic Controller 12-6
Manual Controller 12-6
Triple Indicator 12-7
OPERATION 12-8
Automatic Mode 12-8
Manual Mode 12-9
Depressurization 12-10
SAFETY DEVICES 12-10
QUESTIONS 12-12

ILLUSTRATIONS

Figure	Title	Page
12-1	Pressurization System Schematic	12-2
12-2	Pressurized Areas	12-3
12-3	Pressurization Control Panel	12-5
12-4	Electronic Controller	12-6
12-5	Manual Controller	12-6
12-6	Triple Indicator	12-7
12-7	Automatic Pressurization during Takeoff	12-8
12-8	Automatic Depressurization after Touchdown	12-9
12-9	Pressurization System Controls and Indicators	12-11

CHAPTER 12
PRESSURIZATION



INTRODUCTION

The EMB-120 pressurization system provides crew and passenger comfort and survival throughout the entire aircraft operating envelope. Pressurization is achieved and controlled using outflow valves to restrict the air-conditioning system airflow exiting the cabin.

GENERAL

The pressurization system is designed to maintain the cabin altitude at a comfortable and safe level for the passengers and crew. The system is capable of maintaining a sea-level cabin to 16,800 feet, and an 8,000-foot cabin at 32,000 feet. Normal cabin differential pressure is 7 psi.

Cabin outflow is controlled by an electropneumatic and a pneumatic outflow valve. The pressurization system may be operated in two modes: automatic or manual. In the automatic mode, the electropneumatic outflow valve is used, with the pneumatic outflow valve slaved to it. In the manual mode, the electropneumatic outflow valve is closed, and cabin outlet flow is controlled only by the pneumatic outflow valve.

Both outflow valves are located in the center of the rear pressure bulkhead inside the pressurized cabin. They are each capable of maintaining the proper cabin pressure independently.

Air is directed into the cockpit and the passenger cabin through the air conditioning distribution system. Airflow from the cockpit exits through an air sump exhaust in the cockpit floor, and airflow from the passenger cabin exits through foot-level grills.

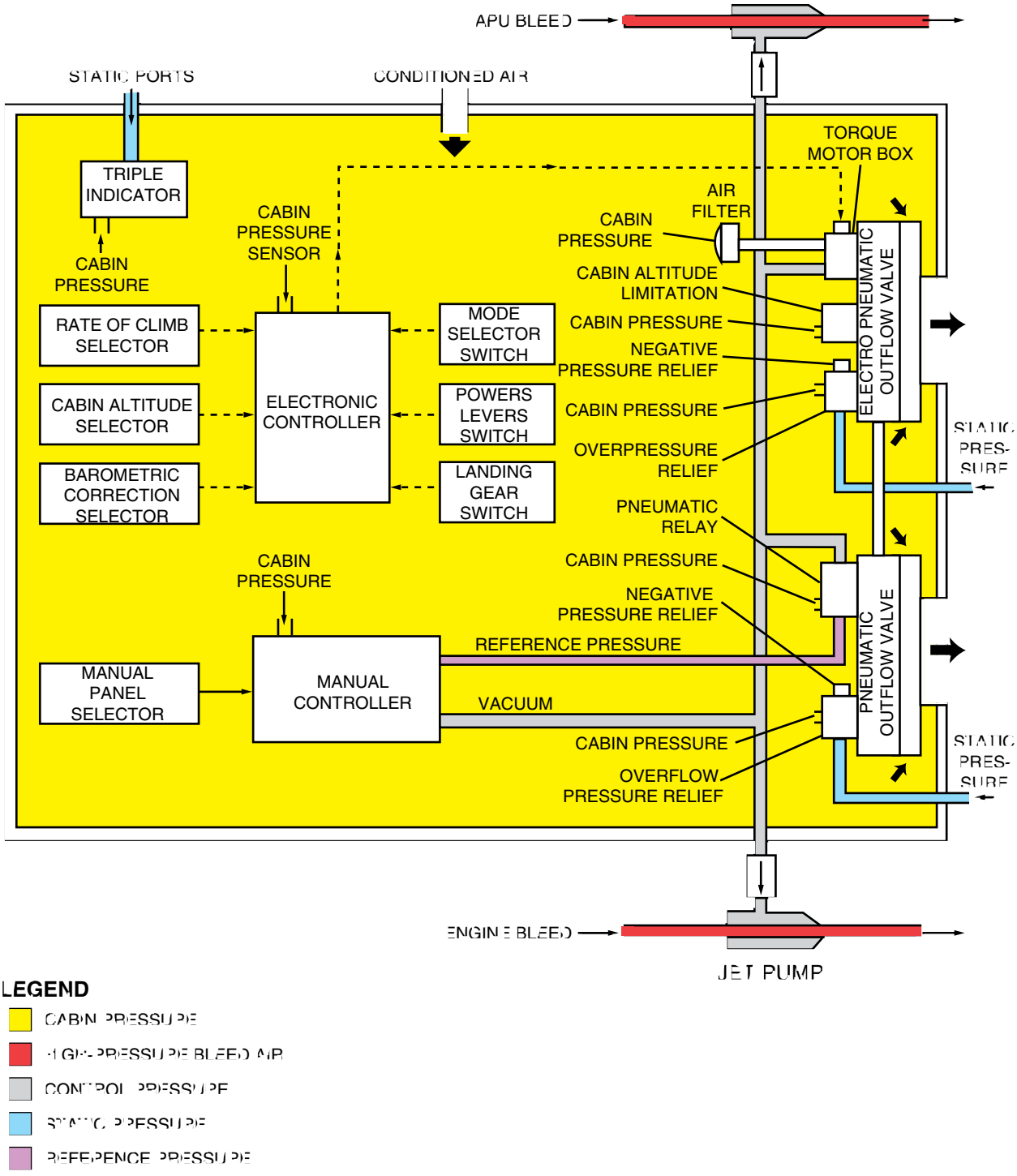


Figure 12-1. Pressurization System Schematic

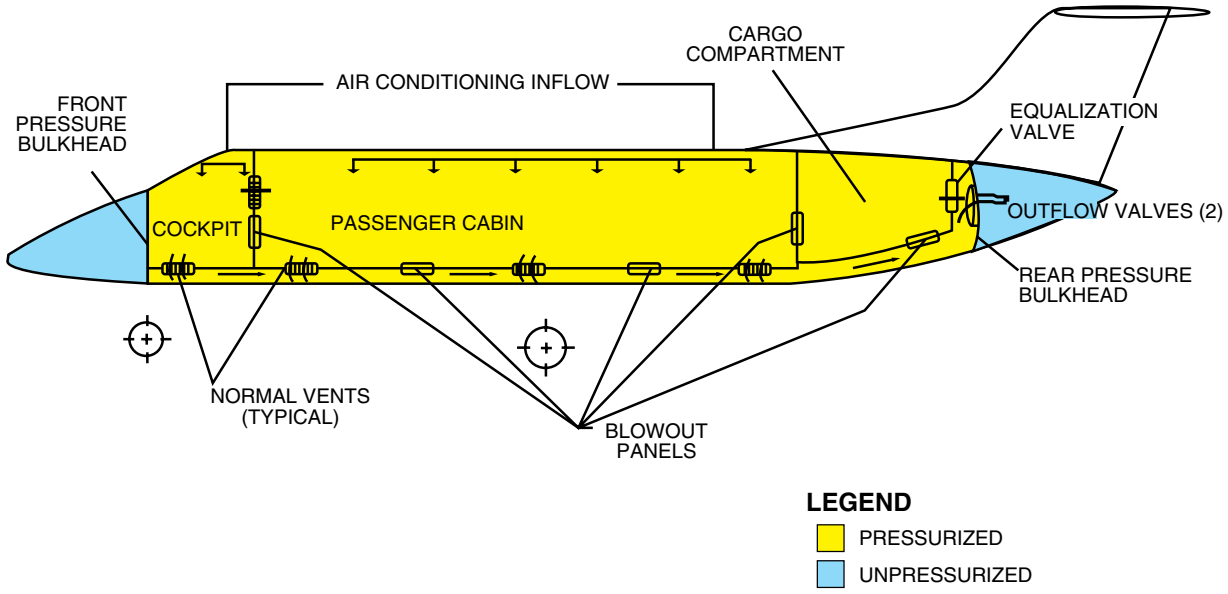


Figure 12-2. Pressurized Areas

The airflow from the cockpit and passenger cabin produces an underfloor flow. This flow is partially recirculated by fans and partially exhausted overboard through the outflow valves.

COMPONENTS

The pressurization system consists of the following components (Figure 12-1):

- Electropneumatic outflow valve
- Pneumatic outflow valve
- Ejector jet pump
- Blowout panels

ELECTROPNEUMATIC OUTFLOW VALVE

The electropneumatic outflow valve, located in the rear pressure bulkhead, controls the outflow of air from the airplane's pressurized zone to the cabin exterior through the unpressurized tail cone (Figure 12-2).

The electropneumatic outflow valve operates only when the pressurization system is in the automatic mode. It is operated electrically by a torque motor that is controlled by the electronic controller.

The diaphragm chamber of the electropneumatic outflow valve is connected to the diaphragm chamber of the pneumatic outflow valve. Therefore, when the electropneumatic outflow valve is operating, the pneumatic outflow valve is also operating and slaved to it.

The electropneumatic outflow valve is equipped with the following safety devices:

- Overpressurization relief valve
- Negative-pressure relief valve
- Cabin altitude limiter

Overpressure Relief Valve

An overpressure relief valve is built into the electropneumatic outflow valve. It senses both cabin and static pressures. When the pressure differential between the two reaches 7.2 psid, it opens the outflow valve to relieve the excess cabin pressure.

Negative-Pressure Relief Valve

A negative-pressure relief valve is also built into the electropneumatic outflow valve. When the pressure differential exceeds –0.3 psid, the valve opens to equalize the cabin pressure.

Cabin Altitude Limiter

The cabin altitude limiter is only on the electropneumatic outflow valve. When cabin altitude reaches approximately 13,000 feet, the limiter closes the electropneumatic outflow valve (and the slaved pneumatic outflow valve), limiting further cabin climb. The cabin altitude will be maintained at 13,000 feet ***only*** if the packs are operating.

If the cabin is depressurized in the manual mode of operation, there is no limitation on how high the cabin will climb.

PNEUMATIC OUTFLOW VALVE

The pneumatic outflow valve, located next to the electropneumatic outflow valve in the rear pressure bulkhead, operates independently only when the manual mode of operation is selected. Otherwise, it is slaved to the electropneumatic outflow valve.

The pneumatic outflow valve is physically similar to the electropneumatic valve except that it uses a pneumatic relay (actuated by pressure differential) rather than a torque motor (powered electrically) for operation.

The pneumatic outflow valve is equipped with the following safety devices:

- Overpressurization relief value
- Negative-pressure relief value

Overpressure Relief Valve

Like the electropneumatic outflow valve, the pneumatic outflow valve has an overpressure relief valve that opens when the pressure differential reaches 7.2 psid.

Negative-Pressure Relief Valve

As on the electropneumatic outflow valve, the negative-pressure relief valve on the pneumatic outflow valve opens to equalize the cabin pressure when the pressure differential exceeds –0.3 psid.

EJECTOR JET PUMP

An ejector jet pump, operated by engine bleed-air pressure, creates the vacuum supplied to the manual controller and used for control of the outflow valves.

An additional ejector jet pump is included when an APU is installed.

BLOWOUT PANELS

Blowout panels are installed to equalize excessive pressure differentials between the pressurized compartments.

In the event of a rupture of the pressure vessel, the blowout panels prevent structural damage to the aircraft by providing quick equalization of pressure between the various pressurized zones.

The blowout panels are located in the forward and aft partitions, aft pressure bulkhead lining, cockpit flooring, and under the skirting panel (Figure 12-2).

The cargo compartment has an equalizer valve installed. It actuates in both directions, maintaining the same pressure in the cargo compartment and the cabin.

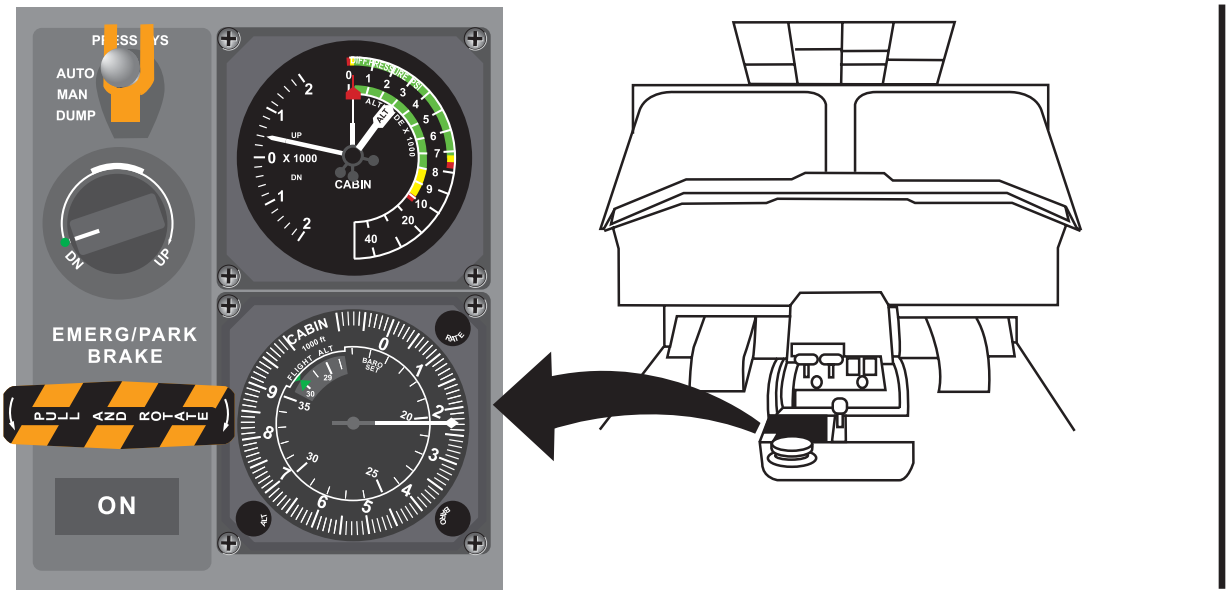


Figure 12-3. Pressurization Control Panel

CONTROL AND MONITORING

The controls and indicator for the pressurization system are located on the center pedestal between the pilots (Figure 12-3).

The controls, used to operate the two outflow valves, are:

- Mode selector switch
- Electronic controller
- Manual controller

All of the pressurization indications are displayed on a triple indicator.

MODE SELECTOR SWITCH

The functions of the three-position mode selector switch (Figure 12-3) are as follows:

- AUTO—Selects the automatic mode of operation. The electronic controller operates the electropneumatic outflow valve which, in turn, drives the slaved pneumatic outflow valve.
- MAN—Selects the manual mode of operation. The manual controller provides inputs to the pneumatic outflow valve only.
- DUMP—(Safety guard must be lifted.) Rapidly depressurizes the aircraft by opening both outflow valves. Cabin rate of climb is approximately 14,400 feet per minute (fpm). Altitude limiter limits cabin altitude to approximately 13,000 feet.



Figure 12-4. Electronic Controller

ELECTRONIC CONTROLLER

The electronic controller (Figures 12-3 and 12-4) is used in conjunction with the AUTO mode. It electrically controls the electropneumatic outflow valve through the torque motor.

Parameters for operating the pressurization system in the AUTO mode may be preset using the three knobs and scales on the electronic controller:

- ALT knob and scales
- RATE knob
- BARO knob and scale

ALT Knob and Scales

The ALT knob on the electronic controller (Figure 12-4) rotates the needle on the instrument to select:

- Cabin altitude on the outer CABIN scale (–1,500 to 10,000 feet), or
- Airplane flight level on the inner FLIGHT ALT scale

The fixed scales display the relationship between cabin altitude and aircraft altitude at the normal operating pressure differential of 7 psid. For example: 0 cabin altitude (outer scale) corresponds to approximately 17,000 feet aircraft altitude (inner scale), and 8,000 feet cabin altitude corresponds to approximately 32,000 feet aircraft altitude.

RATE Knob

The RATE knob selects the cabin rate of climb or descent in the AUTO mode.

Normal Cabin Climb Rate

The detent position of the RATE knob sets 650 fpm cabin rate of climb or 450 fpm cabin rate of descent. This is the best compromise between passenger comfort and time for cabin climb and descent during normal operations.

Maximum Cabin Climb Rate

The fully clockwise position of the RATE knob sets 1,450 fpm cabin rate of climb or 1,000 fpm cabin rate of descent.

Minimum Cabin Climb Rate

The fully counterclockwise position of the RATE knob sets 200 fpm cabin rate of climb or 150 fpm cabin rate of descent.

BARO Knob and Scale

The BARO (barometric) knob adjusts the electronic controller between 29.00 and 31.00 inches Hg. The BARO SET scale is located at the 10 o'clock position of the inner scale on the electronic controller (Figure 12-4).

MANUAL CONTROLLER

The manual controller (Figure 12-5) pneumatically controls the pneumatic outflow valve during operation in the MAN mode. The MAN mode should be used ***only*** in the event of an electrical power failure or AUTO mode failure.

When an AUTO mode failure occurs, the manual controller knob should be placed in the 1 o'clock position before switching the mode selector from AUTO to MAN.

The controller selects variable cabin climb or descent rates between 2,500 fpm rate of climb (UP position) and 1,500 fpm rate of descent (DN position).

The manual controller is not provided with a scale for setting cabin rate or altitude. It requires monitoring the triple indicator cabin rate and cabin altitude indications until the desired rate and altitude are obtained.

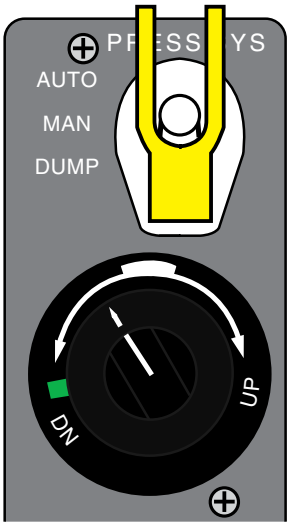


Figure 12-5. Manual Controller

TRIPLE INDICATOR

The triple indicator (Figure 12-6) includes the following indicators:

- Cabin rate needle
- Cabin altitude needle
- Differential pressure needle

The triple indicator also triggers a cabin altitude warning.



Figure 12-6. Triple Indicator

Cabin Rate Needle

The cabin rate needle indicates the cabin rate of climb or descent on the left scale.

Cabin Altitude Needle

The cabin altitude needle, marked ALT, indicates the cabin pressure altitude in thousands of feet on the right inner scale.

Differential Pressure Needle

The differential pressure needle, marked ΔP, indicates the pressure differential (delta P) between cabin pressure and outside air pressure in pounds per square inch on the right outer scale.

Cabin Altitude Warning

An altimetric switch within the triple indicator closes when the cabin altitude reaches approximately 10,000 feet. It sends a signal to the aircraft warning system which triggers the following indications:

- A 3-chime aural alert sounds.
- The voice warning "Cabin" is given.
- The red CABIN ALT light on the MAP illuminates.
- The master WARNING lights flash.

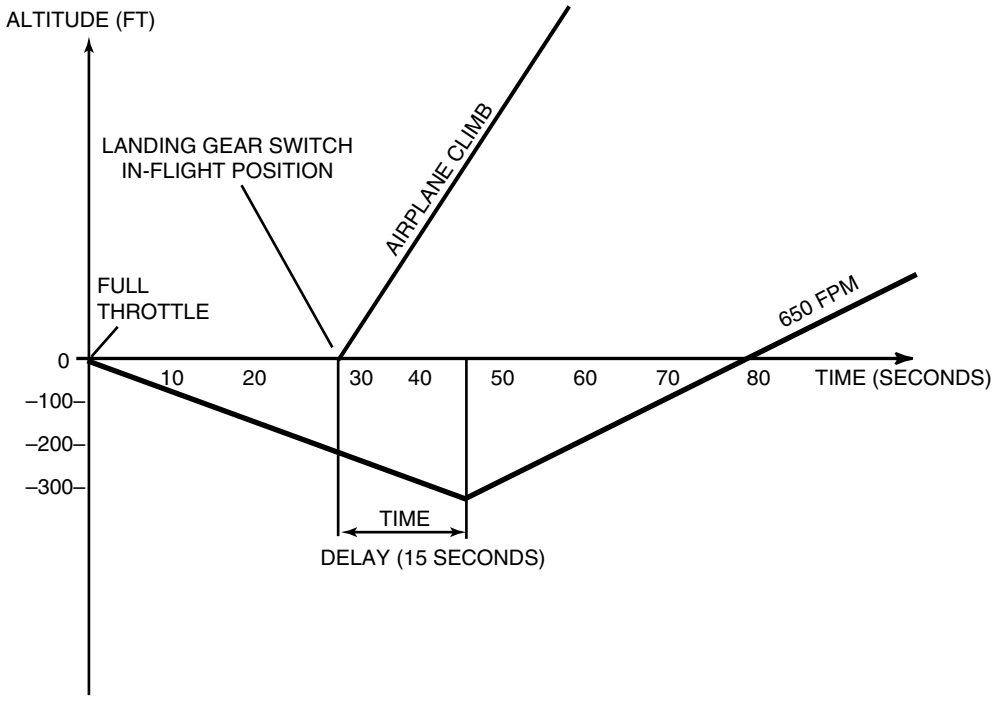


Figure 12-7. Automatic Pressurization during Takeoff

OPERATION

The pressurization system maintains the passenger cabin and cockpit pressure by controlling air-conditioning outflow to the atmosphere.

AUTOMATIC MODE

The electronic controller is powered by DC bus 2 when the mode selector is positioned to AUTO.

On the ground, the controller keeps both outflow valves open (cabin depressurized). The cabin is pressurized automatically during takeoff and depressurized after landing.

Takeoff

During takeoff with both air-conditioning packs operational, the pressurization control panel should be set up as follows:

- Mode selector switch—AUTO
- Manual controller selector—DN
- RATE knob—detent
- BARO knob—Current altimeter
- ALT knob—Assigned cruise altitude

As the power levers are advanced for takeoff, a microswitch in the left control stand sends a signal to the electronic controller. The controller initiates cabin prepressurization at a descent rate of 450 fpm (Figure 12-7). This is done to avoid the sudden cabin pressurization, or "bump," that would occur if the outflow valves rapidly closed at rotation.

When the airplane rotates, the landing gear proximity UP switch sends a signal to the electronic controller to initiate a cabin climb of 650 fpm. The climb begins 15 seconds after rotation (Figure 12-7).

Climb and Cruise

The electronic controller senses cabin pressure as an input. During aircraft climb, it compares the rate of change of cabin pressure to the programmed cabin rate of climb (RATE knob). The controller maintains the cabin climb at the programmed rate

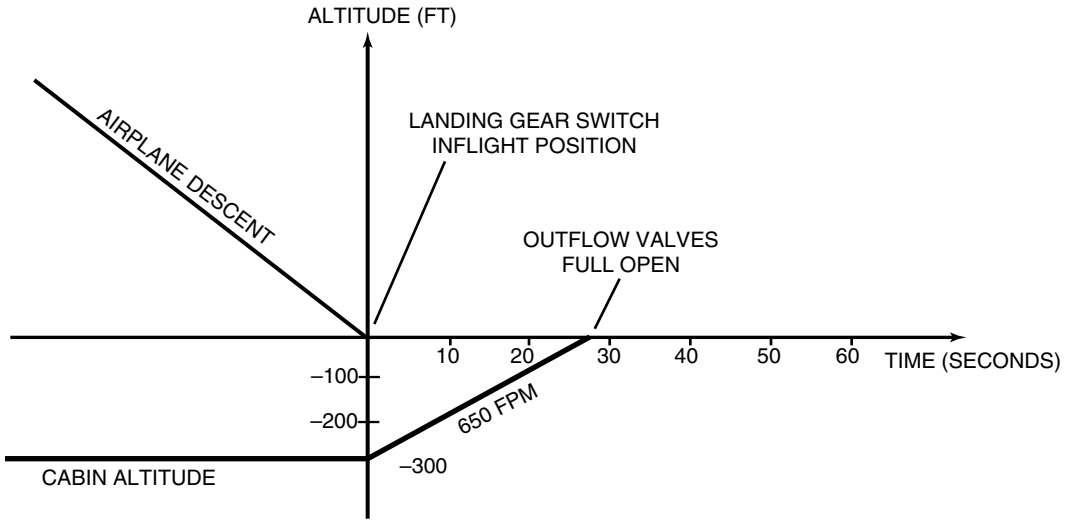


Figure 12-8. Automatic Depressurization after Touchdown

by modulating the signal to the electropneumatic outflow valve torque motor.

Once the cabin altitude set by the ALT knob is reached, it is maintained by the electronic controller.

With a cabin differential pressure of 7.0 psid during cruise and the maximum differential pressure of 7.2 psid, the aircraft may deviate up to 200 feet above the selected flight level without causing a cabin bump.

Descent and Landing

During descent, the ALT knob on the electronic controller should be set to 300 feet below the landing field elevation and the BARO knob should be set to the current altimeter setting. Once set, the electronic controller maintains a cabin rate of descent of 450 fpm (Figure 12-8).

The electronic controller requires two signals to depressurize the cabin after landing. The first is received from a microswitch in the left power lever quadrant when the power levers are reduced during the landing flare. The second is from the landing gear squat switches at touchdown. Once both signals are received, the controller depressurizes the cabin at a rate of 650 fpm (Figure 12-8).

Once the cabin is depressurized (cabin altitude equal to outside altitude), the outflow valves are fully open.

MANUAL MODE

The manual mode should be used ***only*** in the event of an electrical power failure or an AUTO mode failure.

In MAN only, the pneumatic outflow valve operates.

The manual mode is selected as follows:

- Manual controller selector—1 o'clock
- Wait 15 seconds
- Mode selector—MAN

The manual controller is used to select the desired rate of cabin climb or descent. It is selectable between –1,500 fpm (DN) and +2,500 fpm (UP).

Manual mode operation requires continuous monitoring of the triple indicator to keep the desired rate and maintain a comfortable cabin.

DEPRESSURIZATION

AUTO Mode

In the automatic mode, the aircraft is depressurized by positioning the mode selector switch to DUMP. The cabin rate of climb is approximately 14,400 fpm.

At aircraft altitudes above 13,000 feet, cabin altitude is limited to 13,000 ±1,500 feet altitude by action of the cabin altitude limiter on the electropneumatic outflow valve.

MAN Mode

In the manual mode, the aircraft is depressurized by the clockwise actuation of the manual controller to the stop (UP). This opens the pneumatic outflow valve only and depressurizes the cabin at a rate of 2,500 fpm.

There is no cabin altitude limiter installed on the pneumatic outflow valve. Therefore, when the aircraft is depressurized in the manual mode, the cabin altitude will climb until it equalizes with aircraft altitude.

SAFETY DEVICES

Following is a summary of the pressurization system safety devices installed on the electropneumatic and pneumatic outflow valves. Protections are overpressure relief, negative-pressure relief, and cabin altitude limitation.

An overpressurization relief valve is installed on each outflow valve. When the valve senses the cabin maximum differential pressure of 7.2 ±0.1 psid, it opens and climbs (depressurizes) the cabin.

A negative-pressure relief valve is also installed on each outflow valve. Consisting primarily of a check valve, the valve opens when it senses a cabin negative differential pressure of –0.3 psid.

The cabin altitude limiter is installed on the electropneumatic outflow valve only. It actuates when it senses a cabin altitude of 13,000 ±1,500 feet. When the cabin altitude limiter actuates, it closes the electropneumatic outflow valve and, through the pneumatic line connection, the pneumatic outflow valve also.

MODE SELECTOR SWITCH
AUTO— Selects the mode of operation. Electronic controller operates the electropneumatic outflow valve which, in turn, drives the slaved pneumatic outflow valve (normal operating mode).
MAN— Selects the manual mode of operation. Manual controller provides inputs to the pneumatic outflow valve only.
DUMP—(Safety guard must be lifted.) Rapidly depressurizes the aircraft by opening both approximately 14,400 fpm. Altitude limiter limits cabin altitude to approximately 13,000 ft.

CABIN CLIMB NEEDLE—
Indicates cabin rate of climb or descent. Independent of mode operation.

CABIN ALTITUDE NEEDLE—
Indicates cabin pressure altitude on the inner scale. Independent of mode operation.



CABIN ΔP NEEDLE—
Indicates pressure differential between cabin pressure and outside air pressure on the outer scale. Independent of mode operation.

MANUAL CONTROLLER—
Pneumatically controls pneumatic outflow valve in MAN mode.
DN: Cabin rate of –1,500 fpm
UP: Cabin rate of +2,500 fpm

RATE KNOB—
Marked detent position: Selects 650 fpm cabin climb or 450 fpm cabin descent.

Full CCW: –150 fpm cabin descent or +200 fpm cabin climb

Full CW: –1,000 fpm cabin descent or +1,450 fpm cabin climb

BARO SET SCALE—
Indicates the barometric setting.

BARO KNOB—
Sets barometric pressure on the BARO SET scale between 29.00 and 31.00 in Hg.

ALT KNOB—
Rotates needle to select altitude between –1,500 ft and +10,000 ft (outer scale), or aircraft flight altitude (inner scale).

ELECTRONIC CONTROLLER NEEDLE—
Indicates the selected cabin altitude on outer scale, or selected flight altitude on the inner scale.

Figure 12-9. Pressurization System Controls and Indicators

QUESTIONS

1. The airplane is pressurized by means of:
A. Ram air entering the cabin from wing scoops
B. Conditioned air from air-conditioning system
C. Pressurization air pump
D. Gasper fan outlets

2. Pressurization is controlled in:
A. Sequence mode
B. Automatic mode
C. Manual mode
D. Both B and C

3. Which of the following is not pressurized?
A. Flight deck
B. Cabin
C. Baggage compartment
D. Avionics compartment

4. The maximum pressurization differential is:
A. 6.5 psid
B. 7.0 psid
C. 7.2 psid
D. 8.0 psid

5. How many outflow valves control air discharge?
A. One
B. Two
C. Three
D. Four

6. In the AUTO mode, which valves are operational?
A. Electropneumatic outflow valve
B. Pneumatic outflow valve
C. Both A and B
D. Either A or B, depending on altitude

7. In manual mode, which valve(s) are operational?
A. Electropneumatic outflow valve
B. Pneumatic outflow valve
C. Both A and B
D. Either A or B, depending on altitude

8. The CABIN warning light comes on at:
A. 9,000 ft
B. 10,000 ft
C. 12,000 ft
D. 12,500 ft

9. With the engines and electrical system off:
A. The electropneumatic valve is closed and the pneumatic valve is open.
B. The electropneumatic valve is open and the pneumatic valve is closed.
C. Both valves are open.
D. Both valves are closed.

10. The absolute cabin altitude limit is:
A. 11,500 ft
B. 13,000 ft
C. 14,500 ft
D. 15,000 ft

11. The cabin climb rate may be set between:
A. 50–1,000 fpm
B. 100–1,000 fpm
C. 200–1,450 fpm
D. 200–1,500 fpm

12. The cabin descent rate may be set between:
A. 50–1,000 fpm
B. 100–1,000 fpm
C. 150–1,000 fpm
D. 150–1,450 fpm

13. On the ground, engines running, electrical power on, power levers at GND IDLE:

A. The electropneumatic valve is open.
B. The pneumatic valve is open.
C. Both valves are closed.
D. Both valves are open.

14. The AUTO mode of setting the pressurization is always used unless:

A. The flight is to be made below 5,000 ft.
B. There is an electrical failure.
C. The AUTO system fails.
D. Either B or C is correct.

15. During descent, if the controller is set as recommended, the cabin should descend at:

A. 250 fpm
B. 400 fpm
C. 450 fpm
D. 500 fpm

16. Upon landing, the cabin is automatically depressurized.

A. True
B. False

CHAPTER 13
HYDRAULIC POWER SYSTEM

CONTENTS

Page

INTRODUCTION 13-1

GENERAL 13-1

Green System 13-3

Blue System 13-3

HYDRAULIC POWER SYSTEM 13-3

General 13-3

Components 13-3

Hydraulic Reservoir Pressurization System 13-6

CONTROL AND MONITORING 13-7

Hydraulic Power Control Panel 13-7

Hydraulic Shutoff Valve Light 13-10

OPERATION 13-10

Hydraulic Fluid Circuits 13-10

Circuit Operation 13-10

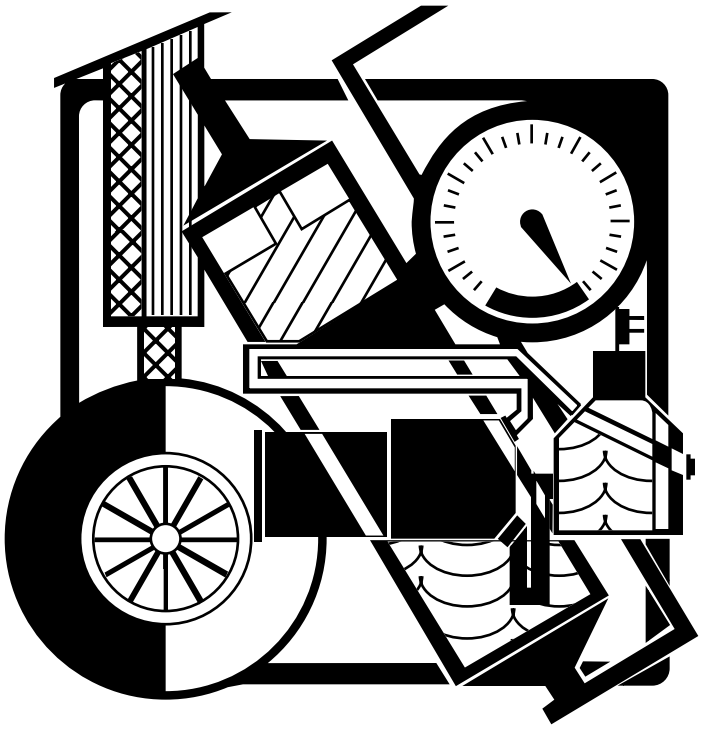
Fluid Specification 13-10

QUESTIONS 13-12

ILLUSTRATIONS

Figure	Title	Page
13-1	Hydraulic System Schematic (Typical)	13-2
13-2	Hydraulic Compartment	13-3
13-3	Hydraulic Reservoir Pressurization	13-6
13-4	Hydraulic Power Control Panel	13-7
13-5	Hydraulic System Operation	13-9
13-6	Hydraulic Power System Controls and Indicators	13-11

CHAPTER 13
HYDRAULIC POWER SYSTEM



INTRODUCTION

The EMB-120 Brasilia has two independent hydraulic systems, each powered by its respective engine-driven pump. An electric pump provides a backup hydraulic power source for each system. Bleed air from the engines or APU is used to pressurize the two hydraulic reservoirs.

GENERAL

The majority of the components for the aircraft's left and right hydraulic systems are located in the left and right engine nacelles. Referred to as green and blue respectively, the two systems are virtually identical.

The primary hydraulic power sources are hydraulic pumps driven by the propeller reduction gearboxes on each engine.

In the event of a failure of the primary pumps, each system incorporates an electric pump as a backup hydraulic power source.

Each hydraulic system powers separate subsystems. The descriptions of the two systems and the subsystems they power follows.

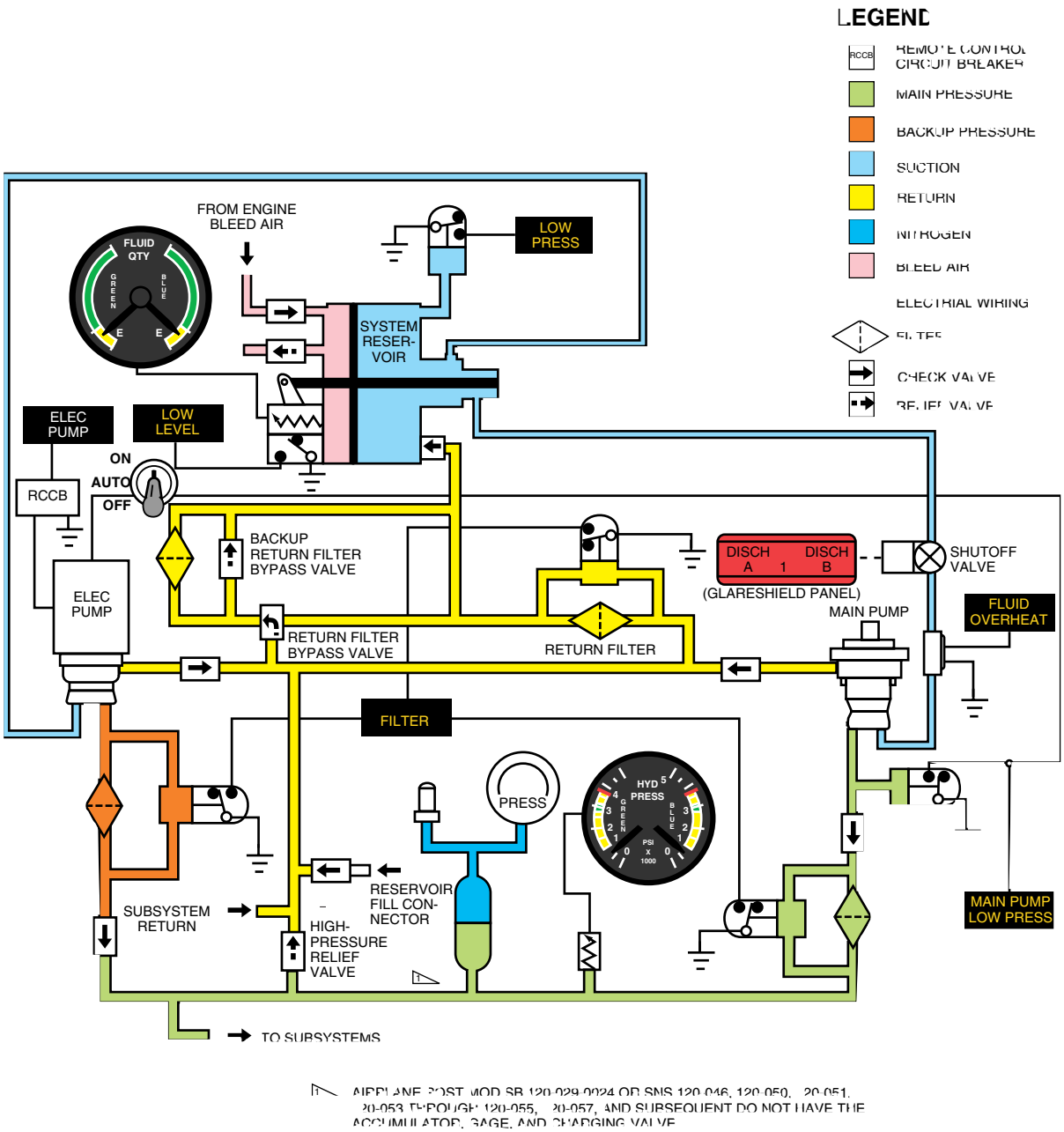


Figure 13-1. Hydraulic System Schematic (Typical)

GREEN SYSTEM

The left, or green, hydraulic system powers the following subsystems:

- Nosewheel steering
- Main entry door
- Landing gear
- Normal brakes (outboard pair)
- Flaps (outboard pair)
- Rudder system 1

BLUE SYSTEM

The right, or blue, hydraulic system powers the following subsystems:

- Normal brakes (inboard pair)
- Emergency brakes (both pairs)
- Flaps (inboard and nacelle pairs)
- Rudder system 2

HYDRAULIC POWER SYSTEM

GENERAL

Both the green and blue hydraulic systems contain the same components (Figure 13-1) and differ only in the subsystems they operate. Therefore, the following information is applicable to both systems.

The majority of the components are found in the respective hydraulic compartment, located in the underwing section of the engine nacelle just aft of the main gear (Figure 13-2).



Figure 13-2. Hydraulic Compartment

COMPONENTS

The following are major components of a hydraulic system:

- Engine-driven hydraulic pump
- Electric hydraulic pump
- Hydraulic reservoir
- Hydraulic control unit
  - Bypass valves
  - Relief valve
- High- and low-pressure filters
- Hydraulic shutoff valve
- Check valves
- Thermal switch
- Drains

Engine-Driven Hydraulic Pump

A hydraulic pump installed on each engine's reduction gearbox accessory section is mechanically driven by the propeller reduction gearbox.

The pumps are axial piston pumps with a self-adjusted outlet pressure of 3,000 psi. In the event of a hydraulic pump overload, or seizure, the pump drive shaft is designed to shear.

Electric Hydraulic Pump

An electric hydraulic pump is installed in the hydraulic compartment (Figure 13-2). It is used as a backup for the engine-driven hydraulic pump.

The electric pump, like the engine-driven pump, is an axial piston pump with a self-adjusted output pressure of 3,000 psi.

The pump is protected from overheat by a thermal switch that shuts off the electric motor if the temperature exceeds 150°C.

Hydraulic Reservoir

A 4.3 liter hydraulic reservoir is located in each hydraulic compartment (Figure 13-2).

The reservoir is pressurized with bleed air to ensure a constant supply of hydraulic fluid is available to the pumps. Either engine or the APU may be used to supply air to both reservoirs.

A single pressure regulating valve keeps both reservoirs at a constant 30 psi. If the pressure exceeds 39 psi, a relief valve opens to relieve the pressure. The valve resets at 33 psi.

A check valve in the pressure line keeps the hydraulic reservoir pressurized in the event of a loss of bleed air.

Hydraulic Control Unit

A hydraulic control unit is integral with each reservoir. It contains the bypass valves and relief valve. The high- and low-pressure filters are attached to the unit (Figure 13-2).

Bypass Valves

The two bypass valves in the reservoir control unit are the return filter bypass valve and the back-up return filter bypass valve (Figure 13-1). If the primary return (low-pressure) filter clogs, a pressure differential across the filter of 65 psid opens the return filter bypass valve allowing flow to the secondary return (low-pressure) filter.

Should the secondary return filter also become clogged, the backup return filter bypass valve opens, also at 65 psid, and permits fluid to return to the reservoir—even if contaminated. Both bypass valves reset at 55 psid.

Relief Valve

The high-pressure relief valve prevents excessive pressure from damaging the hydraulic components.

High - and Low - Pressure Filters

There are four filters, two high pressure and two low pressure (return) in each hydraulic system. The filters, located on the hydraulic control unit, are physically identical although they serve different functions.

The two high - pressure filters are installed in the pressure lines from the engine-driven and electric hydraulic pumps. The two low-pressure return filters, primary and secondary, are installed in the return lines prior to the hydraulic reservoir (Figure 13-1).

Three of the filters, the two high pressure and the primary low-pressure return, are equipped with a pressure switch. If any of the filters are bypassed, its switch illuminates the system FILTER light on the hydraulic control panel.

Hydraulic Shutoff Valve

A hydraulic shutoff valve is installed in each hydraulic compartment.

The valve is used to shutoff the flow of hydraulic fluid to the engine driven hydraulic pump in the event of an engine fire.

This normally open gate valve is operated by a 28-VDC electric motor, powered by the hot battery bus. Actuation is by the engine fire "T" handles.

A CLOSED light on the fire extinguishing panel illuminates when the valve closes after the handle is pulled.

A relief valve, in the shutoff valve body, is adjusted to open at 60 psi to relieve hydraulic fluid pressure due to thermal expansion.

Check Valves

Five check valves prevent opposing pressure from interfering with component operation. A check valve is installed downstream of the pressure and return outlets of each pump, and one is installed at the reservoir fill connector.

Thermal Switch

A thermal switch is installed in the engine-driven pump suction line. It triggers the FLUID OVERHEAT light on the hydraulic control panel.

Drains

Each hydraulic system has three hydraulic fluid drains. They are located downstream of each hydraulic pump and after the low - pressure relief valve.

The pump drains collect hydraulic fluid in the event of pump shaft leakage.

The relief valve drain collects hydraulic fluid from excess reservoir pressure, or from reservoir bleed while filling.

The three hydraulic drains are connected with other engine drains, and vented to the atmosphere through the underwing mast drain at the rear of each nacelle.

HYDRAULIC RESERVOIR PRESSURIZATION SYSTEM

Air for the hydraulic reservoir pressurization system (Figure 13-3) comes from the aircraft pneumatic system; specifically, the deicing duct of each engine.

A restrictor is installed to prevent excessive bleed-air flow from the engine in the event of a reservoir failure.

A check valve is installed downstream of the restrictor to prevent airflow back through the ducting in the event of an engine failure.

The air from each engine, once leaving its check valve, enters a single-pressure regulating valve. The bleed air, at approximately 160 psi, is reduced to 30 psi by the pressure regulating valve prior to entering the reservoirs.

During normal operation, the hydraulic reservoirs are pressurized through the pressure regulating valve by bleed air from both engines. If one engine fails, air from the other engine is sufficient to pressurize both reservoirs.

The APU may also be used to supply air through the pressure regulating valve to pressurize both reservoirs.

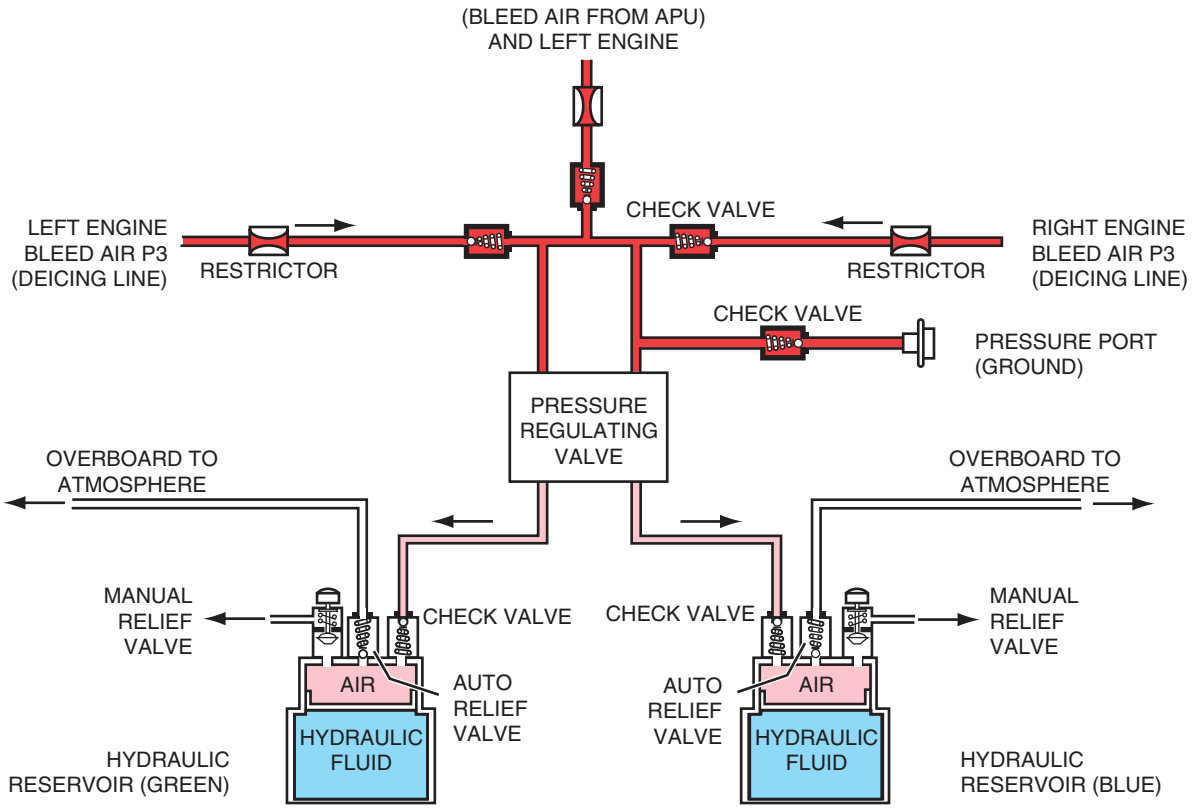


Figure 13-3. Hydraulic Reservoir Pressurization

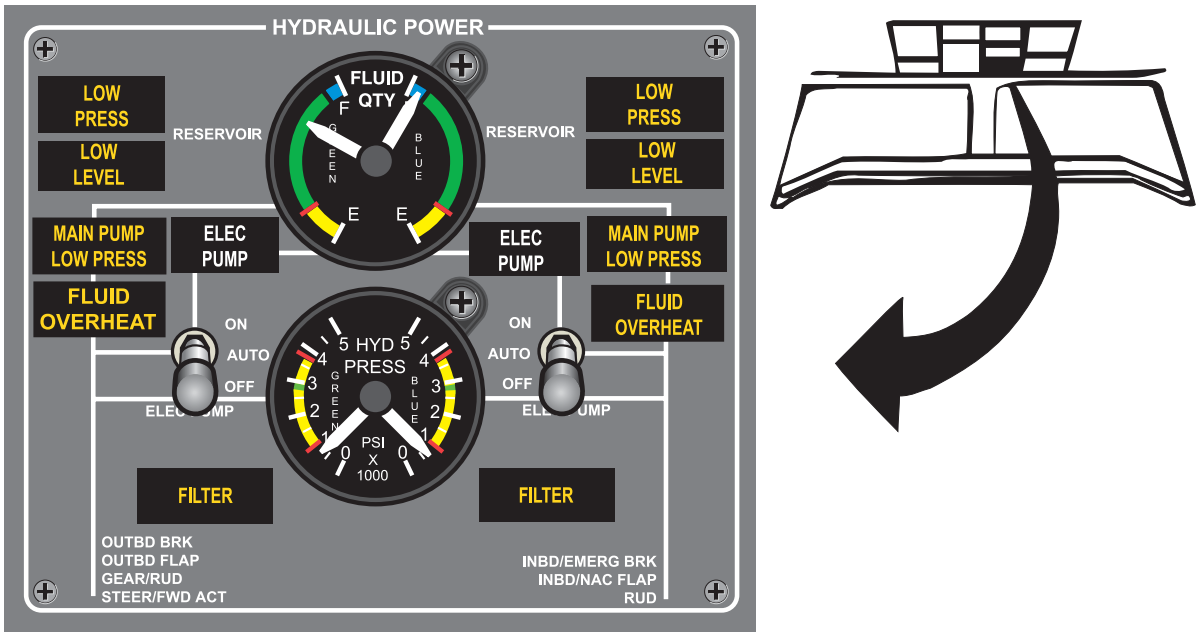


Figure 13-4. Hydraulic Power Control Panel

CONTROL AND MONITORING

The controls and indicators for both the green and blue hydraulic systems are located on the overhead HYDRAULIC POWER control panel (Figure 13-4).

The amber HYD light on the MAP illuminates in conjunction with any of the amber lights on the HYDRAULIC POWER control panel.

HYDRAULIC POWER CONTROL PANEL

The HYDRAULIC POWER control panel contains alarm/indicating lights, electric hydraulic pump switches for each system, and dual fluid quantity and hydraulic pressure indicators for both systems.

The panel is arranged with the green hydraulic system controls and indicators on the left side and the blue system on the right. The two-dual system indicators are located in the center of the panel.

Alarm/Indicating Lights

The hydraulic alarm/indicating lights are:

- RESERVOIR LOW PRESS (amber)
- RESERVOIR LOW LEVEL (amber)
- MAIN PUMP LOW PRESS (amber)
- FLUID OVERHEAT (amber)
- FILTER (amber)
- ELEC PUMP (white)

Illumination of any amber light on the HYDRAULIC POWER panel triggers the following indications and warnings:

- A single chime aural alert sounds
- The amber HYD light on the MAP illuminates
- The master CAUTION lights flash

LOW PRESS Light

The amber LOW PRESS light illuminates when pressure in the related hydraulic reservoir drops below 12 psi.

LOW LEVEL Light

The amber LOW LEVEL light illuminates when fluid volume in the related hydraulic reservoir drops below 0.8 liter.

MAIN PUMP LOW PRESS Light

The amber MAIN PUMP LOW PRESS light illuminates anytime the output of the related engine-driven hydraulic pump is below 1,500 psi. The light goes out when pump output exceeds 1,950 psi.

FLUID OVERHEAT Light

The amber FLUID OVERHEAT light illuminates when the hydraulic fluid temperature in the related system exceeds 120°C. The light remains on until the temperature drops below 110°C.

FILTER Light

The amber FILTER light monitors three of the four filters in a hydraulic system.

Located in parallel with the high- pressure filters of each hydraulic pump (engine driven/electrical) and the primary return filter, pressure switches trigger the light when a filter is bypassed.

ELEC PUMP Light

The white ELEC PUMP light is illuminated when the related electric hydraulic pump is operating.

ELEC PUMP Switch

The three position ELEC PUMP switch has the following functions:

- ON—The related electrical pump operates regardless of system status.

NOTE

The electric pump pressurizes the hydraulic system to the normal 3,000 psi. However, due to a check valve downstream of the main pump, the MAIN PUMP LOW PRESS light remains illuminated.

- AUTO—The related pump automatically comes on if the engine-driven pump output drops below 1,500 psi.
- OFF—The electric pump will not operate.

FLUID QTY Indicator

The FLUID QTY indicator, in the top center of the panel, is a dual gage that indicates hydraulic reservoir fluid level for both systems. The left needle indicates green system level and the right indicates blue.

There are two range markings on the indicator:

- Yellow band—empty to 0.8 liter
- Green band—0.8 to 4.3 liters

HYD PRESS Indicator

The HYD PRESS indicator, in the bottom center of the panel, is a dual gage that indicates hydraulic pressure in both systems. The left needle indicates green system pressure and the right indicates blue.

The range markings on the indicator are:

- Red radials—Indicate maximum (3,800 psi) and minimum (1,000 psi) pressure
- Green band—Normal operating range (2,700 to 3,100 psi)
- Yellow bands:
  - High-pressure caution (3,100 to 3,700 psi)
  - Low-pressure caution (2,700 to 1,000 psi)

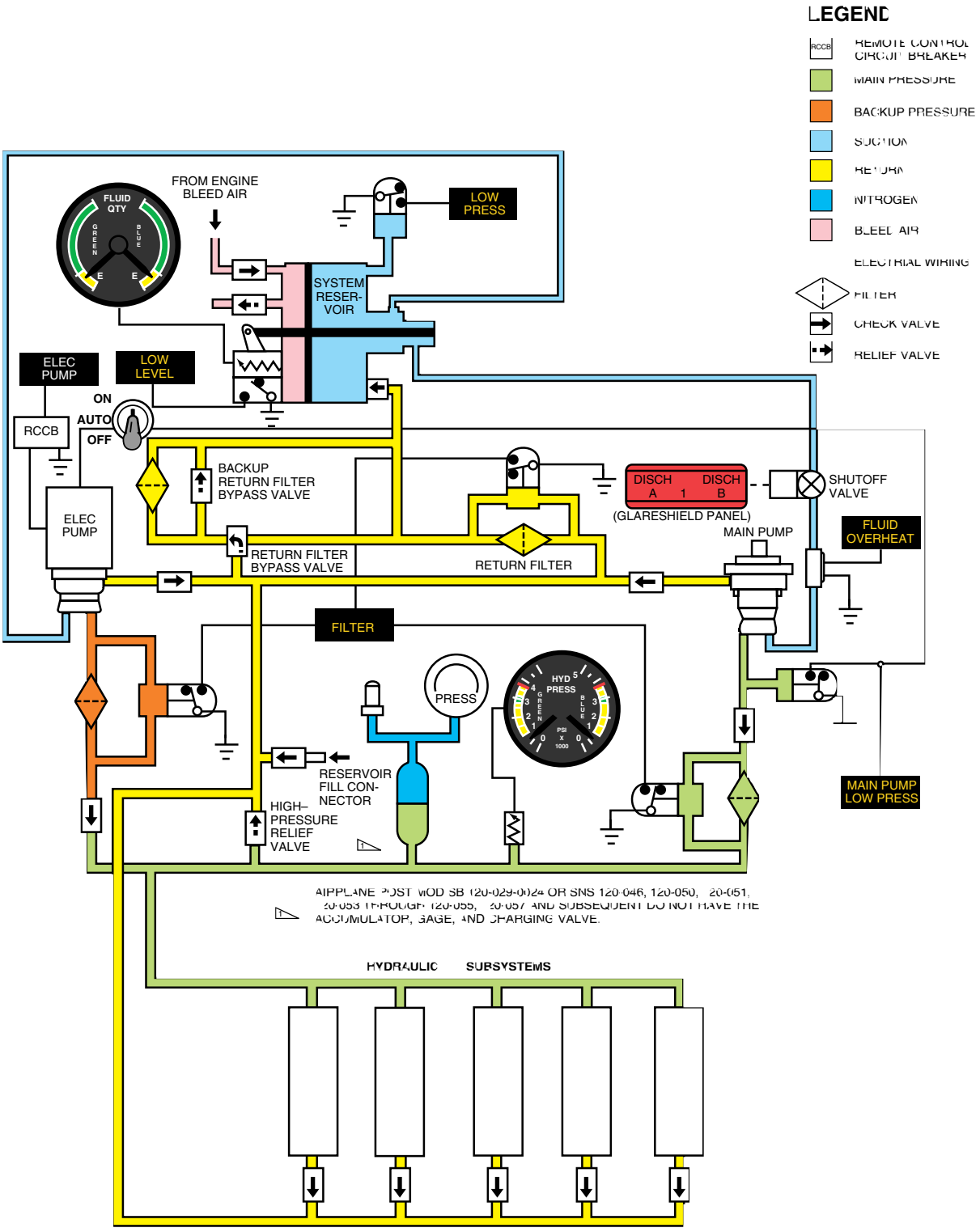


Figure 13-5. Hydraulic System Operation

HYDRAULIC SHUTOFF VALVE LIGHT

The hydraulic shutoff valve CLOSED lights are located on the fire extinguishing panel in the center of the glareshield. They illuminate when the respective hydraulic firewall shutoff valve is closed by the fire handle.

OPERATION

Although the aircraft hydraulic system is made up of two independent hydraulic systems (green and blue), both systems operate in the same manner. Figure 13-5 is used to illustrate hydraulic system operation.

HYDRAULIC FLUID CIRCUITS

The aircraft hydraulic systems are circuits that start and finish at the hydraulic reservoirs

The pressurized reservoirs ensure a positive hydraulic fluid supply to the engine-driven and electric hydraulic pumps.

The hydraulic fluid circuits have suction, pressure, and return lines.

Suction Lines

Each hydraulic reservoir has two suction lines: one to the engine-driven hydraulic pump, through the hydraulic shutoff valve; and the other to the electric hydraulic pump.

Pressure Lines

Both the engine-driven and electric hydraulic pumps increase the pressure of the fluid from the reservoir to 3,000 psi. Under normal conditions, only the engine-driven pumps operate. (In the event of an engine-driven pump failure, the related electric pump automatically comes on, provided the ELEC PUMP switch is in AUTO.)

Leaving either the engine-driven or electric hydraulic pumps, the fluid goes through a check valve and high-pressure filter. The check valve prevents hydraulic fluid flow to the opposite pump, and the high-pressure filter prevents contamination of the hydraulic system in the event of a pump failure.

The pressure lines from the engine-driven pump and the electric pump merge and distribute the pressurized hydraulic fluid to the subsystems.

A high-pressure relief valve is installed to discharge excess pressure to the return system in the event of a hydraulic pump pressure adjustment device failure. The relief valve opens at 3,650 psi and closes again at 3,300 psi.

Return Lines

The return lines from the hydraulic subsystems, the hydraulic pump drain lines, and the high-pressure relief valve, all join after their respective check valves. The fluid is then routed to the primary low-pressure filter.

If the primary low-pressure filter clogs, a bypass valve diverts the fluid to the secondary low-pressure filter. If the secondary filter also clogs, its bypass valve opens and the hydraulic fluid is routed, unfiltered, direct to the reservoir.

CIRCUIT OPERATION

With a hydraulic pump operating but no subsystem in use, the fluid circuit is from the reservoir to the operating pump, then through the pump drain line to the low-pressure filter and back to the reservoir.

When a subsystem is actuated (i.e., landing gear lowered), the fluid circuit is now from the reservoir to the pump, then through the high - pressure filter to the actuated subsystem. From the subsystem, the fluid enters a return line, joins back with the main return system, and returns to the reservoir through the primary return filter.

FLUID SPECIFICATION

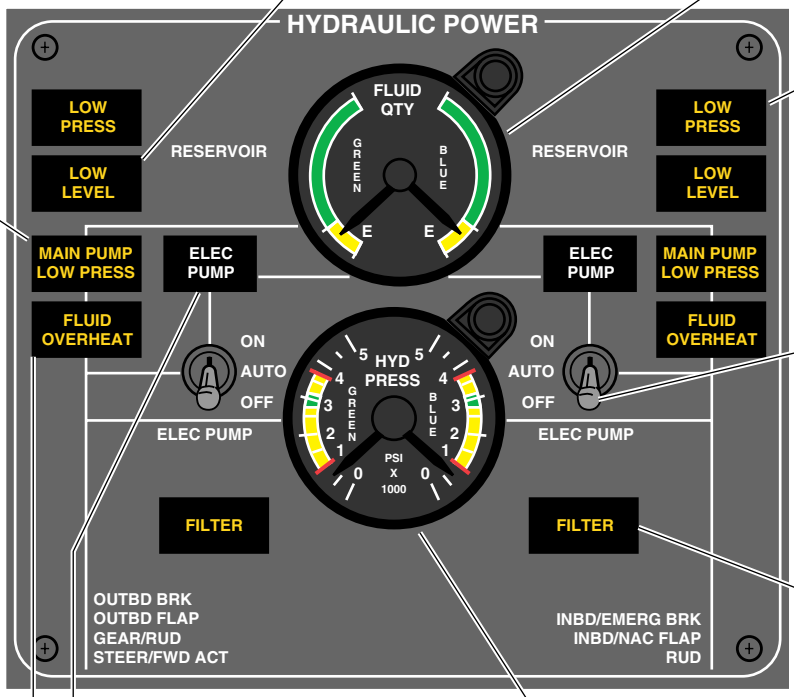
The hydraulic fluid used in the EMB-120 hydraulic system is MIL-H-5606.

GREEN SYSTEM **MAIN PUMP LOW PRESS**URE LIGHT—Illuminates when the engine-driven pump outlet pressure drops below 1,500 psi.

GREEN SYSTEM RESERVOIR **LOW LEVEL** LIGHT—Illuminates when the reservoir level is below 0.8 liters.

HYDRAULIC FLUID QUANTITY INDICATOR—Indicates the respective system fluid quantity. The bands are from full level (4.3 L) to 0.8 L, green range: from 0.8 L to empty, yellow range.

BLUE SYSTEM RESERVOIR **LOW PRESS**URE LIGHT—Illuminates when the reservoir pressure drops below 12 psig.



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BLUE SYSTEM ELECTRIC HYDRAULIC PUMP SWITCH

ON—Turns on the electrically driven hydraulic pump. Limit is 20 seconds of operation, due to high drain on the battery. Use is unlimited when the engine starter/generator is generating.

AUTO—The electrically driven pump is automatically turned on when the corresponding engine-driven pump outlet pressure drops below 1,500 psi.

OFF— Turns off electrically driven pump.

BLUE SYSTEM **FILTER** LIGHT—Illuminates when at least one of the following filters becomes clogged: engine pump pressure filter, electric hydraulic pump pressure filter and return filter.

GREEN SYSTEM **ELEC**TRIC HYDRAULIC **PUMP** LIGHT—Illuminates when the electrically driven pump is functioning.

HYDRAULIC PRESSURE INDICATOR—Indicates the respective system pressure:

- Green band at 2,700 to 3,100 psi
- Yellow band at 1,000 to 2,700 psi and 3,100 to 3,700 psi
- Red radials at 1,000 and 3,800 psi

GREEN SYSTEM MAIN PUMP **FLUID OVERHEAT** LIGHT—Illuminates when fluid temperature at the suction line exceeds 248˚F (120˚C).

Figure 13-6. Hydraulic Power System Controls and Indicators

QUESTIONS

1. The EMB-120 has two independent hydraulic systems.

A. True
B. False

2. The systems are known as:

A. Green system
B. Red system
C. Blue system
D. Both A and C

3. Hydraulic pressure is provided by:

A. Engine-driven pumps
B. Bleed-air-driven pumps
C. Electric pumps only
D. None of the above

4. Hydraulic fluid may be blocked from flowing through the main hydraulic pump by:

A. Turning the electric pump switch OFF
B. Turning the electric pump switch ON
C. Pulling the engine fire "T" handle
D. Turning off the accumulator

5. The two hydraulic systems are basically identical.

A. True
B. False

CHAPTER 14
LANDING GEAR AND BRAKES

CONTENTS

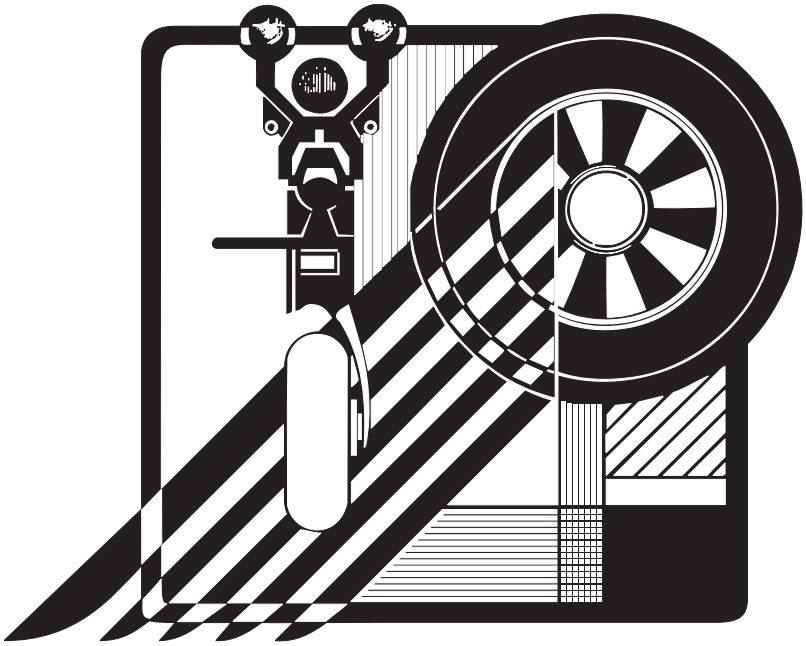
Page

INTRODUCTION 14-1
GENERAL 14-1
LANDING GEAR 14-2
General 14-2
Landing Gear Actuating System 14-3
Landing Gear Actuation 14-8
Landing Gear Indicating System 14-10
Landing Gear Warning System 14-11
BRAKES 14-13
Normal Brake System 14-13
Antiskid System 14-17
Locked Wheel Protection 14-19
Touchdown Protection 14-20
Operational Features 14-20
Emergency/Parking Brake System 14-21
NOSEWHEEL STEERING 14-23
General 14-23
Components 14-23
Nosewheel Steering Operation 14-25
QUESTIONS 14-28

ILLUSTRATIONS

Figure	Title	Page
14-1	Nose Gear and Main Gear	14-2
14-2	Control Module	14-3
14-3	Emergency Gear Compartment	14-4
14-4	Gear and Door Hydraulic Circuit	14-5
14-5	Alternate/Emergency Extension Controls	14-7
14-6	Electrical Override Switch	14-9
14-7	Emergency Extension System Controls	14-10
14-8	Landing Gear Indicating System	14-10
14-9	Retraction Sequence Indications	14-11
14-10	Extension Sequence Indications	14-11
14-11	Brake System	14-13
14-12	Brake Control Panel—Pressure Switch Indication	14-14
14-13	Brake Control Panel—TEST Pushbuttons	14-15
14-14	Normal Brake System	14-16
14-15	Antiskid System	14-17
14-16	Differential Braking Technique Schematic	14-19
14-17	Emergency/Parking Brake System	14-20
14-18	Nosewheel Steering System	14-22
14-19	Landing Gear Controls and Indicators	14-26
14-20	Brakes and Nosewheel Steering Controls and Indicators	14-27

CHAPTER 14
LANDING GEAR AND BRAKES



INTRODUCTION

The EMB-120 has a tricycle landing gear system with dual-wheel assemblies on each gear. Gear position is determined by a set of indicator lights adjacent to the gear handle. Gear warnings are provided by indicator lights, a warning horn, and a synthesized voice.

A steerable nosewheel provides directional control during ground operations.

GENERAL

The landing gear systems are separated into the following subsystems:

- Landing gear
- Brakes
- Nosewheel steering

Landing gear actuation may be achieved by three different systems:

- Normal system—electrically controlled/hydraulically actuated
- Alternate electrical override system—electric/hydraulic
- Emergency free-fall extension system—mechanical

The main wheel brake assemblies are equipped with hydraulic brakes. Normal brake operation is available with or without antiskid.

An emergency/parking brake system is independent of the pedals and actuated by a handle on the cockpit center console.

The hydraulic nosewheel steering is controlled by both a steering wheel on the pilot's left console and the rudder pedals.

LANDING GEAR

GENERAL

The landing gear (Figure 14-1) consists of the following major components:

- Main oleo strut
- Main drag/locking strut
- Secondary drag/locking strut
- Dual-actuating hydraulic cylinder
- Two aluminum alloy wheel assemblies

The two main landing gear also have multidisc brake assemblies.

The nose gear has nosewheel steering actuator.

Each wheel assembly consists of a wheel with appropriate bearings, seals, and a tubeless tire. The main and nose tires are rated to 165 knots.

To protect the tires, each wheel has three thermal fuses designed to open when tire temperatures reach 140°C.

Landing gear doors are hydraulically or mechanically actuated. In flight, all landing gear doors close to improve aerodynamics. The forward doors for each landing gear are hydraulically actuated and close when the gear is extended or retracted. The aft doors are mechanically fixed to, and operate with, the gear.

Landing gear extension and retraction is accomplished by the electrically controlled/hydraulically actuated normal landing gear system. In the event of a failure of the normal system, the landing gear may be extended by an electrical override extension system, also electrically controlled/hydraulically actuated. If both of the normal and override systems fail, the aircraft is equipped with a mechanical emergency free-fall extension system.

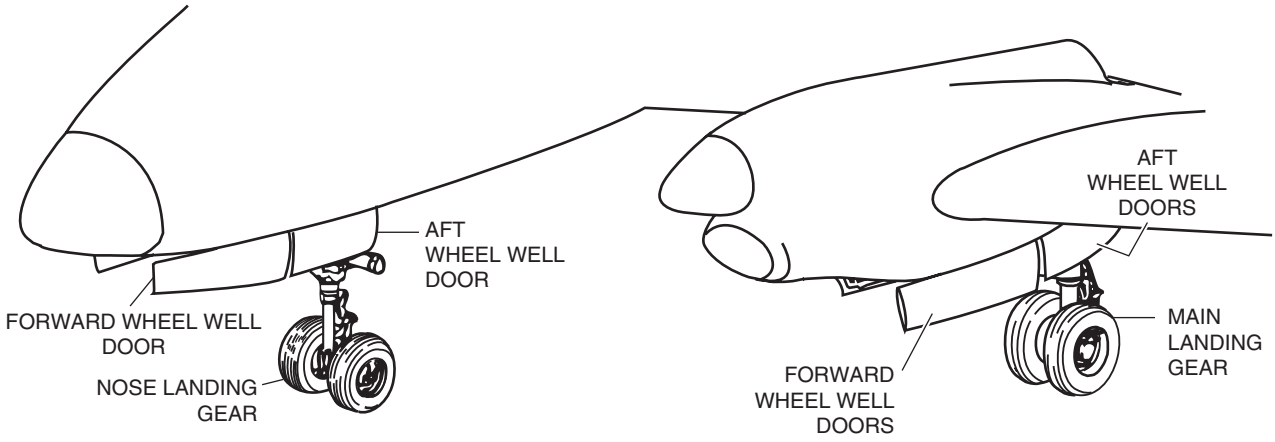


Figure 14-1. Nose Gear and Main Gear

Hydraulic pressure for landing gear extension and retraction is provided by the green hydraulic system.

The braking system used to stop main wheel rotation during retraction gets its pressure from the landing gear retraction line. Two snubbers installed in the nose wheel well stop nose wheel rotation.

The landing gear indicating system allows the crew to monitor the position of the landing gear.

The warning system monitors landing gear, flap, and power lever configurations to prevent gear-up landings.

Both main and nose landing gear have provisions for ground locking pins to prevent inadvertent gear retraction on the ground.

LANDING GEAR ACTUATING SYSTEM

The landing gear actuating system provides the means to retract and extend the landing gear. It consists of the following:

- Electrical control circuit
- Gear and door hydraulic circuits
- Alternate extension system
- Emergency extension system

Electrical Control Circuit

The landing gear electrical control circuit is used to control the landing gear actuating solenoid valves and the forward door solenoid valves. The circuit operates using the following components:

- Control module
- Relay box
- Proximity sensors
- Electrical override switch

Control Module

The landing gear control module, located on the copilot's instrument panel (Figure 14-2), provides a means for the flight crew to operate and monitor the landing gear. It includes the landing gear control lever, the landing gear position indicating lights and test buttons, and a DOWN LOCK REL button. The control module is powered by both emergency DC buses 1 and 2.

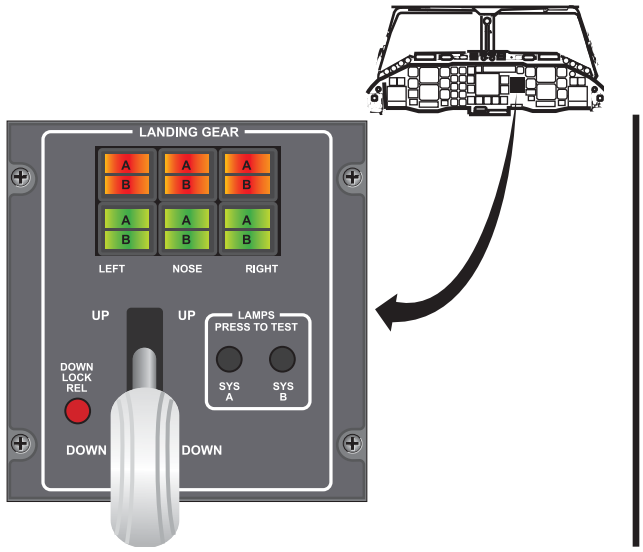


Figure 14-2. Control Module

The landing gear lever activates an internal electrical switch energizing a relay in the landing gear relay box.

When the aircraft is on the ground, an electrical solenoid mechanically locks the gear lever preventing it from being moved to the UP position. If required, the electrical solenoid may be bypassed by pressing the red DOWN LOCK REL button and moving the lever to the UP position.

Circuits internal to the landing gear control module receive signals from proximity sensors and display the position of the landing gear on the appropriate position indicator lights.

There are two separate but identical circuits used to indicate gear position; circuit A and circuit B. As a result, each landing gear has two red and two green position indicating lights, one for each circuit. The LAMPS test buttons, labeled SYS A and SYS B, test the lights only.

The red indicator lights indicate a disagreement between the landing gear position and the landing gear lever position.

The green indicator lights indicate that the corresponding landing gear is down and locked.

Relay Box

The landing gear relay box contains the relays that control the actuating solenoid valves for the landing gear and the gear doors.

Together, the relay box and the landing gear control module provide the electrical circuit to control extension and retraction of the landing gear.

Proximity Sensors

The aircraft is equipped with two uplock and two downlock proximity sensors on each gear (circuit A and circuit B). They provide the input to the landing gear position indicators.

There is also one proximity sensor on each landing gear shock absorber to sense air/ground position and actuate the landing gear control lever lock solenoid.

Electrical Override Switch

An electrical override switch is provided to lower the landing gear should the control module or relay box fail.

The switch is in the emergency gear compartment (Figure 14-3) located under a door in the cockpit floor to the right of the control pedestal.

Gear And Door Hydraulic Circuits

The gear and door hydraulic circuit (Figure 14-4) provides a means to open and close the forward wheel well doors and to extend and retract the landing gear.

For the purpose of discussion, the hydraulic circuit is divided into two subsystems; the gear retraction/extension hydraulic system, and the forward wheel well doors opening/closing hydraulic system.

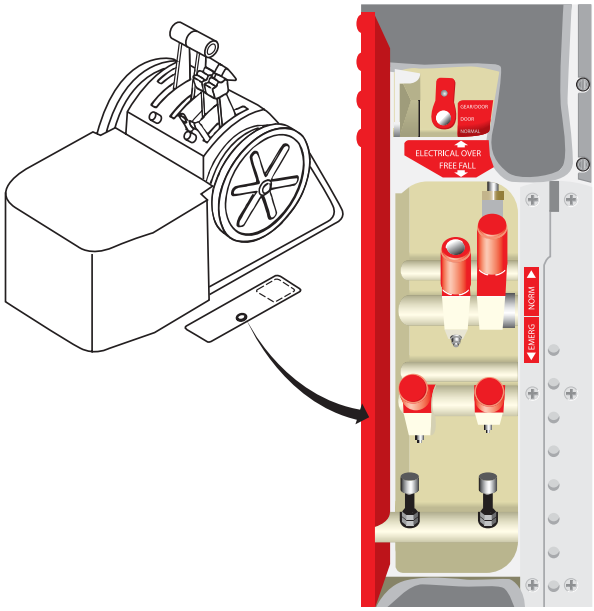


Figure 14-3. Emergency Gear Compartment

Landing Gear Retraction/Extension Circuit

The landing gear retraction/extension system is an electrically controlled, hydraulically actuated system.

Electrical control is provided by the landing gear control module and relay box. Hydraulic power is provided by the green hydraulic system. The landing gear/door hydraulic pressure accumulator is used to actuate the passenger door and to assist the landing gear hydraulic circuit.

The landing gear retraction /extension hydraulic system consists of the following components:

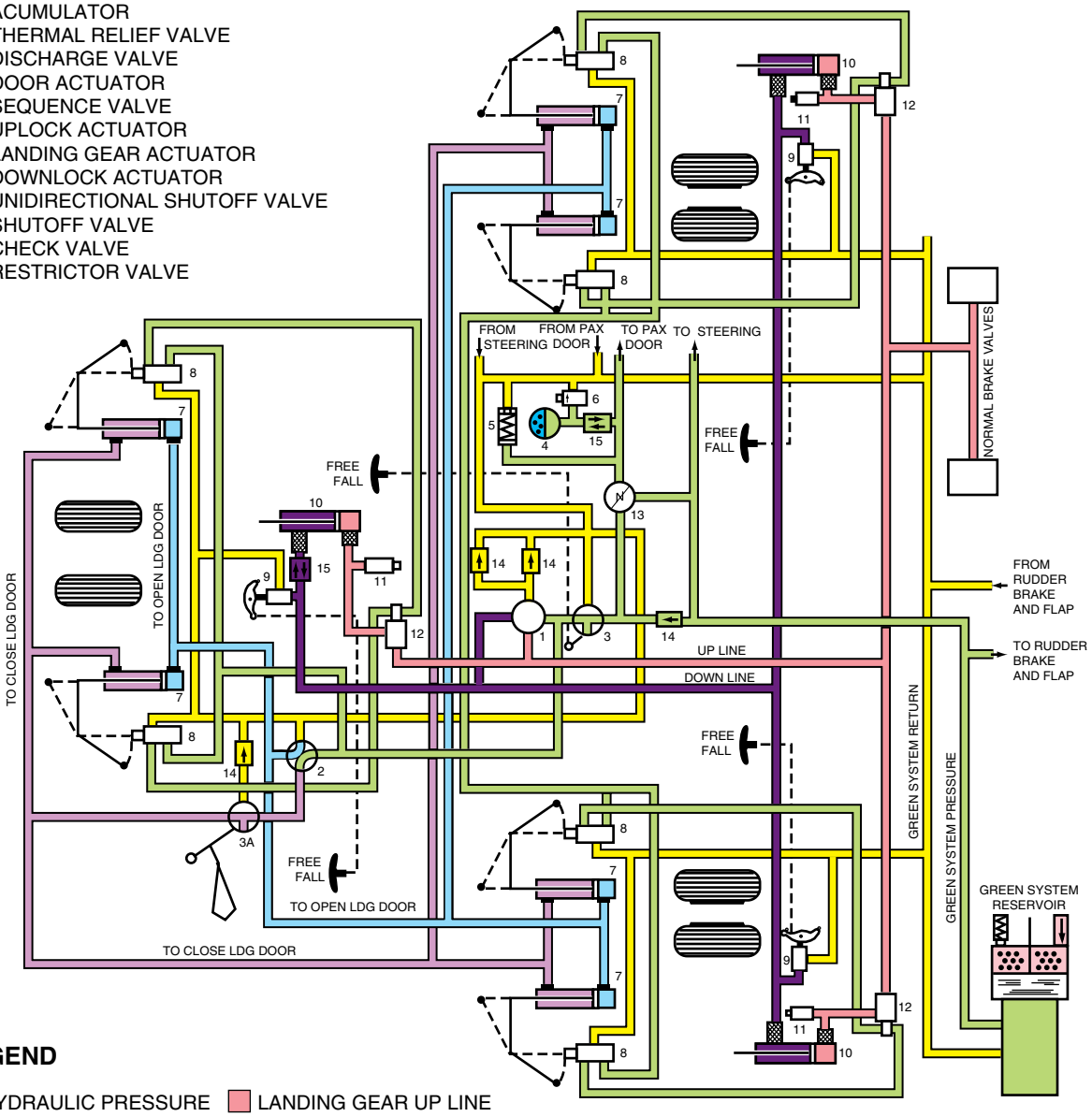
- Check valves
- Emergency selector valve
- Landing gear solenoid valve
- Sequence valves
- Unidirectional shutoff valves
- Downlock and uplock actuators
- Restrictor valve
- Actuating cylinders

COMPONENTS

1. LANDING GEAR SOLENOID VALVE
2. DOORS SOLENOID VALVE
3. EMERGENCY SELECTOR VALVE

3A. MANUAL SELECTOR VALVE

4. ACUMULATOR
5. THERMAL RELIEF VALVE
6. DISCHARGE VALVE
7. DOOR ACTUATOR
8. SEQUENCE VALVE
9. UPLOCK ACTUATOR
10. LANDING GEAR ACTUATOR
11. DOWNLOCK ACTUATOR
12. UNIDIRECTIONAL SHUTOFF VALVE
13. SHUTOFF VALVE
14. CHECK VALVE
15. RESTRICTOR VALVE



LEGEND

- HYDRAULIC PRESSURE
- GENERAL RETURN LINE
- BLEED AIR
- NITROGEN
- LANDING GEAR UP LINE
- LANDING GEAR DOWN LINE
- DOORS OPEN LINE
- DOORS CLOSED LINE

Figure 14-4. Gear and Door Hydraulic Circuit

Check Valves

Two check valves are located in the hydraulic compartment of the aircraft nose section.

One check valve, installed in the emergency selector valve pressure line, prevents the loss of hydraulic pressure in the passenger door accumulator when the green hydraulic system is depressurized.

Two check valves, installed in the landing gear solenoid valve return line, prevents pressure fluctuations from the general return line from unlocking the landing gear uplocks or downlocks.

Emergency Selector Valve

The emergency selector valve is a mechanical, two-position valve (normal and emergency) located in the aircraft nose hydraulic compartment. The valve is actuated from the emergency gear compartment (Figure 14-3). Actuation occurs as part of the emergency free-fall landing gear system operation.

When moved to the emergency position, the valve shuts off the green hydraulic system pressure line. Simultaneously, it connects the wheel well door closing line to the general return line, allowing the weight of the landing gear to open the wheel well doors.

Landing Gear Solenoid Valve

The landing gear solenoid valve is also located in the aircraft nose hydraulic compartment. The valve is operated by the landing gear control lever.

When energized, this valve directs hydraulic pressure to retract or extend the landing gear as selected. After the landing gear has locked up or down, the valve is deenergized. This shuts off the pressure line and simultaneously connects the retraction and extension lines to the general return line.

Sequence Valves

Two sequence valves are located in each wheel well. Operating in conjunction with the shutoff valve, their purpose is to prevent the landing gear from retracting until the wheel well doors are fully open. The sequence valves are mechanically actuated by the wheel well doors.

Unidirectional Shutoff Valves

A unidirectional shutoff valve is located in each landing gear wheel well. The valve blocks hydraulic pressure from the solenoid valve to the downlock actuator and to the retraction chamber of the landing gear actuator cylinder until the gear doors are fully open.

Downlock and Uplock Actuators

A downlock and uplock actuator is attached to each landing gear drag and locking strut. They are hydraulically actuated to unlock the landing gear, and mechanically actuated (by a spring) to lock the gear.

The uplock actuator is attached to the uplock hook that holds the landing gear in the retracted position. The uplock hooks are normally opened hydraulically by the actuator but also may be opened mechanically when using the emergency free-fall system.

Restrictor Valve

A restrictor valve is installed in the nose landing gear extension hydraulic line. Its purpose is to slow nose gear retraction by restricting hydraulic fluid flow. The valve has no effect on the hydraulic fluid flow during landing gear extension.

Actuating Cylinders

An actuating cylinder, one for each landing gear, is located between the main drag and locking strut and the oleo strut. These cylinders are double-action actuators that extend and retract the gear. The main and nose landing gear actuating cylinders operate in the same manner but have a different physical appearance.

Forward Wheel Well Doors Opening/Closing Circuit

The landing gear wheel well doors hydraulic circuit is used to open and close the forward wheel well doors on all three landing gear. The circuit is powered by the green hydraulic system. When using the emergency free-fall system with a hydraulic system failure, the weight of the falling landing gear opens the doors.

The landing gear doors hydraulic circuit consists of the following components:

- Wheel well doors solenoid valve
- Manual selector valve
- Check valve
- Door actuating cylinders

Wheel Well Doors Solenoid Valve

The wheel well doors solenoid valve is a four-way, two-position valve. When energized, it directs hydraulic pressure to the opening side of the door actuators and connects the closing side to general return. When deenergized, the valve directs hydraulic pressure to the closing side of the actuators and connects the opening side to return.

Manual Selector Valve

The manual selector valve, located in the aircraft nose hydraulic compartment, is a maintenance valve used to prevent the landing gear doors from closing during maintenance activities. It is equipped with an interference handle that prevents the hydraulic compartment door from being closed if the valve is left in the MAINT position.

Check Valve

One check valve is located in the wheel well doors hydraulic circuit. Installed in the manual selector valve return line, it prevents the hydraulic fluid in the general return line from closing the gear doors.

Door Actuating Cylinders

The actuating cylinders, one for each wheel well door, are located in the landing gear wheel wells. These cylinders are double-action actuators that open and close the wheel well doors.

Alternate Extension System

The alternate extension system is an electrical override circuit used to extend the landing gear with an electrical failure of the control module or relay box. It provides a secondary means of electrically activating the landing gear hydraulic circuit. The electrical override switch is located in the emergency gear compartment (Figure 14-5).

Emergency Extension System

The emergency free-fall extension system is for use with a failure of both the electrical control circuit and the hydraulic actuating system.

The mechanical controls for this system are in the emergency gear compartment with the electrical override switch (Figure 14-5). The extension lever used to operate the mechanical controls is attached to the copilot's lateral console (Figure 14-5).

There are four mechanical actuating assemblies in this system. One assembly is connected to the emergency selector valve, the other three are connected to their respective landing gear uplocks.

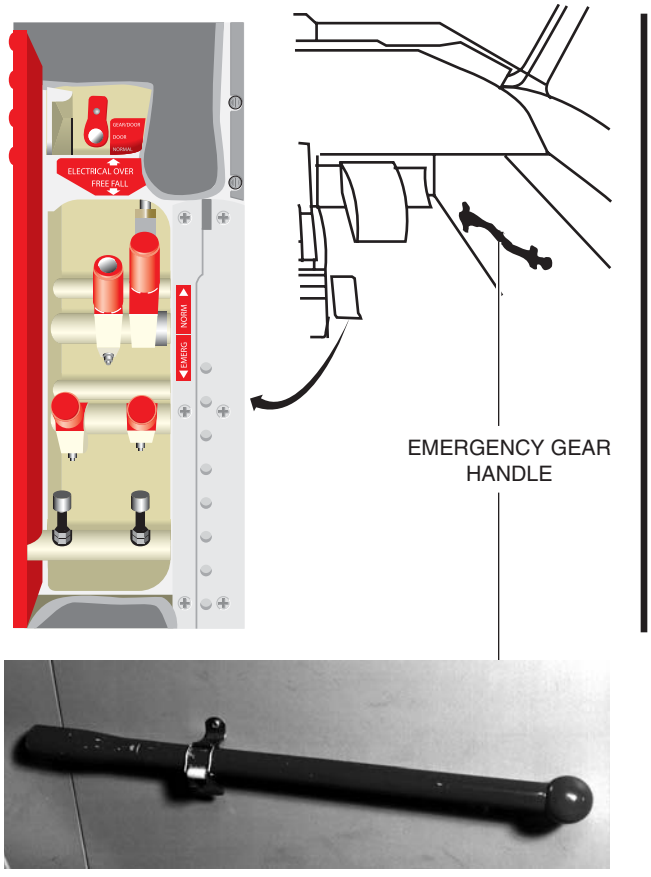


Figure 14-5. Alternate Emergency Extension Controls

LANDING GEAR ACTUATION

Landing gear actuation is achieved by one of three methods:

- Normal retraction and extension
- Alternate electrical override extension
- Emergency free-fall extension

Figure 14-4 may be used to follow gear actuation discussed in this section.

Retraction Sequence

There is no alternate or emergency method to retract the landing gear.

Prior to initiating the normal retraction sequence, the configuration is as follows:

- The landing gear is down and locked.
- The forward wheel well doors on each landing gear are closed.

NOTE

Hydraulic pressure on the closed side of the door actuators keeps the wheel well doors closed. Through the sequence valves, the shutoff valves are also kept closed, blocking pressure to the gear actuator.

After takeoff, when UP is selected on the landing gear lever, the electrical control circuit energizes two solenoid valves. The wheel well door's solenoid valve energizes open, and the landing gear solenoid valve energizes to retract.

The wheel well doors begin to open. (Hydraulic pressure is still blocked from the landing gear actuating cylinders by the closed shutoff valves.)

When the wheel well doors are completely open, they mechanically actuate the sequence valves. The hydraulic pressure that has kept the shutoff valves closed is routed to return, allowing the shutoff valves to open.

Hydraulic pressure, through the shutoff valves, releases the downlocks and pressurizes the up side of the landing gear actuating cylinders, retracting the landing gear. (A restrictor valve, installed in the nose landing gear extension line, controls the rate at which the nose landing gear is retracted to prevent damage to the aircraft when it reaches full up.)

When the landing gear is up and locked, proximity sensors signal the electrical control circuit to deenergize the two solenoid valves. The landing gear solenoid valve deenergizes off, and the wheel well doors solenoid valve deenergizes to close.

Hydraulic pressure on the landing gear actuating cylinders is routed to return.

Hydraulic pressure, through the wheel well doors solenoid, closes the wheel well doors completing the retraction sequence.

Extension Sequence

Any extension sequence starts with the following configuration:

- The landing gear is up and locked.
- The wheel well doors are closed.

NOTE

Hydraulic pressure keeps the forward wheel well doors closed. Through the sequence valves, the shutoff valves are also kept closed, blocking fluid from the gear actuator to return.

Normal Extension

Moving the landing gear lever to the DOWN position signals the electrical control circuit to energize the two solenoid valves. The wheel well door's solenoid valve energizes to open, and the landing gear solenoid valve energizes to extend.

The wheel well doors begin to open. (Hydraulic fluid from the landing gear actuating cylinders to return is still blocked by the closed shutoff valves.)

When the wheel well doors are completely open, they mechanically actuate the sequence valves. The hydraulic pressure that has kept the shutoff valves closed is routed to return, allowing the shutoff valves to open.

Hydraulic pressure, through the shutoff valves, releases the uplocks and pressurizes the down side of the landing gear actuating cylinders, extending the landing gear. (The restrictor valve in the nose landing gear extension line has no effect on nose gear extension).

When the landing gear is down and locked, proximity sensors signal the electrical control circuit to deenergize the two solenoid valves. The landing gear solenoid valve deenergizes off, and the wheel well doors solenoid valve deenergizes closed.

Hydraulic pressure on the landing gear actuating cylinders is routed to return.

Hydraulic pressure, through the wheel well doors solenoid, closes the wheel well doors, completing the extension sequence.

Alternate Electrical Override Extension

The alternate electrical override system is used to electrically activate the landing gear hydraulic circuit when the landing gear control module or relay box has failed.

The electrical override switch (Figure 14-6) is a three-position switch labeled NORMAL–DOORS–GEAR/DOORS.

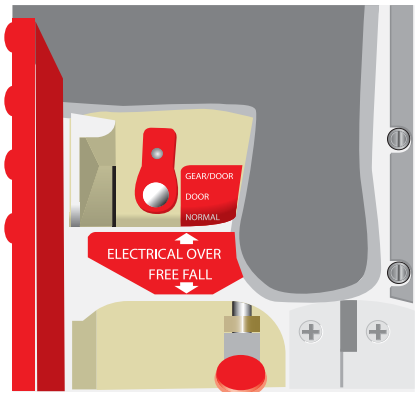


Figure 14-6. Electrical Override Switch

- NORMAL—Permits normal operation of the landing gear control circuits
- DOORS—Energizes the wheel well door's solenoid valve open, pressurizing the "open" side of the door actuators
- GEAR/DOORS—Energizes the wheel well door's solenoid valve open, and the landing gear solenoid valve to extend
  - Doors remain open.
  - Landing gear extends hydraulically.

In the DOORS and GEAR/DOORS positions, the switch electrically bypasses the landing gear control module and relay box, directly actuating the landing gear and wheel well doors solenoid valves.

Following this procedure, if the switch is returned to the NORMAL position, the forward wheel well doors will close.

NOTE

Abnormal checklist procedures are accomplished with the normal gear handle in the down position.

Emergency Free-Fall Extension

The emergency free-fall extension system is used if both the electrical control circuit and the hydraulic actuating system fail.

The mechanical controls for this system are in the emergency gear compartment. The free-fall extension lever, attached to the copilot's lateral console, is used to operate the mechanical controls.

There are four mechanical actuating assemblies in this system (Figure 14-7). One assembly is mechanically connected to the emergency selector valve. The other three assemblies are mechanically connected to their respective landing gear uplock assemblies.

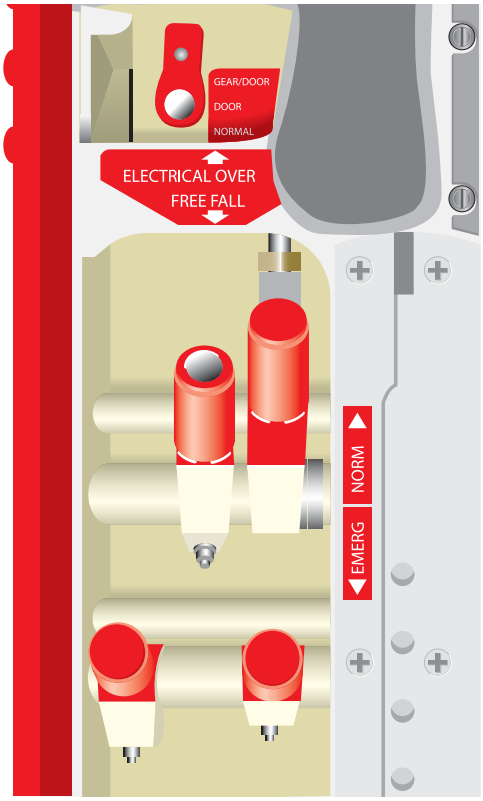


Figure 14-7. Emergency Extension System Controls

When the emergency selector valve is moved (with the handle) to the emergency position, landing gear hydraulic pressure is connected to return. This effectively depressurizes the entire landing gear system.

The handle is then used to mechanically unlock the uplock assemblies for each landing gear. The weight of the falling gear opens the forward wheel well doors and the airstream helps carry the gear aft to the down and locked position.

The forward doors remain open during the remainder of the flight.

NOTE

Abnormal checklist procedures are accomplished with the gear handle up until all gear down and locked, then place handle down.

LANDING GEAR INDICATING SYSTEM

The landing gear position indicating system (Figure 14-8) enables the flight crew to monitor the landing gear position. There is no indication of door position.

The system consists of two fully independent electrical circuits: A and B. These identical circuits operate simultaneously and are powered by emergency DC buses 1 and 2. They are protected by two ampere circuit breakers labeled INDICATION A and INDICATION B.

Each circuit consists of proximity sensors, indicating lights and landing gear control module circuitry.

A lamp test is provided for each circuit. This test only checks the condition of the lamps. There is a push-button for each indicating circuit, labeled SYS A and SYS B. When pressed, all of the indicating lights for that circuit illuminate regardless of landing gear position. These lights do not illuminate during the alarm lights test sequence and must be tested separately.

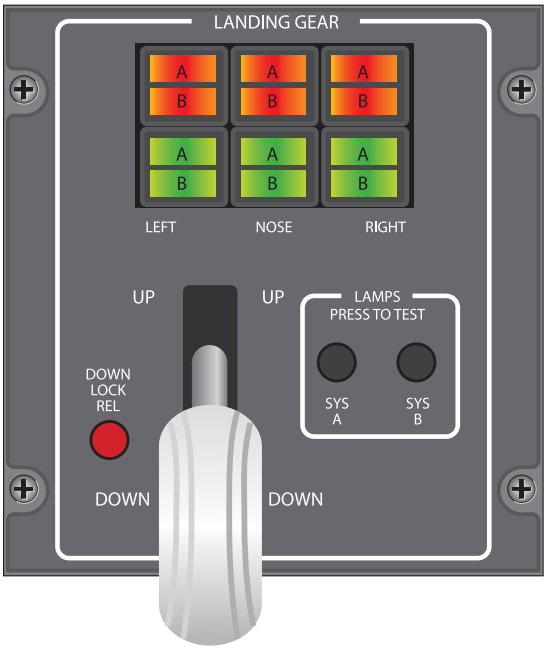


Figure 14-8. Landing Gear Indicating System

Retraction Sequence Indication

The indication sequence for landing gear retraction (Figure 14-9) begins with the gear down and locked, the landing gear lever in the DOWN position, and the green indicator lights illuminated.

When the landing gear lever is moved to the UP position, the red indicating lights illuminate immediately, indicating a disagreement between the landing gear lever and the landing gear position. Because the landing gear does not unlock until the forward wheel well doors are fully open, the green indicating lights remain illuminated momentarily.

After the forward wheel well doors are fully open and the landing gear unlocks from the DOWN position, the green indicator lights extinguish. The red disagreement lights remain illuminated.

When the landing gear reaches the up and locked position, the red indicating lights extinguish. With the gear up and locked, all of the gear position indicating lights are out.

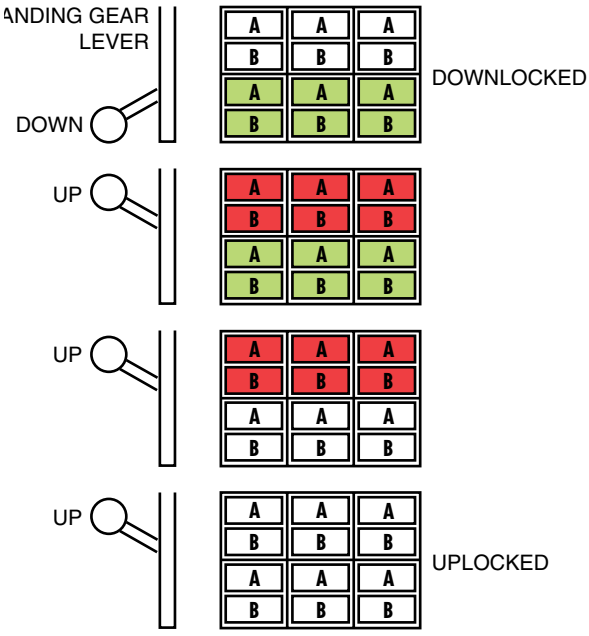


Figure 14-9. Retraction Sequence Indications

Extension Sequence Indication

The indication sequence for landing gear extension (Figure 14-10) begins with the gear up and locked, the landing gear lever in the UP position, and all indicating lights out.

When the landing gear lever is moved to the DOWN position the red indication lights illuminate immediately indicating a disagreement between the landing lever and the landing gear position.

When the landing gear reaches the down and locked position, the green indicator lights illuminate, and simultaneously the red indicator lights extinguish.

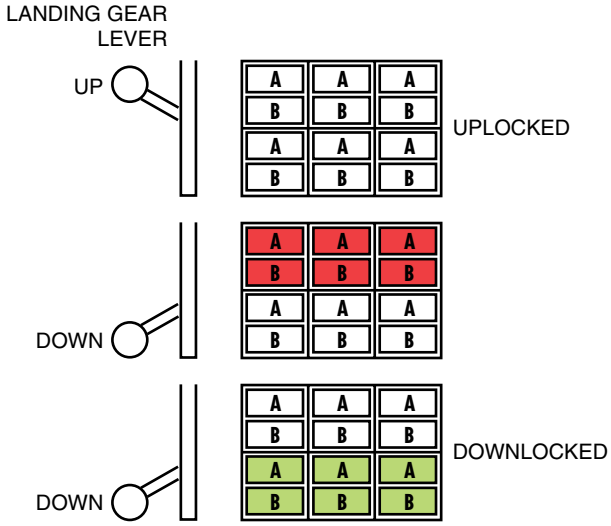


Figure 14-10. Extension Sequence Indications

LANDING GEAR WARNING SYSTEM

The purpose of the landing gear warning system is to prevent gear up landings. Regulations require that the system must present visual and aural warnings if the landing gear is not down and locked, and:

- Power is reduced below the approach position and/or,
- Flaps are selected beyond the approach position.

The landing gear warning system consists of the following indications:

- A three-chime aural alert sounds
- The voice warning "Landing Gear" is given
- The red GEAR light on the MAP illuminates
- The master WARNING lights flash

Components

The warning system receives its inputs from the following components:

- Downlock proximity sensors
- Flap selector microswitch
- Power lever microswitches
- Radio altimeter(s)

Downlock Proximity Sensors

The downlock proximity sensors send signals to the landing gear warning system if any of the three landing gear are not down and locked.

Flap Selector Microswitch

A microswitch is installed at 17° in the flap selector lever to indicate flap selection beyond the 15° approach position.

Power Lever Microswitches

Since the power lever position varies depending upon EEC status, two microswitches are installed in each power lever. One is set at 46° power lever angle (PLA) for EEC ON approaches, and the other is set at 66° PLA for EEC OFF approaches.

Radio Altimeter(s)

There are two additional features of the warning system for aircraft equipped with radio altimeters:

- Power lever induced warnings are inhibited above 1,500 feet AGL.
- A warning is triggered at 200 feet AGL if the gear is not down and locked, regardless of power lever or flap position.

The radio altimeter functions arm at 200 feet AGL on the departure.

Operation

Warnings are provided when any gear is not down and locked, and any of the following conditions exist:

- At 200 feet AGL regardless of flap or power lever position: This warning may not be canceled.
- When either or both power levers are reduced below 46° (EEC ON) or 66° (EEC OFF): The aural and master warnings may be canceled, but the GEAR light on the MAP remains on.
- When the flaps are selected beyond 17° regardless of actual flap position: This warning is unconditional, cannot be canceled, and is not inhibited by radio altitude.

On aircraft equipped with two radio altimeters, failure of one does not affect system function. Failure of both radio altimeters (or the single radio altimeter on aircraft equipped with only one), disables both radio altimeter warning features. All other landing gear warning system features remain operational.

Most Minimum Equipment Lists (MELs) allow dispatch with one or both radio altimeters inoperative.

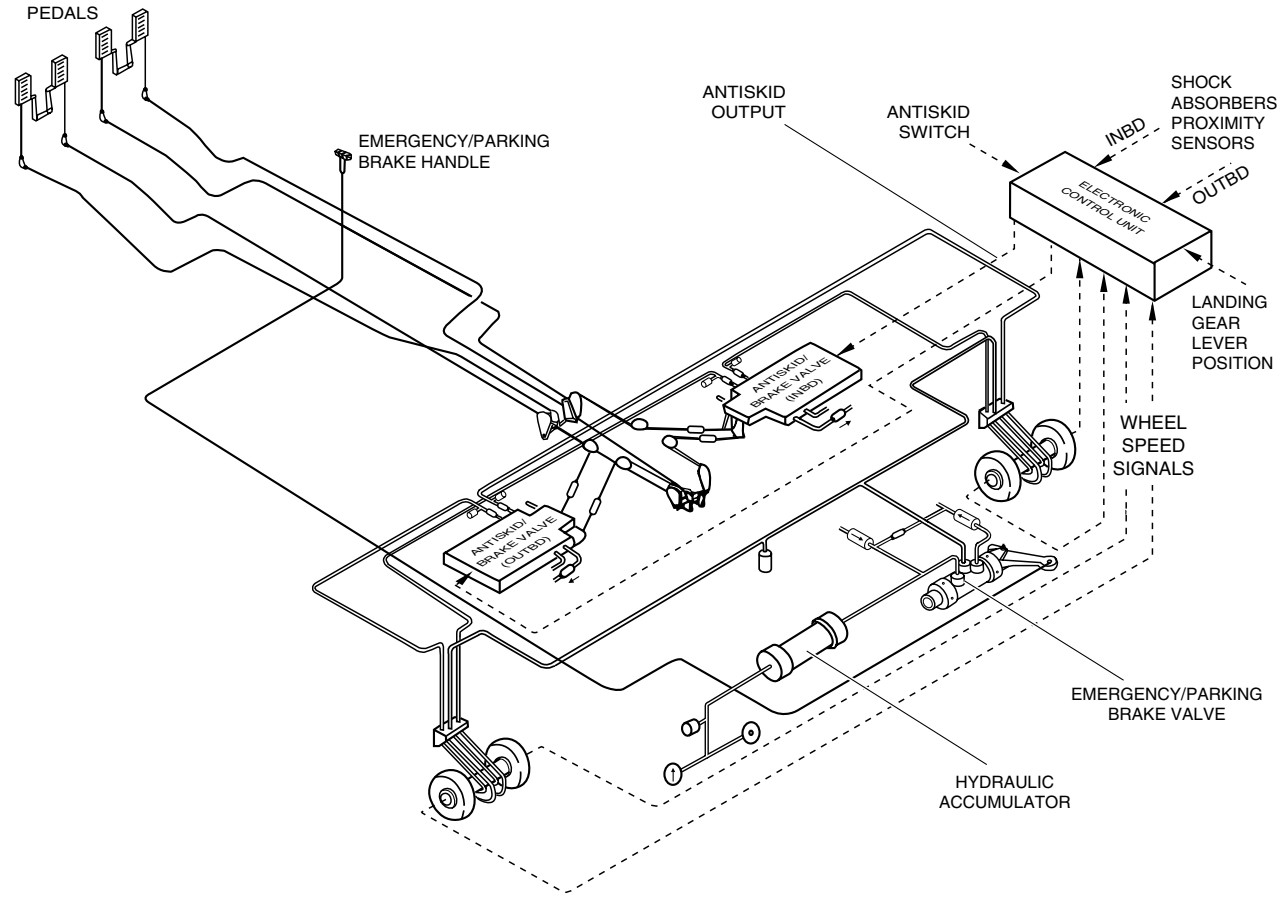


Figure 14-11. Brake System

BRAKES

The aircraft brake system (Figure 14-11) consists of the following:

- Normal brake system
- Antiskid system
- Emergency/parking brake system

NORMAL BRAKE SYSTEM

General

The normal brake system is used to decelerate the aircraft during ground operations.

After takeoff, hydraulic pressure from the landing gear retraction line is applied to the brake valves to stop main wheel rotation prior to the up and locked position. Nosewheel rotation is stopped by two snubbers on the forward wall of the nose gear wheel well.

The normal brake system is hydraulically actuated and mechanically controlled through the pilot's and copilot's rudder pedal toe brakes. The normal brake system hydraulic circuit consists of two totally independent subcircuits:

- Outboard wheels subcircuit (green hydraulic system power)
- The inboard wheels subcircuit (blue hydraulic system power)

Components

The normal brake system consists of the following components:

- Toe brakes
- Antiskid/brake valves
- Hydraulic fuses
- Pressure switches
- Shuttle valves
- Brake assemblies

Toe Brakes

The rudder pedal toe brakes, through cables and bellcranks, actuate the antiskid/brake valves. When a toe brake is depressed, both inboard and outboard antiskid/brake valves direct hydraulic fluid to the corresponding wheels brake assemblies.

Antiskid/Brake Valves

There are two antiskid/brake valves; one for the outboard brakes on each wheel, and the other for the inboard brakes.

The antiskid/brake valve is a double-body valve with each section controlling one brake assembly. A servo valve is attached to provide antiskid capability.

The valves also reduce the 3,000 psi hydraulic pressure input to a maximum of 1,200 psi for brake system operation.

Hydraulic Fuses

There are four hydraulic fuses, one for each brake line, installed next to the antiskid/brake valves. The fuses stop the flow of fluid to the brakes in the event of a leak downstream of the fuse.

Pressure Switches

A pressure switch is installed in each brake line downstream of the hydraulic fuse. The switches are connected to the INBOARD and OUTBOARD lights on the brake control panel (Figure 14-12). The pressure switches activate their respective light when the LINE PRESSURE TEST button is pressed and the line pressure is at least 300 psi.

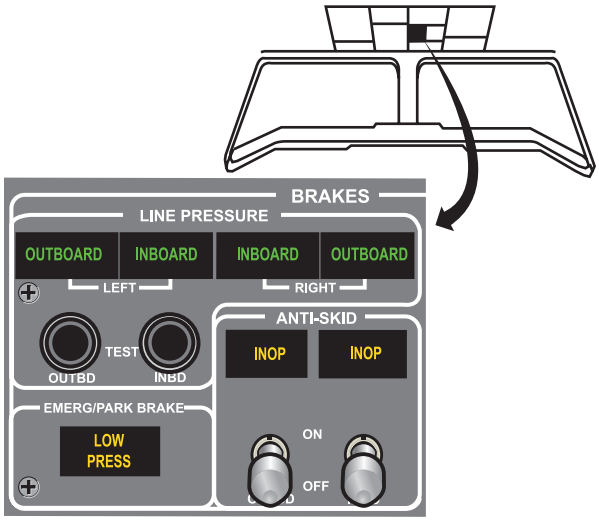


Figure 14-12. Brake Control Panel—Pressure Switch Indication

Shuttle Valves

A shuttle valve on each brake assembly selects either the normal brake system or emergency/parking brake system to provide hydraulic pressure to the brake assembly. The system with the highest pressure is directed to the brake.

Brake Assemblies

Each brake assembly consists of a series of fixed and rotating discs. There is one brake assembly for each of the four main landing gearwheels.

Small slave cylinders are mounted around the circumference of each assembly. When hydraulic pressure is applied to the cylinders, the plates are squeezed together to inhibit wheel rotation. Braking action is proportional to the pressure applied.

A brake wear indicator for the pressure/wear plate assembly is mounted on each of the brake assemblies. The indicator pin protrudes from the piston housing approximately 3/8 inches when the brake assembly is new. If the pin is flush with the housing, maintenance should be contacted.

Brake Control Panel

The brake control panel (Figure 14-12) is located on the cockpit overhead panel. It contains the following controls and indicators for the aircraft brake systems:

- Line pressure TEST pushbuttons
- Line pressure INBOARD/OUTBOARD lights
- Antiskid control switches
- Antiskid INOP lights
- Emergency/parking brake LOW PRESS light

Line Pressure TEST Pushbuttons

The line pressure LEFT and RIGHT TEST pushbuttons are labeled OUTBD and INBD (Figure 14-13). To perform the line pressure test, press and hold both test buttons while depressing both toe brakes. The INBOARD/OUTBOARD lights illuminate if pressure in their respective line is at least 300 psi. The test may be conducted in flight provided the gear handle is up.

Line Pressure INBOARD/ OUTBOARD Lights

The green INBOARD and OUTBOARD indicator lights represent each main landing gear inboard and outboard brake line pressure. These lights receive a signal from the pressure switch installed in the respective brake line only when the line pressure test pushbuttons are pressed and brake pressure is above 300 psi.

Antiskid Control Switches

Two antiskid ON/OFF control switches, labeled OUTBD and INBD, are located in the antiskid section of the brake control panel. Each switch controls a pair (outboard or inboard) of wheel brakes.

Antiskid INOP Lights

The amber antiskid INOP lights, above each antiskid control switch, illuminate when:

- The switch is OFF.
- The respective antiskid system is deenergized.
- The control switch is ON and the antiskid system is malfunctioning.

The INOP lights illuminate only with the gear lever down.

Emergency/Parking Brake LOW PRESS Light

The amber LOW PRESS light on the brake control panel, activated by a switch in the emergency/parking brake accumulator, illuminates when the pressure in the nitrogen side of the accumulator is less than 1,500 psi.

If the accumulator precharge is below normal or the hydraulic pressure stored in the accumulator is depleted, emergency and parking brake operation is doubtful. It may be possible to restore hydraulic pressure in the accumulator by turning on the blue system electric hydraulic pump.

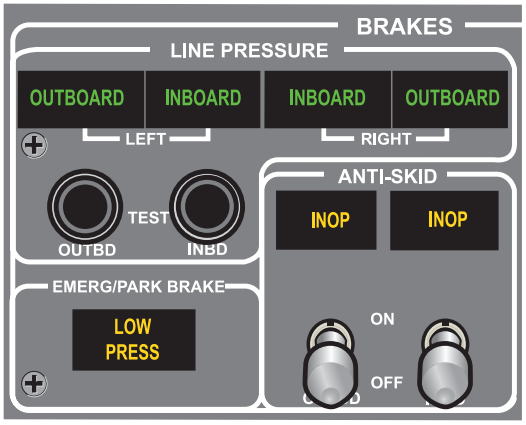


Figure 14-13. Brake Control Panel—TEST Pushbuttons

Indications and Warnings

Illumination of either amber INOP light or the amber LOW PRESS light triggers the following warning system indications:

- A single-chime aural alert sounds.
- The amber BRAKES light on the MAP flashes.
- The master CAUTION lights flash.

When ALARM CANCEL is pressed, the CAUTION lights extinguish and the BRAKES light stops flashing but remains on as long as the INOP or LOW PRESS light is illuminated.

Normal Brakes Operation

Normal brake operation begins with a crew member applying pressure to the toe brakes. This motion is transmitted to the antiskid/brake valves by cables and bellcranks (Figure 14-14).

The antiskid/brake valves control the hydraulic pressure, proportional to brake deflection, transmitted to the brake assemblies. The maximum pressure transmitted is 1,200 psi. If a single brake pedal is depressed, both inboard and outboard brake

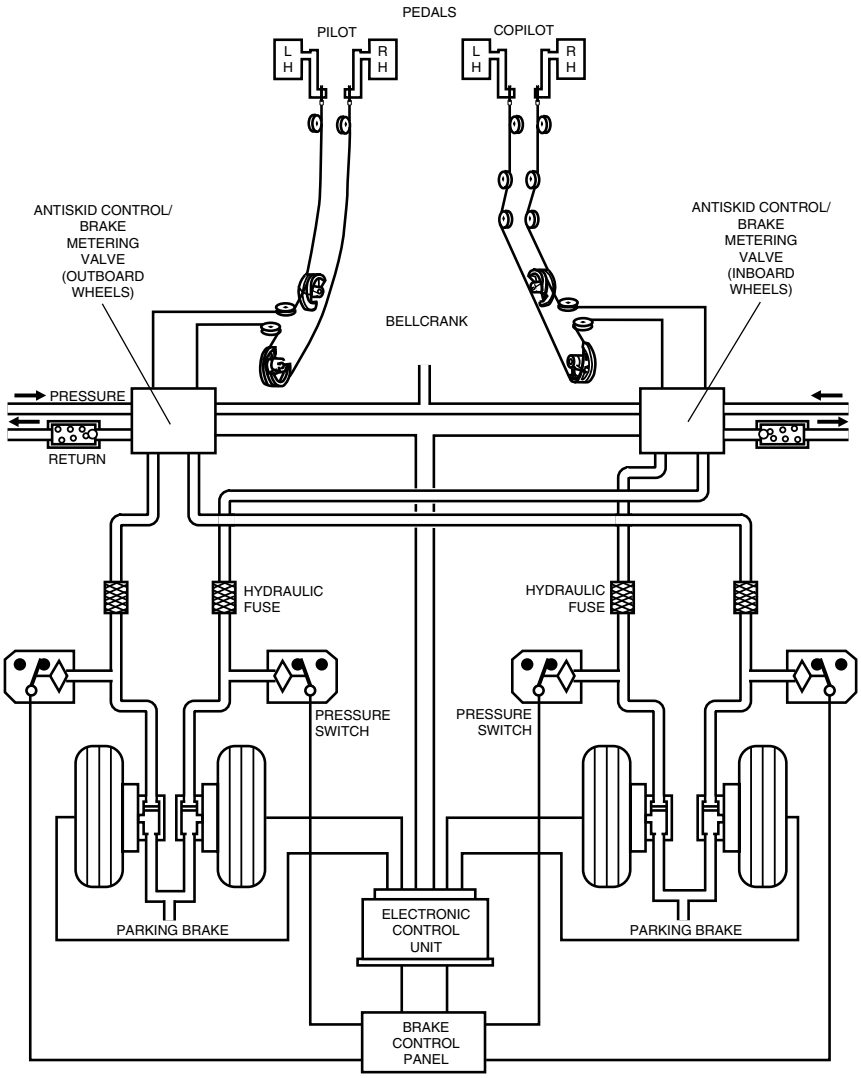


Figure 14-14. Normal Brake System

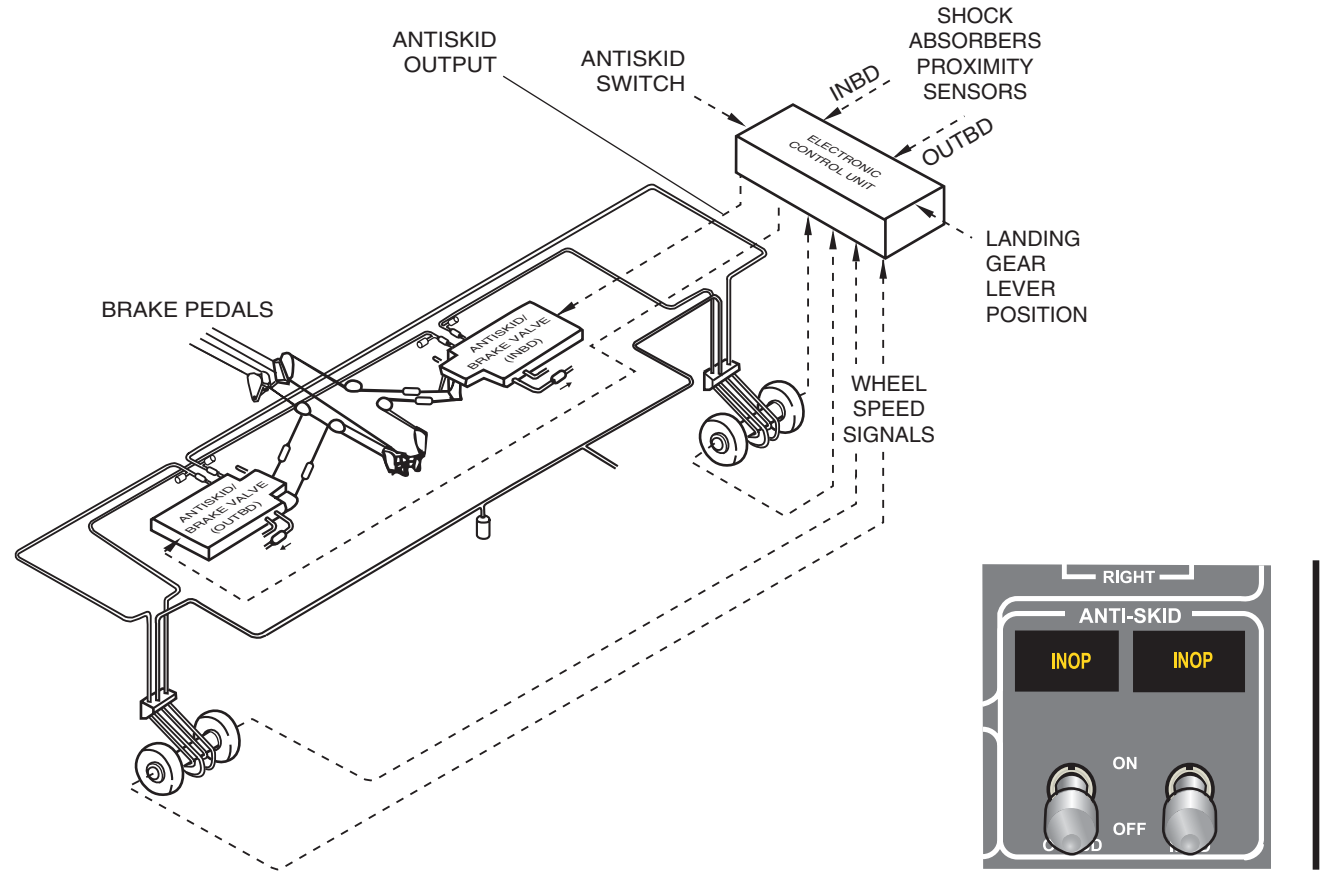


Figure 14-15. Antiskid System

assemblies on that main landing gear are actuated. This permits the use of differential braking for limited directional control.

If an excessive leak occurs between the hydraulic fuse and the brake assembly, the flow of hydraulic fluid is stopped by the fuse. This ensures operation of the other brake assembly controlled by the same antiskid/brake valve.

After takeoff, when the landing gear lever is moved to UP, hydraulic fluid from the landing gear retraction line acts on the antiskid/brake valve applying the brakes. This stops the main landing gearwheels spinning prior to the up and locked position and prevents damage to the aircraft.

ANTISKID SYSTEM

General

The aircraft is equipped with an antiskid system (Figure 14-15) to obtain maximum efficiency from the braking system. Maximum braking efficiency is obtained just prior to a skid.

The antiskid system senses and analyzes wheel rotation speed and electronically controls the output of each antiskid/brake valve.

The antiskid system operates only when the antiskid OUTBD and INBD switches are ON and the landing gear lever is down.

Components

The antiskid system consists of the following components:

- Wheel speed transducers
- Servo valves
- Electronic control unit

Wheel Speed Transducers

The wheel speed transducers are the wheel speed sensing devices. They provide an AC voltage to the electronic control unit proportional to the rotational speed of each wheel. There are four wheel speed transducers, one installed in each main landing gear wheel hub.

Servo Valves

A servo valve, in each antiskid/brake valve, controls the antiskid/brake valve when the antiskid system is energized.

When the antiskid system is off, the servo valve allows hydraulic pressure (proportional to brake pedal displacement) to pass unaffected to the brake assembly. With the antiskid system on and a wheel in a skid, the servo valve reduces the hydraulic pressure to the brake assembly as a function of the signals from the electronic control unit.

Electronic Control Unit

The electronic control unit controls the operation of the antiskid system. It consists of two separate circuits, one to control the outboard antiskid servo valve and the other to control the inboard. The outboard control circuit is powered by DC bus 1 and the inboard is powered by DC bus 2.

The control unit receives the signals from the transducers and compares them to a self-generated signal. If a wheel is in a skid, the control unit sends a correction signal to the corresponding servo valve, which reduces the hydraulic pressure to that brake assembly. When the wheel is back up to speed, the correction signal is eliminated and the brake pressure is now, again, proportional to the input from the brake pedals.

The electronic control unit also has a self-test capability. This self-test feature illuminates the appropriate INOP light whenever:

- A transducer fails.
- A servo valve fails.
- Pressure to the brakes is reduced for more than 1.5 seconds.

Antiskid System Operation

The anti-skid system limits the hydraulic pressure applied to each brake valve by inputs from the pedals. The anti-skid system provides maximum stopping performance, thus minimizing wheel skidding and, consequently, enabling minimum tire wear under any runway condition. The system is primarily composed of four wheel speed transducers, installed at main landing gear wheel axles, one electronic control box, and two brake valves, one for the outboard pair of wheels and other for the inboard pair.
The system is enabled when the following conditions are met:
– Anti-skid switch set to ON,
– Landing gear lever set to DOWN,
– At least one main landing gear shock absorber compressed (airplane on the ground).

When the anti-skid system is actuating, the skid control circuitry computes the initial wheel speed signal, received from the speed transducers, to determine an airplane reference speed. This reference speed is continuously updated and compared with the average speed from the wheels, to determine whenever the airplane is skidding or not. If a skidding is sensed, the brake valves are commanded to relieve the pressure applied by the pilots through pedals, thus eliminating the skidding. It is important to emphasize that the anti-skid system does not apply pressure on the brakes, but only relieves it when one or more wheel slips.

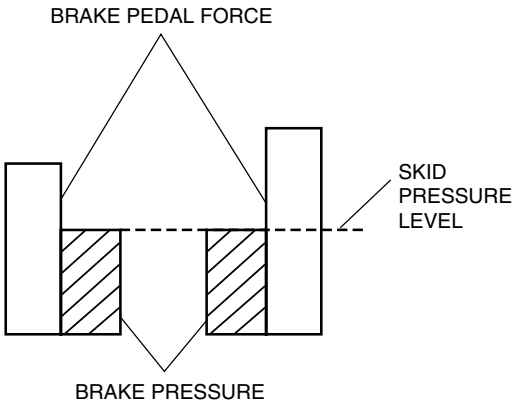
Thus, there are some instances when the conventional differential braking technique may not be applicable. In these cases, when it is necessary to apply differential braking, the pilots may have to reduce the pedal force on the opposite to the turn.

The anti-skid system incorporates the locked wheel protection and touchdown protection features.

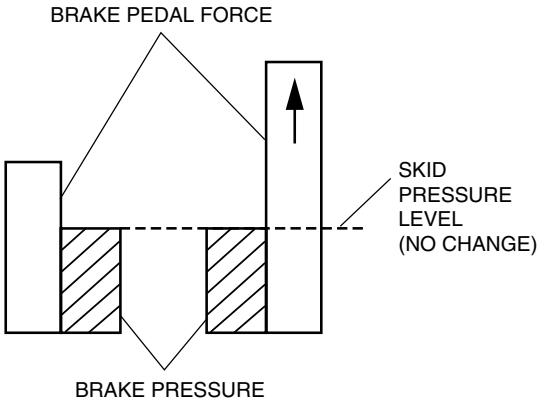
LOCKED WHEEL PROTECTION

For wheel speeds above 22 kt, the anti-skid system activates the locked wheel protection between left and right wheels of the same pair. The anti-skid compares the wheel speed signals from both wheel of inboard (outboard) pair. If the slower wheel speed is less than or equal to 40% of the faster wheel speed, the skid control circuitry sends a corrective signal to the associated brake valve. The brake valve then commands a full brake pressure relief to both

THE ANTISKID SYSTEM MODULATES THE BRAKE PRESSURE TO A LEVEL WHICH PREVENTS THE WHEELS FROM SKIDDING. IF ONE OF THE WHEELS LOCKS, THE ANTISKID SYSTEM REDUCES THE BRAKE PRESSURE OF THE ASSOCIATED PAIR, ELIMINATING THE SKIDDING.



IF THE PILOT MODULATES THE PEDAL FORCE IN AN ATTEMPT TO STEER THE AIRPLANE THROUGH DIFFERENTIAL BRAKING, THE ANTISKID SYSTEM MAINTAINS THE SKID PRESSURE LEVEL, PRECLUDING THE AIRPLANE FROM TURNING.



THE CORRECT ACTION CONSISTS IN A REDUCTION OF THE OPPOSITE PEDAL FORCE TO A LEVEL BELOW THE SKID LEVEL, PERMITTING THE REDUCTION OF THE CORRESPONDING BRAKING PRESSURE.

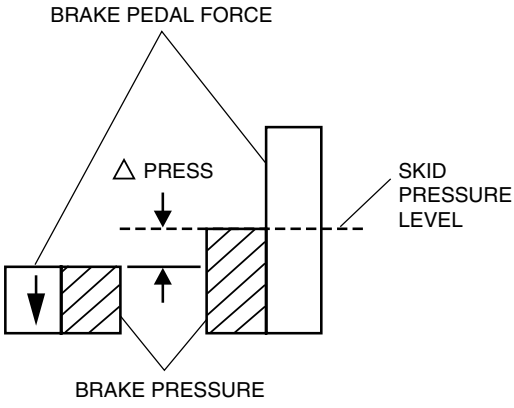


Figure 14-16. Differential Braking Technique Schematic

paired wheels, allowing wheel speed recovery. The 40% tolerance between wheel speeds is provided to permit an amount of differential braking, for steering purposes.

For wheel speeds below 22 kt, the locked wheel protection is deactivated, and anti-skid system actuates normally, but without the wheel speed comparator. For wheel speeds below 6 kt, the anti-skid protection is deactivated, allowing the pilot to lock and pivot on a wheel.

TOUCHDOWN PROTECTION

The touchdown protection is a feature that prevents brake actuation before main wheels spin up, when the airplane is still airborne, during landing. This protection permits the brake actuation only after 3 seconds have elapsed since the latest touchdown or after the wheels have spun-up to 50 kt. To provide the touchdown protection, the anti-skid system receives signals from both main landing gear proximity sensors. In the event one landing gear proximity sensor fails at the air position, the anti-skid system will operate normally. However, if both landing gear proximity sensors fail at the air position, thus generating a false signal to the anti-skid system , braking capacity will be available only for wheel speeds above 50 kt. Below 50 kt, a total loss of brake capacity will occur, and no anti-skid inoperative indication light will illuminate. In this case, turning the anti-skid system off will restore the braking capability, but without anti-skid protection

OPERATIONAL FEATURES

When the landing gear lever is moved to UP, the anti-skid system is deactivated. This permits pressure from the landing gear retraction circuit to be applied to the brake valves to stop wheel rotation. Although the system is deactivated, the INOP lights do not illuminate.

Differential braking is not available when brake pedal force is greater than that required to achieve maximum antiskid braking. A technique for maximum antiskid braking and differential braking is shown in Figure 14-16.

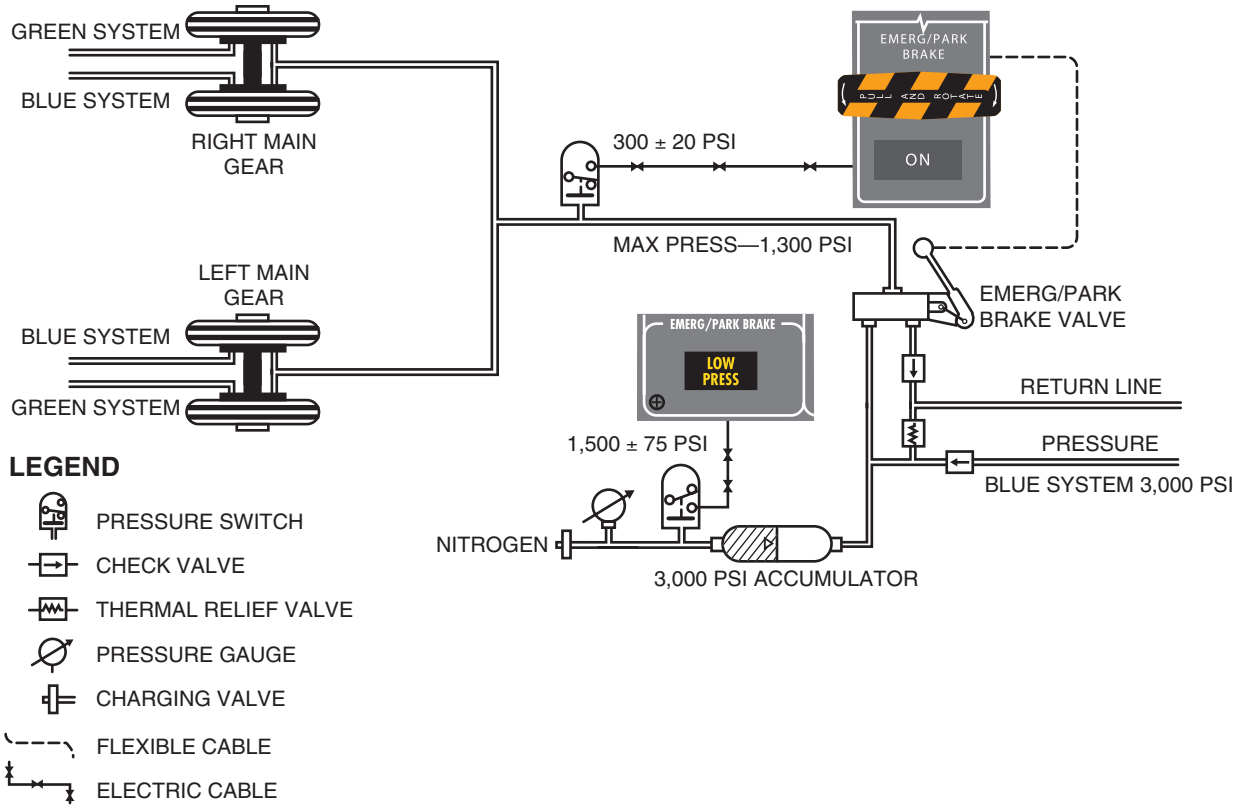


Figure 14-17. Emergency/Parking Brake System

EMERGENCY/PARKING BRAKE SYSTEM

General

The emergency/parking brake system (Figure 14-17) is normally used as a parking brake. It is available as an emergency brake if the normal brake system loses hydraulic pressure. The emergency/parking brake is activated by pulling up on the parking brake handle.

Charged by the blue hydraulic system, the emergency/parking brake system is completely independent of the normal brake system. An accumulator stores hydraulic pressure enabling operation should both hydraulic systems fail.

When actuated, the system provides 1,300 psi hydraulic pressure simultaneously to all main landing gear brake assemblies. No antiskid or differential braking is available with the emergency/parking brake system.

Components

The emergency/parking brake system consists of the following components:

- EMERG/PARK BRAKE control panel
- Emergency/parking brake control valve
- Brake ON light
- Brake accumulator
- LOW PRESS light
- Thermal relief valve
- Check valves

EMERG/PARK BRAKE Control Panel

The emergency/parking brake control panel, labeled EMERG/PARK BRAKE, is located on the aft panel on the pilot's side of the center console. It contains the brake handle and the brake ON light.

Emergency/Parking Brake Control Valve

The emergency/parking brake control valve provides hydraulic pressure to all main landing gear brakes simultaneously. It receives 3,000 psi pressure and reduces it to a maximum of 1,300 psi.

When the handle is pulled, the valve routes hydraulic pressure to the brake assemblies proportional to handle displacement. When released, hydraulic pressure is routed to return.

Brake ON Light

The white parking brake ON light is located beneath the brake handle on the EMERG/PARK BRAKE control panel. It illuminates when hydraulic pressure to the brake assemblies from the emergency/parking brake system exceeds 300 psi.

On some aircraft, a PARKING BRAKE light is located on the right side of the center instrument panel.

Brake Accumulator

The brake accumulator ensures a 3,000 psi pressure source for the brake control valve. Located in the left wing root, the accumulator is internally divided by a floating piston A nitrogen precharge on one side of the piston, and the hydraulic fluid on the other provide the source of emergency brake pressure and fluid should normal hydraulic system pressure be lost. A fully charged accumulator has pressure and fluid sufficient for 10 complete parking brake applications.

LOW PRESS Light

The amber LOW PRESS light on the overhead brake control panel, as discussed previously, is activated by a switch in the emergency/parking brake accumulator. It illuminates when the pressure in the nitrogen side of the accumulator is less than 1,500 psi.

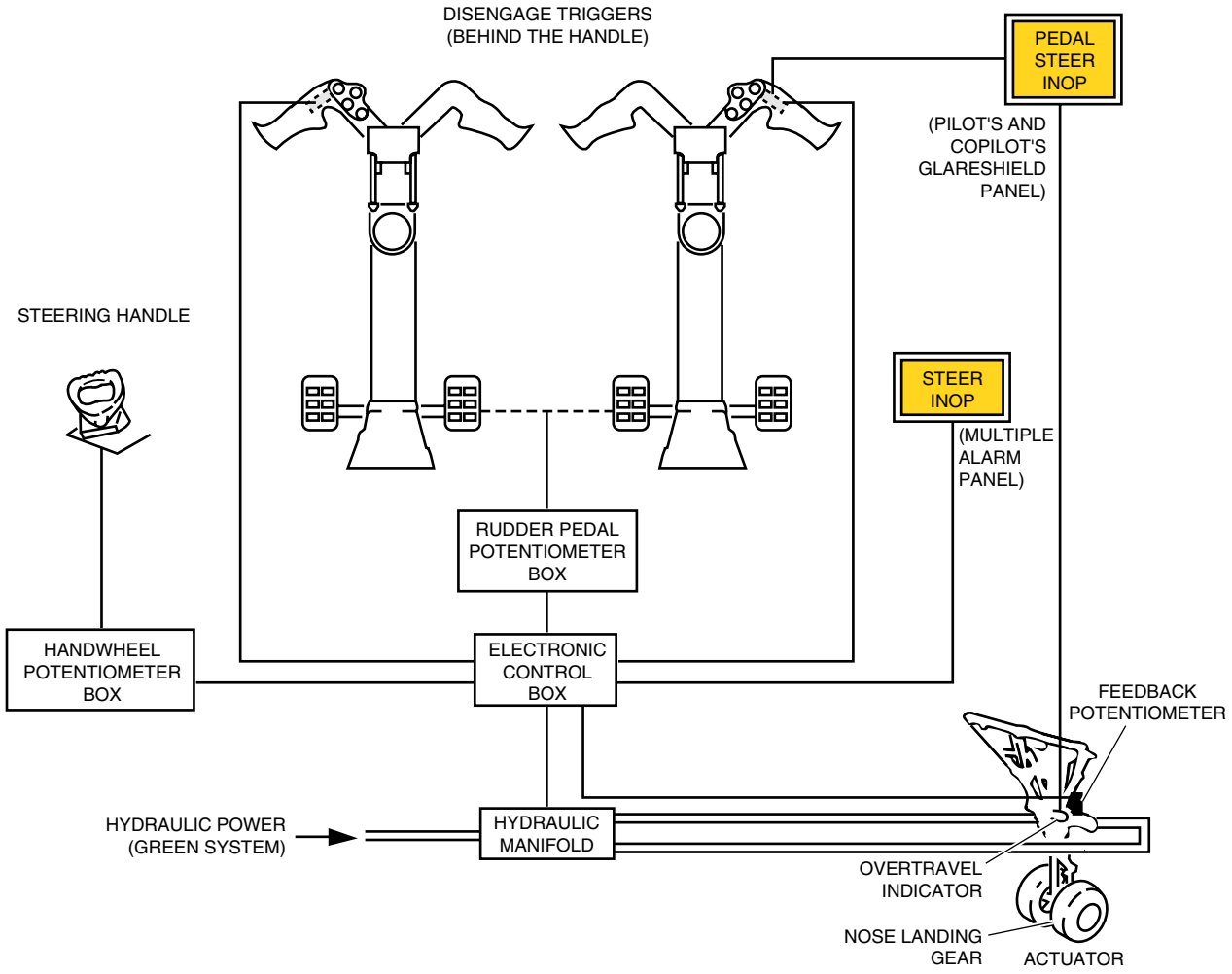


Figure 14-18. Nosewheel Steering System

Thermal Relief Valve

The thermal relief valve protects the emergency/parking brake system from damage as a result of thermal expansion of the hydraulic fluid in the brake assemblies. The valve opens and diverts hydraulic fluid to return when the pressure exceeds 3,600 psi.

Check Valves

Two check valves are installed in the system. One prevents pressure in the general return line from actuating the brakes. The other allows the accumulator to remain pressurized in the event of a blue hydraulic system failure.

Emergency/Parking Brake Operation

The emergency brake is applied by pulling up on the handle. To set the parking brake, the handle is pulled up and rotated 90°.

Blue system hydraulic fluid, under 3,000 psi pressure, flows through the check valve to both the accumulator and the emergency/parking brake control valve. When the handle is pulled, the emergency parking brake control valve opens and hydraulic pressure, proportional to valve actuation, is applied to the four brake assemblies. The maximum pressure to the brakes is 1,300 psi.

When setting the parking brake, be sure that the white ON light illuminates. This indicates there is sufficient brake pressure to prevent aircraft movement when the engines are started.

If the power levers are advanced for takeoff with the emergency/parking brake set, the following warnings are triggered:

- A triple-chime aural alert sounds.
- The voice warning "Takeoff Brakes" is given.
- The master WARNING lights flash.

NOSEWHEEL STEERING

GENERAL

The nosewheel steering system provides a means of directional control during ground operations. The system is electrically controlled and hydraulically actuated (green hydraulic system). The nosewheel steering may be controlled either by a handle on the left console or by the pilot's or copilot's rudder pedals.

The following conditions must be met for the nosewheel steering system to operate:

- The nose landing gear must be down and locked.
- The nose and right landing gear shock absorbers must be compressed.

COMPONENTS

The steering system consists of the following major components and indicators (Figure 14-18):

- Steering handle
- Rudder pedals
- Disengage triggers
- Feedback potentiometer
- Electronic control box
- Hydraulic manifold assembly
- Steering actuating cylinder
- PEDAL STEER INOP lights
- STEER INOP light

Steering Handle

A steering handle (sometimes referred to as a "tiller") is located on the left console. The handle controls the deflection of the nosewheel up to 50° left or right.

Slight downward pressure on the handle engages the nosewheel steering. Once engaged, a proportional signal is sent to the electronic control box when the steering handle is turned. When released, the handle automatically disengages and centers. With the handle disengaged, nosewheel steering is through rudder pedal input.

Rudder Pedals

The rudder pedals are primarily used for nosewheel steering during takeoff and landing. This avoids potential "hard over" nosewheel commands when using the steering handle. The deflection range for rudder pedal steering is ±7°.

Rudder pedal steering is normally available whenever the steering handle is not in use. The pilot's and copilot's rudder pedals are mechanically linked, allowing either crewmember to control the nosewheel steering. Rudder pedal input is transmitted to a potentiometer that provides a signal, proportional to the amount of rudder pedal deflection, to the electronic control box.

Disengage Triggers

A rudder pedal disengage trigger is located on each control wheel. The triggers are used to disconnect the rudder pedal nosewheel steering during control checks, push-backs, and towing when the engines or electric hydraulic pumps are running.

Feedback Potentiometer

A feedback potentiometer provides an electrical signal, indicating the position of the nosewheel, to the electronic control box. The feedback potentiometer is located on the nose landing gear.

Located within the potentiometer box is a position sensor activated by the potentiometer shaft. This sensor provides a signal when the nosewheel angle exceeds ±9°.

Electronic Control Box

The electronic control box receives and processes the signals from the rudder pedal disengage triggers, steering handle, rudder pedals, and feedback potentiometer. The processed signal is then sent to the hydraulic manifold assembly. The electronic control box also senses an electrical failure in any of the system potentiometers.

The electronic control box is powered by emergency DC bus 1, through the STEERING circuit breaker.

Hydraulic Manifold Assembly

A hydraulic manifold assembly controls the hydraulic fluid pressure to the steering actuating cylinder. The manifold is installed in the aircraft nose hydraulic compartment. The hydraulic manifold contains a filter, check valves, a solenoid valve, a servo valve, a bypass valve, and a compensator valve.

Steering Actuating Cylinder

The steering actuating cylinder controls the deflection of the nosewheel. It is located on the nose landing gear and receives hydraulic pressure from the hydraulic manifold.

A red overtravel indicator pin protrudes from the actuating cylinder housing. It provides a visual warning if the nosewheel has been turned more than 90° during towing. If the overtravel indicator is visible, maintenance action is required.

PEDAL STEER INOP Lights

The two amber PEDAL STEER INOP lights are mounted on each side of the glareshield and illuminate when pedal steering capability is inoperative.

The PEDAL STEER INOP lights illuminate when either rudder pedal disengage trigger is pressed or when the nosewheel angle exceeds +9° with the steering handle centered. The steering handle must be pressed to center the nosewheel and turn the PEDAL STEER INOP lights out.

STEER INOP Light

The amber STEER INOP light on the MAP illuminates when the electronic control box senses an electrical failure in the potentiometer circuits or when the control box has lost power.

NOSEWHEEL STEERING OPERATION

Nosewheel steering is accomplished by using either the pilot's or copilot's rudder pedals or the pilot's steering handle.

When a rudder pedal is depressed, the rudder pedal potentiometer sends a proportional signal to the electronic control box. This signal is compared to the nosewheel position signal from the feedback potentiometer. The resultant signal is transmitted to the solenoid and servo valves in the hydraulic manifold releasing hydraulic fluid pressure to the steering actuator.

The electronic control box maintains the nosewheel in the selected position as long as the rudder pedals are not moved.

The maximum deflection of the nosewheel obtained by the use of the rudder pedals is ±7°.

If the nosewheel should deflect beyond ±9° with the steering handle in the center position, the position sensor in the feedback potentiometer box disengages the system and the PEDAL STEER INOP light illuminates. To reset the system, depress the steering handle with the nosewheel angle less than ±9°.

The rudder pedal disengage trigger is used to momentarily interrupt rudder pedal steering. This permits steering with differential braking or rudder only, and illuminates the PEDAL STEER INOP lights. Release the trigger to reengage rudder pedal steering.

When depressed, the steering handle activates the steering handle reset switch. This activates a nonlinear potentiometer providing a signal to the electronic control unit proportional to the rotation of the handle. From this point, the system operates the same as if it were being controlled by the rudder pedals—except the nosewheel may be controlled through a ±50° arc.

Because of the nonlinear potentiometer, the steering handle has reduced sensitivity in the central region. This feature, more commonly known as variable ratio steering, provides for smoother steering.

The steering angle may be increased to ±57° by the simultaneous use of the rudder pedals and steering handle.

Whenever the steering system disengages, either automatically (electrical failure or nosewheel exceeds ±9° during rudder pedal control), manually (rudder pedal disengage trigger depressed), or through loss of hydraulic power, the nosewheel is allowed to free castor. Steering is then accomplished by differential braking or rudder commands, and towing of the aircraft is permissible within the towing limit of ±90°.

The electronic control box has the capability to sense an electrical circuit failure. If a failure is detected, or there is no electrical power to the control box, the amber STEER INOP light on the MAP and the master CAUTION lights flash.

The nosewheel automatically centers when weight is removed from the nose gear or when hydraulic pressure is lost.

When the aircraft is airborne, steering handle and rudder pedal commands are disabled (nose gear is not down and locked, and nose and right landing gear shock absorber is extended).

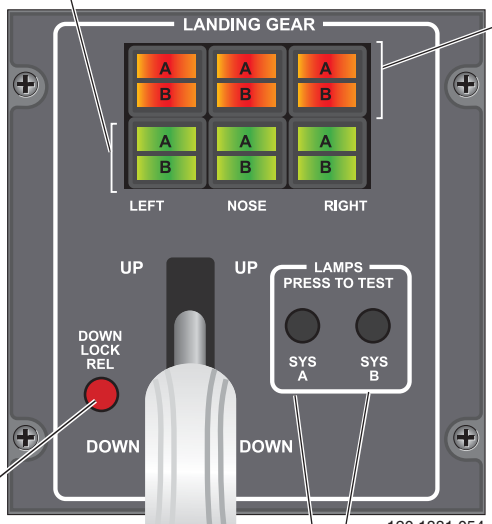
Operational Notes

Always use pedal steering for takeoff and landing. Using the steering handle causes a tendency to overcontrol and no hardover protection is provided.

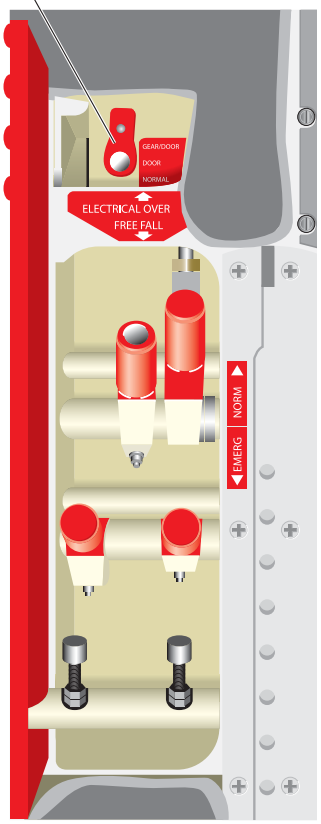
When making sharp turns onto the runway, always make sure that the aircraft is straight before releasing the steering handle. If the handwheel is released when the nosewheel angle is greater than ±9°, the PEDAL STEER INOP lights illuminate and the nosewheel steering is disengaged. The illumination of these lights may not be apparent in bright sunlight. If a static runup is performed prior to brake release, the aircraft will veer sharply when the brakes are released. When tubeless tires are installed, the possibility also exists of having a flat tire if enough pressure is exerted on the tire during runup to separate the tire bead from the rim.

INDICATION SYSTEM A AND B GREEN LANDING GEAR LIGHTS—When respective gear is down and locked.

INDICATION SYSTEM A AND B RED LANDING GEAR LIGHTS—When respective gear is not according to the lever position.



ELECTRICAL OVERRIDE GEAR ACTUATION SWITCH
GEAR DOORS—Lets down and locks the landing gear.
DOORS—Allows to open the doors.
NORMAL—Allows to close the doors.

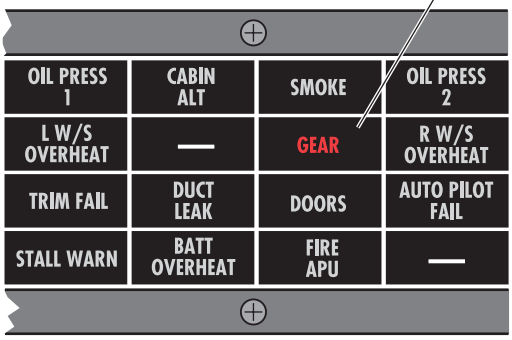


DOWNLOCK RELEASE BUTTON PRESSED—Releases the landing gear lever downlock to permit the retraction of the landing gear on the ground.

INDICATION SYSTEMS PUSH BUTTONS LAMP TEST PRESSED—All lights turn on, independently of the gear position or circuit failure.

LANDING GEAR LEVER—Operates solenoid valve to raise (UP) or lower (DOWN) the landing gear. Landing gear control lever latch prevents movement to the UP position when the airplane is on the ground.

NOTE:
Two landing gear position indication systems are provided (systems A and B). A disagreement in the position indication of the landing gear will be defined when systems A and B give conflicting indication. Refer to ABNORMAL PROCEDURES to determine which is the correct indication systems.



GEAR LIGHTS (RED)—Illuminates when gear is not down and:
- Flaps are selected beyond 17˚
- PLA is reduced below 46˚ (EEC ON)
- PLA is reduced below 66˚ (EEC OFF)
- Descent is below 200 feet AGL

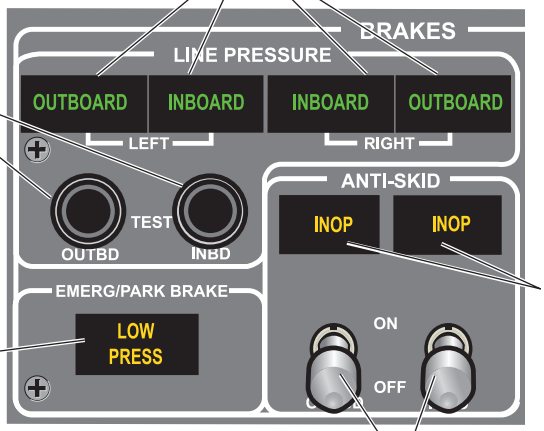
The light is accompanied by:
- Three-chime aural alert
- "Landing gear" voice warning
- Flashing master WARNING lights

Figure 14-19. Landing Gear Controls and Indicators



INBD AND **OUTBD** TEST SWITCHES—Both pressed in conjunction with application of the left and right brakes pedals; all line pressure lights should illuminate.

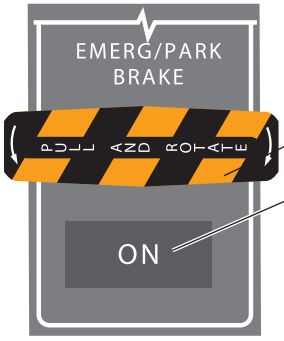
BRAKES LINE PRESSURE LIGHTS (GREEN)—Illuminate when the brakes are applied through the pedals and the relevant test. Switches are pressed, indicating that positive pressure is applied in the respective line.



ANTISKID **INOP** LIGHT (AMBER) — Illuminates when:
- Switch is OFF.
- The system is deenergized.
- Switch is ON and system malfunctions.

EMERG/PARK BRAKE **LOW PRESS** LIGHT (AMBER)—Illuminates when accumulator pressure is less that 1,500 psi.

ANTISKID SYSTEM CONTROL SWITCHES—
- ON: Tums on the relevant antiskid system.
- OFF: Tums off the relevant antiskid system.



EMERGENCY/PARKING BRAKE CONTROL (AFT PANEL)

EMERGENCY/PARKING BRAKE HANDLE

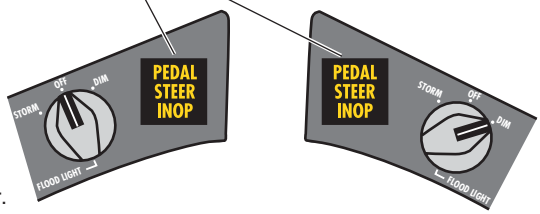
PARKING BRAKE LIGHTS (WHITE)—Illuminate when the emergency/parking brake is applied.



(CENTER PANEL)

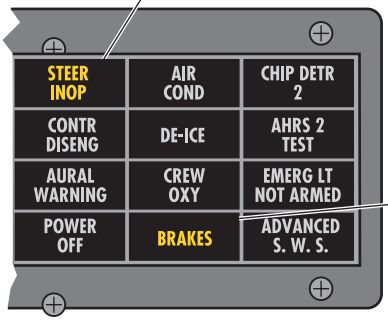
For airplanes Mod SB 120-032-0021, or SNs 120046, 120050, 120053, 120054, 120057, and subsequent

PEDAL STEER INOP LIGHT (AMBER)—Illuminates when:
- Either rudder pedal disengage trigger is pressed.
- Nosewheel angle exceeds ± 9˚ with the steering handle centered.



STEER INOP LIGHT (AMBER)—Illuminates when:
- Electronic control box senses potentiometer circuit failure.
- Electronic control box loses power.

The light is accompanied by:
- Single-chime aural alert
- Flashing master CAUTION lights



BRAKES LIGHT (AMBER)—Illuminates with either:
- Antiskid INOP light
- Emerg/park brake LOW PRESS light

The light is accompanied by:
- Single-chime aural alert
- Flashing master CAUTION lights

Figure 14-20. Brakes and Nosewheel Steering Controls and Indicators

QUESTIONS

1. The landing gear may be extended by:
- A. Normal system
- B. Alternate electrical system
- C. Free-fall system
- D. All of the above

2. Position status of the landing gear is provided by:
- A. Limit switches
- B. Proximity switches
- C. G-force sensors
- D. Sight glass in the cockpit

3. The landing gear system is powered by:
- A. Red hydraulic system
- B. Green hydraulic system
- C. Blue hydraulic system
- D. Both B and C

4. If the free-fall emergency method of lowering the gear is necessary, what unlocks the uplocks?
- A. Electrical current
- B. Hydraulic pressure
- C. Mechanical cables
- D. None of the above

5. How many landing gear position indications systems are there?
- A. One
- B. Two
- C. Three
- D. Four

6. The landing gear warning system will sound if:
- A. Flaps are selected beyond 17° with gear up.
- B. Power lever angle is reduced below 66° PLA with gear up.
- C. Airplane descends below 200 feet AGL.
- D. All of the above

7. Hydraulic pressure to the brake system is:
- A. 1,000 psi
- B. 2,000 psi
- C. 3,000 psi
- D. 4,000 psi

8. The EMB brakes have an antiskid feature.
- A. True
- B. False

9. When the parking brake is applied, how many brakes are actuated?
- A. Two
- B. Four
- C. Six
- D. None

10. If, during takeoff, the pilot forgot to release the parking brake:
- A. The applied power will overcome and release the brakes.
- B. A flashing light on the MAP will warn the pilot.
- C. A voice message will warn the pilot.
- D. The parking brake automatically releases as you taxi onto the runway.

11. Brakes are automatically applied when the landing gear is raised.
- A. True
- B. False

12. The nosewheel has its own brake system.
- A. True
- B. False

13. How many indications does the pilot have to warn him that the emergency/parking brake is on?

A. One
B. Two
C. Three
D. Four

14. The antiskid system may be turned ON or OFF.

A. True
B. False

15. The nosewheel steering is operated:

A. By a cable and drum assembly
B. Hydraulically through the blue hydraulic system
C. Hydraulically through the green hydraulic system
D. By a combination of hydraulics and cables.

16. The nosewheel is steered by means of rudder pedals and steering handle.

A. True
B. False

17. When entering the cockpit and powering up the electrical system, the pilot sees the PEDAL STEER INOP warning light illuminated. This indicates to the pilot that:

A. A circuit breaker has popped.
B. Hydraulic fluid is low in the system.
C. The nosewheel has been turned past its limit.
D. A rudder pedal cable has broken.

18. The nosewheel may be steered through a _____ with the steering handle.

A. 50° arc
B. 75° arc
C. 100° arc
D. 125° arc

19. Rudder pedals control nosewheel steering through a _____ arc.

A. 5°
B. 7°
C. 14°

20. Should steering overtravel occur, the pilot may detect it on the preflight by means of a tell-tale indicator.

A. True
B. False



CHAPTER 15
FLIGHT CONTROLS

CONTENTS

Page

INTRODUCTION 15-1
GENERAL 15-1
FLIGHT CONTROLS 15-2
Elevators 15-2
Ailerons 15-4
Rudders 15-6
Gust Lock 15-10
FLAPS 15-10
General 15-10
Components 15-10
Control and Monitoring 15-12
Operation 15-14
STALL WARNING SYSTEM 15-16
General 15-16
Components 15-18
Control and Monitoring 15-18
Operation 15-19
System Test 15-22
QUESTIONS 15-23

ILLUSTRATIONS

Figure	Title	Page
15-1	Flight Controls	15-2
15-2	Elevator Control System	15-3
15-3	Elevator Trim Control and Indication	15-3
15-4	Aileron Control System	15-5
15-5	Rudder Control System	15-6
15-6	Rudder Pedal Adjustment Mechanism	15-7
15-7	Rudder Control Panel	15-8
15-8	Gust Lock	15-10
15-9	Flap System	15-11
15-10	FLAP Annunciator/Control Panel	15-12
15-11	Flap Position Indicator	15-13
15-12	Flap Fault Indication Lights	15-15
15-13	Stall Warning System	15-17
15-14	STALL WARNING Control Panel	15-18
15-15	Fast/Slow Indicator	15-19
15-16	Flight Control System Controls and Indicators	15-20
15-17	Stall Warning System Controls and Indicators	15-21

CHAPTER 15
FLIGHT CONTROLS



INTRODUCTION

The primary flight controls of the EMB-120 are manually actuated elevator and ailerons, and a hydraulically-actuated rudder.

The secondary flight controls consist of the flaps plus the aileron and elevator trim tabs. There is no rudder trim tab; rudder trim is achieved by repositioning the hydraulic center of the rudder power control unit.

GENERAL

The primary flight controls are operated by mechanically interconnected control columns, "rams horn" control wheels, and rudder pedals. In case of jamming, elevator and ailerons may be disconnected between pilot and copilot allowing operation by only one side.

The control wheels and columns, and their corresponding control surfaces, may be locked with the gust lock installed on the control stand forward of the power levers. The rudder is provided with hydraulic gust damping only.

The ailerons and elevator are mechanically actuated by cables, pulleys, bellcranks, and pushrods. The rudder is normally hydraulically actuated and, as a backup, may be mechanically actuated.

The flaps consist of three panels on each wing. Each panel is individually actuated by a hydraulic jack.

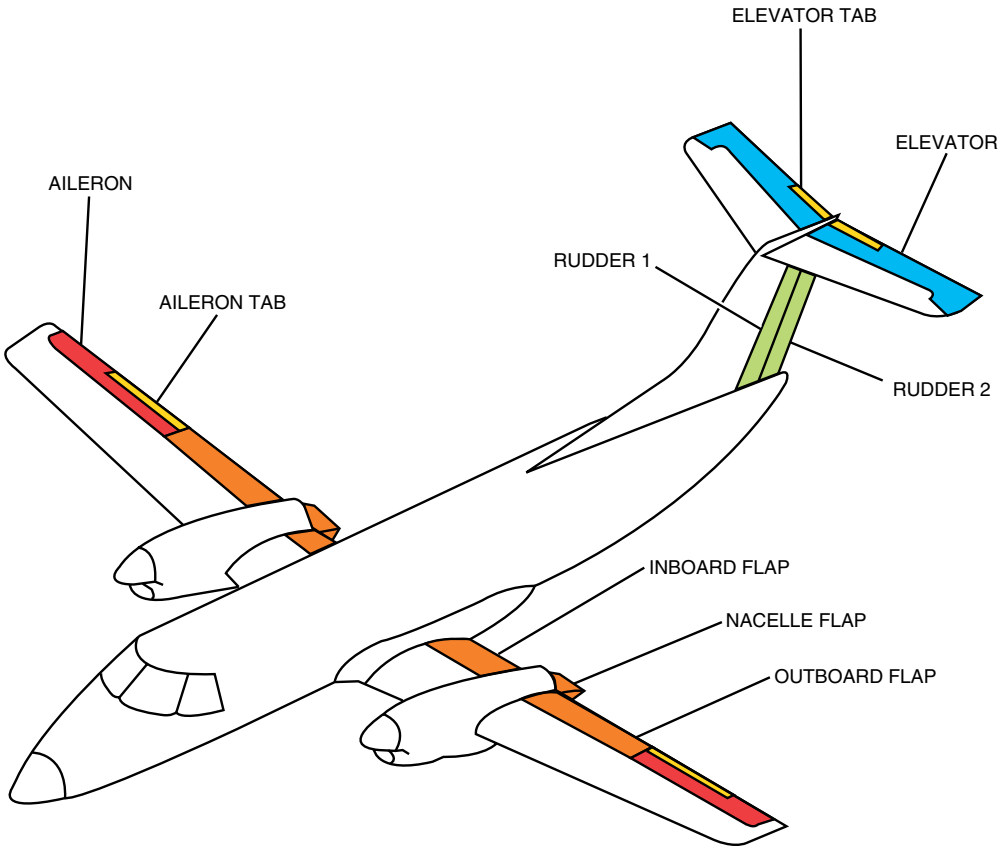


Figure 15-1. Flight Controls

Elevator trim tabs are adjusted by control wheels on each side of the control pedestal. The aileron trim tab (located on the right aileron) and the rudder trim are adjusted with trim wheels on the aft left and aft end of the center console.

FLIGHT CONTROLS

ELEVATORS

General

The left and right elevators (Figure 15-1) provide pitch control for the airplane. Though not physically connected, the elevators work in tandem because the control systems that actuate them are mechanically interconnected.

Elevator Controls

The elevators are actuated by forward and aft movement of either the pilot's or copilot's control column. The control columns are mechanically connected to the elevators by pulleys, cables, pushrods, and bellcranks.

The elevator control system (Figure 15-2) is made of two independent subsystems that are mechanically interconnected.

The left elevator control subsystem additionally contains the autopilot pitch control servo. The right ele-vator control subsystem also contains the stick pusher for both stall warning systems.

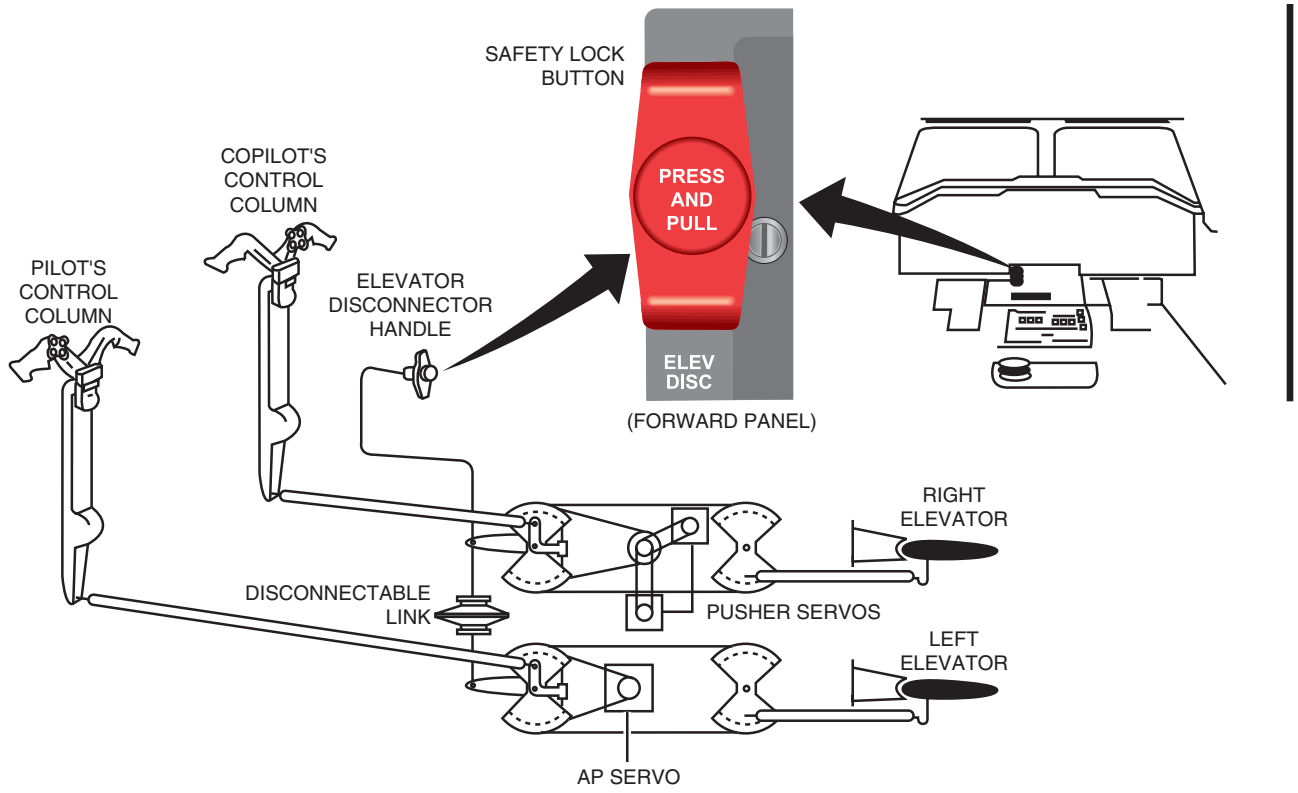


Figure 15-2. Elevator Control System

Elevator Trim

The elevator trim system provides the flight crew with the ability to minimize the elevator control forces required to maintain a desired pitch attitude.

Mechanical Pitch Trim

The elevator trim system is mechanically actuated by two interconnected pitch trim control wheels, located on each side of the control pedestal (Figure 15-3). Through a single system of chains, cables, and pushrods, the wheels actuate a trim tab on each elevator.

Electric Pitch Trim

The elevator trim system may be electrically actuated by pitch trim control switches operating the autopilot pitch trim servo.

The switches, labeled TRIM, are located on the outboard side of each control wheel. The pilot's pitch

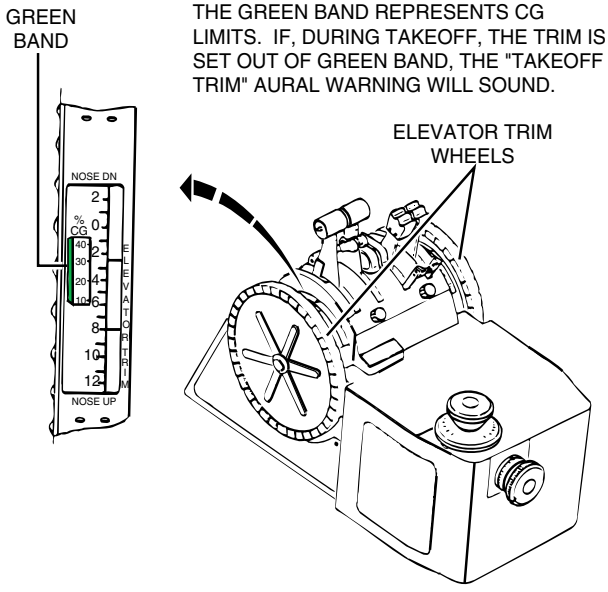


Figure 15-3. Elevator Trim Control and Indication

trim switch has priority over the copilot's. Both parts of the split switch must be depressed at the same time for the pitch trim to operate.

NOTE

The electric pitch trim must not be used if only one-half of either trim switch will operate the system. Check as detailed in normal procedures.

Trim Indication

The elevator pitch trim control wheel operates an indicator on a scale graduated in both trim units and % MAC (Figure 15-3). A green takeoff band is marked on the scale.

Trim Fail Warning

If the electric pitch trim split switch is not operated simultaneously, or the electric trim system fails, the following warnings are triggered:

- A 3-chime alert sounds.
- The voice warning "Trim Fail" is given.
- The red master WARNING lights flash.

Takeoff Trim Warning

If the power levers are advanced for takeoff with the elevator trim out of the green takeoff band, the trim proximity switches (one on each trim tab) send a signal that triggers the aural warning system as follows:

- A 3-chime alert sounds.
- The voice warning "Takeoff Trim" is given.
- The red master WARNING lights flash.

Elevator Disconnect System

An elevator disconnect system is provided to disconnect the left elevator control from the right elevator control if either subsystem should jam.

The elevator disconnect system is controlled by a red control grip, labeled ELEV DISC, located on the left side of the control pedestal forward panel (Figure 15-2). The handle is provided with a locking button labeled PRESS AND PULL.

When disconnected, the left elevator is controlled only by the pilot's control column or the autopilot servo, and the right elevator is controlled only by the copilot's control column or the stall warning system control pushers.

Once disconnected, the elevator control system may not be reconnected in flight.

Elevator Disconnect Warning

When the disconnect system is actuated the following warnings are triggered:

- A single-chime alert sounds.
- The amber CONTR DISENG light on the MAP illuminates.
- The amber master CAUTION lights flash.

AILERONS

General

The aileron control system (Figure 15-4) provides roll control for the airplane. It consists of separate left and right control subsystems that are mechanically interconnected.

Aileron Controls

The aileron control system is actuated by side-to-side movement of either the pilot's or copilot's "rams horn" control wheels.

The aileron surfaces on each outboard wing are mechanically actuated by chains, control cables, and bellcranks.

The left aileron control subsystem contains the autopilot roll control servo.

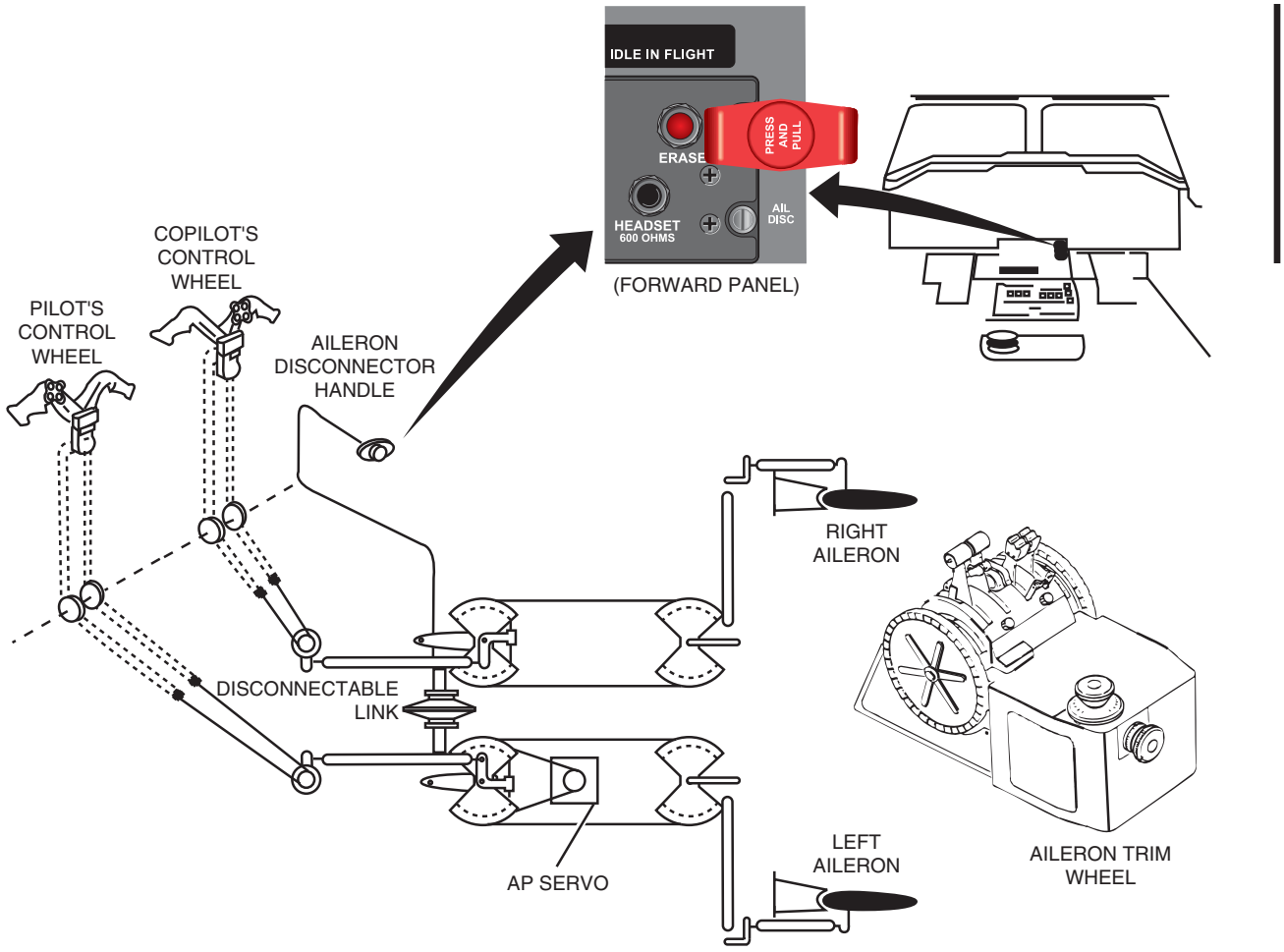


Figure 15-4. Aileron Control System

Aileron Trim

The aileron trim system provides the flight crew with the ability to minimize the aileron control forces required to maintain a desired roll attitude.

The right aileron trim tab surface is mechanically actuated by cables, chains, and rods, and is a trim/balance-type tab.

The left aileron is a balance type tab only and cannot be trimmed by the aileron trim system.

Trim Indication

The aileron trim control wheel, located on the aft center console (Figure 15-4), incorporates an indicator and a scale graduated in trim units.

Aileron Disconnect System

An aileron disconnect system is provided to disconnect the left aileron control from the right aileron control if either subsystem should jam.

The aileron disconnect system is controlled by a red control grip, labeled AIL DISC, located on the right side of the control pedestal forward panel (Figure 15-4). The handle is provided with a locking button labeled PRESS AND PULL.

When disconnected, the left aileron is controlled only by the pilot's control wheel or the autopilot servo, and the right aileron is controlled only by the copilot's control wheel.

Once disconnected, the aileron disconnect system may not be reconnected in flight.

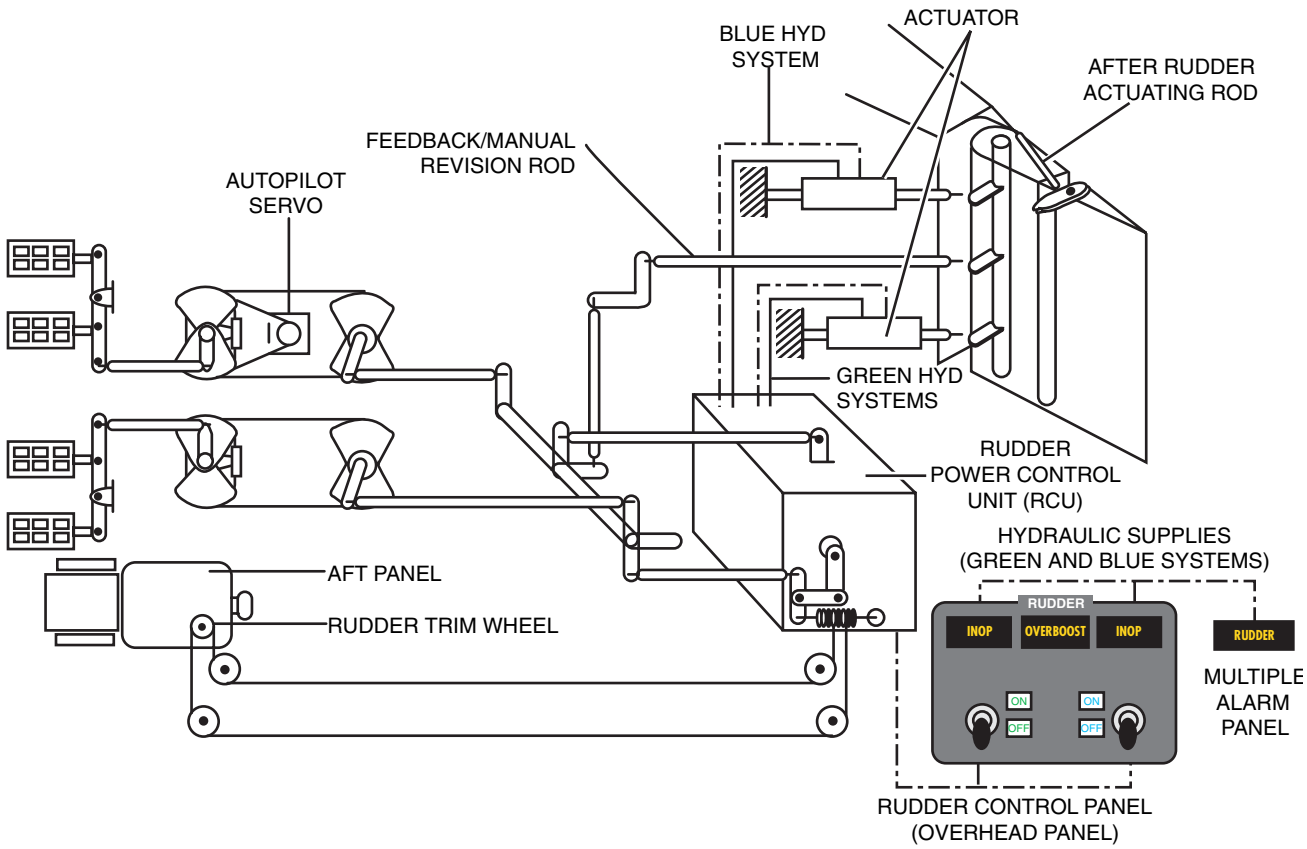


Figure 15-5. Rudder Control System

Aileron Disconnect Warning

When the aileron disconnect system is actuated, the following warnings are triggered:

- A single-chime aural alert sounds.
- The amber CONTR DISENG light on the MAP illuminates.
- The amber CAUTION lights flash.

RUDDERS

General

The rudder control system (Figure 15-5) provides directional control and stability around the aircraft vertical (yaw) axis.

The rudder is mounted behind the vertical stabilizer fin and is divided into two sections called rudder 1 and rudder 2.

Rudder 2, behind rudder 1, is mechanically connected to rudder 1 and deflects in the same direction with 75% more travel. The tandem mounted rudder acts as a large antibalance tab.

The maximum deflections are:

- Rudder 1—20° right and 17° left
- Rudder 2—35° right and 29.4° left

Greater deflection to the right is provided because the propellers are not counterrotating. Due to the clockwise rotation of both propellers, the left engine is the critical engine. Therefore, if the left engine fails, more right rudder is necessary to maintain directional control at V_{MC}.

Rudder 1 is normally actuated by hydraulic actuators and a feedback rod. If the hydraulic system fails, rudder 1 is actuated only by the feedback rod.

Rudder Control System

The rudder control system consists of the following major components:

- Rudder pedals
- Rudder power control unit (RPCU)
- Hydraulic actuators

Rudder Pedals

There are two sets of rudder pedals mechanically connected to the hydraulic rudder power control unit and rudder 1 by pushrods, torque tubes, control cables, and bellcranks.

The rudder control system is actuated by fore and aft movement of either the pilot's or copilot's rudder pedals, or by the autopilot yaw channel servo.

The rudder pedals also operate the nosewheel steering during ground operation.

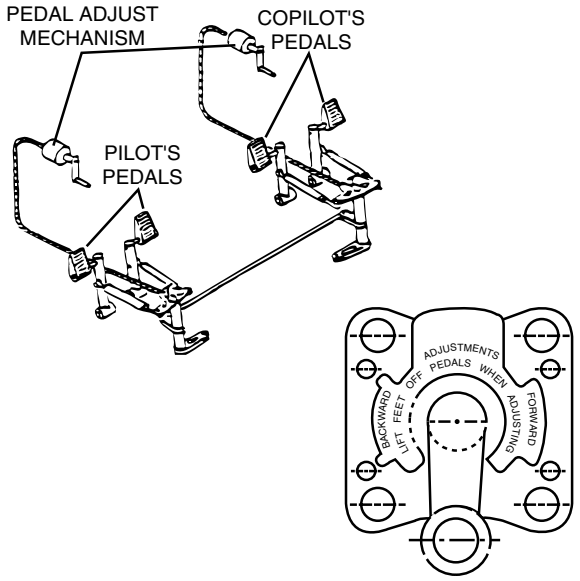


Figure 15-6. Rudder Pedal Adjustment Mechanism

The mechanically interconnected rudder pedals are not provided with a disconnect system.

Rudder Pedal Adjustment. Each pedal assembly is provided with an adjustment mechanism (Figure 15-6) that allows the pedals to be adjusted fore and aft for crew comfort. The adjustment crank is located below the instrument panel in front of each control column.

Rudder Power Control Unit (RPCU)

The rudder power control unit (RPCU) provides green and blue hydraulic system segregation and redundancy. It also accepts the rudder trim input and includes a load feel mechanism that provides a progressively increasing artificial feel to the rudder pedals as they are depressed.

The RPCU receives hydraulic pressure through isolation solenoid valves from both the blue and green hydraulic systems. The servo valves receive control inputs from the pilot's and copilot's rudder pedals. Bypass valves within the RPCU permit operation of the rudder by either system.

The RPCU controls the hydraulic fluid pressure to the hydraulic rudder actuators.

Hydraulic Actuators

Two hydraulic actuators are used to move rudder 1. They are double action actuators and each are supplied independently, one by the blue hydraulic system and the other by the green.

Rudder Trim System

The rudder trim system provides the flight crew with the ability to minimize the rudder control forces required to maintain a desired rudder deflection.

The rudder does not have a trim tab. The rudder itself is deflected by adjusting the hydraulic center of the rudder power control unit. This also causes the rudder pedal neutral position to adjust accordingly.

Since rudder trim is dependent upon the RPCU, rudder trim capability is lost in the event of a total hydraulic failure.

The rudder is trimmed via the RUDDER TRIM wheel on the top rear of the control pedestal. The wheel incorporates an indicator an a scale graduated in trim units.

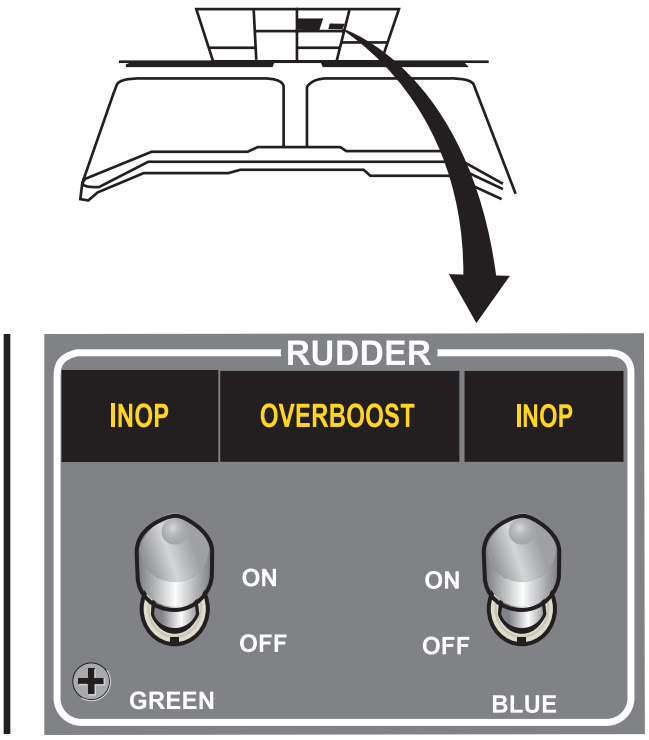


Figure 15-7. Rudder Control Panel

Control and Monitoring

Control and monitoring of the rudder system is accomplished with the switches and indicator lights on the rudder control panel and the indicator lights on the MAP.

Rudder Control Panel

The overhead RUDDER panel (Figure 15-7) contains the following controls and indicator lights for the rudder system:

- Rudder control switches
- INOP lights
- OVERBOOST light

Rudder Control Switches. The two rudder control switches, labeled BLUE and GREEN, control the rudder hydraulic system isolation valves. The solenoid operated valves are normally open and electrically closed.

When the switch is ON, the valve is open (unpowered) allowing hydraulic fluid to be supplied to the RPCU. When the switch is OFF, the solenoid valve is electrically held in the closed position.

Power for the green and blue hydraulic isolation solenoid valves are emergency DC bus 1 and 2 respectively.

Electrical control for the rudder system is through a system of relays. The relays receive signals from both airspeed indicators and the pressure sensing switches in the RPCU.

INOP Lights. The amber INOP lights indicate a drop in hydraulic system pressure. The lights illuminate when:

- The switch is OFF.
- The switch is ON and hydraulic pressure drops.

An INOP light triggers the following warnings:

- A single-chime aural alert sounds.
- The amber RUDDER light on the MAP illuminates.
- The amber master CAUTION lights illuminate.

At airspeeds above 120 knots, a single-rudder hydraulic system failure will not trigger an INOP light.

However, if both rudder hydraulic systems fail above 120 knots, both INOP lights illuminate and trigger the same warnings.

OVERBOOST Light. For enhanced rudder authority and redundancy, it is desirable to have both hydraulic systems supply power to the rudder at low airspeeds.

It is not desirable to have both hydraulic systems supply power at high airspeeds, due to the possibility of overstressing the airframe with full rudder input. Therefore:

- Airspeed < 120 knots—Both green and blue hydraulic systems supply hydraulic pressure to the rudder.
- Airspeed > 120 knots—The green system is automatically shut down.

The amber OVERBOOST light on the rudder control panel illuminates to indicate the failure of the rudder green hydraulic system to shut down when the airspeed is above 120 knots. The OVERBOOST light triggers the following warnings:

- A single chime aural alert sounds.
- The amber RUDDER light on the MAP illuminates.
- The amber master CAUTION lights illuminate.

To prevent full rudder input, the crew must switch one system OFF.

RUDDER SPEED SW Light

The amber RUDDER SPEED SW light on the MAP illuminates when there is a disparity in the signals received from both airspeed indicators as the airspeed approaches 120 knots.

When a disparity exists, the system may not automatically switch OFF the green hydraulic system. Therefore, the crew must switch OFF one system to preclude a potential overboost condition.

Rudder System Operation

In normal operation, the rudder system is mechanically controlled, hydraulically actuated, and electrically monitored. With a total hydraulic failure, the rudder system may be mechanically actuated with the rudder pedals.

The rudder system operates in one of three modes:

- Hydraulically by both green and blue hydraulic systems simultaneously
- Hydraulically by either green or blue hydraulic system separately
- Mechanically reverted (without hydraulic power assistance)

At airspeeds less than 120 knots with both switches on the rudder control panel ON, the electrical relays deenergize (open) the hydraulic isolation solenoid valves pressurizing both the blue and green rudder hydraulic system.

When the airspeed exceeds 120 knots with both switches ON, the electrical relays energize (close) the green hydraulic system isolation solenoid valve. The rudder is now actuated by the blue hydraulic system only.

The electrical relays are designed to give priority to the blue rudder hydraulic system. If the blue system should fail, or be switched OFF, the green system automatically provides hydraulic pressure.

If both the blue and green hydraulic systems fail, or both switches are OFF, the rudder system is manually reverted. It is operated directly through the control tubes, cables, and linkage.

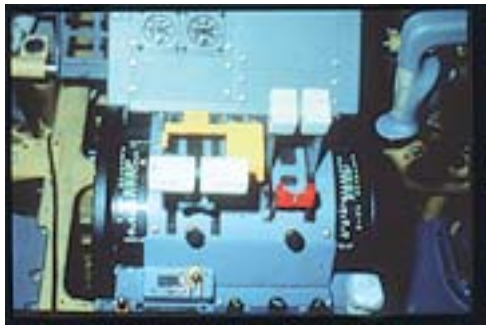




Figure 15-8. Gust Lock

GUST LOCK

The ailerons and elevators may be immobilized to protect against wind gust damage by use of the gust lock located on the control pedestal (Figure 15-8).

To engage the gust lock, the following conditions must first be met:

- Engine power levers below FLT IDLE
- Control column full forward
- Control wheel full left

The gust lock is then engaged by pulling the gust lock lever full aft.

The control columns lock in the full forward position and the ailerons in full left roll. The gust lock also limits power lever travel to FLIGHT IDLE and below, preventing inadvertent takeoffs with the control locks on.

The rudder is not locked by the gust lock. Adequate damping is provided by the rudder hydraulic systems.

FLAP SYSTEM

GENERAL

The EMB-120 flap system is installed to reduce the stall speed by providing airfoil boundary layer control and wing camber change. This permits a safe speed reduction during takeoff and landing, allowing shorter takeoff and landing distances.

Each wing has three flap sections: outboard, nacelle, and inboard (Figure 15-9). The outboard and inboard flaps are double-slotted Fowler flaps. The nacelle flaps are standard.

The flaps are electronically controlled and hydraulically actuated.

The outboard flaps are actuated by the green hydraulic system. The inboard and nacelle flaps are actuated by the blue system.

COMPONENTS

In addition to the flap panels, the flap system consists of the following major components:

- Electronic control unit
- Actuators

Electronic Control Unit

The electronic control unit, located in the avionics compartment, provides automatic digital flap control.

The unit is a microprocessor with three independent control channels, one for each flap pair (Figure 15-9).

It processes signals from the selector lever (enable), the flap panel (requested position), and the transducers on each actuator (actual flap position), and commands the solenoid control valves to extend or retract the flaps.

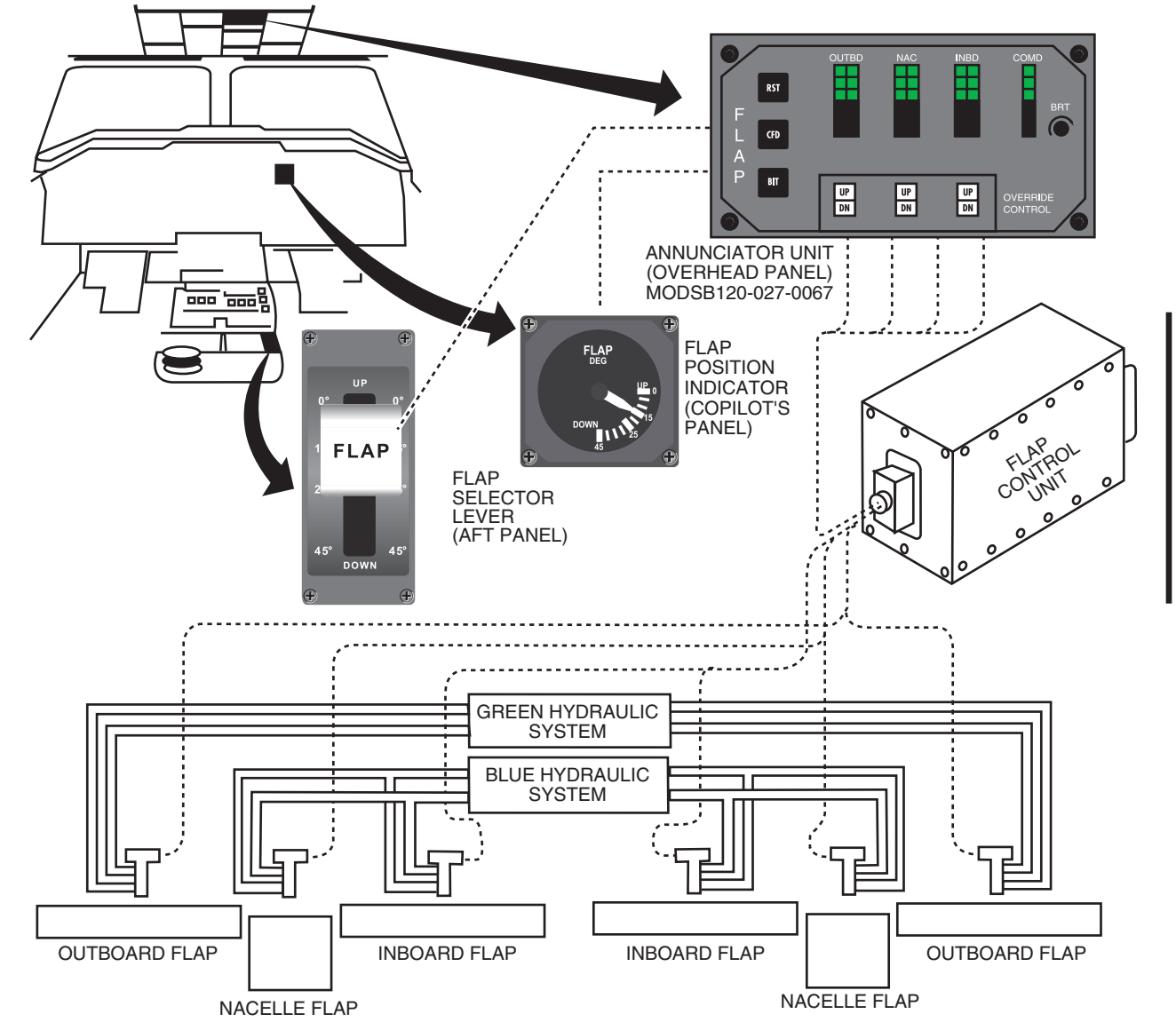


Figure 15-9. Flap System

Actuators

There are six flap actuators, one for each flap, used to hydraulically extend and retract the flaps.

The actuators are all similar in design and function. Each actuator has:

- An integral manifold with two solenoid control valves—The valves allow the larger piston chamber to be either pressurized or vented, extending or retracting the actuator.
- An unbalanced piston area with the smaller area under constant pressure—This ensures the actuator is driven to the retract position whenever the larger piston area chamber is vented to return.
- A piston transducer signaling the flap position to the control unit.

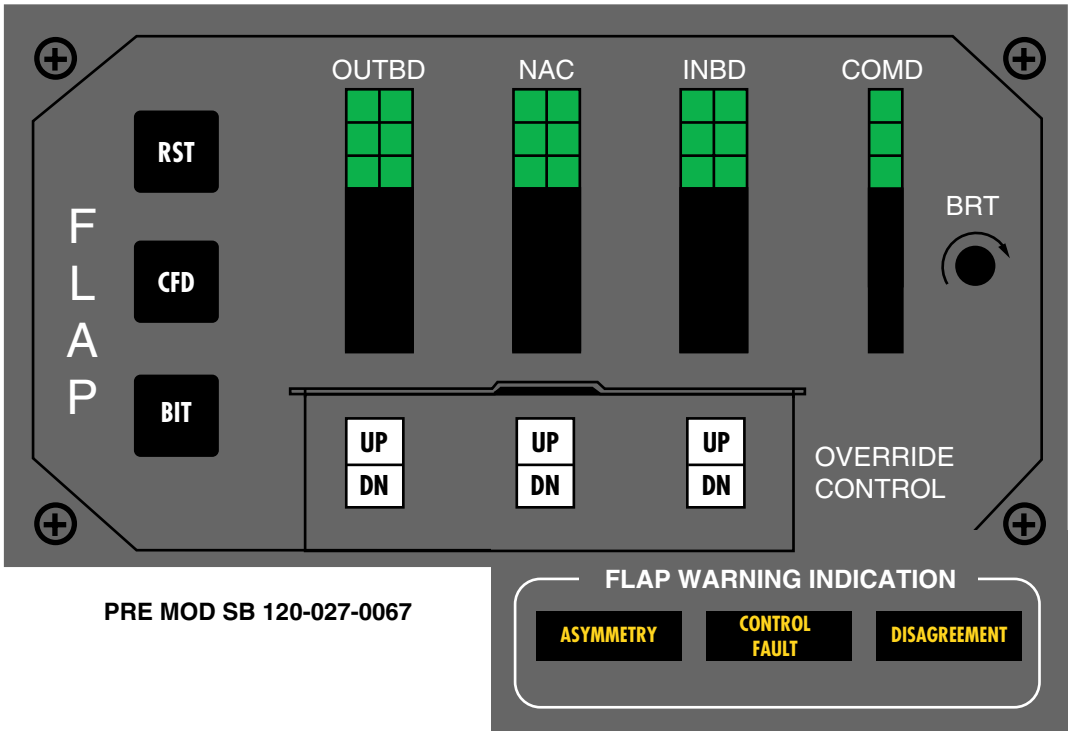


Figure 15-10. Flap Annunciator/Control Panel

CONTROL AND MONITORING

Selector Lever

The selector lever is used to command flap position. The lever is located on the right aft side of the center console.

The selector lever is labeled and detented at the following flap positions:

- 0° (UP)—Normal enroute configuration
- 15°—Takeoff/approach configuration:

  This position has a mechanical maze gate stop to prevent accidental selection of 0° flaps on a go-around.

  To pass the 15° gate, the lever must be lifted, selected to the 15° flap position, released, lifted again, and moved in the desired direction.
- 25°—Normal landing configuration
- 45°—Alternate landing configuration

Annunciator Control Panel

The flap annunciator/control panel, labeled FLAP (Figure 15-10), is located on the cockpit overhead panel.

The flap annunciators provide a digital light bar indication of flap and selector lever position as follows:

- Three double-light bar columns, labeled OUTBD, NAC, and INBD, represent each flap pair.
- A single-light bar column, labeled COMD, represents the commanded position of the flap selector lever.

Each light represents 5.625° of flap. There are eight lights in each column to represent the total 45° of flap.

A brightness control knob, labeled BRT, controls the intensity of the light bars.

The FLAP panel also incorporates the following additional system controls:

- Three momentary pushbuttons on the left side of the panel:
  - Reset control (RST)
  - Cancel fault display control (CFD)
  - Built-in-test control (BIT)
- A set of three rocker switches labeled:
  - OVERRIDE CONTROL, UP–DN

Reset Control (RST)

The reset control (RST) pushbutton clears all memories and faults, initializes all systems, and reinstates automatic control. If RST is pressed after a control fault, the last valid display is deleted. The system also resets itself when power is applied.

Cancel Fault Display Control (CFD)

A cancel fault display control (CFD) pushbutton stops the flashing of the light bars ***except*** during the built-in-test. The light bars revert to steady illumination when CFD is pushed.

Built-in-test Control (BIT)

A built-in-test control (BIT) pushbutton initiates a three-second system self-test. The flap selector lever must be in the 0° (UP) position to perform this test. (The ASYMMETRY light illuminates when the BIT button is pressed, and goes out when the button is released).

A successful test is indicated by:

- The steady illumination of all light bars
- The illumination of the CONTROL FAULT light
- Flashing the FLAP light and ADVANCED SWS (stall warning system) light on the MAP
- The pointer on the analog flap position indicator on the instrument panel raising approximately 1/4 inch above the 0° position.

An unsuccessful test is indicated by the flashing of the affected channel's light bars.

At the completion of the test, the RST pushbutton is used to reset the system and restore automatic control.

Override Control Switch

Within the OVERRIDE CONTROL area of the FLAP panel are three rocker switches labeled UP–DN.

The switches disengage automatic control of a flap pair and directly control the flap actuator solenoids. The switches are positioned under a hinged cover and below the light bars of the respective flap pair (OUTBD, NAC, INBD) they control.

Flap Warning Indication

Three warning lights are located in the FLAP WARNING INDICATION section of the flap annunciator panel (Figure 15-10). Discussed in more detail later, the lights are:

- ASYMMETRY
- CONTROL FAULT
- DISAGREEMENT

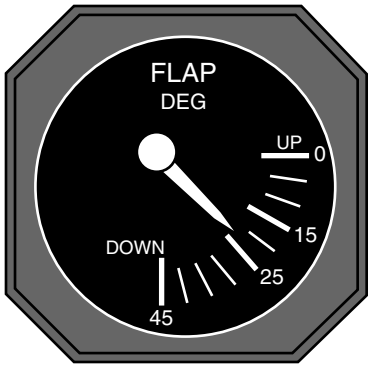


Figure 15-11. Flap Position Indicator

Position Indicator

The analog flap position indicator (Figure 15-11) is located on the instrument panel in front of the

copilot. It provides an indication of the weighted average aerodynamic value of flap position.

The FLAP position indicator scale is graduated in flap positions of 0° (UP), 15°, 25°, or 45°.

OPERATION

General

Flap system normal operation begins when the flight crew commands flap actuation by moving the selector handle into any one of the detented positions: 0°, 15°, 25°, 45°.

Within the handle assembly, an enable switch signals the annunciator panel and the control unit that the selector lever is in a detented position. This authorizes the actuation of the flaps; the flaps will not move unless the handle is in a detented position.

Once the handle is set, a potentiometer sends a signal to the FLAP annunciator panel. The signal is interpreted and displayed as a selected position on the COMD light bar on the FLAP panel.

The FLAP annunciator panel signals the selector handle position to the control unit. The control unit then sends the signal to the flap actuator solenoid control valves through three independent channels.

As the flap actuators extend/retract, piston transducers send a position signal through the control unit to the FLAP annunciator panel. These position signals are compared to the commanded position signal, and adjustment command signals are sent by the control unit to the solenoid control valves.

The flap position signals are also used to indicate flap position on the FLAP panel light bars (OUTBD, NAC, INBD).

The FLAP annunciator panel also determines a single, weighted, aerodynamic average flap position. It sends this signal to be displayed on the analog FLAP position indicator.

Synchronous Flap Movement

The system is programmed to equally and synchronously deflect flap pairs so that all pairs reach the selected position at the same time.

Due to mechanical configuration, each pair of flaps have a different rate of travel.

To have the flap pairs arrive at the selected position synchronously, the FLAP panel commands each pair's motion in increments of no greater than five degrees.

Since the flap pairs are commanded independently, flap pair position disagreements up to 7° while in transit are considered normal and expected.

For example: the outboard flap pair may be extended up to seven degrees greater than the nacelle flap pair at any point in the extension process.

Flap Symmetry Control

The flap system provides for automatic retraction of a flap panel when the aerodynamic loads on that panel exceed the limits. Due to the symmetry priority function, the other flap panel within the pair follows the affected panel so that symmetry is maintained.

System design and programming command symmetrical flap pair actuation. (The system cannot command an asymmetric condition within a pair.) If an asymmetric condition occurs, the control unit provides solenoid commands using the following symmetry priority.

If the asymmetry is:

- Less than 2°—Both flaps of the pair move toward the selected position.
- Greater than 2° but less then 3.5°—The flap closest to the selected position waits for the other flap to catch up.

- Greater than 3.5° but less than 7°—The system disregards the selected position command and moves the flaps toward one another.

- Greater than 7°—The affected pair's automatic control is disengaged and an alarm is activated.

Once the flaps reach their final position, a 1.2° deadband is established to avoid unnecessary solenoid activation.

In an emergency, each flap pair may also be manually extended/retracted. The solenoids for each pair of flaps are directly connected to a FLAP annunciator panel OVERRIDE control rocker switch.

When an override switch is actuated in either direction, the corresponding control unit channel is deactivated. Position indicating on both the FLAP annunciator panel and analog indicator remains active. The use of the override switches does not activate an alarm.

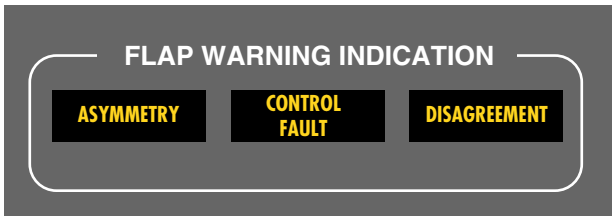


Figure 15-12. Flap Fault Indication Lights

Fault Indications

The control unit provides for alarms and warnings for the following three types of failures:

- Asymmetry fault

- Disagreement fault

- Control fault

The corresponding light on the FLAP WARNING INDICATION section of the FLAP annunciator panel (Figure 15-12) illuminates when a failure is detected.

WARNING

If a fault condition is detected, the crew should take no action to correct the fault without reference to the appropriate checklist. Failure to comply with the approved checklist procedures may result in extreme asymmetries and difficulty with aircraft control.

Asymmetry Fault

Flap asymmetry is defined as a different degree of extension within a flap pair. Asymmetry is the most hazardous flap fault due to potential uncommanded aircraft roll.

An asymmetry fault occurs when the flaps within a pair differ in position by more than seven degrees.

When the system senses this condition, the affected flap pair is disengaged from the automatic control system and, along with a positive roll tendency, the following indications are presented:

- ASYMMETRY fault light illuminates.

- The affected flap pair light bar flashes.

- A single-chime aural alert sounds.

- The amber FLAP light on the MAP illuminates.

- The master CAUTION lights flash.

Pressing the CFD button on the FLAP annunciator panel cancels the flashing light bar.

Disagreement Fault

A disagreement fault occurs when a position difference greater than seven degrees exists between two or more pairs of flaps. With this fault, the flap pairs are still symmetrical.

When the FLAP annunciator panel detects a disagreement fault, the following indications are presented:

- DISAGREEMENT fault light illuminates.
- All light bars on the annunciator flash.
- A single-chime aural alert sounds.
- The amber FLAP light on the MAP illuminates.
- The master CAUTION lights flash.

Although the system detects and displays this fault, no corrective action is taken.

Pressing the CFD button cancels the flashing light bars.

If the system self-corrects, the FLAP light on the MAP deenergizes.

Control Fault

A control fault is a failure in:

- Any of the three control unit channels
- The annunciator panel
- The selector lever

When a control fault is detected in a control unit channel, the affected flap pair is disengaged from automatic control and its light bar flashes.

When a control failure is detected in the annunciator panel or the selector lever, all three flap pairs are disengaged and all flap position light bars flash.

Once a control fault has occurred, the disagreement fault is masked to prevent one failure from triggering two alarms.

When a control fault occurs, the following indications are presented in addition to the flashing light bar(s):

- CONTROL FAULT light illuminates.
- A single-chime aural alert sounds.
- The amber FLAP and ADVANCED SWS (stall warning systems) lights on the MAP illuminate.
- The master CAUTION lights flash.

The amber ADVANCED SWS light indicates:

- Flap position data is unreliable.
- The stall warning system has been reset to actuate the following:
  - Stick shaker as if the flaps were greater than 35°.
  - Stick pusher as if flaps were at 0°.

NOTE

Aircraft Mod SB 120-027-0067 have an alphanumeric display window on the FLAP annunciator panel for additional flap position information.

Some Pre Mod aircraft may have the above Mod annunciator panels. In these cases, the alphanumeric feature is not functional.

STALL WARNING SYSTEM

GENERAL

The stall warning system is designed to alert the pilots to an impending stall condition. It prevents the aircraft from attaining attitudes that could result in full aerodynamic stall and potential loss of control.

System operation is based on angle of attack.

Sensors along the fuselage sides detect the direction of local airflow. This information, corrected for slip, is continuously presented visually (fast/slow indicator) to the crew as the actual angle of attack in reference to 1.3 V_{SO}.

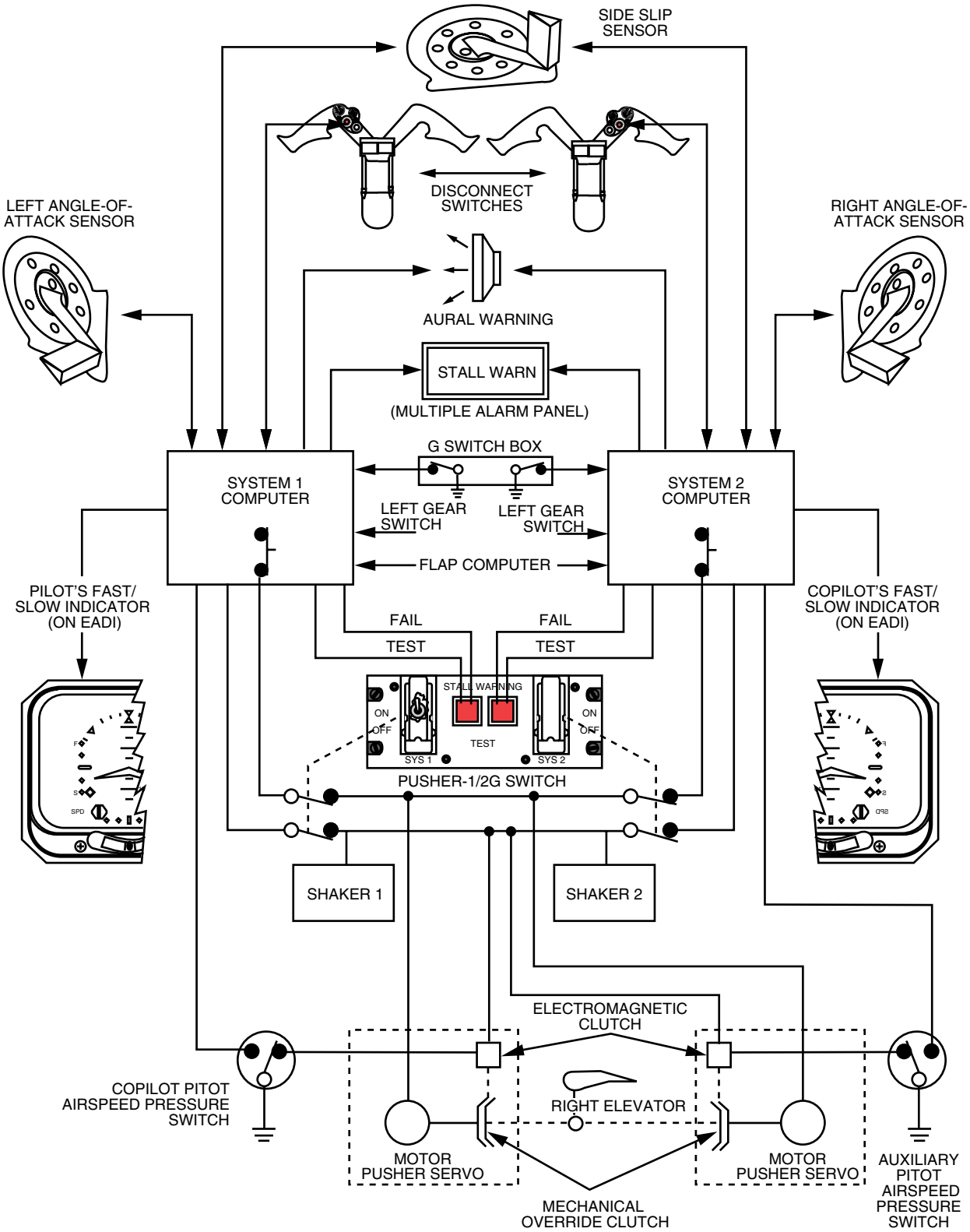


Figure 15-13. Stall Warning System

COMPONENTS

The system contains two identical subsystems (Figure 15-13) that function independently to drive control column stick shakers, stick pushers, and the aural warning system.

Failure of either subsystem has no effect on the remaining subsystem.

Each subsystem contains the following:

- A heated angle of attack sensor
- A computer
- A fast/slow indicator
- A column shaker

Both subsystems share the following:

- A slip sensor
- Two pusher servos

CONTROL AND MONITORING

AP/PUSHER Disconnect Switches

Two AP/PUSHER disconnect switches for each subsystem are located on each control column. When either disconnect switch is held, the computers will inhibit both stick pusher servos allowing the elevator control system to operate freely.

STALL WARNING Control Panel

The STALL WARNING control panel is located on the aft section of the center pedestal (Figure 15-14). It contains two red guarded disconnect switches, labeled SYS 1 and SYS 2, and two red failure warning light TEST buttons.

The disconnect switches give the pilots the capability to disconnect a malfunctioning computer from its respective shaker and pusher servos.

The two red failure warning light/TEST buttons illuminate with a subsystem malfunction and are used to initiate each subsystem self-test.

The red STALL WARN light on the MAP illuminates anytime a failure warning light TEST button is on.

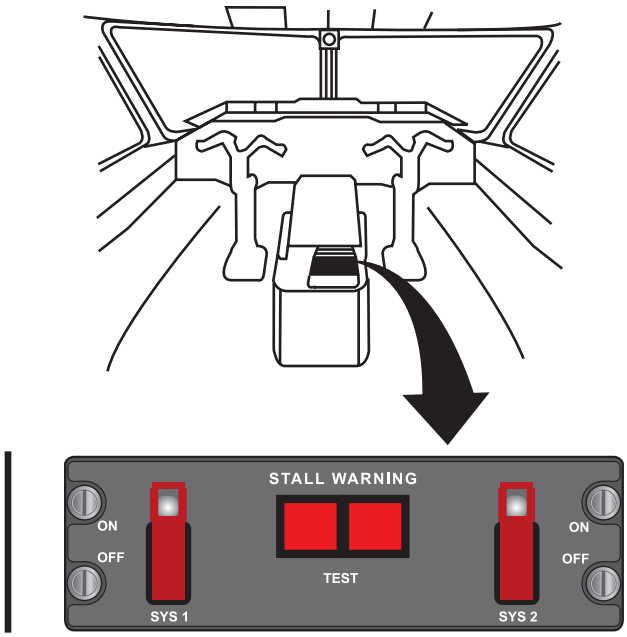


Figure 15-14. STALL WARNING Control Panel

Fast/Slow Indicators

A diamond shaped pointer and fast/slow scale is included on each pilot's attitude indicator (Figure 15-15). The pointer, when centered, indicates that the airplane pitch attitude corresponds to the best angle of attack for a determined airspeed.

With the pointer centered, the proper airspeed is established regardless of airplane gross weight, loading, configuration, or environmental conditions.

To produce the desired flight condition at a given airspeed:

- If the pointer moves toward the slow side (S), either reduce pitch attitude or add power (or both) to recenter the pointer.
- If the pointer moves toward the fast side (F), either increase pitch attitude or reduce power (or both) to recenter the pointer.

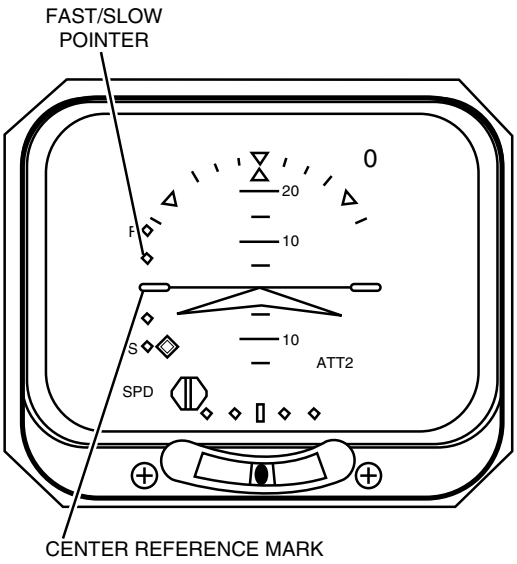


Figure 15-15. Fast/Slow Indicator

OPERATION

Stall warning system operation is enabled 5 seconds after aircraft takeoff to prevent unwanted operation on rotation.

Stall warning actuation is calibrated as a function of angle of attack, its rate of change, and flap position. (In the absence of flap position information, the FLAP and ADVANCED S.W.S. lights illuminate on the MAP, and the computer assumes flaps 45° for shaker actuation and flaps 0° for pusher actuation.)

Warning of an impending stall develops in the following sequence:

- Control column shaker (shaker actuation also disengages the autopilot)
- Aural warning (the clacker)
- Stick pusher forward

The stick pusher maintains forward pressure until the aircraft reaches a normal acceleration of 1/2 G, or the pilot disconnects the system. Pusher actuation is inhibited below 1/2 G.

In addition to the AP/PUSHER disconnect switches on each control wheel, a mechanical clutch allows the pilot to override pusher actuation.

To prevent a sudden pitch down during cruise, pressure switches in the pitot static system provide an input that inhibits pusher actuation above 173 knots and enables it below 153 knots.

Airplanes Pre Mod SB 120-027-0063: For flaps 0° and 45°, the fast/slow indication is calibrated for an airspeed of 1.3 V_S. The fast/slow indication is not valid for flaps 15 and 25°.

In the event of a flap control fault (indicated by flashing ADVANCE S.W.S., FLAP, and CONTROL FAULT lights), the fast/slow indication assumes flaps 0°. The fast/slow information is not valid.

Airplanes-Mod SB 120-027-0063: The fast/slow indication is valid for flaps 0°, 15°, 25°, and 45°. For flaps 0°, 15°, and 45°, the fast/slow indication is calibrated for an airspeed of 1.3 V_S. For flaps 25°, it is calibrated for an airspeed of 1.37 V_S.

The fast/slow indication disappears in the following cases:

- Flaps 0° and landing gear up and locked
- Flap control fault

DIGITAL LIGHT BAR **(COMD)** POSITION INDICATOR — Indicates position of flap selector lever.

RESET **(RST)** BUTTON —
- Clears all memories and faults
- Initializes all system
- Reinstates automatic control

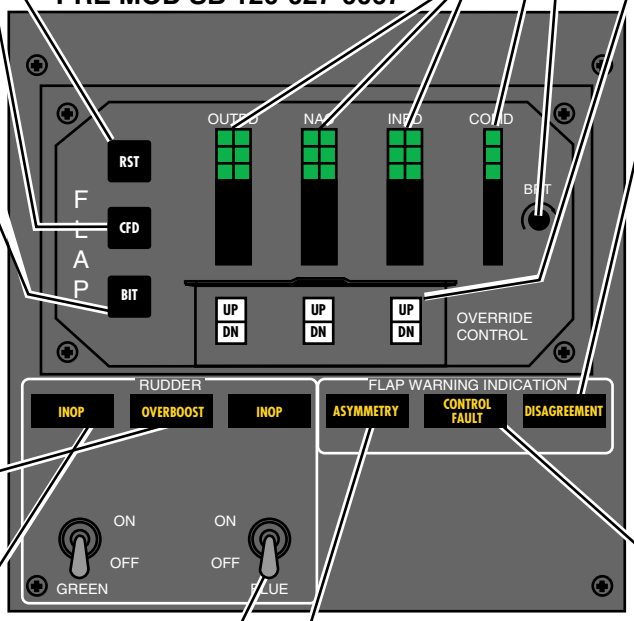
DIGITAL LIGHT BAR **(OUTBD, NAC, INBD)** POSITION INDICATOR — Indicates position of flap pair. Each light represents 5.625° of flap.

BRIGHTNESS **(BRT)** CONTROL KNOB — Controls intensity of light bars.

OVERRIDE CONTROL ROCKER SWITCHES (UP–DN)—Disengage automatic control of a flap pair and directly controls the flap actuator solenoids.

CANCEL FAULT DISPLAY **(CFD)** BUTTON — Stops light bar flashing (except during BIT)

PRE MOD SB 120-027-0067



BUILT IN TEST **(BIT)** BUTTON — Initiates 3-second system self-test (flap lever must be UP). A successful test is indicated by:
- Steady illumination of all light bars
- Illuminating of CONTROL FAULT light
- Flashing FLAP and ADVANCED SWS lights on MAP
- Pointer on analog flap position indicator raising approximately 1/4 inch above 0° position

DISAGREEMENT FAULT CAUTION LIGHT (AMBER)—Illuminates when flaps pairs differ in position by more than 7°. When illuminated :
- All light bars flash.
- Single-chime aural alert sounds.
- Amber FLAP light on MAP illuminates.
- Master CAUTION lights flash.

PRESSING the **CFD** button cancels flashing light bars.

OVERBOOST CAUTION LIGHT (AMBER) — Illuminates when both hydraulic actuators remain pressurized above 120 knots. Illuminates in conjunction with the RUDDER light on the MAP.

CONTROL FAULT CAUTION LIGHT (AMBER) — Illuminates when indicating failure in:
- Any of the three control unit channels
- The annunciator panel
- The selector lever

When illuminated:
- Affected light bar(s) flashes.
- Single-chime aural alert sounds.
- Amber FLAP and **ADVANCED SWS** lights on MAP illuminate.
- Master CAUTION lights flash.

GREEN SYSTEMS INOP LIGHT (AMBER) — Illuminates when corresponding switch is in **OFF** position or when the system is depressurized below 120 knots. It illuminates in conjunction with the RUDDER light on the MAP.

BLUE HYDRAULIC SYSTEM ISOLATION SWITCH —
- **ON:** Blue hydraulic system pressure is supplied to the rudder power system.
- **OFF:** Hydraulic pressure is supplied to the rudder power system.

ASYMMETRY FAULT CAUTION LIGHT (AMBER) — Illuminates when flap pair differs in position by more than 7°.
When illuminated:
- Affected flap pair light bar flashes.
- Single-chime aural alert sounds
- Amber FLAP light on MAP illuminates.
- Master CAUTION lights flash.

Pressure the **CFD** button cancels flashing lights bar.

NOTE:
Both **INOP** lights will only be illuminated above above 120 knots. if both rudder systems become depressurized

NOTE:
Once a control fault has occurred, the disagreement fault is masked to prevent one failure from triggering two alarms.

WARNING

If a fault condition is detected, the crew should take no action to correct the fault without reference to the appropriate checklist. Failure to comply with the approved checklist procedures may result in extreme asymmetries and difficulty with aircraft control.

PRE MOD SB 120-027-0067

Figure 15-16. Flight Control System Controls and Indicators



STALL WARNING **SYS 1/SYS 2** SWITCHES —
- ON: Activates stall warning system.
- OFF: Deactivates system. Computer lines to the shaker, electromagnetic clutch, and pusher servomotor are interrupted.

SYSTEMS 1/2 FAILURE WARNING LIGHT AND **TEST** SWITCH (RED) —
Illuminate when:
- Failure of affected system is detected (associated shaker and pusher are inhibited) with switch **ON** or **OFF**.
- The airplane is energized and the system has not been tested.
- The switch is **ON** and the system malfunctions. Illuminates in conjunction with the red STALL WARN light on MAP.

When pressed momentarily, it starts the test cycle. After 2.5 seconds:
- Fast/slow indicator moves toward slow.
- Shaker actuates.
- Aural warning (clacker) sounds.
- Pusher actuates.

Test cycle takes 5 seconds. If the test passed:
- STALL WARN and failure warning lights will extinguish.

If the test failed:
- STALL WARN and failure warning lights remain illuminated.
- Voice message “STALL WARNING” is given.
- Shaker and pusher are inhibited.

NOTE:
On airplanes Mod 120-031-0003 or 120.035 and subsequent, the STALL WARNING voice message is inhibited on the ground.
If the test is not performed on the ground, failure warning and STALL WARN lights will remain illuminated during the flight, but the system will function normally.

AP/PUSHER DISCONNECT SWITCH (RED) —
When pressed, both pusher servos are inhibited.

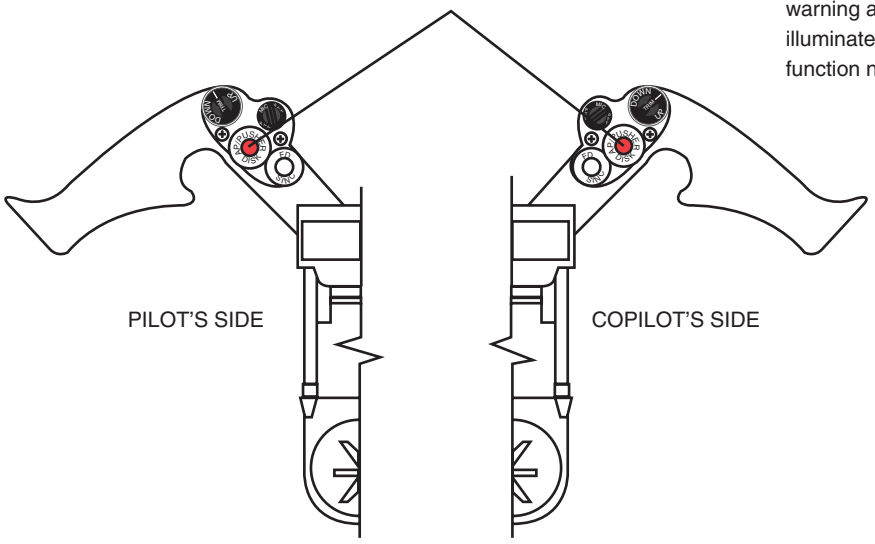


Figure 15-17. Stall Warning System Controls and Indicators

SYSTEM TEST

The stall warning subsystems are tested individually by depressing the respective TEST button on the stall warning panel.

The TEST buttons are lighted if a self-test has not been performed since the last power-up. They also illuminate with a stall warning system failure or fault.

Pressing a TEST button initiates the following self-test sequence after 2.5 seconds:

- Fast/slow indicator moves toward slow
- Column shaker actuates
- Aural warning (clacker) sounds
- Stick pusher actuates

If a self-test is passed, the TEST button and STALL WARN lights extinguish.

Indications if subsytem fails self-test are:

- TEST button and STALL WARN lights remain illuminated
- Voice message “Stall Warning” is given
- Column shaker and stick pusher are inhibited

System test is possible only on the ground due to landing gear switch logic. Each subsystem must be tested individually.

Should either subsystem fail in flight:

- The respective TEST button and STALL warning light on the MAP illuminate
- Voice message “Stall Warning” is given

QUESTIONS

1. The EMB-120 has _____ rudders.
A. One
B. Two
C. Three
D. Two plus one

2. The rudder system may be actuated by:
A. The green and blue hydraulic systems
B. The green or blue hydraulic system
C. Mechanically by a push rod
D. All of the above

3. The hydraulic rudder system may be turned ON or OFF by the pilot.
A. True
B. False

4. During normal operation above 120 knots, the ____ hydraulic system operates the rudder.
A. Blue
B. Green

5. If airspeed is above 120 knots and both hydraulic systems remain connected to the rudder, what cockpit indication will the pilot see?
A. RUDDER INOP light
B. OVERBOOST light
C. No light at all
D. Both A and B

6. On the ground, the rudder is held securely in place with the gust lock.
A. True
B. False

7. Rudder trim is:
A. A hydraulic function
B. A mechanical function
C. Electric motor driven
D. No rudder trim available

8. Ailerons are actuated by means of:
A. A hydraulic assembly
B. A cable and rod assembly
C. An electric motor
D. Combination of A and B

9. The left and right aileron may be operated independently by disconnecting one from the other in case of a stuck or jammed aileron.
A. True
B. False

10. With the disconnect system actuated, the left aileron jammed, and the autopilot engaged, the aircraft is controllable with the autopilot.
A. True
B. False

11. The autopilot servo is connected to:
A. The right aileron actuator
B. The left aileron actuator
C. Both aileron actuators
D. Neither aileron actuator

12. With the disconnect system actuated, the copilot's control wheel operates:
A. The left aileron
B. The right aileron
C. Either left or right aileron (depending on altitude)
D. Neither left or right aileron

13. The elevator is actuated by means of:

A. A hydraulic assembly
B. A cable and rod assembly
C. A combination of A and B
D. Bells and whistles

14. If the elevator was jammed and the disconnect system actuated and only the pilot's side was free, the stall warning system would still be completely operational.

A. True
B. False

15. Trim is available in:

A. Roll
B. Pitch
C. Yaw
D. All of the above

16. The airplane is equipped with ____ flap panels per wing.

A. One
B. Two
C. Three
D. Four

17. Flaps are actuated by:

A. A hydraulic assembly
B. A cable and rod assembly
C. An electric motor/rod assembly
D. Both A and B

18. Flap setting positions are:

A. 0 and 15°
B. 15 and 25°
C. 25 and 45°
D. All of the above

19. The outboard pair of flaps is actuated by the blue hydraulics system.

A. True
B. False

20. An asymmetrical flap condition occurs when:

A. One pair of flaps does not respond to a control signal.
B. Two pairs of flaps do not respond to a control signal.
C. Two flap panels in the same pair differ in position of more than 7°.
D. One flap pair jams on the way down or up.

21. A flap disagreement condition occurs when:

A. One pair of flaps does not respond to a control signal.
B. Two pairs of flaps do not respond to a control signal.
C. Two pairs of flaps differ by more than 7°.
D. All of the above

22. What method or indication does the pilot have to determine a flap position malfunction?

A. View the analog flap position indicator.
B. A horn sounds in the cockpit.
C. View the FLAP annunciator panel.
D. Both A and C

CHAPTER 16
AVIONICS

CONTENTS

Page

INTRODUCTION 16-1
GENERAL 16-1
FLIGHT INSTRUMENTS 16-3
- General 16-3
- Pitot-Static Instruments 16-3
- Electric Instruments 16-8
- Miscellaneous Instruments 16-10
- Electronic Flight Instrument System (EFIS) 16-11

ATTITUDE ANDHEADING REFERENCE SYSTEM (AHRS) 16-22
- General 16-22
- Components 16-22
- System Operation 16-23
- System Failures 16-24

NAVIGATION SYSTEM 16-25
- General 16-25
- Components 16-25
- Navigation System/EFIS Operation 16-38

COMMUNICATIONSYSTEM 16-47
- General 16-47
- Components 16-47

AUTOFLIGHT 16-52

General 16-52
Components 16-55
Operating Modes 16-60
Points to Remember 16-65
GROUND PROXIMITY WARNING SYSTEM 16-67
General 16-67
GPWS Modes 16-67
GPWS Aural Warning Message Priorities 16-72
GPWS Controls and Indicators 16-72
TRAFFIC ALERT AND COLLISION AVOIDANCE SYSTEM (TCAS I)—OPTIONAL 16-73
General 16-73
System Description 16-73
TCAS Self-Test 16-75
TCAS Control Display Unit 16-75
Display Symbols 16-75
FLIGHT RECORDERS 16-75
Cockpit Voice Recorder (Optional) 16-75
Flight Data Recorder System (Optional) 16-77
ELECTRONICS BAY VENTILATION SYSTEM 16-79
Components 16-79
Operation 16-81
QUESTIONS 16-82

ILLUSTRATIONS

Figure	Title	Page
16-1	EMB-120 Avionics	16-2
16-2	Pitot-Static System Schematic	16-4
16-3	Airspeed Indicator	16-5
16-4	Encoding and Nonencoding Altimeters	16-6
16-5	Altitude Alerter Preselect Panel	16-7
16-6	Altitude Alerter Operation	16-7
16-7	Vertical Speed Indicator	16-8
16-8	Standby Horizon	16-8
16-9	Turn-and-Bank Indicator	16-9
16-10	Standby Compass	16-10
16-11	Outside AirTemperature Indicator	16-10
16-12	Digital Clock Timer	16-10
16-13	Electronic Flight Instrument System (EFIS)	16-11
16-14	EADI Normal Display	16-12
16-15	EADI Composite Format	16-13
16-16	EADI Failure Indications and Flags	16-13
16-17	EHSI Full Compass Rose (HSI) Format	16-14
16-18	EHSI ARC Format	16-15
16-19	EHSI MAP Format	16-16
16-20	EHSI Failure Indications and Flags	16-16
16-21	Multifunction Display (MFD)	16-17
16-22	MFD—Page Checklist (Typical)	16-18
16-23	EFIS CONTROL PANEL	16-19

16-24 Display Control Panel (DCP) 16-20
16-25 Weather Radar Panel (WXP) 16-21
16-26 Course Heading Panel (CHP) 16-21
16-27 AHRS Schematic 16-22
16-28 Compass Compensator Unit 16-23
16-29 Navigation System 16-25
16-30 Course Heading Panel (CHP) 16-26
16-31 Weather Radar Panel (WXP) 16-27
16-32 VHF NAV Radio Control 16-29
16-33 ADF Radio Control 16-32
16-34 Radio Magnetic Indicator (RMI) 16-34
16-35 DME Indicator 16-35
16-36 Transponder Control 16-37
16-37 EHSI ARC Format 16-41
16-38 EHSI MAP Format 16-42
16-39 EHSI Failure Indications and Flags 16-43
16-40 EADI Normal Format 16-44
16-41 EADI Composite Format 16-46
16-42 EADI Failure Indications and Flags 16-46
16-43 Communication System 16-47
16-44 VHF COM Radio Control 16-49
16-45 Control Wheel MIC Switch 16-51
16-46 Attendant's Interphone 16-51
16-47 Observer's Audio Panel 16-51
16-48 Ramp Connection 16-51
16-49 SELCAL System 16-52

16-50 HF Panel 16-52
16-51 Autoflight System 16-53
16-52 Flight Control Panel 16-55
16-53 Autopilot Panel 16-57
16-54 Pitch Trim Switch 16-58
16-55 Autopilot/Pusher Disconnect Button 16-59
16-56 Autopilot SYNC Button 16-59
16-57 AP Engage/YAW Annunciators 16-60
16-58 Autopilot Transfer Switches 16-60
16-59 GPWS Mode 1 16-67
16-60 GPWS Mode 2 16-68
16-61 GPWS Mode 3 16-69
16-63 GPWS Mode 4B 16-70
16-64 GPWS Mode 5 16-71
16-65 GPWS Controls and Indicators 16-71
16-66 GPWS FLAP OVRD Button/INOP Annunciator 16-72
16-67 TCAS Schematic 16-73
16-68 TCAS Control Display Unit (CDU) 16-74
16-69 Cockpit Voice Recorder 16-77
16-70 Flight Data Entry Panel 16-78
16-71 Electronics Bay Ventilation System 16-80

TABLES

Table Title Page
16-1 Range Rings 16-75
16-2 Display Symbols 16-76

CHAPTER 16
AVIONICS



INTRODUCTION

The EMB-120 avionics covered in this chapter includes the flight and navigation instruments, the standard Collins communication package available for installation, and the fully integrated Collins flight control system. Also included is the ground proximity warning system (GPWS), cockpit voice and flight data recorder systems, and a description of the electronics bay ventilation system.

The user should consult applicable AFM supplements and vendor manuals for additional material or information on systems not included in this chapter.

GENERAL

All of the avionics displays and control panels are located on the forward instrument panel, either in front of the pilot and copilot or on the panel center section (Figure 16-1).

The autopilot controls are located on the center console, aft of the throttle quadrant.

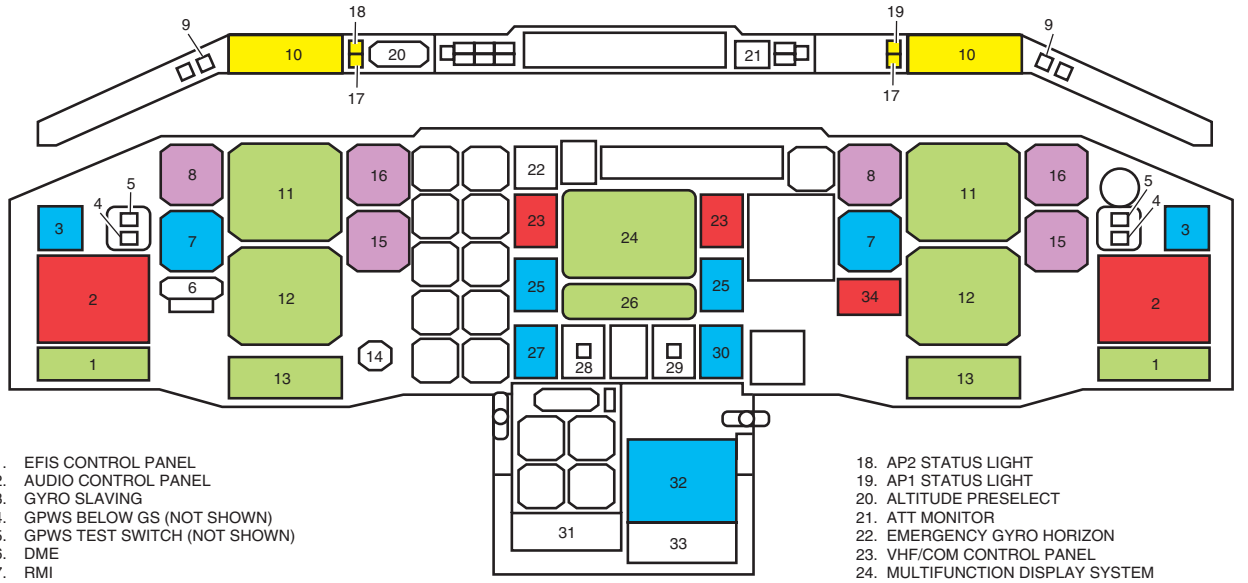
The Collins EFIS-85 electronic flight instrument system provides the primary attitude and heading reference information as well as displaying navigation information.

The communication system provides for voice communications between crewmembers, passengers, and ground stations.

The standard automatic flight control system integrates the air data system, autopilot and flight director, the EFIS and attitude heading reference system (AHRS), and the standby instruments. A conventional flight control system without EFIS is optional but is not covered in this chapter. With either system, the flight director function may be used with or without the autopilot coupled.

The yaw damper system operates independently of the autopilot.





1. EFIS CONTROL PANEL
2. AUDIO CONTROL PANEL
3. GYRO SLAVING
4. GPWS BELOW GS (NOT SHOWN)
5. GPWS TEST SWITCH (NOT SHOWN)
6. DME
7. RMI
8. AIRSPEED INDICATOR
9. CAT II SWITCH
10. AP/FD MODE SELECTOR ANNUNCIATOR
11. ELECTRONIC ATTITUDE DIRECTOR INDICATOR (EADI)
12. ELECTRONIC HORIZONTAL SITUATION INDICATOR (EHSI)
13. DISPLAY CONTROL PANEL (DCP)
14. OUTSIDE AIR TEMPERATURE INDICATOR
15. VERTICAL SPEED INDICATOR
16. ALTIMETER
17. AP TRANSFER LIGHT
18. AP2 STATUS LIGHT
19. AP1 STATUS LIGHT
20. ALTITUDE PRESELECT
21. ATT MONITOR
22. EMERGENCY GYRO HORIZON
23. VHF/COM CONTROL PANEL
24. MULTIFUNCTION DISPLAY SYSTEM
25. VHF/NAV CONTROL PANEL
26. WEATHER RADAR CONTROL UNIT
27. ADF CONTROL PANEL
28. GPWS FLAP OVRD BUTTON (NOT SHOWN)
29. GPWS INOP ANNUNCIATOR (NOT SHOWN)
30. TRANSPONDER
31. FLIGHT RECORDER ENTRY PANEL—OPTIONAL
32. OMEGA/RNAV CONTROL PANEL—OPTIONAL
33. VOICE RECORDER—OPTIONAL (NOT SHOWN)
34. SELCAL—OPTIONAL (NOT SHOWN)

LEGEND

- PITOT STATIC
- EFIS
- NAVIGATION
- COMMUNICATION
- AFCS

Figure 16-1. EMB-120 Avionics

FLIGHT INSTRUMENTS

GENERAL

The EMB-120 flight instruments may be grouped into the following four major categories based on their source of input and method of operation:

- Pitot-static instruments
- Electric instruments
- Miscellaneous instruments
- (EFIS) Electronic Flight Instrument System instruments

PITOT-STATIC INSTRUMENTS

Pitot-Static System

The pitot-static system (Figure 16-2) provides the impact (dynamic) and static air pressures for flight instruments and airplane systems that operate on the principle of differential air pressure.

The system consists of three independent pitot-static circuits, each connected to one of the following pitot tubes:

- Pilot's pitot tube—located on the left side of the aircraft nose
- Copilot's pitot tube—located on the right side of the aircraft nose
- Auxiliary pitot tube—located on the right side of the aircraft nose beneath the copilot's pitot tube

Each electrically heated pitot tube contains two static pressure sources in addition to its dynamic pressure source. The tubes are aerodynamically balanced to minimize system errors.

Pitot Lines

The pilot's pitot line provides dynamic pressure for the pilot's airspeed indicator.

The copilot's pitot line provides dynamic pressure for the following:

- Copilot's airspeed indicator
- Two air data sensors (ADSs)
- One airspeed pressure switch

The auxiliary pitot line provides dynamic pressure for the following:

- Two ADSs
- One airspeed pressure switch
- Flight data recorder (not shown)
- TAS computer or ADS (if installed)

Static Lines

The static system consists of six static pressure lines: a primary and secondary on each of the three pitot tubes.

The pilot's primary static line supplies static pressure for the pilot's:

- Airspeed indicator
- Altimeter
- VSI
- Rad-bar altimeter (if installed)

The copilot's primary static line supplies static pressure for the copilot's:

- Airspeed indicator
- Altimeter
- VSI
- Rad-bar altimeter (if installed)

The auxiliary primary static line supplies static pressure for the following:

- Two ADSs
- Airspeed pressure switches

- Altitude switches
- Flight data recorder (not shown)
- TAS computer or ADS (if installed)

The secondary static lines (one from each pitot tube) provide static pressure for the following:

- Two ADSs
- Cabin differential pressure
- Airspeed pressure sensors
- Altitude switches

Altitude/Airspeed Pressure Switches

Four altitude pressure switches are connected to the static lines, two set for 10,000 feet and two set for 14,000 feet. The 10K and 14K altitude switches are used by the EEC for controlling HMU solenoid operation and warning.

The two airspeed pressure switches provide output signals to the landing gear and stall warning systems and are also used in controlling the rudder hydraulic system.

Air Data Sensors (ADSs)

Two air data sensors (ADSs) are connected to both pitot and static lines to provide altitude and indicated airspeeds to the autopilot computer. A third air data sensor is installed when the airplane is equipped with a flight data recorder system.

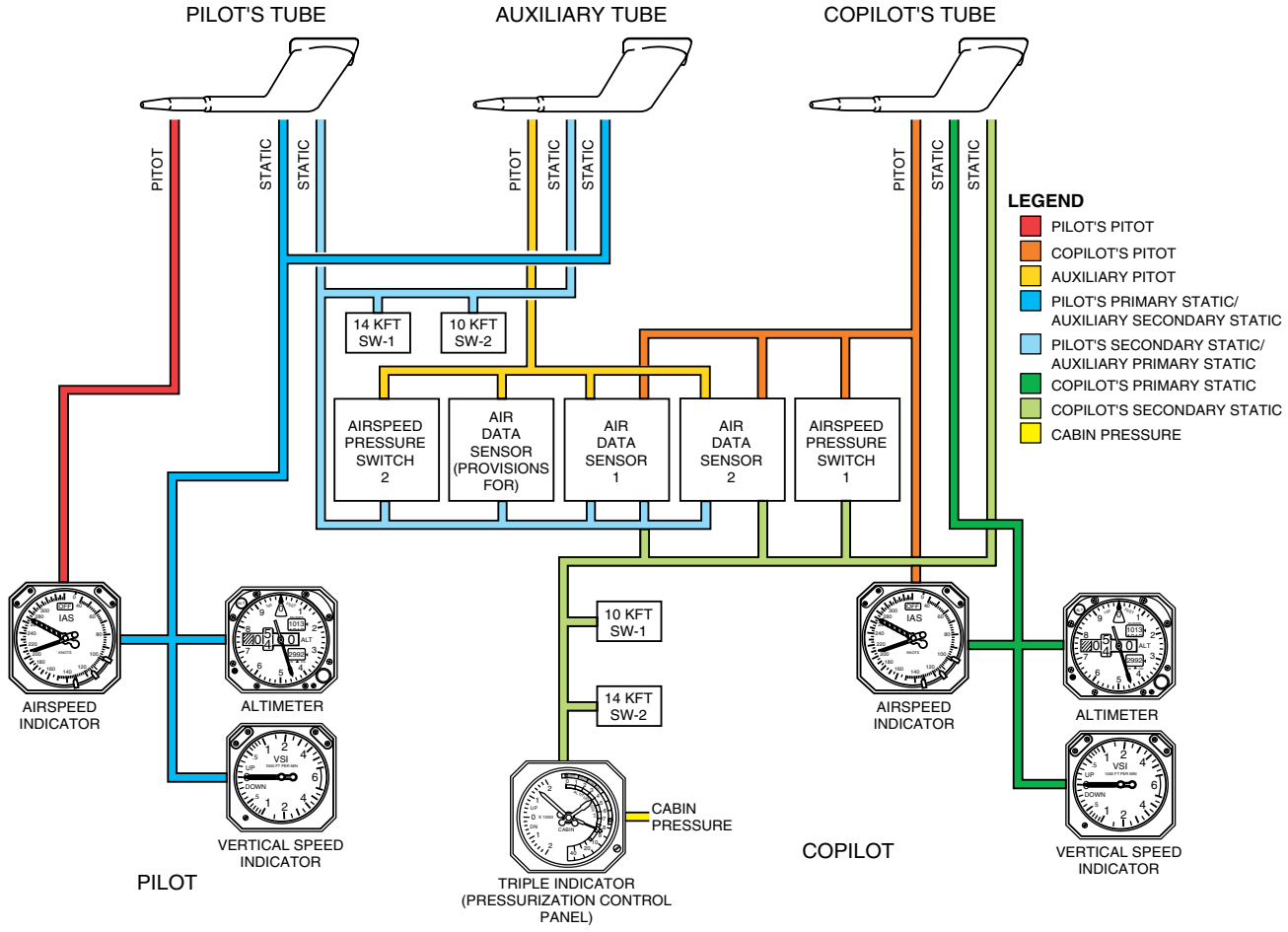


Figure 16-2. Pitot-Static System Schematic

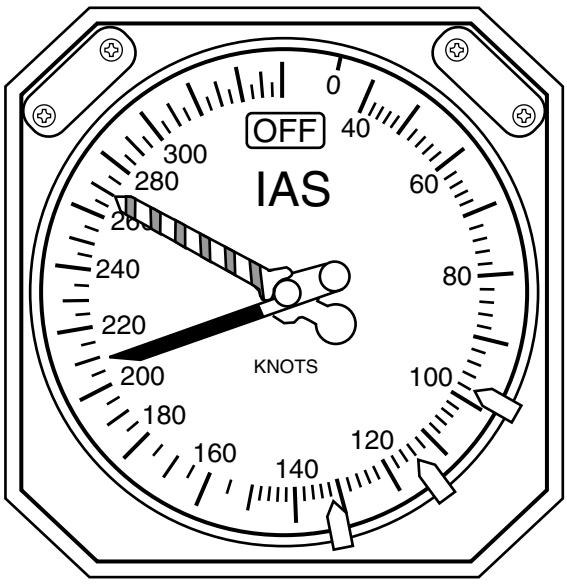


Figure 16-3. Airspeed Indicator

Airspeed Indicators

The aircraft is equipped with two identical airspeed indicators (Figure 16-3), each installed in the pilot's and copilot's panel.

The indicators contain an airspeed pointer, VMO indicator, airspeed markers, and an OFF flag.

The pointer is a needle that rotates to indicate airspeed in knots on a dial-face scale. The scale is in 2-knot increments from 40 to 150 knots, and 5-knot increments from 150 to 360 knots. The airspeed is labeled at 20 knot increments.

The VMO indicator is a striped pointer that also rotates. Its position varies with altitude and provides the maximum allowable airspeed at any given altitude.

The airspeed markers (bugs) are used to mark reference airspeeds. They are located on the bezel of the dial face and are manually set.

If the indicator loses power, the word OFF appears in the OFF flag window.

Altimeters

General

The aircraft incorporates two types of altimeters in its flight instrument package, an encoding (servo) altimeter, and a nonencoding altimeter (Figure 16-4).

The pilot's panel contains the encoding altimeter and the copilot's panel contains the nonencoding altimeter. As an option, an encoding altimeter may be installed in the copilot's panel.

Encoding altimeters provide a Mode C output to the ATC transponder.

Encoding Altimeter

The encoding altimeter contains the following controls and indicators:

- Altimeter setting control
- Inches-of-mercury setting window
- Millibars setting window
- 100-foot pointer
- Digital counter
- Altitude alert light
- Failure warning flag

Altimeter Setting Control

The altimeter setting control, in the lower right corner of the instrument, is used to set the barometric setting into the altimeter.

Inches-Of-Mercury Setting Window

The inches-of-mercury setting window, located below the digital counter, displays the set barometric pressure in inches of mercury.

Millibars Setting Window

The millibars setting window, located above the digital counter, displays the set barometric pressure in millibars.

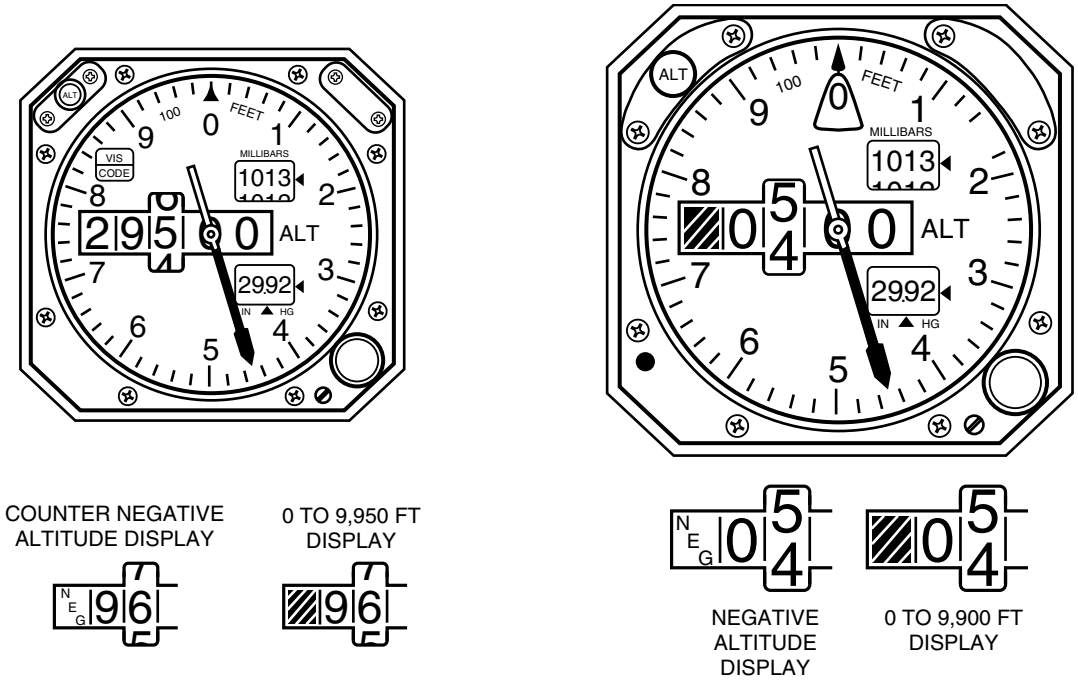


Figure 16-4. Encoding and Nonencoding Altimeters

100-Foot Pointer

The 100-foot pointer is the needle that rotates on the altimeter face. The scale on the altimeter face is graduated in 20-foot increments and numbered each 100 feet.

Digital Counter

The digital counter, located in the center of the dial face, contains five digits used to provide the altitude readout.

Reading from left to right, the first two digits display the altitude in thousands of feet, and the third digit displays the altitude in hundreds of feet. The two numbers on the extreme right are fixed at 00.

The counter indicates negative altitude by displaying the letters NEG in the first window. At altitudes from 0 to 9,950 feet, the first window displays a series of bars.

Altitude Alert Light

The illumination of the altitude alert light is described in the altitude alerter system section. It is redundant with ALT lights on the nonencoding altimeter and the altitude preselect indicator.

Failure Warning Flag

The failure warning flag appears if there is a loss of the vibrator or a loss of the encoding information.

The warning flag, consisting of the words VIB and CODE, is located in the upper left corner of the dial face.

Nonencoding Altimeter

Nonencoding altimeters contain the same controls and indicators as the encoding altimeters except the warning flag is colocated with the zero altitude indices. The warning flag appears to indicate loss of power or abnormal servo operation.

The altimeter may also include a press-to-test button which, when pressed, causes an increase of 550 ±250 feet in the altimeter indication and displays the failure warning flag.

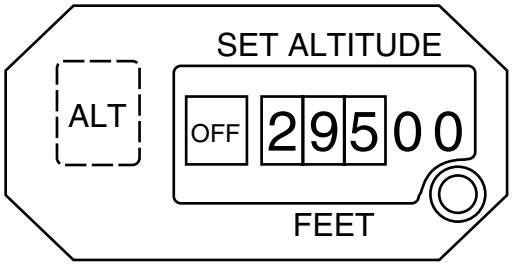


Figure 16-5. Altitude Alerter Preselect Panel

Altitude Alerter System

The altitude alerter system provides visual and aural warnings to alert the pilot as the airplane approaches a preselected flight altitude.

The system receives altitude information from the servo altimeter and compares it with the altitude set on the altitude alerter preselect panel (Figure 16-5).

The visual warnings include altitude alerter lights and altimeters.

The aural warning includes:

- Three 2,900 Hz tones (Pre Mod SB 120-031-0018) or,
- "Altitude Alert" voice message (Mod SB 120-031-0018 or SN 120.114 and subsequent)

These visual and aural warnings are triggered when:

- The aircraft is 400 feet from the selected altitude. (The lights remain on until 200 feet from the selected altitude.)
- The aircraft altitude deviates approximately 200 feet from the selected altitude. The lights reilluminate and the audio signal is reactivated. (The lights remain on until the airplane reenters the 200-feet range, or a new altitude is selected.)

Figure 16-6 depicts the altitude alerter system operational sequence.

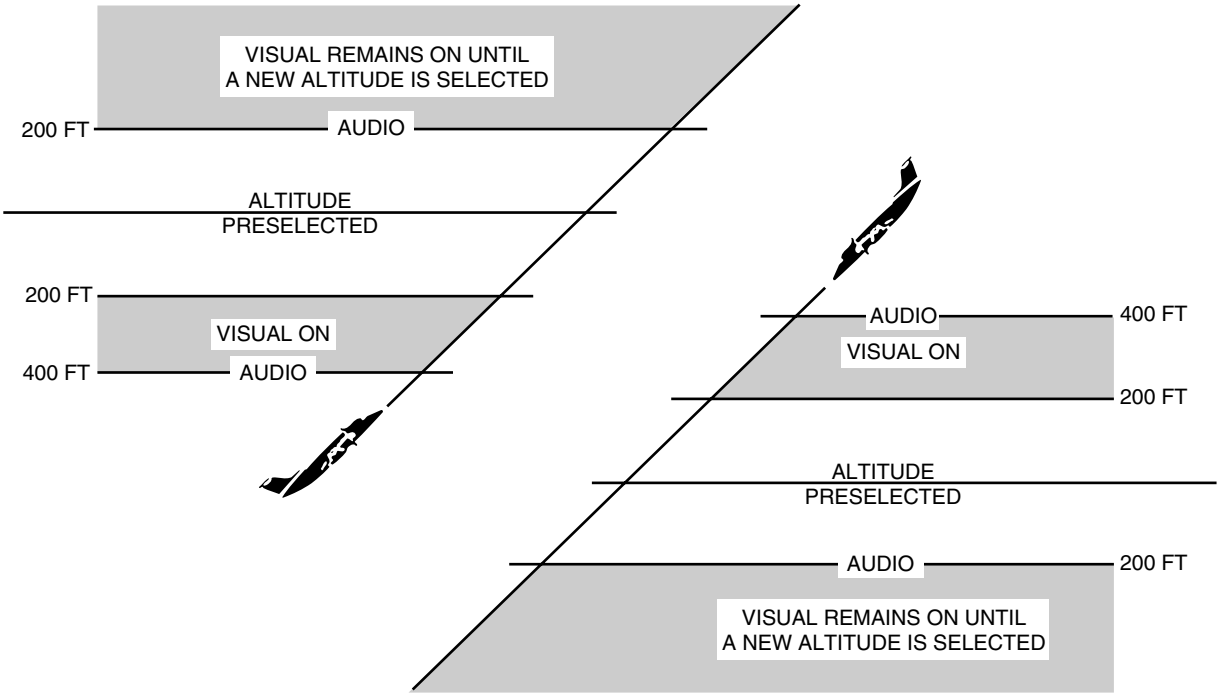


Figure 16-6. Altitude Alerter Operation

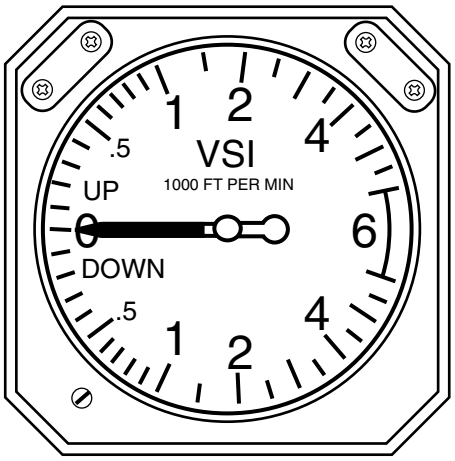


Figure 16-7. Vertical Speed Indicator

Vertical Speed Indicator (VSI)

General

The aircraft may be equipped with either of two vertical speed indicators in its flight instrument package: a standard vertical speed indicator (VSI) (Figure 16-7) or an instantaneous vertical speed indicator (IVSI).

The difference between the two is an accelerometer incorporated in the IVSI to reduce pointer response time.

Normal configuration is a VSI installed in both the pilot's and copilot's panel. The IVSI is optional.

VSI and IVSI Components

The VSI and IVSI are nearly identical in appearance. The VSI has the letters "VSI" printed on the dial face, while the IVSI has the word "CLIMB."

Both instruments incorporate a zero adjustment screw and a vertical speed indicator pointer.

The zero adjustment screw, in the lower left corner of the instrument, is used to calibrate the vertical speed indicator pointer to zero.

The vertical speed indicator pointer is a needle that deflects up or down from 0. The scale on the dial face goes from 0 to 6,000 feet in both directions. It is in 100-foot increments from 0 to 1,000 feet, and 500-foot increments from 1,000 to 6,000 feet.

ELECTRIC INSTRUMENTS

Standby Horizon

General

The standby horizon (Figure 16-8) provides a visual indication of airplane attitude. It functions as a back-up in the event the main attitude system fails.

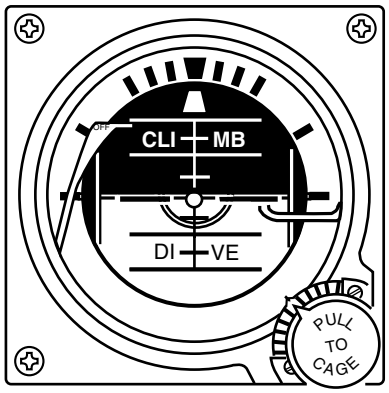


Figure 16-8. Standby Horizon

Located on the main instrument panel, the standby horizon is powered by 28 VDC from Emergency DC Bus 1. It is also connected to the emergency battery, which supplies power for at least 30 minutes in the event of a total electrical generation system failure.

The standby horizon incorporates a rotor speed and mechanical erection system that enables it to provide at least nine minutes of valid attitude information in the event of a complete electrical failure.

Power Warning Flag

When the red power warning flag is in view, it indicates either of the following:

- Caged condition
- Open motor winding
- Power failure

Operation

It is recommended that the standby horizon be cage-locked before energizing the airplane. (Pulling the caging/pitch-trim knob cages the gyro. Once full aft, rotating the knob clockwise "cage-locks" the gyro.) It should be uncaged after the airplane is energized.

The indication should be completely stabilized within 3 minutes after releasing the knob.

When fully released, rotating the knob adjusts the miniature airplane's pitch attitude.

Caging during flight, although not normally required, should be done in level flight.

The standby horizon should be cage-locked after the airplane is deenergized during shutdown.

NOTE

Never cage-lock an operating standby horizon while the airplane is in motion.

Turn-and-Bank Indicator

General

The turn-and-bank indicator (Figure 16-9) provides turn rate and slip-skid information. It is a conventional ball-and-pointer instrument calibrated for two-minute turns.

The turn-and-bank indicator is located on the pilot's panel below the vertical speed indicator and is powered by 28 VDC from DC Bus 1.

Warning Flag

When the turn-and-bank indicator is not energized, the warning flag is orange. When the indicator is energized the flag is black.

Radio Altimeters

General

The EMB-120 normally has a radio altitude system incorporated within the electric flight instrument system (EFIS). Operation of this system is covered in the EFIS section.

When the standard flight instrument system (FIS) is installed, the aircraft may be equipped with one or two radio altimeters, located on the related pilot's and copilot's instrument panels. Refer to vendor manuals for specific instrument operation.

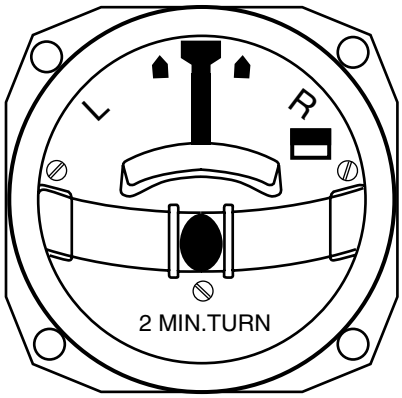


Figure 16-9. Turn-and-Bank Indicator

MISCELLANEOUS INSTRUMENTS

Standby Compass

The standby compass (Figure 16-10) is a standard magnetic compass mounted along with the compass correction card on the windshield center post. An alternate compass correction card is located on the lower part of the pilot's main panel for use during electrical emergency conditions.

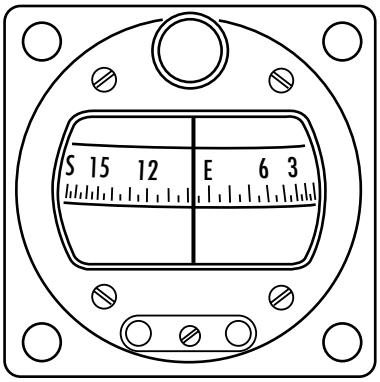


Figure 16-10 Standby Compass

Outside Air Temperature Indicating System

The outside air temperature indicating system uses a thermistor, located on the external skin of the right forward front fuselage, to sense outside air temperature.

The temperature is displayed in degrees centigrade on an LCD indicator (Figure 16-11), located on the pilot's instrument panel.

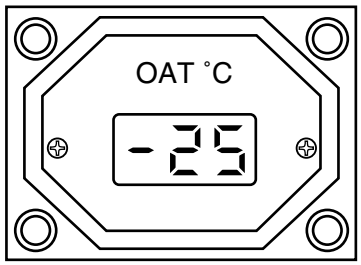


Figure 16-11. Outside Air Temperature Indicator

Clocks

General

The aircraft is equipped with two digital clock-timers that indicate the time, date, and operate as a chronometer (Figure 16-12).

The airplane may also be equipped with two optional mechanical clocks, or a Davtron-800 digital clock.

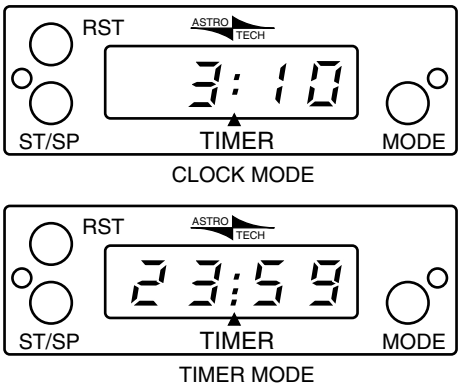


Figure 16-12. Digital Clock Timer

Digital Clock-Timers

The digital clock-timers are mounted on the pilot's and copilot's control wheels. They utilize a four digit LCD and include: a MODE button to select clock or timer operation and an ST/SP button used with an RST button to set the time and date.

Mechanical Clocks

In addition to the digital clock-timers, optional mechanical clocks may be installed on the pilot's and copilot's instrument panel.

Digital Clock (Davtron M-800)

The optional Davtron M-800 digital clock uses an LCD display to indicate local time, GMT, and elapsed time. It is located on the pilot's instrument panel.

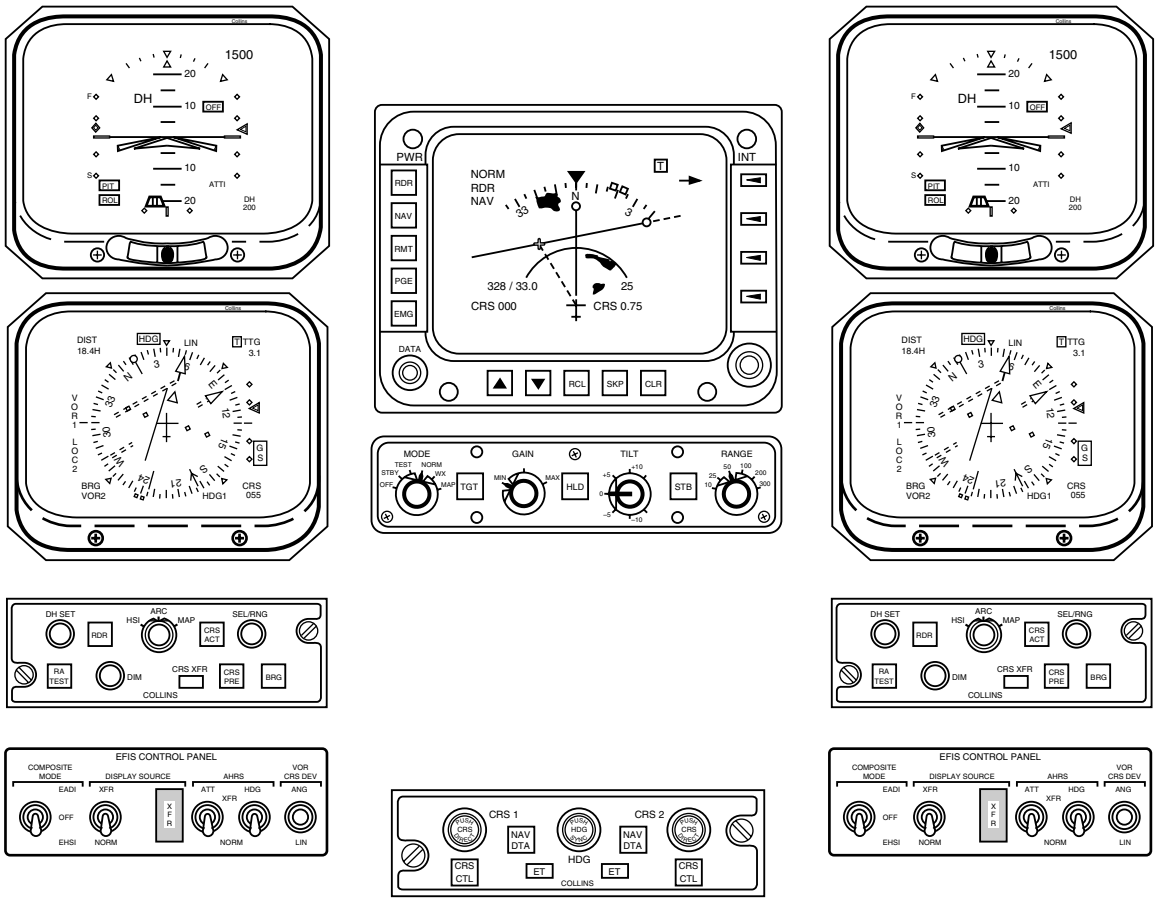


Figure 16-13. Electronic Flight Instrument System (EFIS)

ELECTRONIC FLIGHT INSTRUMENT SYSTEM (EFIS)

General

The electronic flight instrument system (EFIS) is a digital flight instrument system that utilizes cathode ray tubes for display (Figure 16-13).

In the EFIS, the conventional electro-mechanical ADI and HSI are replaced by cathode ray tubes and become an electronic ADI (EADI) and an electronic HSI (EHSI).

Another cathode ray tube replaces the standard radar display, and is further used to display navigation information and flight data.

Components

The EFIS consists of five displays:

- Four identical panel-mounted cathode ray tubes, or electronic flight displays (EFDs) (used as the EADIs and EHSIs)
- A larger cathode ray tube, the multi-function display (MFD) mounted on the center panel (used to display the radar, navigation, and flight information)

The EFIS also includes six control panels:

- Two EFIS CONTROL PANELs
- Two display control panels (DCPs) (controlling displays on the individual EADIs and EHSIs)

- A weather radar panel (WXP)
- A course heading panel (CHP)

The EFIS operates via three remotely-mounted (electronic equipment bay) processor units that receive and process all data inputs for display:

- Two display processor units (DPUs), one for each EADI/EHSI pair
- A multifunction processor unit (MPU) for the multifunction display (MFD)

Displays

The EADI, EHSI, and the MFD are each capable of displaying information from multiple sources. This initial discussion of the EFIS displays covers only the basic instruments and their primary displays.

Navigation inputs to the EFIS and the associated displays are covered separately in the Navigation section of this chapter.

The EADI and EHSI are multicolor cathode ray tube displays. They display airplane attitude and a plan view of the horizontal navigation situation, respectively.

The MFD is a multicolor cathode ray tube display unit that mounts in the instrument panel in the space normally provided for the weather radar indicator. Standard functions displayed by the unit include weather radar, a pictorial navigation map, and page data.

EADI

The EADI has two basic formats: normal (Figure 16-14) and composite (Figure 16-15), including multiple failure indications and flags (Figure 16-16). The composite format displays a portion of the EHSI display on the EADI in the event of an EHSI failure.

Attitude sensor annunciation is displayed at all times in the lower right portion of the EADI (ATT1/ATT2). On-side sensor annunciation is white and cross side is yellow.

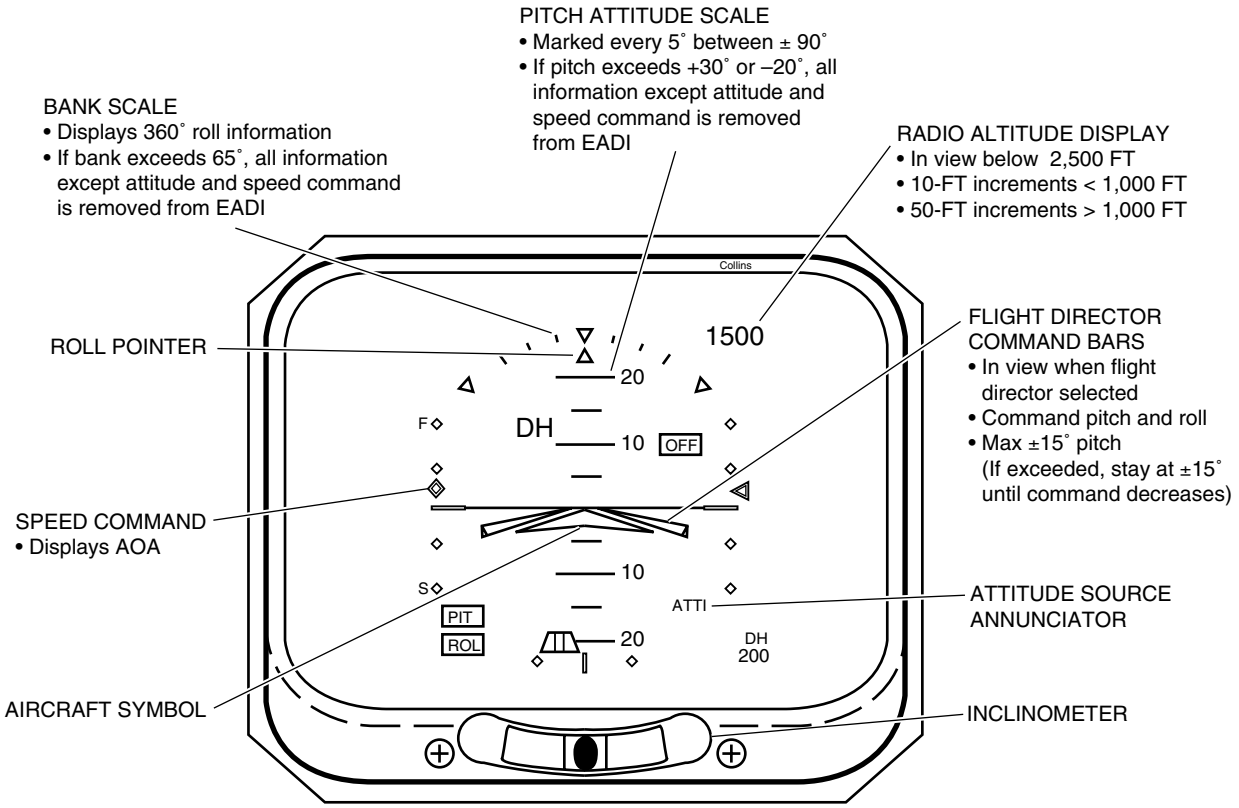


Figure 16-14. EADI Normal Display

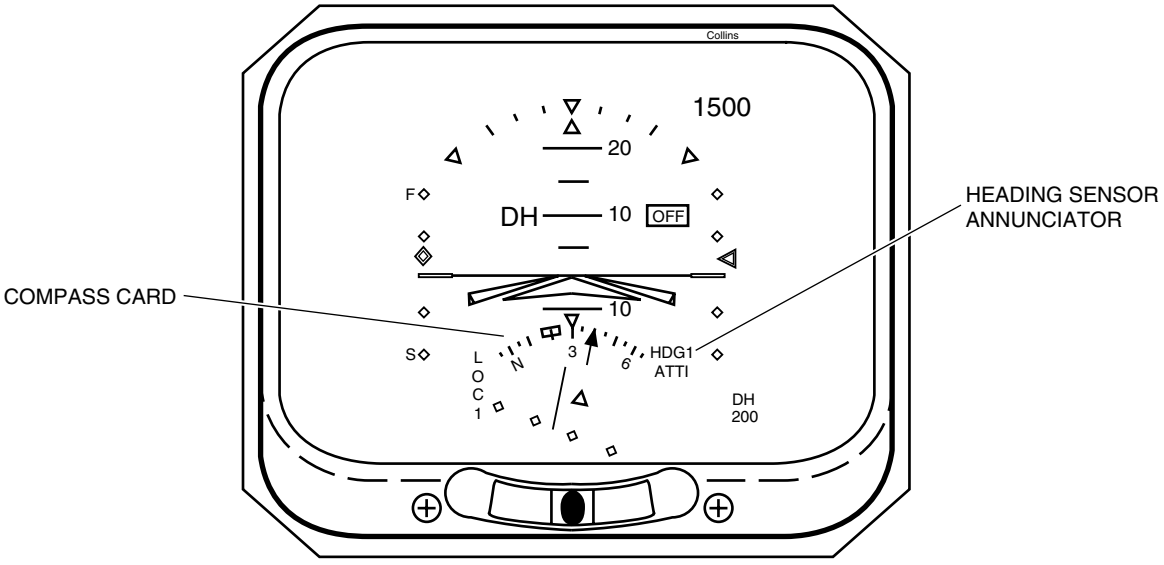


Figure 16-15. EADI Composite Format

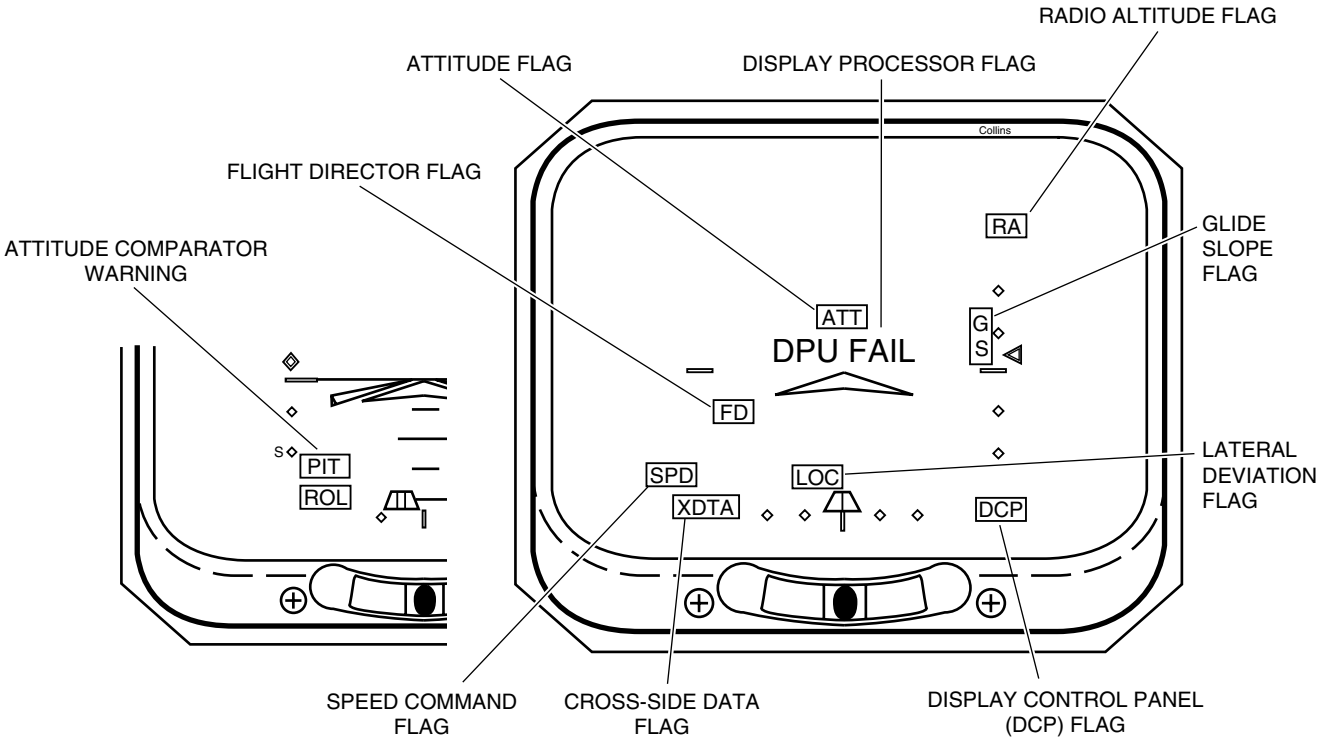


Figure 16-16. EADI Failure Indications and Flags

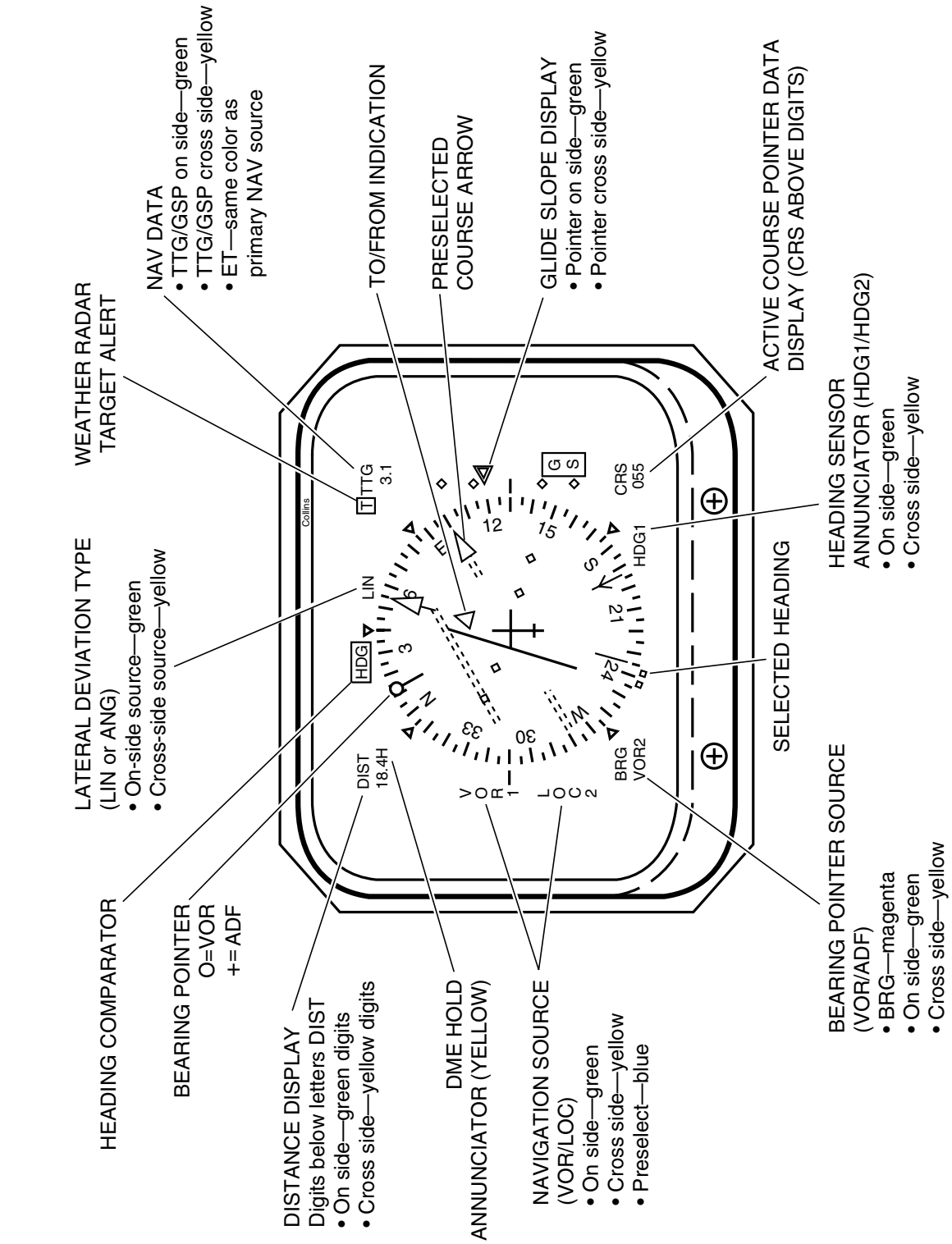


Figure 16-17. EHSI Full Compass Rose (HSI) Format

NOTE

For airplanes Mod SB 120-034-0064 or SN 128 and subsequent, the attitude sensor annunciator is displayed only when the attitude information is provided by the opposite attitude and heading reference system (AHRS). When selected, the yellow annunciator (ATT1 or ATT2) appears on both EADIs.

ATTITUDE COMPARATOR WARNING—If a pitch or roll error in excess of 6° is detected, the PIT or ROL boxed inscription appears (flashing) in the lower left corner of the EADI with the following additional indications:

- A single-chime alert sounds.
- The amber ATTITUDE DISPLAY light on the MAP flashes.
- The master CAUTION lights flash.

Pressing the ALARM CANCEL button or the ATT/HDG MONITOR button on the glareshield stops the lights flashing; however, they remain on until the failure is eliminated.

EHSI

The EHSI displays indications in three formats as selected on the display control panel:

- HSI—full compass rose (Figure 16-17)
- ARC—expanded compass sector (Figure 16-18)
- MAP—expanded compass sector with weather radar information superimposed (Figure 16-19)

EHSI failure indications and flags are shown in Figure 16-20.

On both the EHSI and the EADI, all flags are red and, except for DPU FAIL, flash for 10 seconds before becoming steady. The DPU FAIL flag remains steady when activated.

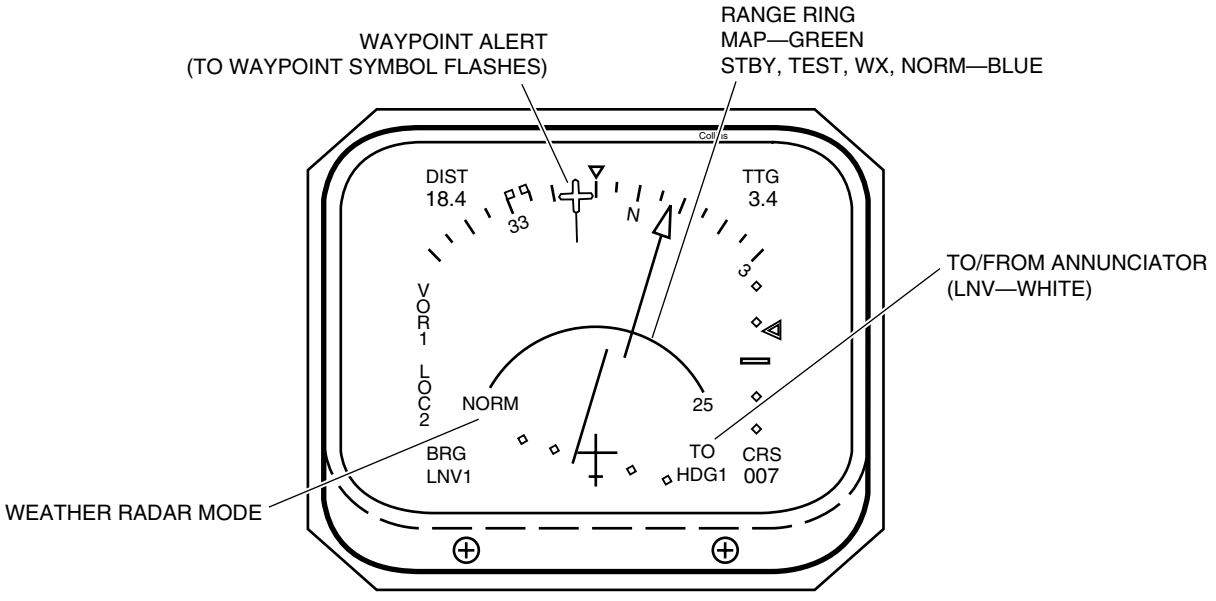


Figure 16-18. EHSI ARC Format

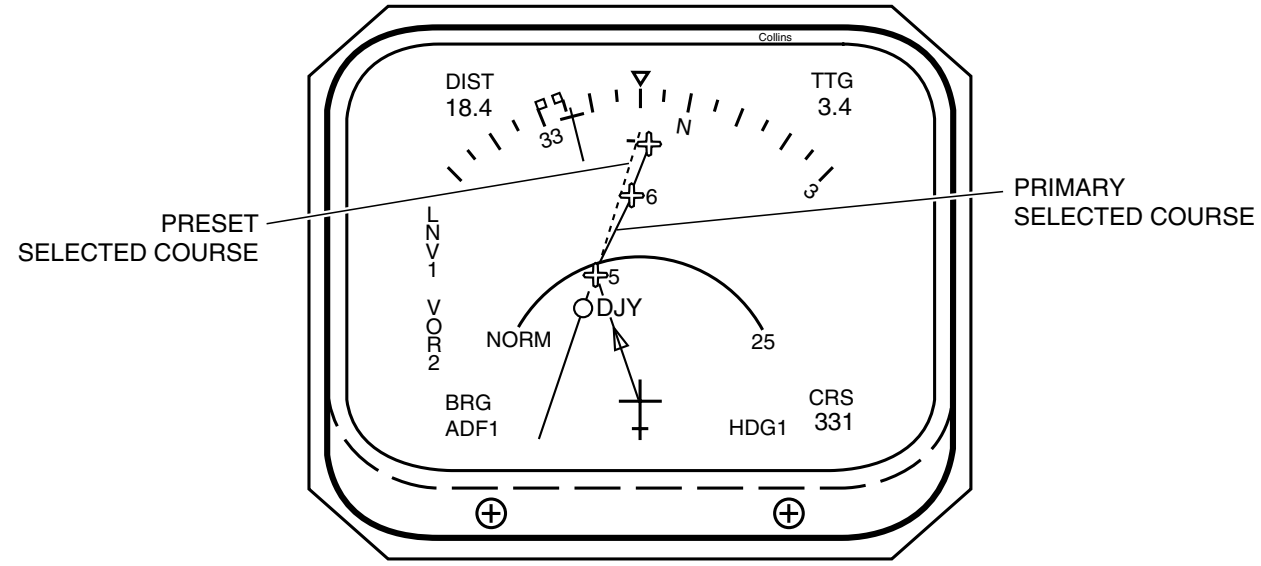


Figure 16-19. EHSI MAP Format

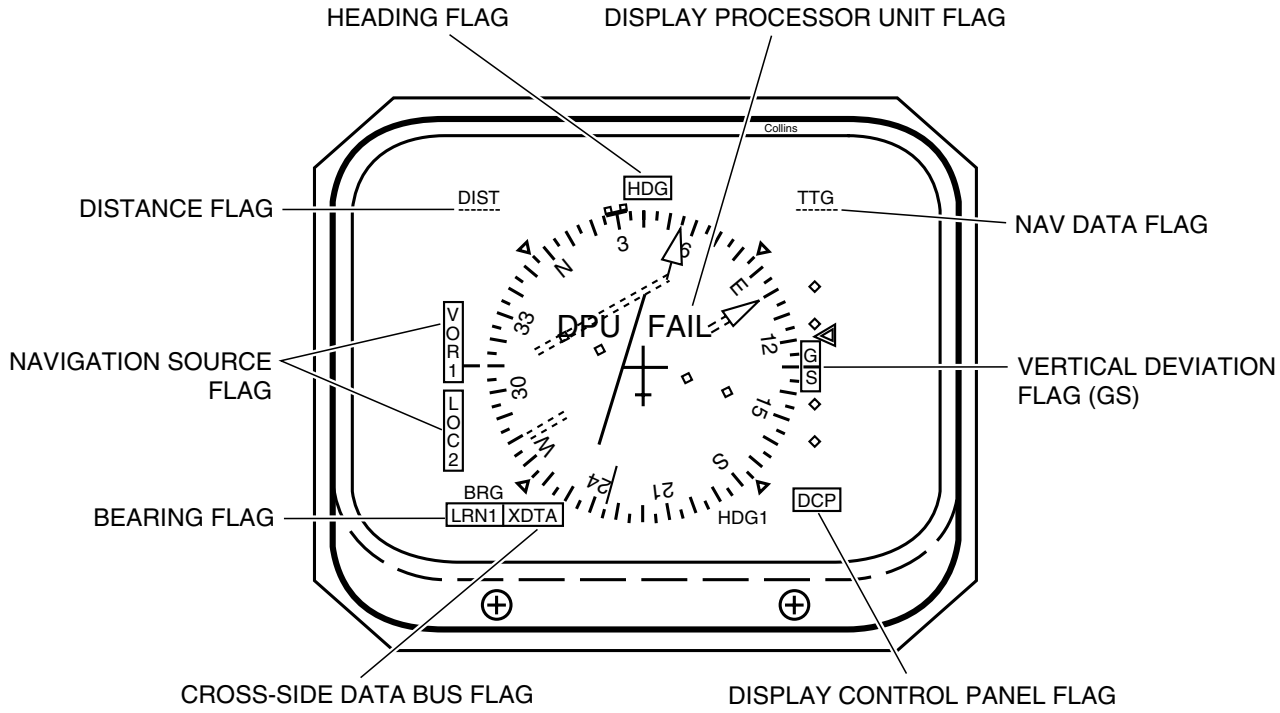


Figure 16-20. EHSI Failure Indications and Flags

MFD

The multifunction display is the fifth, and largest, cathode ray tube display in the EFIS. It provides standard weather radar information in addition to navigation and page data information.

A pictorial navigation map shows the location of VOR stations or the location of waypoints provided by the long-range navigation system.

The weather radar information may be shown separate or in conjunction with navigation information (Figure 16-21).

The page data display (Figure 16-22) consists of 100 information pages organized into chapters for display of normal, emergency, and operational information as designated by the operator.

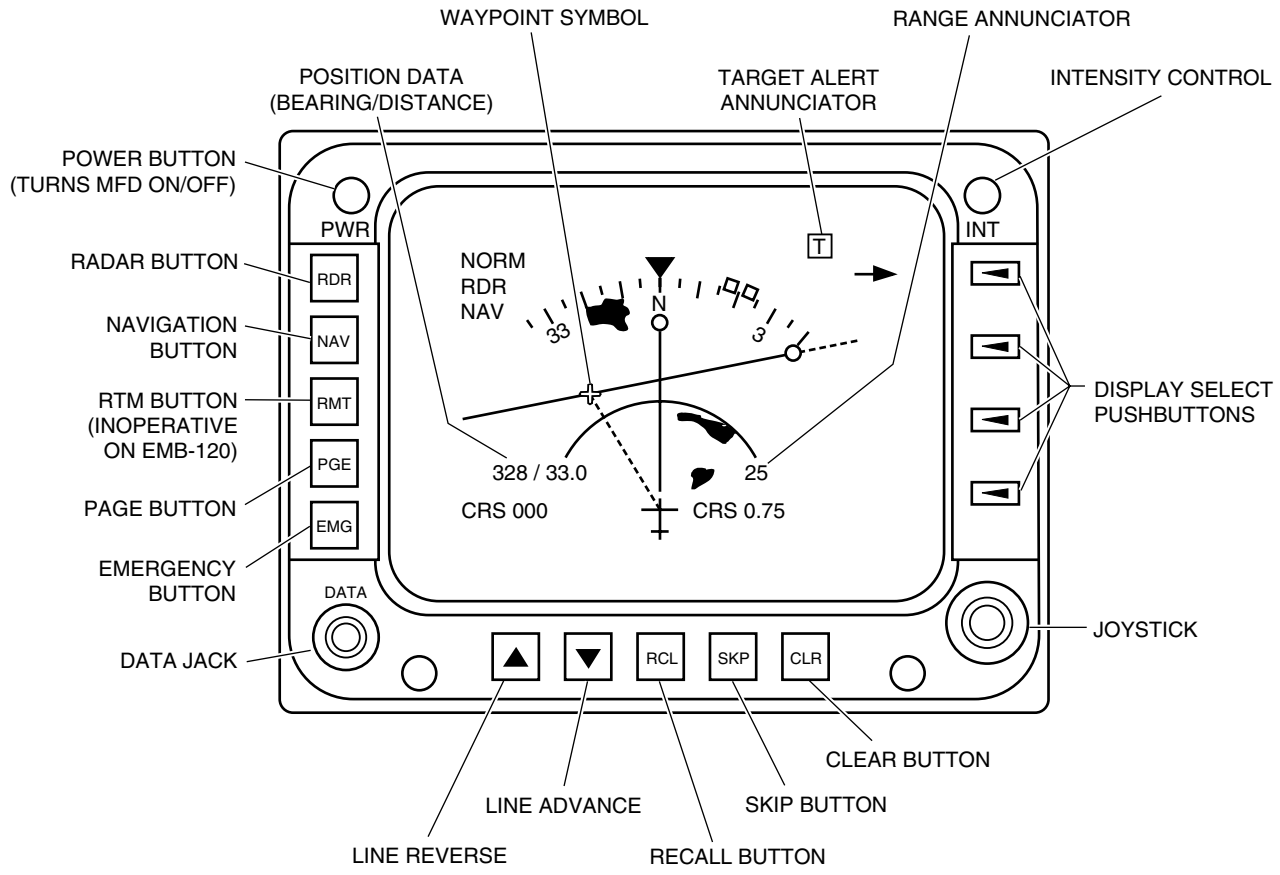


Figure 16-21. Multifunction Display (MFD)

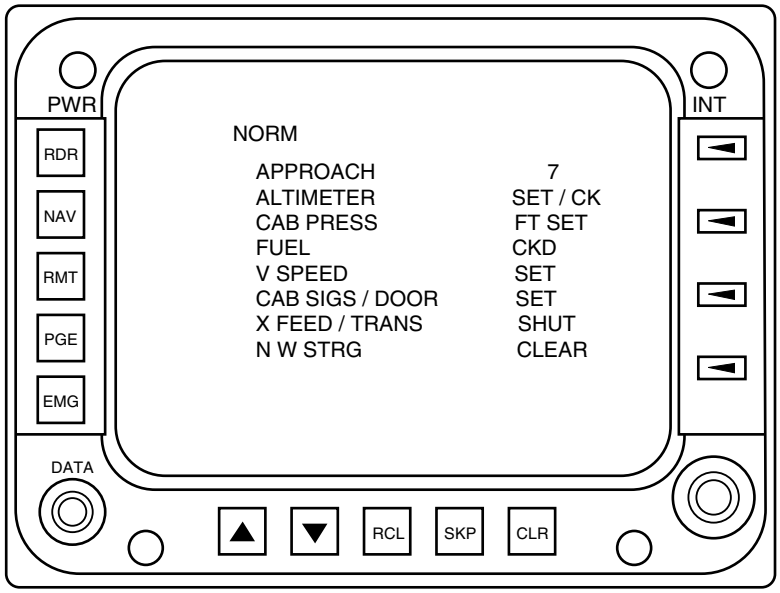


Figure 16-22. MFD—Page Checklist (Typical)

MFD Controls

RDR—Displays radar alone or in conjunction with NAV mode.

NAV—Displays navigation data.

PGE/EMG—Allows user selection, control, and entry of alphanumeric information.

DATA—Jack used for entry of remote data.

LINE ADVANCE(▼)—Moves cursor ahead (down the list) in PGE/EMG modes. Used in NAV mode to increase range.

LINE REVERSE(▲)—Backs up cursor one line without acknowledgment. Used in NAV mode to decrease range.

SKP—Used in PGE/EMG modes to move cursor past a step without changing its color.

CLR—Used in PGE/EMG modes to reset all lines to yellow, and display to first page of chapter.

JOYSTICK—Used in NAV, PGE, and EMG modes.

- NAV—Used to locate a defined waypoint. When moved, a green waypoint symbol appears with its position from the airplane symbol shown in lower left center of display. A green dashed line connects waypoint to airplane symbol.
- PGE/EMG—Downward movement displays next four chapters, upward movement displays previous four. Within a selected chapter, it moves display to next (down) or prior (up) page. Left/right movement displays first page of last/next chapter.

DISPLAY SELECT PUSHBUTTONS—Provide additional display control in NAV, PGE, and EMG modes.

- NAV—Used for source selection
- PGE/EMG—Used to select the first page of the appropriate chapter.

INT—Controls brightness of the display.

MFD Warning Flags

HDG (red)—Indicates failure of heading sensor. Appears to the right of lubber line. (If failure is from an internal synchro or bus parameters monitor, the compass card is frozen.)

HDG (yellow)—Indicates heading comparator is mismatched. Appears to the left of lubber line.

MPU FAIL (red)—Indicates MPU failure. Appears in the center of the MFD.

Navigation sensor failures are indicated by the respective annunciation on the MFD (VOR, VLF, etc.), changing color to red.

With Omega operation, if either navigation sensor is incompatible with the type of heading displayed (Mag or True), the appropriate navigation annunciator flashes until a compatible selection is made.

Control Panels

EFIS Control Panel

An EFIS CONTROL PANEL (Figure 16-23) sometimes referred to as EFIS transfer panel, is provided for each pilot. The panel provides flexibility and backup capability should a failure occur in an EFIS component or the attitude and heading reference system (AHRS).

COMPOSITE MODE transfer switch - Selects the composite mode display for the EADI or the EHSI. This switch is used with the failure of the EADI or EHSI. The composite image may also be displayed on the MPU.

DISPLAY SOURCE transfer switch—This switch changes the processor used to drive the EADI and EHSI from the normal data processing unit (DPU) to the multifunction processing unit (MPU). It is used in conjunction with a failure of the DPU (DPU FAIL flag on EADI or EHSI).

XFR light—An amber light that illuminates anytime DISPLAY SOURCE transfer switch is in the XFR position.

AHRS transfer switches—The ATT and HDG switches select the attitude or heading from the opposite (off side) attitude and heading reference system (AHRS). They are used in conjunction with an ATT or HDG flag on the display. The switches are also discussed in the AHRS section.

VOR CRS DEV switch—Selects either angular (ANG) or linear (LIN) deviation for the active VOR pointer.

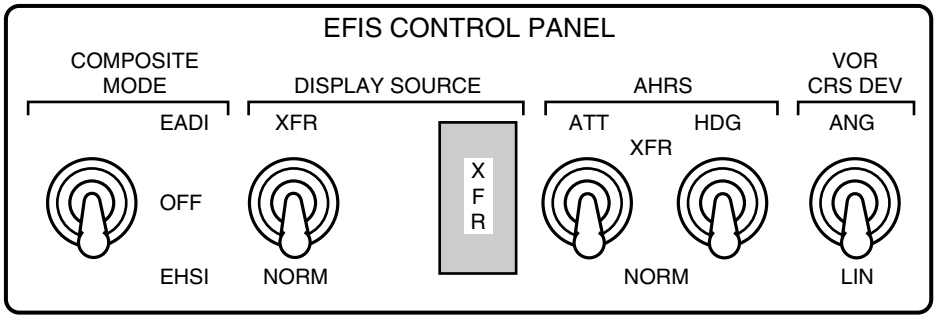


Figure 16-23. EFIS CONTROL PANEL

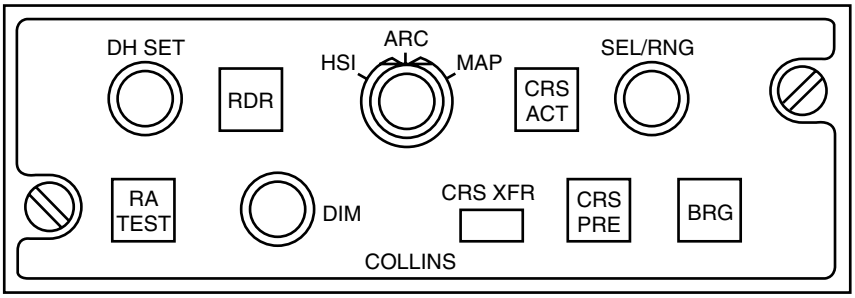


Figure 16-24. Display Control Panel (DCP)

Display Control Panel (DCP)

Each display control panel (DCP), selects the operation and function modes of its EADI and EHSI. The panel receives course and heading inputs from the course heading panel. It stores this information in its memory and transmits it to the EADI/ EHSI and the MFD processors, for display.

DH SET knob—Used to set the decision height displayed on the EADI.

RA–TST pushbutton—Momentarily tests the radio altimeter. When pressed:

- 50-feet radio altitude is displayed on EADI.
- DH annunciation is displayed if decision height is set greater than test altitude.

DIM control—Two concentric knobs used to dim the EADI and EHSI. The knobs are sized to turn together but may be operated separately.

Format switch—This three position rotary switch selects HSI, ARC, or MAP formats on EHSI.

CRS ACT button (active course)—Works in conjunction with the SEL/RNG knob to select the NAV source for the active course arrow. Pressing the button once activates the menu mode and displays the NAV source (VOR1, VOR2, VLF1, LOC1, LOC2) in a flashing white box on the EHSI. Pressing the button a second time activates the selected NAV source.

CRS PRE button (preset course)—Used for selecting a second navigation sensor for display. The procedure is similar to that for active course. If the button is pressed with no preset course displayed, the last selected sensor is displayed in the menu box. If the button is pressed while in the menu mode without changing the selection, the preset course is removed from the display. This allows presetting a course and recalling it later.

BRG—Used to select the desired NAV sensor (ADF1, ADF2, VOR1, VOR2, LRN) to be connected to the bearing pointer. Selection technique is the same as for preset course.

SEL/RNG knob—Used for NAV sensor selection and range selection. Operates in conjunction with CRS ACT, CRS PRE, and BRG pushbuttons as a source selector in the menu mode, and as a range selector with ARC or MAP format selected.

Range selection is valid only if no weather information is selected on the DCP. If radar is selected, ranges are controlled on the weather radar panel (WXP).

With ARC or MAP format selected and CRS ACT, CRS PRE, and BRG buttons not activated, the SEL/RNG knob selects ranges of 5, 10, 25, 50, 100, 200, 300, and 600 NM. One half of the full scale range is displayed at the right end of the range arc on the EHSI.

CRS XFR button—Selects the preset course as the active course. The previous active course sensor is removed from the display.

RDR button—Adds weather radar information to the EHSI when in ARC or MAP format.

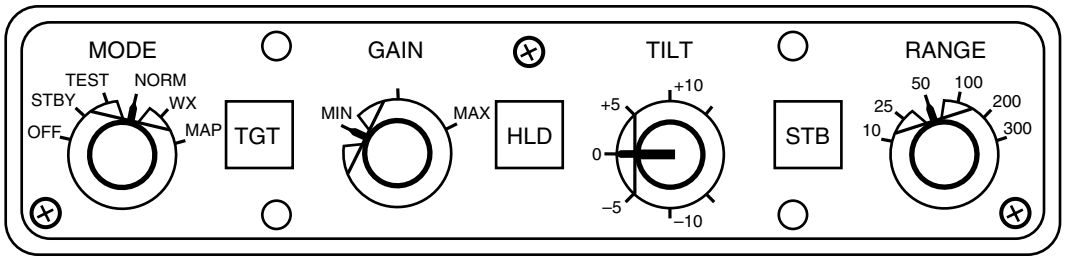


Figure 16-25. Weather Radar Panel (WXP)

Weather Radar Panel (WXP)

The weather radar panel (WXP) (Figure 16-25) provides the data processing, mode control, range selection, and system operating control facility for the display of weather radar information on the MFD or the EFDs.

The radar is discussed in more detail in the Navigation section of this chapter.

Course Heading Panel (CHP)

The course heading panel (CHP) (Figure 16-26) provides course and heading selection for the EFIS.

Details of panel operation are discussed in the Navigation section also.

Processor Units

The processor units are electronic components remotely-mounted in the electronic equipment bay. Each unit has a thermal sensor switch that actuates the amber EFIS OVERTEMP light on the multiple alarm panel.

Display Processor Unit (DPU)

A display processor unit (DPU) receives information from the navigation system and from other EFIS components, converts the data into digital form, and drives an EADI and EHSI. There are two DPUs, one for each EADI/EHSI pair.

Multifunction Processor Unit (MPU)

The multifunction processor unit (MPU) receives information from the same sources as the DPU, converts it into digital form, and drives the MFD.

The MPU also serves as a back up for the DPUs. In the event of a failure, DPU replacement is accomplished by placing the DISPLAY SOURCE switch on the EFIS CONTROL PANEL to the XFR position.

When the MPU replaces a DPU, the multifunction display shows the same indication as the EHSI.

In the event both DPUs fail, the pilot and copilot would each select XFR. The MPU then drives for all four EFDs with repeated indications.

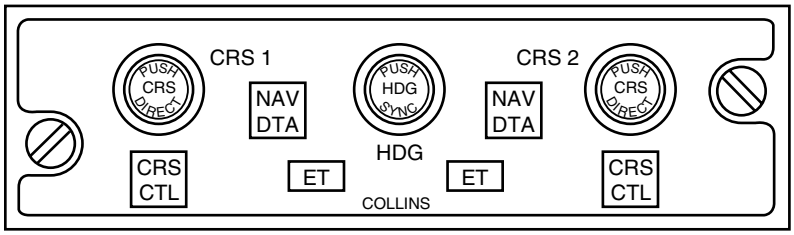


Figure 16-26. Course Heading Panel (CHP)

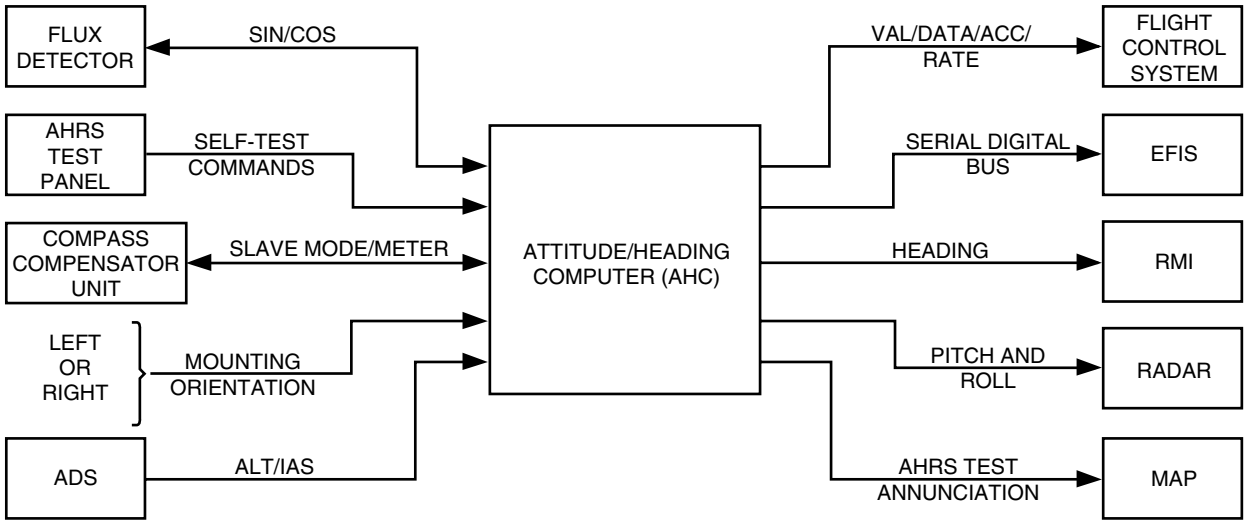


Figure 16-27. AHRS Schematic

ATTITUDE AND HEADING REFERENCE SYSTEM (AHRS)

GENERAL

The attitude and heading reference system (AHRS) is a microcomputer-based system that provides the attitude and heading information for the aircraft systems (Figure 16-27).

The EMB-120 has two identical, and independent, AHRSs.

Electrical power is supplied to AHRS 1/2 as listed below. Refer to Chapter 2, Electrical Power System schematic.

- Primary DC—Relay Box Emergency DC Bus 1/2
- Primary AC—26 VAC Emergency/ Essential Bus
- Secondary DC (loss of both AUX generators) —Relay Box DC Bus 1/DC Bus 2
- Secondary AC (loss of both AUX generators) —Same as primary source

NOTE

The backup battery supplies 28 VDC power to the AHC in the event of power transients during engine start (provided the electrical emergency switch is in NORMAL and the backup battery switch is in the ARM position).

COMPONENTS

Each AHRS consists of the following major components:

- Flux detector
- Attitude/heading computer (AHC)
- Compass compensator unit

Flux Detector

The flux detector senses the earth's magnetic lines of flux. Located remotely in the left/right wing tip to minimize interference from the aircraft structure, the flux detector is the source of magnetic heading information for the AHRS.

Attitude/Heading Computer (AHC)

The attitude/heading computer (AHC) is a strapdown reference (mounted solidly to system chassis – no gimbals) that contains rate and acceleration multisensors, digital and analog input/output, and the computation channels required to generate the system outputs.

The AHC provides all attitude and heading measurements, as well as angular rates and linear accelerations about the three aircraft axes.

Initialization occurs automatically with application of electrical power. In the NORMAL mode, initialization is accomplished within 70 seconds. In the DG mode, initialization requires 10 minutes.



Figure 16-28. Compass Compensator Unit

Compass Compensator Unit

Mounted on the pilot/copilot's instrument panel, the compass compensator unit (Figure 16-28) provides control for the remote compass compensation. It contains the following controls and indicators:

DG MODE pushbutton—The directional gyro (DG) pushbutton selects or deselects the DG mode.

- In the DG mode (pushbutton illuminated), flux detector information is not used.
- In the slaved mode (normal operating mode), the gyro is coupled to the flux detector.

GYRO SLAVING meter—The meter provides a visual indication of the slaving status of the gyro with respect to the magnetic heading sensed by the flux detector.

- In the slaved mode, the slaving meter needle oscillates about the center position.
- In the DG mode, the needle moves to the right or to the left, indicating the direction of error.

SLEW pushbuttons (left/right arrowheads)—These pushbuttons are operational only in the DG mode and are used to periodically correct gyro deviations.

SYSTEM OPERATION

After approximately 70 seconds, the ATT and HDG flags go out of view and valid attitude and heading information is provided (10 minutes in DG mode).

The system may be operated with the gyro either slaved or not slaved to the flux detector. The mode is selected with the DG pushbutton on the compensation control unit.

Operation With Gyro Slaved

In normal operation, the system is coupled to the flux detector. In this slaved mode, the DG pushbutton is not illuminated and the computer uses signals from the flux detector.

The gyro slaving indication is shown by the slaving meter needle, which must oscillate about the center position.

Operation With Gyro Not Slaved

In the gyro not-slaved or DG mode, the DG pushbutton is illuminated and the DG annunciation is displayed on the EHSI.

The AHC does not use the heading information sent by the flux detector.

When the gyro is not slaved, it presents errors which must be corrected periodically. The errors are displayed by the needle on the slaving meter, and the corrections are performed using the slew pushbuttons.

- The left pushbutton is used if the needle is to the right.
- The right pushbutton is used if the needle is to the left.

SYSTEM FAILURES

EFIS Control Panel—(AHRS System Use)

The EFIS control panel enables the pilot or copilot to use attitude and heading information from the cross-side AHRS.

The AHRS ATT and HDG switches are used in conjunction with ATT and HDG flags on the EFIS displays.

- In the NORM position, the EFIS display receives heading/attitude information from its respective (on-side) AHRS.
- In the XFR position, the EFIS display receives heading/attitude information from the cross-side AHRS.

Attitude Failure

Attitude failure is indicated on the EADI by the red ATT flag in a rectangle, above the aircraft symbol. The background sky/earth indication, roll and pitch scales, roll pointer, and control bars go out of view on the display.

The AHRS ATT switch, on the EFIS control panel of the system with the attitude failure, must be positioned to XFR. The attitude signals are now provided by the cross-side AHRS.

The ATT flag goes out of view on the display, and the attitude annunciator (ATT-1/ATT-2) changes from white to yellow, indicating that the attitude source of the cross-side AHRS is being used.

The yellow ROL and/or PIT comparator warnings are displayed on the EADI if there is a discrepancy in excess of 6° between the pilots indications.

Heading Failure

Heading failure is indicated on the EHSI by the red HDG flag in a rectangle, in place of the lubber line above the compass card.

The AHRS HDG switch, on the EFIS control panel of the system with the heading failure, must be positioned to XFR. The heading signals are now provided by the cross-side AHRS.

The HDG flag goes out of view on the display, and the heading annunciator (HDG1/ HDG2) changes from white to yellow, indicating that the heading source of the cross-side AHRS is being used.

The yellow HDG comparator warning is displayed on the EHSI if there is a discrepancy in excess of 6° between the pilot's indications.

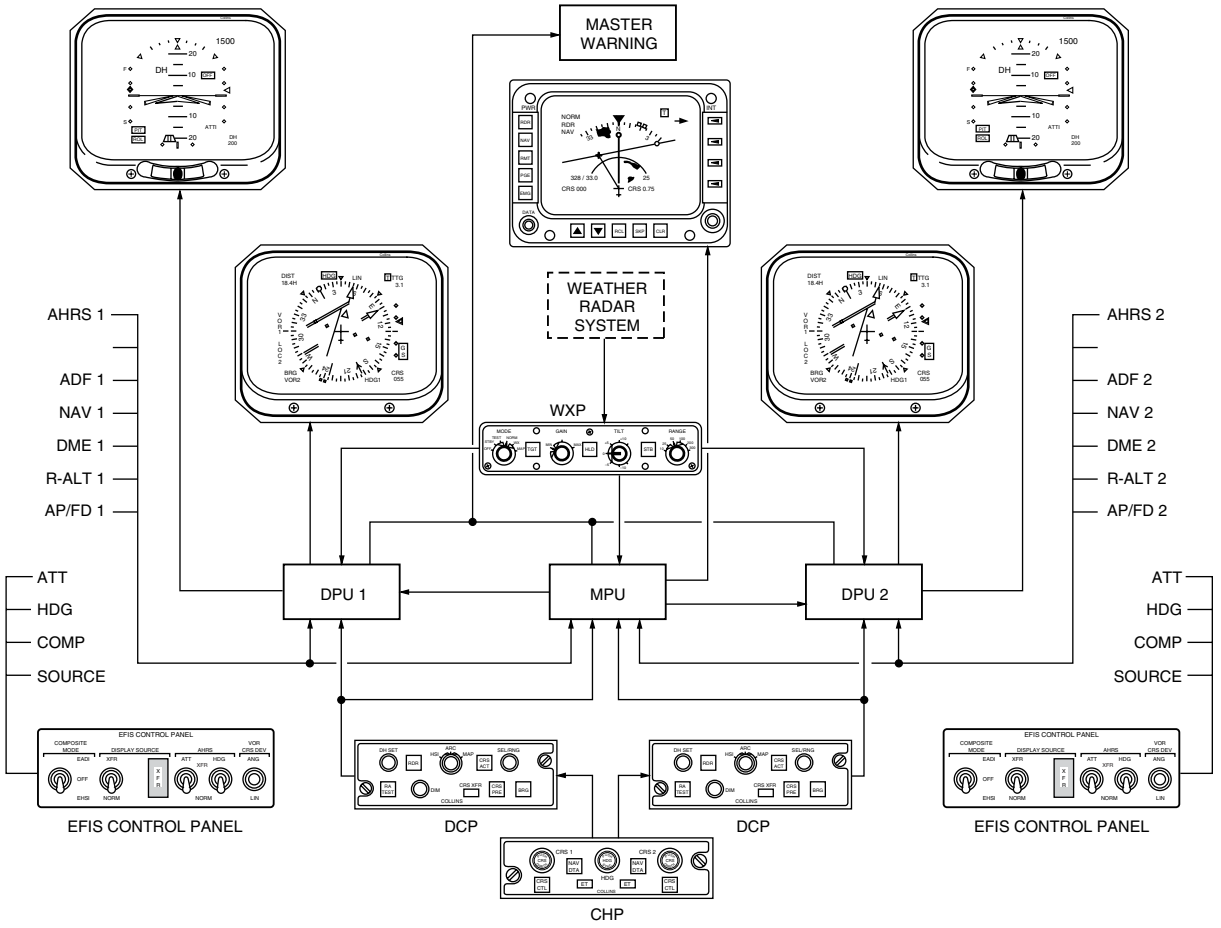


Figure 16-29. Navigation System

NAVIGATION SYSTEM

GENERAL

The EMB-120 navigation system (Figure 16-29) incorporates the EFIS and the two AHRSs with thefollowing navigation equipment:

- A weather radar
- Two VHF NAV radios
- One ADF radio
- Two radio magnetic indicators (RMIs)
- Two distance measuring equipment systems (DMEs)
- Two ATC transponders
- An optional OMEGA/VLF system

COMPONENTS

Course Heading Panel

General

The course heading panel (CHP), located on the right side of the center console behind the condition levers (Figure 16-30), provides the selection of course and heading for the EFIS. It is powered from Radio Master DC Bus 1C.

The CHP output is to both display control panels (DCPs).

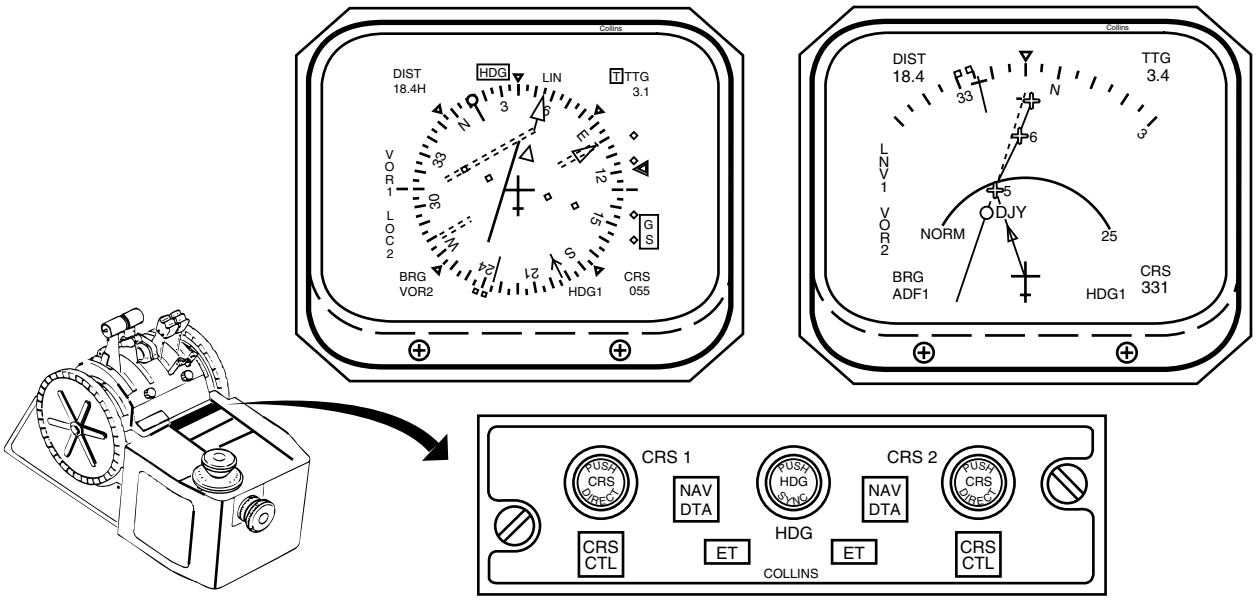


Figure 16-30. Course Heading Panel (CHP)

Controls

CRS knob—Two course select knobs (CRS 1 and CRS 2) are provided to allow independent course selection on the pilot and copilot EHSI. The knob rotates the selected EHSI course arrow to the desired course.

Course PUSH DIRECT buttons—Each CRS knob includes a center pushbutton which, when pressed, causes the respective EHSI course arrow to rotate until the selected navigation source provides a "direct to" indication.

CRS CTL buttons—Two independent (pilot's or copilot's) course control push buttons determine what course arrow (active or pre-select) is being controlled by the respective course select knob. When pressed, the cursor toggles to the opposite course.

HDG knob—The heading select knob controls the heading marker (bug) on both EHSIs simultaneously. The bug may be rotated in either direction. The heading marker on the MFD moves simultaneously if the display is selected to the EHSI format.

Heading PUSH SYNC button—The HDG knob includes a center pushbutton which, when pressed, causes the heading bug to rotate and match the airplane heading under the lubber line of the EHSIs and, if selected, MFD.

ET buttons—The elapsed time buttons control the elapsed time displayed at the right upper corner of each EHSI. At power on, the display initializes at 00:00 (minutes). The first press starts the timer. The second press stops the count, and a third press resets the counter to zero.

NAV DTA buttons—These buttons provide selection of navigation data displayed in the same place as ET information, without disturbing the ET count.

- First press: TTG (time-to-go) data appears on the display. This is computed time-to-go, to or from the station selected as the active course.
- Second press: SPD (ground speed) data appears on the display. This is computed ground speed, to or from the station selected as the active course.
- Third press: ET data displayed again.

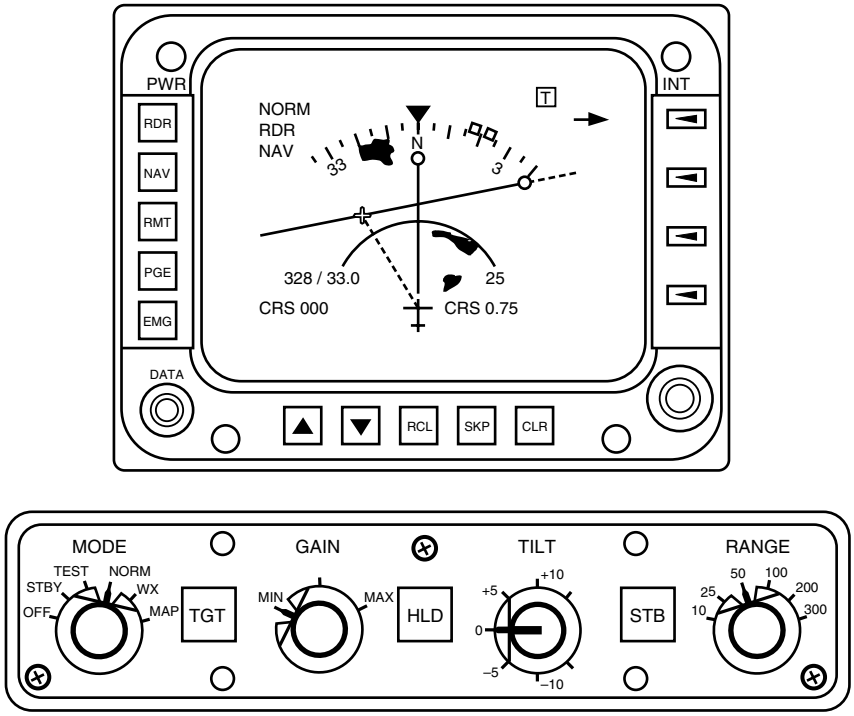


Figure 16-31. Weather Radar Panel (WXP)

Weather Radar Panel (WXP)

The weather radar display is controlled by the weather radar panel (WXP) (Figure 16-31).

MODE Control

The MODE control is a rotary switch used to select the various operating modes of the weather radar. These modes are annunciated on the upper left corner of the MFD (except OFF) in all MFD formats. The modes are listed as follows:

OFF—Removes power from the weather radar receiver-transmitter. The MFD has a separate PWR switch that must be on for the MFD to operate.

STBY—Power is applied to the radar system and the 60-second warm-up period is initiated. The weather display, radar transmitter, and radar antenna scan drive circuits are inhibited during the 60-second warm-up period, or anytime STBY is selected.

TEST—Initiates the radar self-test function. The radar self-test display appears after the 60-second warm-up period has elapsed.

For a normal-size test pattern, the 25 NM range should be selected. Selecting other ranges increases or decreases the test pattern size, resulting in an incomplete test pattern. TEST is displayed in the upper left side of the display.

NORM—Selects the weather detection mode. The radar antenna begins to scan and, after the 60-second warm-up period has elapsed, the radar transmitter is energized. Detectable weather appears as one of four colors: black, green, yellow, or red (least to most reflective) in addition to the black screen. NORM is displayed.

WX—Same as NORM mode except automatic cyclic contour circuits are enabled. Contouring storm cells alternate between red and black at a one-cycle-per-second rate. WX is displayed.

On airplanes Mod SB 120-034-0022 or SNs 120.042 and subsequent, weather information appears in green, yellow, red, and magenta colors (least to most reflective), and the path attenuation correction (PAC) feature is automatically enabled.

MAP—Ground targets appear in cyan, yellow, and magenta (least to most reflective). Disables the automatic cyclic contour circuits. MAP is displayed.

GAIN Control

The GAIN knob is a 5-position switch that controls the gain of the radar receiver. The highest gain occurs in MAX position, with each of the other positions reducing receiver sensitivity to MIN. Gain may be used with NORM, WX, and MAP modes.

The ability of a target to be displayed is dependent upon the gain setting. If any position other than MAX is selected, the mode annunciator alternates between GAIN and the selected mode (e.g., GAIN– MAP–GAIN–MAP) to remind the user that the system is operating at a reduced gain and some targets may not be displayed.

TILT Control

The TILT control manually adjusts the pitch axis of the radar antenna from approximately +15° to -15° to allow the best target presentation.

RANGE Control

The RANGE control is a rotary switch that is used to select the radar display range. It is selectable in ranges of 10, 25, 50, 100, 200, and 300 nautical miles. A range mark is shown in the center of the display with range annunciation.

TGT Pushbutton

The latching TGT button selects/deselects the target alert feature. This feature notifies the user when a return signal of contour strength is detected within 60 to 160 nautical miles, and +15° of current heading (regardless of range selected).

A yellow box with a yellow T appears when target alert is enabled on the WXP and the mode selector is in other than OFF or STBY. The annunciation flashes when a target is detected.

HLD Pushbutton

Pressing the momentary HLD button freezes the weather display on the MFD until the button is pressed again, or power is removed from the system.

Target updating does not occur in the HLD mode. The weather radar mode annunciator alternates between HOLD and the selected mode (e.g., HOLD–WX–HOLD–WX) to remind the user that the system is in the HOLD mode and the weather display is not being updated.

STB Pushbutton

The latching STB switch enables/disables the radar antenna pitch and roll stabilization circuits. The switch is normally pushed in to select the on position.

The switch is selected to the off position in the event of a gyro input signal failure to the radar antenna.

WARNING

DO NOT operate radar near fuel spills or people.

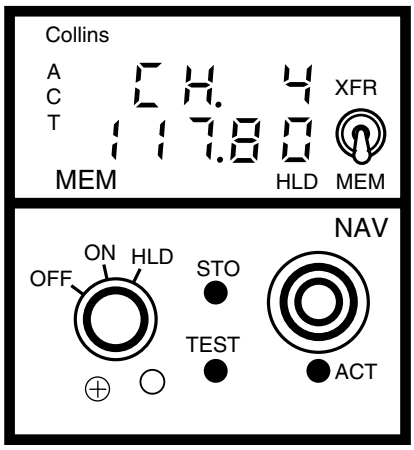


Figure 16-32. VHF NAV Radio Control

VHF Navigation Radio

General

The EMB-120 has two VHF NAV radios. The CTL-32 NAV controls (Figure 16-32) are located on the center panel, either side of the weather control panel. Each radio has 200 VOR/LOC channels, from 108.00 through 117.95 MHz, spaced 0.05 MHz apart. In addition, there are 40 glide-slope channels automatically paired with the localizer channels. The NAV radios also incorporate a marker beacon receiver.

Displays

Active frequency display—The upper display is the active frequency (frequency to which the equipment is tuned) and also displays diagnostic messages.

Preset frequency display—The lower display is the preset (inactive) frequency and also displays diagnostic messages.

Controls and Indicators

Power and mode switch—ON and OFF positions control system power. The HLD position allows the NAV frequency to be changed but holds the DME to the current active frequency.

Frequency select knobs—Two concentric knobs control the preset and active frequency displays. During normal operation, the frequency select knobs change the preset frequency display.

The larger knob changes the three digits to the left of the decimal point in 1-MHz steps. The smaller knob changes the two digits to the right of the decimal point in 0.05-MHz steps.

Frequencies roll over at the upper and lower limits. The two frequency select knobs are independent, so the upper and lower roll-over of the 0.1 MHz digit does not cause the 1.0 MHz digit to change.

XFR–MEM switch—This is a 3-position toggle switch, spring-loaded to the center neutral position. When held to the XFR position, the preset frequency is transferred up to the active display and the NAV/DME retunes. The previously active frequency becomes the new preset frequency displayed in the lower window.

When this switch is held to the MEM position, one of the four frequencies stored in the memory is recalled into the preset display. The storage location (CH1 through CH4) for the frequency is displayed in the upper window while the switch is held. All four stored frequencies may be displayed one at a time by repeatedly toggling the switch to the MEM position.

After the desired stored frequency has been recalled to the preset display, it may be transferred to the active display by positioning the switch to XFR.

ACT button—The ACT button is used to enable the frequency select knobs to directly retune the active frequency.

To retune the active frequency:

1. Push the ACT button for approximately 2 seconds.
   - The bottom window displays dashes.
   - The upper window continues to display the active frequency.

2. Dial in the desired active frequency using the frequency select knobs.

   - The ACT annunciator flashes as the frequency is dialed in, indicating that the receiver is being retuned.

   - If the ACT annunciator continues to flash after the frequency has been selected, it indicates that the receiver is not tuned to the displayed active frequency)

3. Push the ACT button a second time to return the control to the normal display.

ACT annunciator—Momentarily illuminates when frequencies are being changed. Flashes if the receiver is not tuned to the active frequency displayed.

MEM annunciator—Illuminates whenever a frequency is being displayed in the lower window.

HLD annunciator—Indicates that the DME is held to the active frequency at time of selection; the NAV frequency may be changed. The upper window displays the NAV frequency and the lower window displays the held DME frequency.

Store (STO) button—Allows up to four preset frequencies to be selected and entered into the non-volatile memory. To program the memory:

1. Dial the frequency to be stored into the preset display.
2. Press the STO button.

   - The upper window displays the channel number of available memory (CH1 through CH4).

   - The lower window continues to display the frequency to be stored (for approximately 5 seconds).

3. The MEM position may be used to advance through the channel numbers without changing the preset display.
4. Push the STO button a second time to commit the preset frequency to memory in the selected location.

The second actuation of the STO button must be done within 5 seconds of selecting the channel number or the first actuation of the STO button. If more than 5 seconds elapse, the control reverts to the normal mode of operation.

TEST button—Press the TEST button to initiate the radio self-test diagnostic routine.

Self-Test

During self-test, VOR, ILS, and marker beacon test outputs are provided. The intensity of the upper and lower displays modulate from minimum to maximum to indicate that self-test is in progress.

VOR Self-Test

1. Select any VOR frequency on the NAV control. A signal on that frequency does not interfere with the self-test.
2. Rotate the CRS knob on the course control panel to set the course arrow on the EHSI to approximately 000°.
3. Press the TEST button.

   a. On the EHSI:

      - The NAV flag comes into view. (Goes out of view after approximately 2 seconds.)

      - The lateral deviation bar is approximately centered.

      - The TO indication appears.

   b. On the RMIs, the pointers connected to the equipment indicate a magnetic bearing of approximately 000°.

The equipment returns to normal operation after approximately 15 seconds, even if the TEST button is held.

ILS (Localizer and Glide Slope) Self-Test

1. Select any localizer frequency on the NAV control.
2. Press the TEST button on the EHSI:
   - The NAV and GS flags come into view. (Go out of view after approximately 3 seconds.)
   - The lateral deviation bar deflects right approximately 2/3 of full scale.
   - The glide slope pointer deflects down approximately 2/3 of full scale.

The equipment returns to normal operation after approximately 15 seconds, even if the TEST button is held.

Marker Beacon Self-Test

The marker beacon is tested automatically when the self-test is actuated and either a VOR or localizer frequency is selected. Proper operation is indicated by all three marker beacon lamps flickering at a 30 Hz rate, and a 30 Hz marker audio output tone.

Diagnostic Display

An extensive self-test diagnostic routine is initiated within the NAV radio when the TEST button is pressed.

NOTE

The diagnostic display is an extension of the self-test capabilities. The associated flags, deviation of the indicators, and proper responses for the self-tests should first be observed. If an out-of-limit condition exists, the problem may be verified in more detail by the diagnostics.

If a fault condition existed prior to the self test, a 2-digit diagnostic code associated with the fault is displayed in the preset display for approximately 2 seconds after the TEST button is pressed.

If a fault is detected during self-test, a 2-digit fault code is displayed in the active display, along with either of the following words:

- Four Dashes (----)—The four dashes are displayed with the code 00, indicating normal operation, and no trouble found.
- FLAG—FLAG is displayed with a 2-digit code when something is abnormal but a failure has not occurred (e.g., low signal level).
- DIAG—DIAG is displayed along with a code to indicate a failure has been detected.

Completion of the self-test is indicated by a return to the normal frequency displays, or a fault code indication.

Record any codes (except 00) to help the service technician locate the problem.

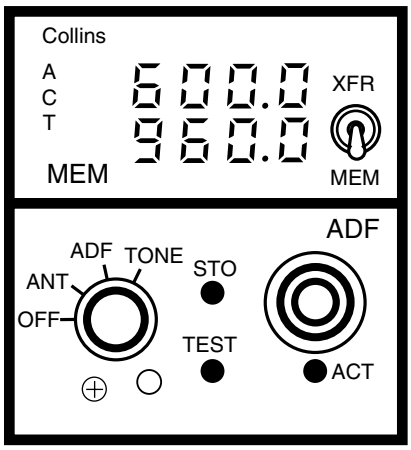


Figure 16-33. ADF Radio Control

ADF Radio

General

The EMB-120 normally has one ADF radio. The CTL-62 ADF control (Figure 16-33) is located on the center panel, just below the left NAV radio control. It has a frequency range of 190 to 1,749.5 kHz and three functional modes of operation.

Displays

Active frequency display—The upper display is the active frequency (frequency to which the equipment is tuned) and, in the self-test mode, displays the word "DIAG".

Preset frequency display—The lower display is the preset (inactive) frequency and, in the self-test mode, displays the diagnostic code.

Controls and Indicators

Power and mode switch—The power and mode switch applies power to the ADF system and establishes the system mode of operation. The switch has four detented positions:

- OFF—Removes system power.
- ANT—The ADF receiver functions as an aural receiver, providing only an aural output of the received signal.
- ADF—Functions as an automatic direction finder receiver in which relative bearing to the station is presented on the associated indicator, and an aural output of the received signal is provided.
- TONE—Provides a 1,000-Hz aural output tone when a keyed CW signal is being received.

Frequency select knobs—Two concentric knobs control the preset or active frequency displays. During normal operation, the frequency select knobs change the preset frequency display.

The larger knob changes the 1,000th and 100th kHz digits in 100-kHz steps. The smaller knob changes the 10th unit and tenths kHz digits in 1-kHz steps (with the exception that the first two detent positions following a change in the rotational direction cause a 0.5-kHz change). Rapid rotation of the smaller knob results in frequency changes greater than 1 kHz as a function of the rate of rotation.

Frequencies roll over at the upper and lower limits. The two frequency select switches are independent of each other such that the upper and lower roll-over of the 10th-kHz digit does not cause the 100th-kHz digit to change.

XFR–MEM switch—This is a 3-position toggle switch, spring-loaded to the center neutral position. When held to the XFR position, the preset frequency is transferred up to the active display and the ADF retunes. The previously active frequency becomes the new preset frequency displayed in the lower window.

When this switch is held to the MEM position, one of the four frequencies stored in the memory is recalled into the preset display. The storage location (CH1 through CH4) for the frequency is displayed in the upper window while the switch is held. All four stored frequencies may be displayed one at a time by repeatedly toggling the switch to the MEM position.

ACT button—The ACT button is used to enable the frequency select knobs to directly retune the active frequency.

1. Push the ACT button for approximately 2 seconds.
   - The bottom window displays dashes.
   - The upper window continues to display the active frequency.
2. Dial in the desired active frequency using the frequency select knobs.
   - The ACT annunciator flashes as the frequency is dialed in, indicating that the receiver is being retuned.
   - If the ACT annunciator continues to flash after the frequency has been selected, it indicates that the receiver is not tuned to the displayed active frequency.
3. Push the ACT button a second time to return the control to the normal display.

ACT annunciator—Momentarily illuminates when frequencies are being changed. Flashes if the receiver is not tuned to the active frequency displayed.

MEM annunciator—Illuminates whenever a frequency is being displayed in the lower window.

Store (STO) button—Allows up to four preset frequencies to be selected and entered into the non-volatile memory. The procedure for programming the memory is the same as the VHF NAV.

TEST button—Press the TEST button to initiate the radio self-test diagnostic routine.

Self-Test

Position the ADF power and mode switch to ADF and tune a nearby NDB, outer marker, or broadcast station. Press the TEST button. The RMI pointer rotates 90° clockwise from the previous valid indication. If the signal received is weak or of poor quality, the bearing pointer rotation will be slow.

Release the TEST button and verify that the RMI pointer returns the initial indication.

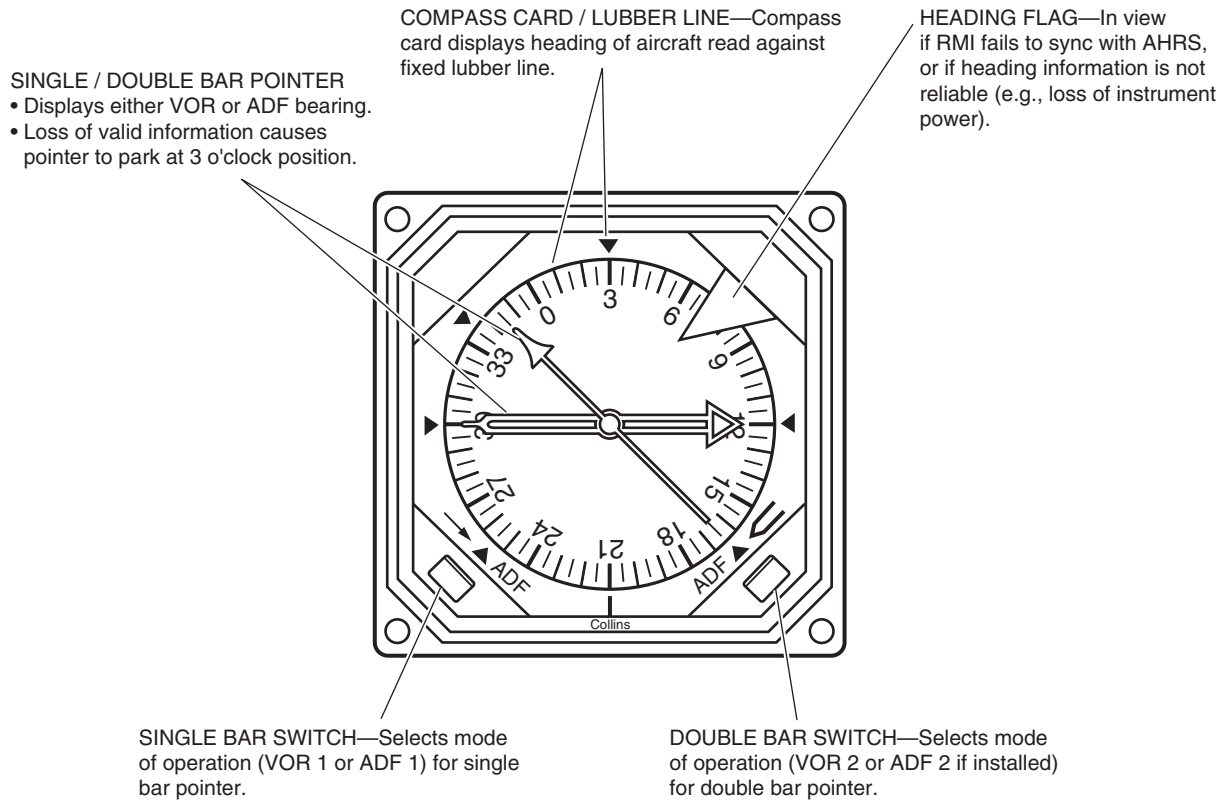


Figure 16-34. Radio Magnetic Indicator (RMI)

Radiomagnetic Indicator (RMI)

The RMI (Figure 16-34) provides aircraft heading information on a calibrated driven compass card read against a fixed lubber line and navigation data on two rotating pointers.

The heading inputs from the AHRS computer sensor units are connected to the pilot's and copilot's RMI in a cross-configuration. RMI No. 1 is connected to AHRS computer sensor unit No. 2, and RMI No. 2 is connected to AHRS computer sensor unit No. 1.

The RMIs receive navigation data from the VOR and ADF, and display it on two pointers. The single bar pointer (No. 1 needle) displays information from the No. 1 VOR and ADF; the double bar pointer (No. 2 needle) displays information from the No. 2 VOR and ADF (if installed).

DME Indicator

General

In addition to the EFIS displays, DME information may also be displayed on the IND-42A DME indicator (Figure 16-35). Frequency selection and self-test are accomplished using the VHF NAV control.

The DME maximum range is 300 nautical miles.

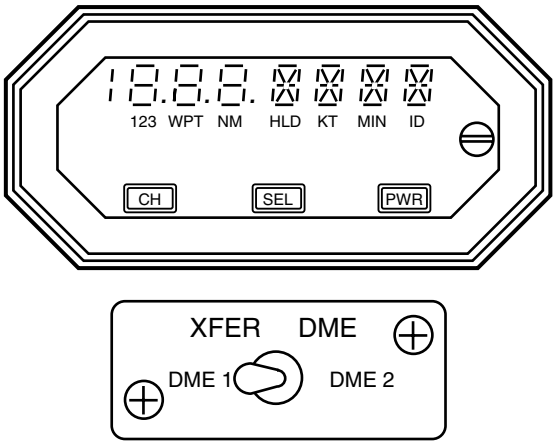


Figure 16-35. DME Indicator

Displays

Numeric display—The left display is numeric. It presents the slant range DME distance, and the diagnostic code (when self-test is performed).

Alphanumeric display—The right display is alphanumeric. It presents information selected by the SEL switch, and the diagnostic identifier (when self-test is performed).

Controls and Indications

Power switch (PWR)—The latching pushbutton on/off PWR switch controls the power applied to the indicator.

Mode selector switch (SEL)—The nonlatching pushbutton SEL switch selects the information to be displayed in the alphanumeric display.

When power is initially applied to the indicator, NM is the active numeric display and ID is the active alphanumeric display.

Pressing the SEL switch sequentially selects KT, MIN, and ID as the alphanumeric display.

Channel switch (CH)—The momentary pushbutton CH switch sequentially selects the frequency channel and lights the appropriate channel annunciator:

- 1—NAV 1 active frequency
- 2—NAV 2 active frequency
- 3—NAV 1 preselected frequency

If one of the 3 channels has invalid data, dashes are displayed.

Transfer switch—Selects which DME transceiver is being used.

1,2,3 annunciators—Sequentially controlled by the CH switch. Indicates the DME channel providing the information displayed.

WPT annunciator—Not used

NM annunciator—Automatically illuminates after power on when valid DME data is available. Indicates that the numbers displayed in the numeric display are slant range DME distance in nautical miles.

HLD annunciator—Indicates that DME hold has been selected on the NAV control panel.

KT annunciator—Sequentially controlled by the SEL button. Indicates that the alphanumeric display is the computed ground speed in knots.

MIN annunciator—Sequentially controlled by the SEL button. Indicates that the alphanumeric display is the computed time-to-station in minutes.

ID annunciator—Automatically illuminates after power on. Sequentially controlled by the SEL button. Indicates that the alphanumeric display is the 2, 3, or 4 letter DME station identifier. (The DME identifier is transmitted once every 30 seconds, therefore it is possible that 2 minutes could elapse before the station identifier is displayed in the alphanumeric display.)



NOTE

If the DME is put into hold, the ID display is automatically selected with the illumination of the HLD annunciator. The pilot may still select other modes (KT or MIN), but when he stops pushing the SEL button the display reverts to the ID mode (after approximately 5 seconds). This safety feature not only reminds the pilot that the DME is in hold, but also shows which station is being held.

Self-Test

A 10-second self-test diagnostic routine is initiated when the TEST button on the VHF NAV control panel is pressed. After initiating self-test:

1. All display segments and annunciators on the DME indicator illuminate for a lamp test.
2. If NM and ID are selected, the numeric display shows a test distance of 100. The alphanumeric display shows:
   - A test ground speed of 100 if KT is selected
   - A test time of 60 if MIN is selected
3. If no faults are detected, the alphanumeric display shows "AOK" at the completion of the test routine. (The DME aural output is be the Morse code characters AOK (• –, – – –, – • –).

Completion of the self-test is indicated by a return to normal display, or the word DIAG and a diagnostic code displayed on the DME indicator.

NOTE

Due to the approximate 10-second self-test cycle time, the test should be made as a preflight check and not during critical flight times.

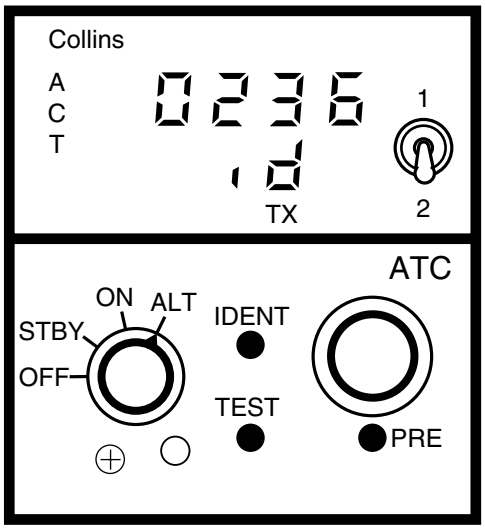


Figure 16-36. Transponder Control

ATC Transponder System

General

The EMB-120 is equipped with two transponders. Both are controlled by a single CTL-92 control (Figure 16-36), located on the center instrument panel just below the right NAV radio control.

Uncorrected barometric altitude, transmitted in mode C operation, is provided by the pilot's encoding altimeter.

Displays

Active code display—The active code to which the equipment is tuned and diagnostic messages in the self-test mode are displayed in the upper window.

During normal operation only a single display (active code) is shown in the upper window.

The normally blank, lower display window is active only during self-test.

Controls and Indicators

Power and mode switch—The power and mode switch applies power to the transponder system and establishes the mode of operation. The switch has four detented positions:

- OFF—Removes system power.
- STBY—Applies power to the equipment but prevents it from transmitting replies.
- ALT—Normal operating position. Allows the transponder selected to reply to the interrogation pulses, as well as to transmit uncorrected barometric altitude when interrogated in mode C. (This position should be used only if the aircraft is equipped with an encoding altimeter or a central air data system.)
- ON—Deletes the altitude code. Normally used only when requested by ATC.

Code select knobs—Two concentric knobs control the active code display. The left pair of digits and the right pair of digits are independent of each other. The larger knob changes the left pair and the smaller knob changes the right pair.

Slow rotation of the knob increases or decreases the right-hand digit of each pair one number per detent. Rapid rotation of the knob causes a digit change proportional to the rate of rotation. Roll over of the right-hand digits occurs at 0 and 7, causing the left-hand digits to increase or decrease.

Select switch 1 or 2—Selects one of the two transponders controlled by the CTL-92.

ACT annunciators—Momentarily illuminates when the code is being changed. Flashes if the actual reply code is not identical to the code shown in the active code display.

TX annunciator—Illuminates each time the transponder replies to an interrogation.

IDENT button—The IDENT button triggers the transponder to transmit a special identification pattern that is displayed on the ground controller's radar scope. It is pressed only when requested to "squawk ident" by the ground controller.

PRE button—The PRE button is used to select a code for storage. To store a code, push and hold the PRE button while turning the code select knobs. The code is stored in non-volatile memory and may be recalled by momentarily pressing the PRE button again. Only a single code is stored in memory.

TEST button—The TEST button initiates the self-test routine. The 1/2 select switch determines which transponder is tested.

Self-Test

1. Position the power and mode switch ON.
2. Set the select switch to 1.
3. Set desired code using code select knobs.
4. Press the TEST button.
   - The active code display intensity modulates from minimum to maximum.
   - If the equipment is functioning properly, AL is displayed in the upper window and the altitude in thousands of feet in 100-foot increments (i.e., 7.4 is 7,400 feet) is displayed in the lower window.
5. In dual-transponder installations, set the select switch to 2 and repeat the self-test.

NAVIGATION SYSTEM/EFIS OPERATION

General

The EFIS receives and displays information from the various components of the navigation system. Inputs from the AHRS weather radar, VHF NAV radios, and the ADF and DME systems may be presented in various formats on any of the EFIS displays.

The display format is controlled by the relevant display control panel (DCP), the course heading panel (CHP), the weather radar panel (WXP), and the controls on the multifunction display (MFD).

An in-depth discussion of the use of navigation information during the various phases of flight is beyond the scope of this chapter. However, a discussion and review of information available to the pilots and the manner in which it may be displayed follows.

NOTE

The on-side or cross-side conditions in the sources selection depends on panel side only. In other words, all number 1 sources are relative to the pilot's side and all number 2 are relative to the copilot's side.

If the pilot selects a number 2 source, like an active course or a bearing, the system considers a cross-side condition. The same is considered for copilot if he selects a number 1 active course or bearing.

EHSI Navigational Display

Formats

As discussed earlier, the EHSI displays (full compass rose, expanded compass sector, and expanded compass sector with weather radar information superimposed) are selectable on each DCP as HSI, ARC, and MAP respectively.

Full compass rose (HSI) format—In the HSI format (Figure 16-17), the EHSI is driven by the AHRS and shows a full 360° compass rose.

White markings are provided every 45° around the periphery of the card to aid in procedure turns. The compass card, reference marks, lubber line and reference aircraft symbol are also white.

Active selected course—The active course, selected on the course heading panel (CHP), is shown by the active (solid) course arrow. Deviation from course is indicated by the center portion of the course arrow which moves laterally with respect to the airplane symbol and the white deviation dots.

A full scale deflection of the lateral deviation bar (2 dots) is equivalent to 10 miles in LIN mode and 10° in ANG mode .

All active course information is green when the on- side NAV source is selected, and yellow when the cross-side source is selected.

Active course digital display—The active course is repeated digitally in green, in the lower right corner below the letters CRS. The CRS letters automatically change to B/C (back course) when a localizer frequency is tuned and the difference between airplane heading and course arrow exceeds approximately 105°.

Preselect Course Pointer—Preselect course appears as a cyan-colored, double-dashed line or course arrow when the CRS PRE button is pushed on the DCP. Course control is achieved with controls on the CHP.

Navigation Source—Two sources of navigation input may be displayed to the left side of the compass card. The upper annunciator is the active source and the lower is the preselected source. Only one source is displayed if the preselect function has not been selected. The source is selected on the DCP.

The annunciator changes color according to the tuned frequency. It appears as VOR/LOC 1 or 2 (green on side/yellow cross side) or VLF 1 or 2, (white on side/yellow cross side).

Only the active course is available to the flight control system. (The preselected course is present and active, but is not available.)

A circle to the right of the navigation source annunciator indicates which course pointer is being controlled by the course heading panel CRS knob and CRS CTL button.

Lateral deviation type—The letters LIN or ANG appear to the right of the lubber line to indicate linear or angular deviation relative to a VOR station.

LIN appears when a VOR station is selected as a navigation source, and LIN is selected on the EFIS control panel. The system reverts automatically to the angular mode (ANG) whenever: DME is lost, DME hold is selected, or a LOC frequency is selected. When this occurs LIN becomes ANG, flashes for 10 seconds, then becomes steady.

Bearing pointer—When BRG is selected on the DCP, a magenta, shape-coded bearing pointer is displayed along with the sensor annunciation.

If the sensor is VOR, the pointer is an octagon. The ADF bearing pointer is a cross, and LNV and RNV bearing pointers are star-shaped. The bearing pointer tail is a line with a V pointing to the station.

Bearing pointer source—The bearing pointer source is displayed in the lower left corner, beneath the magenta letters BRG. The sources can be VOR/ADF 1 or 2 (green on side/yellow cross side) and VLF 1 or 2 (white on side/blue cross side).

Heading sensor annunciator—Located on the lower right side, the HDG1 or HDG2 annunciation indicates the AHRS heading input (white on side/yellow cross side) selected by the AHRS HDG switch on the EFIS CONTROL PANEL.

Glide-slope display—The vertical deviation pointer, on the right side of the compass card, represents the center of the glide-slope beam or RNAV track. The center of the scale represents the airplane position relative to beam center. Each of the inner scale dots represents 1/4° displacement, and each of the outer dots 1/2°.

The pointer is in view (green on side/yellow cross side) only when the NAV receiver is tuned to a LOC frequency. B/C (back course) appears in the glide-slope scale when the difference between the selected course and aircraft heading is greater than 105°.

Distance display—Distance to the selected VOR or waypoint is shown in the upper left of the display, below the letters DIST. Resolution is to 0.1 nm below 100 nm and 1 nm above 100 nm. The digits are green for on-side distance data, and yellow for cross- side distance data. No computed data is shown as dashes (green on side/yellow cross side).

DME hold annunciator—Appears as a yellow H next to the DME display when HLD is selected on the VOR NAV control panel.

NAV data—When selected using the NAV DTA button on the CHP, data consisting of elapsed time, ground speed, or time-to-go is displayed below the letter ET, GSP, or TTG, as applicable. The source of TTG or GSP is that source to which the active course arrow is connected. The ground speed and time-to-go display are green for the on-side source and yellow for the cross-side source. The elapsed time display is green.

To/From indication—The To/From indication is a triangle the same color as the active course arrow. It indicates if the selected VOR course is To or From the station. It may also indicate the course to the next waypoint, or the course from an overflown waypoint when the LRN is the navigation source data.

When the HSI format is changed to ARC format, the triangle symbol changes to TO or FR, appearing at the lower right of the display just above HDG 1 or 2.

To/From indications disappear when a localizer frequency is selected or when the MAP format is presented.

The MAP format shows the VOR, VLF station as a waypoint with a solid course line to the station and a dashed line from the station.

Weather radar target alert—A flashing T appears when a target is detected by radar within 60 to 160 nm and +15° of the present heading (TGT selected on the WXP and mode selector not OFF or STBY).

DR (dead reckoning)—This annunciation is relative to LRN operation only. DR appears at the upper right side of the display when the Omega station signal is insufficient.

ARC format—The ARC format (Figure 16-37) selects a ±40° compass segment at the top of the display with the aircraft symbol at the bottom.

The active course is displayed as an arrow with the deviation bar and scale close to the aircraft symbol. A digital active course is shown at the lower right. Active course information is green for on-side source and yellow for cross-side source.

Annunciation of AHRS heading input is identical to the compass rose format (HSI).

When the selected heading bug is off scale, a digital heading readout appears at the appropriate end of the compass sector.

The preselect course pointer is replaced by a pictorial presentation of the selected sensor. The symbol is octagonal for VOR, and star-shaped for VLF/LRN.

When weather information is not being displayed in ARC format, range is selected via the SEL/RNG knob on the DCP. It is displayed as a dashed 120° blue arc halfway between the compass segment and aircraft symbol. Half of the full scale range is displayed digitally at the right end of the range arc. Selectable full scale ranges are 5, 10, 25, 50, 100, 200, 300, and 600 nm.

Weather information may be superimposed on the ARC format by pressing the RDR button on the DCP. When weather is being displayed, control is achieved through the weather radar control panel (WXP). The weather radar mode selected on the WXP is shown in blue at the left end of the range arc. The range ring is now a solid blue arc controlled by the range control on the WXP.

If the WXP is turned off, or a loss of data from the WXP is detected with weather displayed, the range ring and weather displays are removed and the range is set to 300 nm.

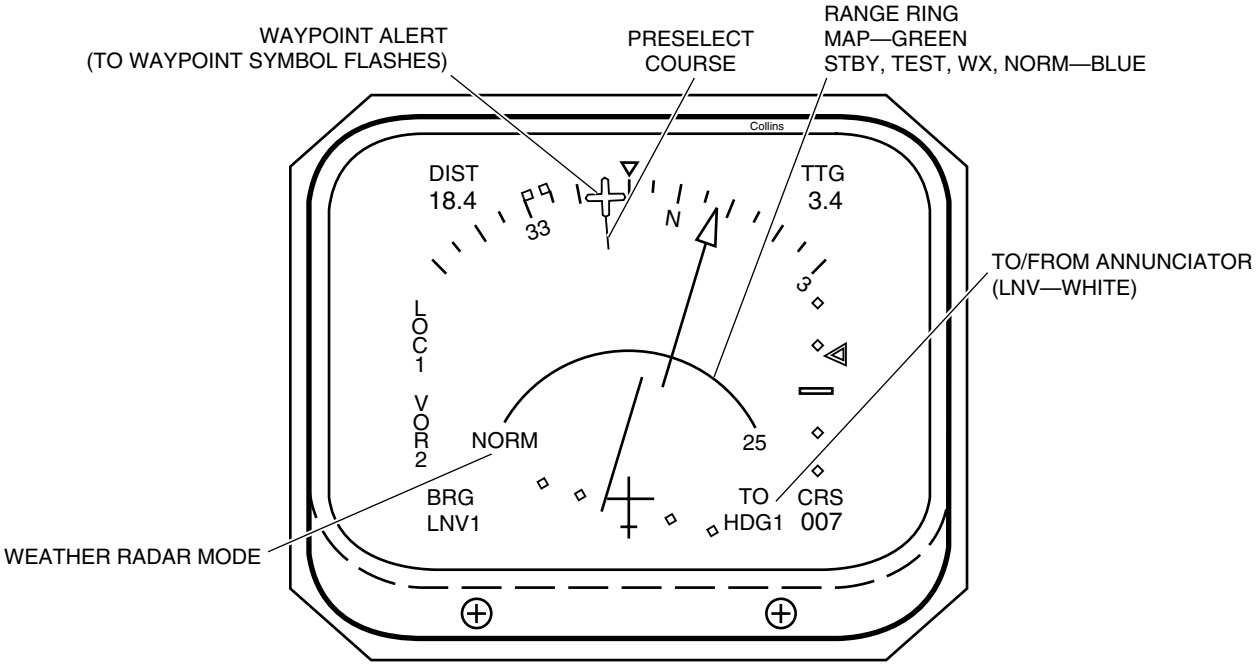


Figure 16-37. EHSI ARC Format

MAP format—The MAP format (Figure 16-38) is basically the same as the ARC format except the active course arrow pointer is also replaced by the pictorial presentation of the selected sensor.

If the NAV sensor is a VOR, the presented symbol is an octagon and includes a line which rotates to the course angle. This line is solid on the To side and dashed on the From side.

The station identification is shown close to the symbol. If the VOR symbol is off scale with respect to selected range and/or bearing, the course line is drawn with an arrow pointing toward the station, and the identifier is displayed on the line.

The preset course is displayed in blue and its sensor is annunciated in blue below the primary course annunciator.

The VOR symbol, course line, identifier, and sensor annunciation are green for on-side and yellow for cross-side sensors.

If the NAV sensor is VLF, a star-shaped waypoint symbol is displayed, identified by numbers close to the symbol and connected by track lines. If the To waypoint is off scale, a WPT number is displayed on the line. The waypoint symbol and sensor symbol are white for on-side and yellow for cross-side sensors.

Loss of VLF validity removes To/From indication.

Loss of VOR validity removes nothing.

Loss of VLF or VOR bearing data removes deviation bar and To/From indication.

Loss of VLF track data freezes the VLF course arrow.

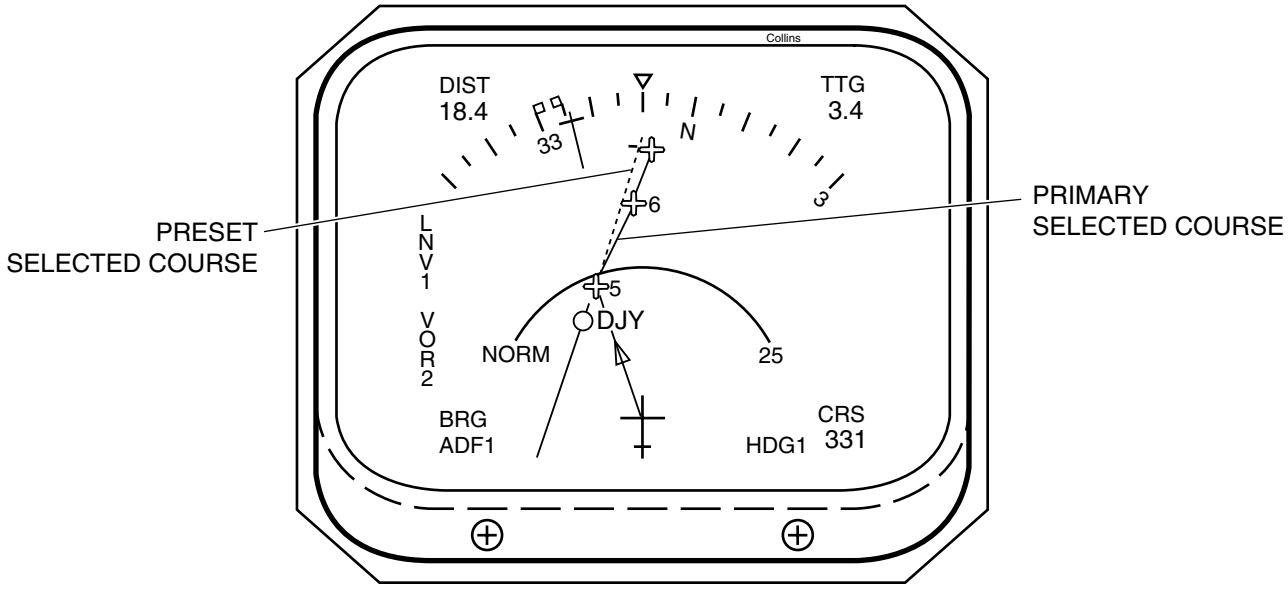


Figure 16-38. EHSI MAP Format

Failure Indications and Flags

All flags are red (Figure 16-39) and all of them, except the DPU FAIL flag, flash for 10 seconds before becoming steady. The DPU FAIL flag remains steady when activated.

Heading flag—HDG appears, replacing the lubber line, with a failure of the selected compass system.

Navigation source flag—The navigation source annunciation becomes red and boxed if the navigation data source fails, or with a loss of signal.

Bearing flag—The bearing source annunciation becomes red and boxed with a bearing source failure. In addition, the bearing pointer is removed from view.

Vertical deviation flag—The GS flag appears in the event of a source failure, or if a LOC signal is tuned when a LRN is being used in a vertical NAV mode. The glide-slope pointer and scale are removed with a source failure or loss of signal.

Distance flag—Failure of the DME is indicated by the distance digits being replaced by red dashes.

NAV DTA Flag—A source failure in the NAV DTA is indicated by red dashes replacing the digits.

Data processing unit flag—The DPU FAIL flag indicates a DPU failure. If the flag remains more than 5 seconds, the display blanks except for the fail message.

Display control panel flag—The DCP flag indicates either a display control panel or course heading panel failure.

Cross-side data bus flag—The XDTA flag appears with a failure of the cross-side data bus. This removes any cross-side data.

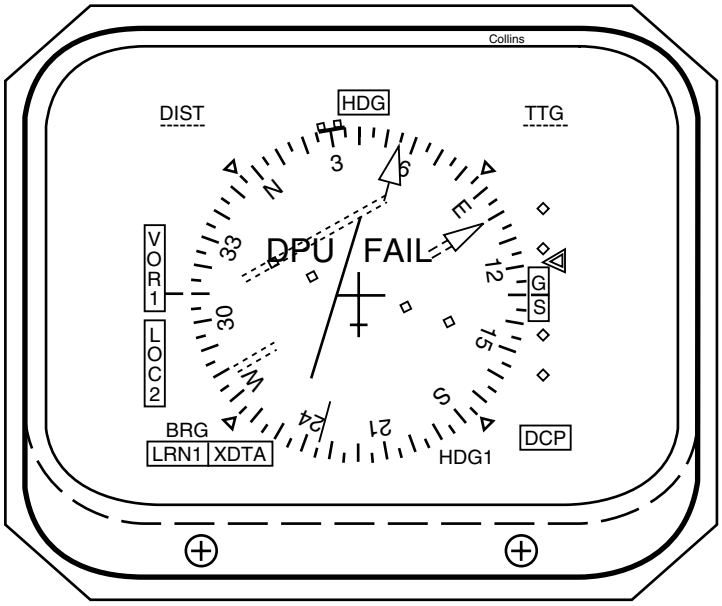


Figure 16-39. EHSI Failure Indications and Flags

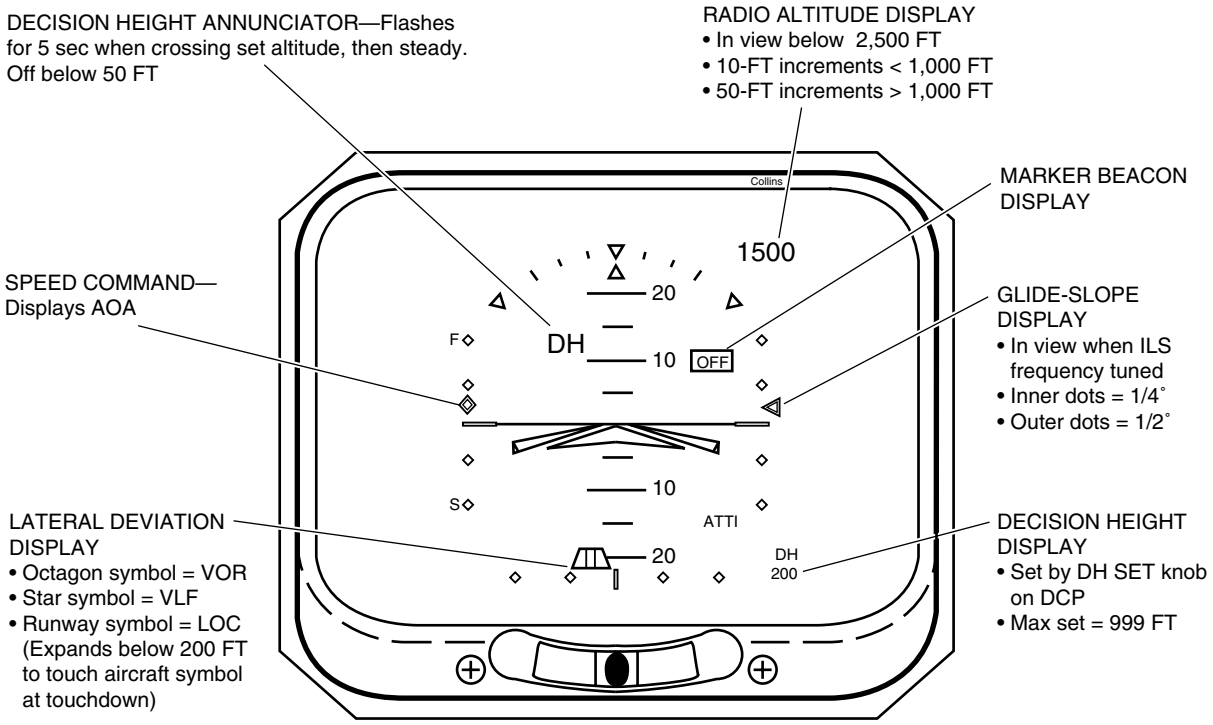


Figure 16-40. EADI Normal Format

EADI Navigational Display

Formats

The following applies to the EADI normal and composite formats as they relate to the display of navigation information.

Normal format—The normal EADI format (Figure 16-40), displays navigation information for the approach phase.

Glide-slope display—The glide-slope deviation is displayed with a scale and pointer on the right side of the EADI. The scale consists of two white dots above and below a white center index. The inner scale dot represents a 1/4 degree displacement, and the outer scale dot represents 1/2 degree. The triangle pointer (green on side/yellow cross side) represents the center of the glide-slope beam.

The glide-slope scale and pointer appear on the display when an ILS frequency is tuned on the NAV radio.

Lateral deviation display—The lateral deviation display at the bottom center of the EADI consists of a scale of four dots and a symbol which moves laterally right or left. The symbol is octagonal for VOR, star-shaped for VLF, and a trapezoid (runway) for LOC.

Both VOR and LOC symbols are green on side/yellow cross side. The VLF symbol is white on side/yellow cross side.

During the final phase of the approach, the LOC symbol (runway) expands both vertically and laterally. It begins to expand at 200 feet radio altitude, and continues until zero radio altitude where the top edge of the symbol touches the bottom edge of the aircraft symbol.

Radio altitude display—The radio altitude is only displayed below 2,500 ft. From 2,500 to 1,000 feet it changes in 50-foot increments. Between 1,000 feet and SL, it changes in 10 foot increments.

Decision height display—The decision height, selected with the DH SET knob on the DCP, is displayed in the lower right corner of the EADI. The maximum selection is 999 feet.

Decision height annunciator—When the aircraft crosses the selected DH, the DH letters flash for 5 seconds and then become steady. The DH extinguishes when radio altitude is 50 feet or less.

Speed command—The speed command, representing angle-of-attack input, is indicated by a scale and a pointer on the left side of the display. The pointer, if centered, represents 1.3 V_S. (For more information, refer to stall warning system description in Chapter 15, "Flight Controls.")

Marker beacon display—Marker beacon passage is indicated by a flashing symbol inside a box on the right center of the EADI. The symbols are as follows:

- OM—outer marker (blue box)
- MM—middle marker (yellow box)
- Inner marker and airways—white box, no inscription

Composite format—The composite format (Figure 16-41) is used in the event of an EHSI failure. The EHSI display is transferred to the EADI as an ARC, displayed in the lower section of the full EADI display.

The composite format is selected by placing the COMPOSITE MODE switch on the EFIS CONTROL PANEL to EADI. The image may be presented on the EADI, EHSI, or the MFD.

Failure Indications and Flags—As with the EHSI, the EADI flags (Figure 16-42) are red, flash for 10 seconds, then become steady. DPU FAIL comes on steady.

Attitude flag—ATT appears when attitude information from the on-side AHRS is invalid or lost. When displayed, the pitch and roll scales, roll pointer, sky and ground raster, and command bars are all removed.

Glide-slope flag—GS appears with glide-slope failure, or when no ILS reception is sensed.

Flight director flag—A failure of the flight director system results in FD displayed below the left wing of the airplane symbol, and removal of the command bars.

Lateral deviation flag—VOR or LOC is displayed with a failure of the VOR or localizer receiver. LOC also appears with a localizer signal interruption. Loss of VOR or VLF data removes the relative pointers.

Radio altitude flag—RA appears when the radio altimeter fails. It removes radio altitude information, DH, and DH set capability.

Speed command flag—With a stall warning system failure, the display is removed and SPD is displayed.

Display processor flag—DPU FAIL appears with a failure of the DPU. After 5 seconds, the rest of the display blanks. If driven by the MPU, the flag is MPU FAIL.

Display control panel flag—DCP appears with a failure of either the display control panel or course/heading panel. Also removes DH and DH set capability.

Cross-side data flag—XDTA appears with a failure of the cross-side data bus. Displays requiring cross-side data are flagged.

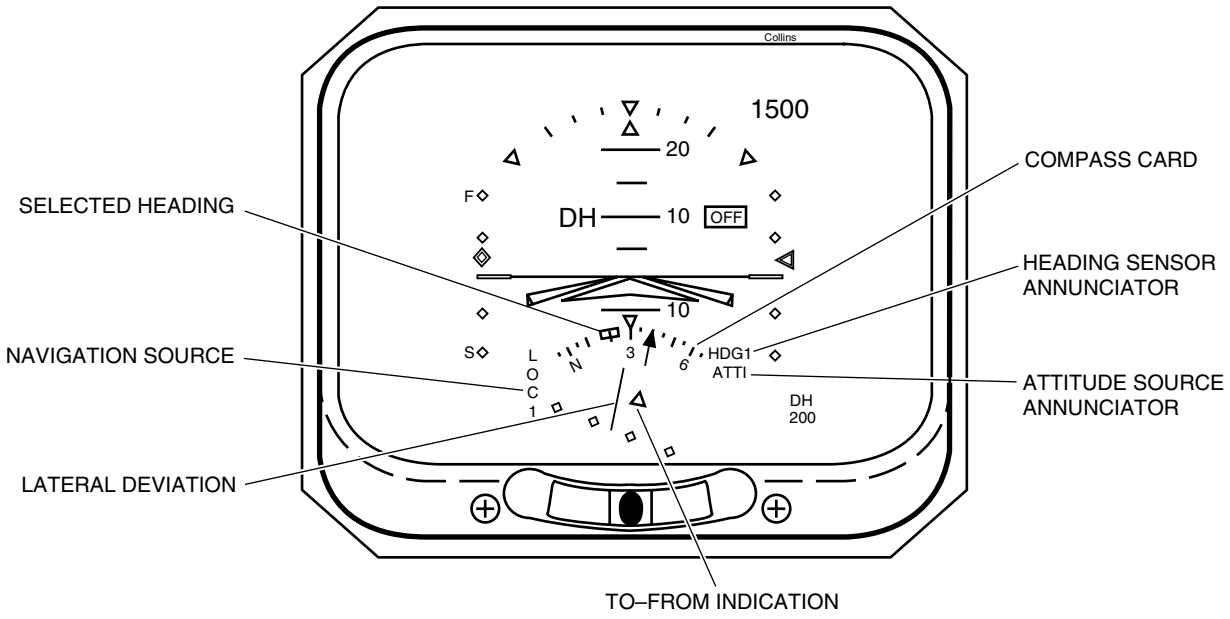


Figure 16-41. EADI Composite Format

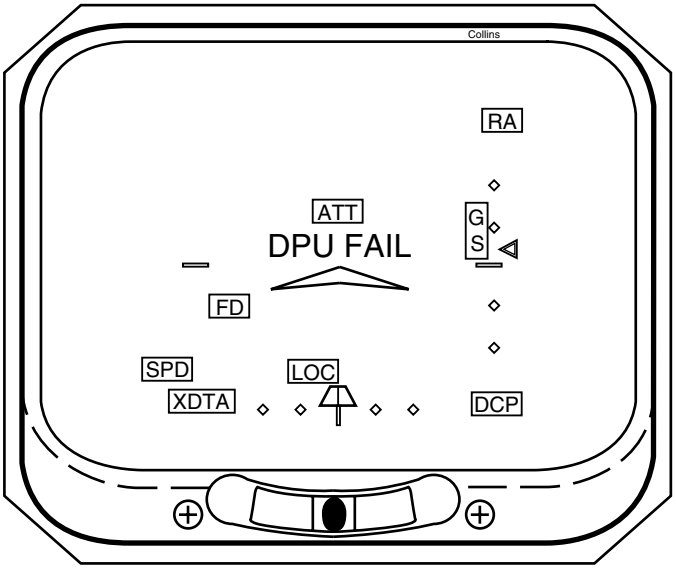


Figure 16-42. EADI Failure Indications and Flags

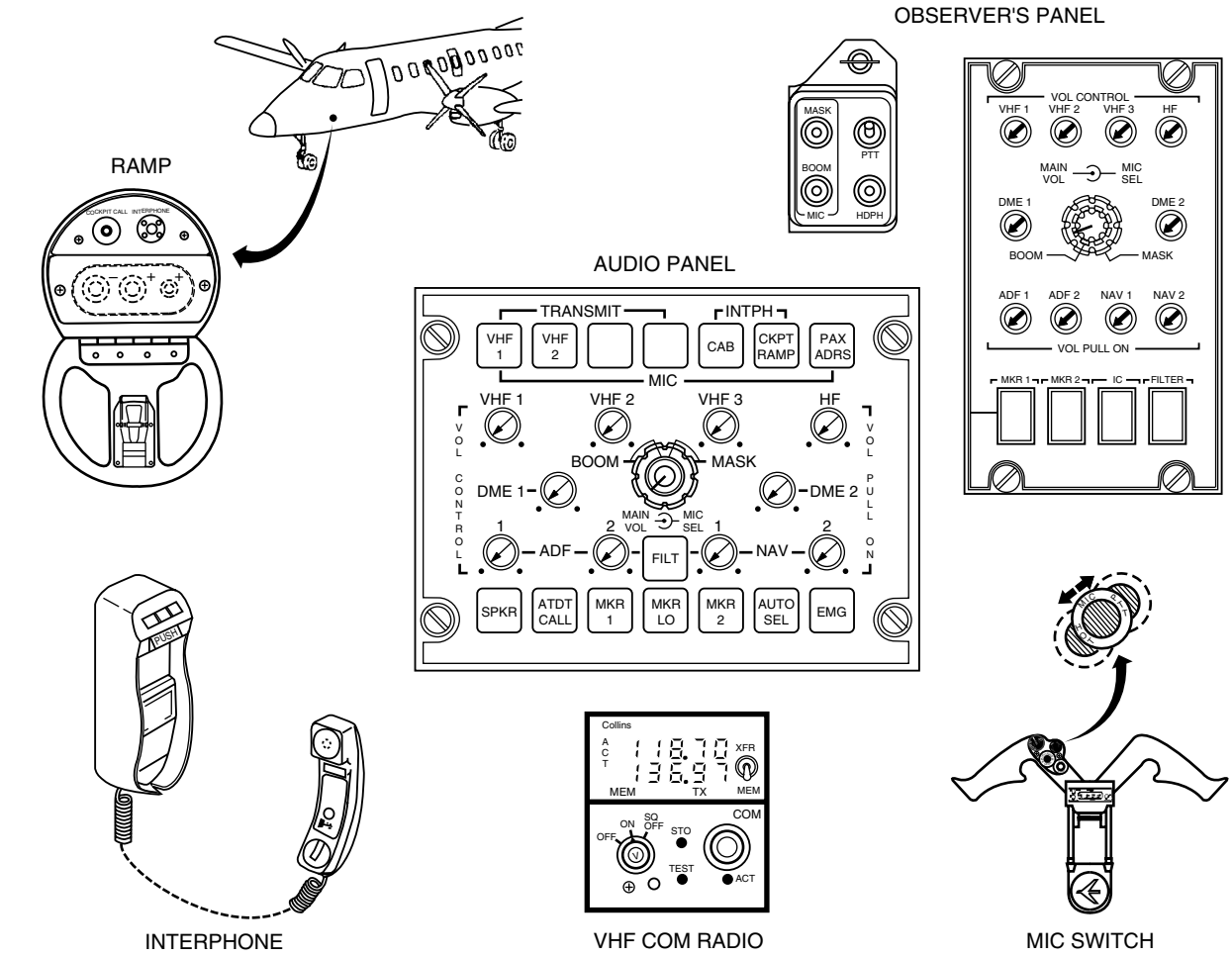


Figure 16-43. Communication System

COMMUNICATION SYSTEM

GENERAL

The EMB-120 communication system provides the aircrew with the capability to talk to:

- Ground stations and other aircraft, using the VHF radio
- Each other and the flight attendant on the intercom
- The passengers, using the passenger address system
- Ground-handling personnel on the ramp interphone

There are also optional capabilities for private company communications using a SELCAL system and a long range HF system.

COMPONENT

The major components that make up the communications system are listed below:

- Audio panel
- VHF COM radios
- SELCAL system (optional)
- HF COM system (optional)

The remaining components are the various switches and connections in the cockpit/cabin and the observer's and external panels.

Audio Panel

The EMB-120 has two audio panels, one for the pilot and copilot, mounted on the left and right side of the forward instrument panel.

The audio panel (Figure 16-43) functions as a mixer box to select the desired transmitter/receiver, interphone, or passenger address system; as well as monitor any of the other radio signals received.

Microphone Selector Switches

VHF 1 and VHF 2—Selects respective VHF for transmission.

CAB—Generates a HI–LO tone in the passenger address system and flashes in conjunction with the CAB switch on the attendant's handset.

CKPT RAMP—Provides for communication between the ramp station and the cockpit. To talk, the control wheel MIC switch must be used.

PAX ADRS—Connects the respective microphone/headset to the passenger address system, and generates a HI tone on the system. The control wheel MIC switch must be used.

Receiver Switches

VOL CONTROL/VOL PULL ON—Provides audio on/off and volume control for VHF 1, VHF 2, DME 1, DME 2, ADF 1, NAV 1 and NAV 2.

The receiver switches are all the pull-on type. The illumination increases in intensity when the switch is pulled. If two or more switches are pulled simultaneously their signals are superimposed. Twisting left or right decreases or increases the volume.

NOTE

VHF 3 and HF switches are deactivated.

MAIN VOL–MIC SEL—A stacked knob in the center of the panel. The inner knob is the master volume control for the headphones and cockpit speakers. The outer knob selects either the BOOM or MASK microphone.

SPKR switch—Routes the audio signal to the cockpit speaker in addition to the headphone. The volume is controlled by the MAIN VOL knob.

ATDT CALL switch—Signals the passenger address to emit a HI–LO tone (to call attendant to the cockpit).

MKR 1 and MKR 2 switches—Selects the respective marker beacon.

MKR LO switch—The marker beacon sensitivity switch changes the sensitivity of the marker beacon. Normally the marker sensitivity is low. Pressing the switch changes the sensitivity to high. Changing the sensitivity on one audio panel also changes the sensitivity on the other.

AUTO SEL switch—Provides automatic selection of the audio receiver for the selected transmitter.

EMG switch—In the event one audio panel fails, the emergency switch causes the headsets to parallel and the microphone to be connected to VHF1 for the pilot or VHF2 for the copilot.

If both audio panels fail, or power is lost, the pilot's microphone and headphone is connected to VHF1 and ADF1 and the copilot's to VHF2 and NAV 1.

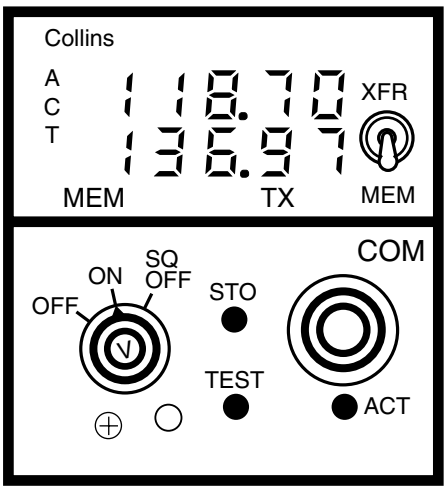


Figure 16-44. VHF COM Radio Control

VHF COM Radio

General

The EMB-120 has two VHF COM radios. The CTL-22 COM controls (Figure 16-44) are located on the center instrument panel, either side of the MFD. Each radio has 720 channels, from 118.000 through 135.975 MHz, spaced 0.025 MHz apart.

Displays

The frequencies displayed on the COM control show only five of the six digits. The sixth digit is always a 0 (when the right digit is 0 or 5), or a 5 (when the fifth digit is a 2 or 7). Therefore, the sixth digit does not need to be displayed.

Active frequency display—The upper display is the active frequency (frequency to which the equipment is tuned) and also displays diagnostic messages.

Preset frequency display—The lower display is the preset (inactive) frequency and also displays diagnostic messages.

Controls and Indicators

Power and mode switch—The ON and OFF positions control system power. The SQ OFF disables the receiver squelch circuits, so noise should be heard. This position is used to set volume control or, if necessary, to try to receive a very weak signal that cannot override the squelch circuits.

Frequency select knobs—Two concentric knobs control the preset or active frequency displays. During normal operation, the frequency select knobs change the preset frequency display.

The large knob changes the three digits to the left of the decimal point in 1-MHz steps. The smaller knob changes the two digits to the right of the decimal point 0.05-MHz steps (or in 0.025-MHz steps for the first two steps after the direction of rotation has been reversed). Numbers roll over at the upper and lower frequency limits.

XFR–MEM switch—This is a 3-position toggle switch, spring loaded to the central neutral position. When held to the XFR position, the preset frequency is transferred up to the active display and the radio retunes. The previously active frequency becomes the new preset frequency and is displayed in the lower window.

When the switch is held to the MEM position, one of the six frequencies stored in the memory is recalled into the preset display. The storage location for the frequency (CH1 through CH6) is displayed in the upper window while the switch is held. All six stored frequencies may be displayed, one at a time, by repeatedly toggling the switch to the MEM position.

ACT button—The ACT button is used to enable the frequency select knobs to directly retune the active frequency.

1. Push the ACT button for approximately 2 seconds.
   - The bottom window displays dashes.
   - The upper window continues to display the active frequency.
2. Dial in the desired active frequency using the frequency select knobs.
   - The ACT annunciator flashes as the frequency is dialed in, indicating that the receiver is being retuned.
   - If the ACT annunciator continues to flash after the frequency has been selected, it indicates that the receiver is not tuned to the displayed active frequency.
3. Push the ACT button a second time to return the control to the normal display.

ACT annunciators—Momentarily illuminates when frequencies are being changed. Flashes if the radio is not tuned to the active frequency displayed.

MEM annunciators—Illuminates whenever a frequency is being displayed in the lower window.

TX annunciators—Illuminates whenever the radio is transmitting.

Store (STO) button—Allows up to six preset frequencies to be selected and entered into the non-volatile memory. To program the memory:

1. Dial the frequency to be stored into the preset display.
2. Press the STO button.
   - The upper window displays the channel number of available memory (CH1 through CH6).
   - The lower window continues to display the frequency to be stored (for approximately 5 seconds).
3. The MEM position may be used to advance through the channel numbers without changing the preset display.
4. Push the STO button a second time to commit the preset frequency to memory in the selected location.

The second actuation of the STO button must be done within 5 seconds of selecting the channel number or the first actuation of the STO button. If more than 5 seconds elapse, the control reverts to the normal mode of operation.

TEST button—The self-test diagnostic routine is initiated in the transceiver by pushing the TEST button.

Self-Test

The intensity of the upper and lower displays modulate from minimum to maximum to indicate that self-test is in progress.

After pressing the TEST button, several audio tones are heard from the audio system while the self-test routine is being executed.

At the completion of the self-test program:

- Normal operation, no trouble found is indicated by dashes in the active (upper) display and 00 in the preset (lower) display.
- Out-of-limit conditions are indicated by DIAG in the active display and a 2-digit fault code in the preset display.

Record any fault codes displayed to assist the service technician in locating the problem.

Control Wheel Communication Switches

Both control wheels are equipped with a MIC switch (Figure 16-45). The fixed HOT position provides continuous communication between pilot and copilot. The momentary PTT position keys VHF transmission and passenger address through the microphone.

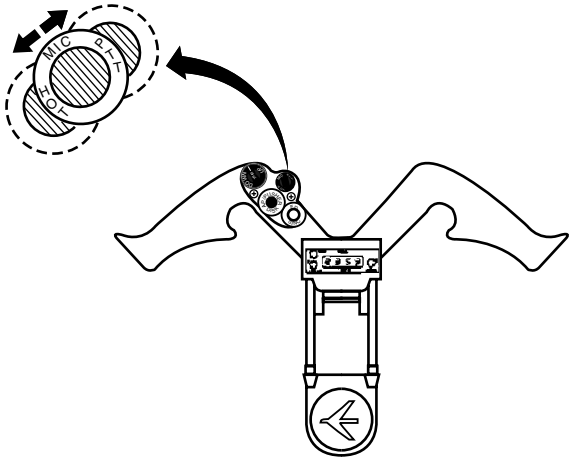


Figure 16-45. Control Wheel MIC Switch

Attendant's Interphone

The attendant's interphone (Figure 16-46) is a hand-held phone installed at the flight attendant's station. It permits communication between attendant and pilot or copilot, as well as passenger address.

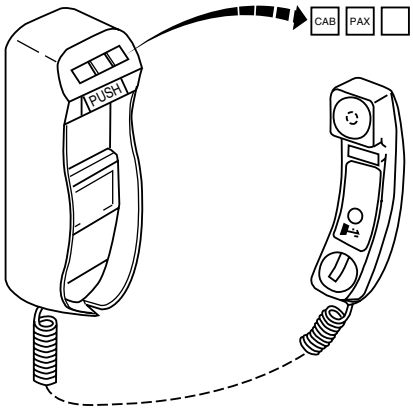


Figure 16-46. Attendant's Interphone

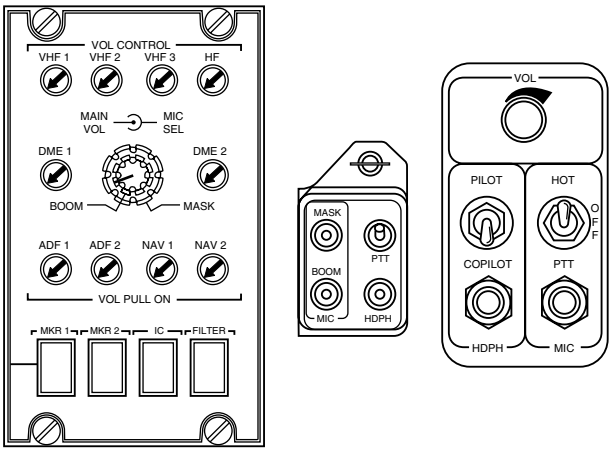


Figure 16-47. Observer's Audio Panel

Observer's Panel

The observer's audio panel (Figure 16-47) is installed on the right side of observer seat. It allows the observer various levels of communication depending upon customer option.

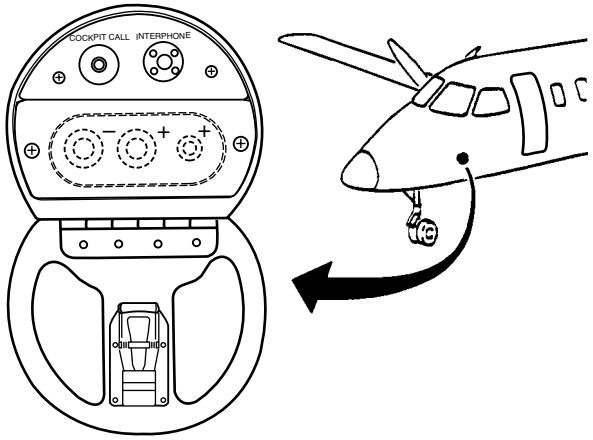


Figure 16-48. Ramp Connection

Ramp Connection

The ramp connection (Figure 16-48) provides for communication between crew and ground attendant. It consists of a cabin call button and headphone/mike jack, located on the external power panel on the aircraft nose.

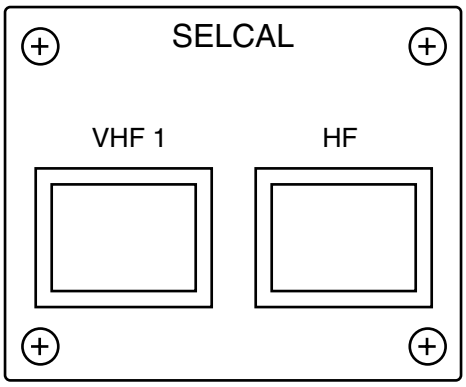


Figure 16-49. SELCAL System

SELCAL System (Optional)

The airplane may be equipped with the optional SELCAL system (Figure 16-49). The system operates together with VHF1 or HF. It consists of a SELCAL panel, installed on the copilot's panel, and a voice message through the aural alarm system.

HF System (Optional)

The aircraft may be equipped with a high frequency (HF) communications system (Figure 16-50) for long-range communication. The HF equipment also operates in conjunction with the SELCAL system.

AUTOFLIGHT

GENERAL

The autoflight system (Figure 16-51), is a fully integrated three axis flight control system including manual electric trim. It consists of two sub-systems:

- Flight director system (FDS)
- Autopilot system (APS)

Each of two FDSs uses the command bars on the EADI to display vertical and horizontal steering commands from its flight guidance computer.

The APS provides three axis control of the aircraft and, when used in conjunction with a FDS, automatically controls the aircraft as directed by the flight guidance computer commands.

Flight Director System (FDS)

In addition to the EADIs and EHSIs, each of the two FDSs consists of the following major components:

- Flight control panel
- Flight guidance computer

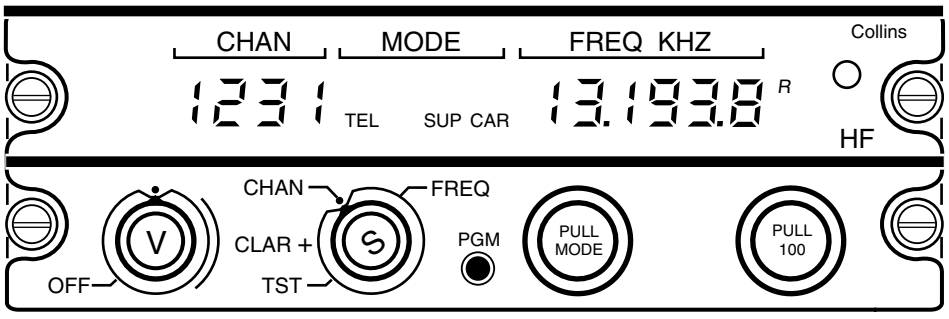


Figure 16-50. HF Panel

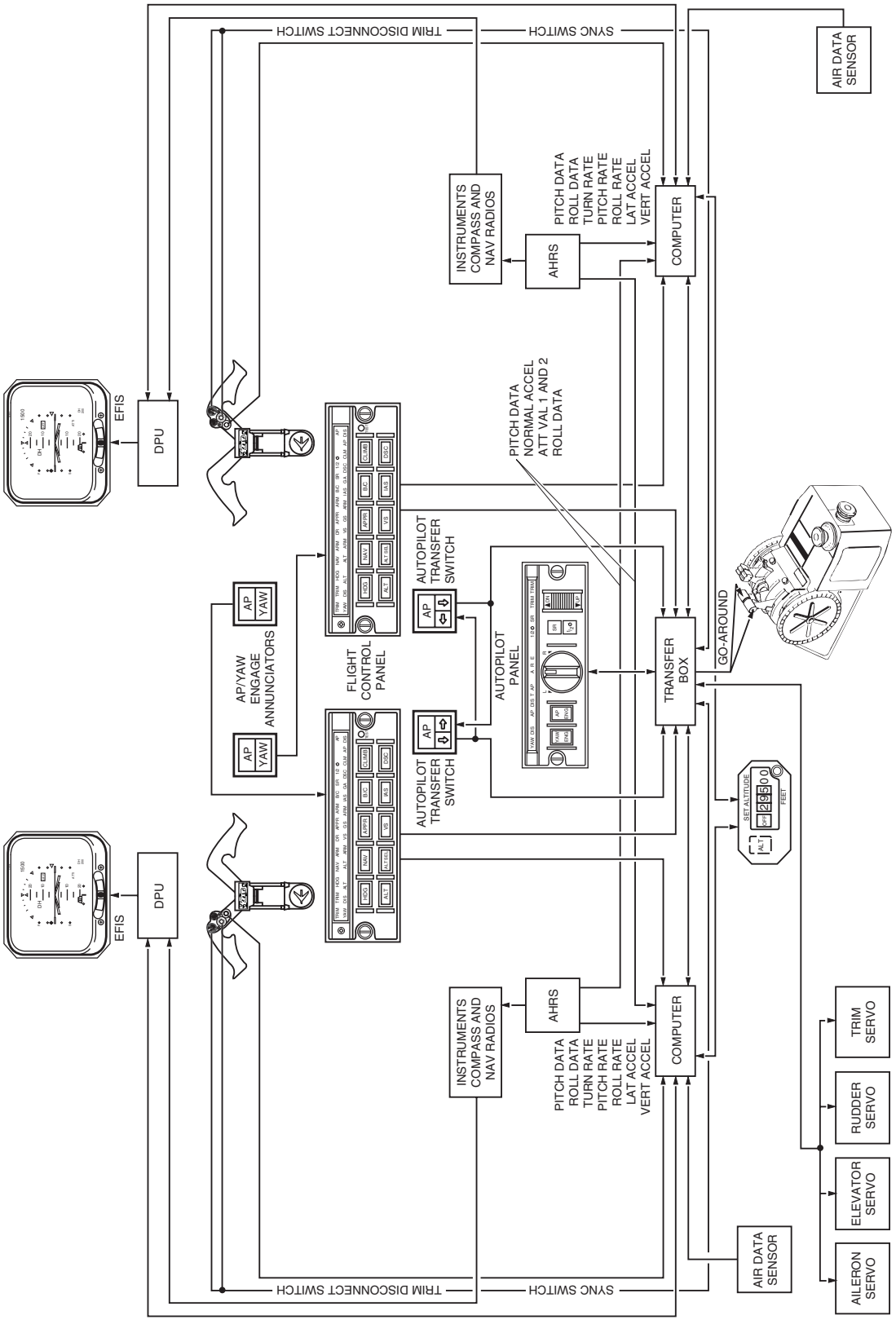


Figure 16-51. Autoflight System

The EADIs display attitude, raw radio information, and steering from the computers. The EHSIs display the navigation situation.

The flight director modes are selected and monitored on the flight control panel.

The flight guidance computer receives inputs from various sources, including the flight control panel. It processes these inputs and provides both lateral and vertical steering commands to the EADI command bars.

Autopilot System

The APS is made up of, or receives input from, the following:

- Two flight guidance computers
- Autopilot panel
- Transfer box
- Air data sensors
- Four servos (one trim and three primary)
- AHRS

The APS also requires navigation system inputs from the avionics systems.

The APS uses the same guidance computers as the FDS. The computers drive the four autopilot servos through a transfer box.

The transfer box integrates the autopilot panel, flight control servos, and FDS. It is controlled by the transfer switches on the glareshield. The selected side provides the component input to be used with the autopilot panel input for autoflight. The nonselected side serves as an independent flight director.

The air data sensors provide pressure altitude and IAS outputs for the ALT hold, IAS hold, and VS hold modes.

The AHRS provides the pitch and roll input for the computer. In addition, it provides the computer with turn and roll rates, as well as lateral and vertical acceleration for proper flight director/autopilot computation.

The altitude preselect alerter provides the computers with altitude preselect error for the altitude preselect mode computation.

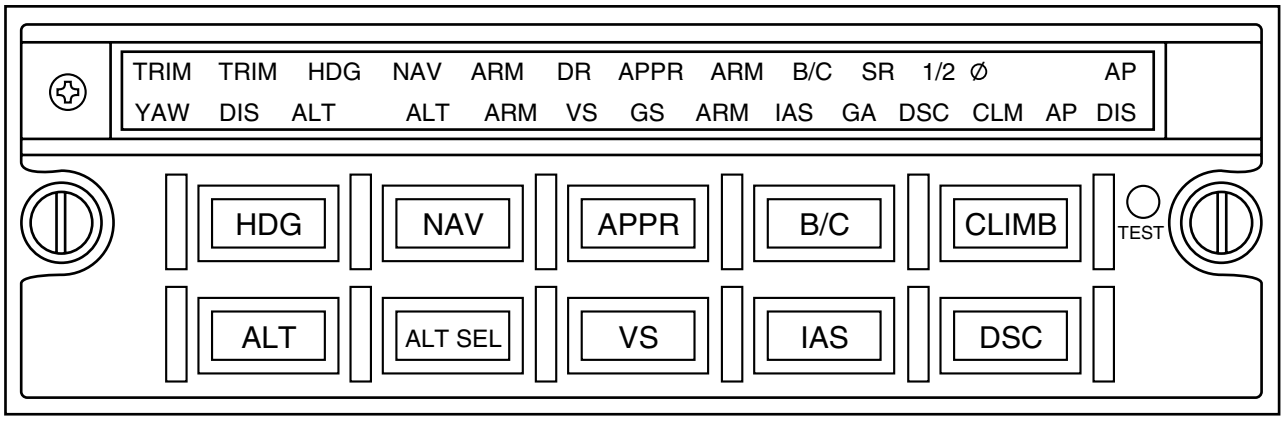


Figure 16-52. Flight Control Panel

COMPONENTS

Flight Control Panel

General

The flight control panel (Figure 16-52) is used to select modes of operation for the FDS or APS. The panel also displays all system communication and is the controlling point for system tests.

Mode Select Buttons

The mode select buttons are momentary, on/off pushbuttons, used to select the operational modes of the autoflight system. Mode selection is interlocked to ensure that only compatible modes are selected at the same time. When no lateral modes are selected, the command bars on the EADI are biased out-of-view. Mode selection is annunciated on the flight control panel.

TEST Button

The TEST button is a momentary-action pushbutton that selects the diagnostic mode.

Mode Annunciators

TRIM (White)—Trim system is operating.

TRIM (Red)—Trim system fails. The red TRIM FAIL light on the MAP and a “Trim Fail” voice message accompany this light.

HDG (Green)—Heading mode is selected. (Also illuminates when NAV, APPR, or BC is selected but not captured.)

NAV (Green)—NAV mode is selected and captured.

ARM (NAV) (White)—NAV mode is selected but not captured.

DR (Green)—System is in dead-reckoning mode.

APPR (Green)—APPR mode is selected and captured.

ARM (APPR) (White)—APPR mode is selected (after an ILS, VOR, or LOC frequency is tuned), but not captured.

B/C (Green)—BIC light illuminates after APPR and B/C modes are selected.

SR (Green)—Soft ride is selected.

1/2 Ø (Green)—Half bank mode is selected.

AP (Red)—Autopilot failure. The red AUTO PLT FAIL light on the MAP and an "Autopilot" voice message accompany this light.

YAW (Green)—Yaw damper engaged.

DIS (YAW) (Amber)—Yaw damper is disengaged. Flashes for 7 seconds, then extinguishes. Also illuminates when yaw damper is engaged and the SYNC button is pressed.

ALT (Green)—Altitude hold mode selected, or ALT SEL is selected and captured.

ALT (Green) ARM (White)—Altitude preselect mode is armed for automatic capture.

VS (Green)—Vertical speed hold mode is selected.

GS (Green)—Glide slope is captured in APPR mode (front-course approach).

ARM (GS) (White)—APPR mode is selected, LOC frequency tuned, and glide-slope valid received (front-course approach).

IAS (Green)—Indicated airspeed hold mode is selected.

GA (Green)—Go-around mode is selected.

DSC (Green)—Descent mode is selected.

CLM (Green)—Climb mode is selected.

AP (Green)—Autopilot is engaged.

DIS (AP) (Amber)—AP is disengaged. It flashes for 7 seconds, then extinguishes.

Flashing Mode Annunciators

Flash only when mode is selected.

HDG (Green)—This light indicates loss of attitude or compass.

NAV (Green)—This indicates loss of attitude, compass, or NAV radio input.

APPR (Green)—This indicates loss of attitude, compass, or NAV radio input.

DIS (YAW) (Amber)—Yaw damper is disengaged. Flashes for 7 seconds, then extinguishes.

ALT (Green)—Loss of attitude or air data sensor is indicated.

ALT (Green) and ARM (White)—Loss of attitude or preselector.

VS (Green)—This light indicates loss of attitude or glide slope.

IAS (Green)—This indicates loss of attitude or air data sensor.

GA (Green)—Loss of attitude is indicated.

DIS (AP) (Amber)—AP is disengaged. It flashes for 7 seconds, then extinguishes.

Self Test

Pressing the button initiates the following:

- All annunciators illuminate and go out.
- The GA annunciator illuminates again, indicating no fault. (Illumination of any other annunciator is a fault indication.)

Press the TEST button again. All annunciators illuminate and go out.

When the test is performed, the red AUTO PILOT FAIL light on the MAP illuminates and the "Auto-pilot" message is heard.

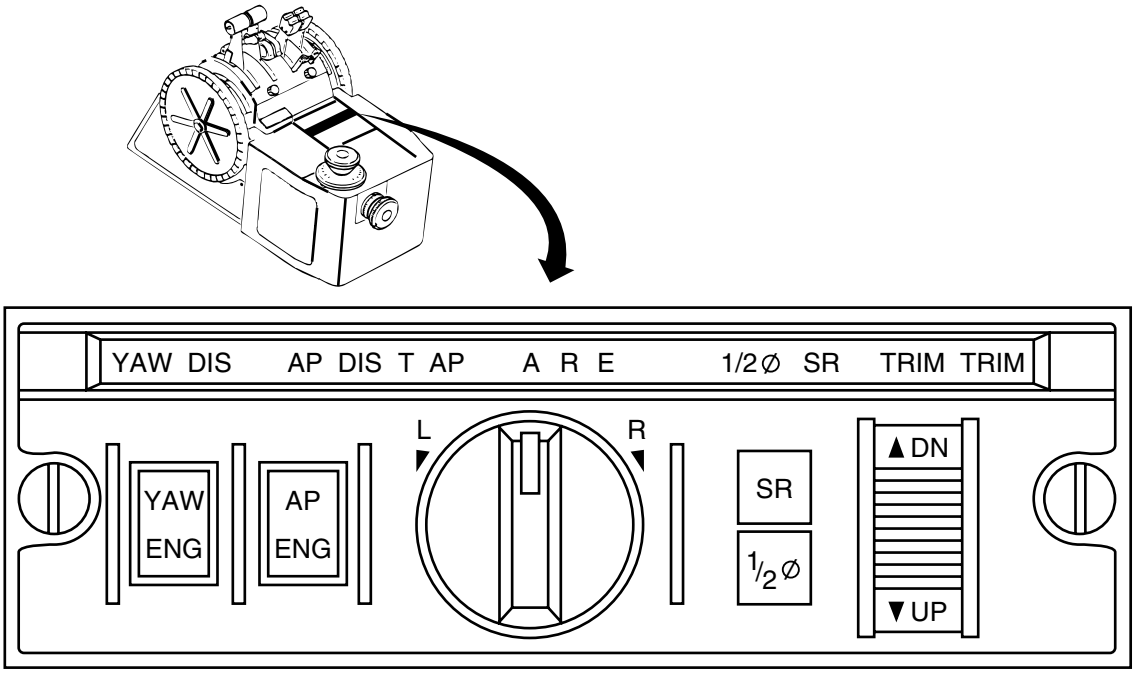


Figure 16-53. Autopilot Panel

Autopilot Panel

General

The autopilot panel (Figure 16-53), controls the autopilot function in the autoflight system.

Annunciators

YAW (Green) - Yaw damper is engaged.

DIS (YAW) (Amber)—Yaw damper is disengaged. Flashes for 7 seconds, then extinguishes. Also illuminates when yaw damper is engaged and the SYNC button is pressed.

AP (Green) - Autopilot is engaged.

DIS (AP) (Amber)—Autopilot is disengaged. Flashes for 7 seconds, then extinguishes. Also illuminates when autopilot is engaged and the SYNC button is pressed.

T (Amber)—Trim servo failure is indicated.

AP (Red)—Autopilot failure is indicated.

A (Amber)—Aileron servo failure is indicated.

R (Amber)—Rudder servo failure is indicated.

E (Amber)—Elevator servo failure is indicated.

1/2 Ø (Green)—Half bank mode is selected.

SR (Green)—Soft ride is selected.

TRIM (Red)—Trim system failure is indicated.

TRIM (White)—Trim system is operating.

Autopilot Controls

YAW ENGage button—This is a momentary action, on/off pushbutton. When pushed, it engages the yaw channel of the autopilot and the rudder servo. The second push disengages the rudder servo only.

AP ENGage button—This also is a momentary action, on/off pushbutton. When pushed, it engages all autopilot servos, including the rudder and trim servos. Pushing a second time disengages all servos except the rudder servo. (The rudder servo is disengaged with the YAW ENG button or the AP disengage button on the control wheel).

Turn control (L/R)—A left/right (L/R) bank-angle control knob is used to manually control the aileron channel when the autopilot is engaged. There is a center detent position at the 0 degree (wings level) position. The control remains at any position between the end stops when released.

Operation of the control cancels any previously selected lateral modes, except APPR. The bank angle commanded by the control is proportional to the displacement of the control from the center detent position. The turn knob becomes inactive if it is out of the detent position when the SYNC button is pressed. The autopilot will not engage if the turn knob is out of detent.

SR Button—This is a momentary action, on/off pushbutton. It selects the soft-ride mode, which reduces autopilot gains to increase passenger comfort during turbulent conditions.

1/2 Ø bank button—A momentary action, on/off pushbutton, it selects the half-bank mode, which limits the maximum commanded roll angle to one-half of the normal value in the heading mode.

Vertical Trim Switch—The UP–DN vertical trim switch is a center-off, spring loaded rocker switch that provides manual control of the elevator channel.

Operation of the switch gives an initial 0.5° pitch change and then a constant pitch rate (up to the pitch attitude limit if the switch is held).

If a vertical mode (ALT, IAS, or VS) is selected, the vertical trim switch gives a preset step change in that parameter.

When the switch is released, the autopilot maintains the new pitch attitude.

The vertical trim switch is not operational after glide- slope capture.

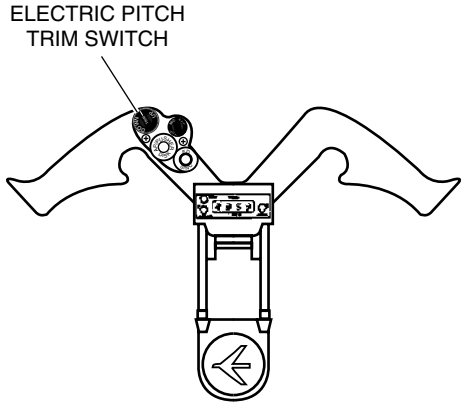


Figure 16-54. Pitch Trim Switch

Remote Switches and Annunciators

Electric Pitch Trim Switch

The electric pitch trim switch (Figure 16-54) is a split, 3-position (UP–OFF–DN) thumb switch mounted on each control column. It operates the autopilot pitch trim servo when the autopilot is not engaged. (Elevator trim is automatic when the autopilot is engaged.) Operation of the switch with the autopilot engaged causes the autopilot to disengage.

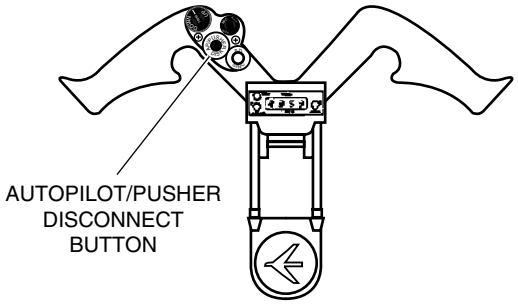


Figure 16-55. Autopilot/Pusher Disconnect Button

AP/PUSHER DISConnect Button

Located on the control column, the autopilot and pusher disconnect button (Figure 16-55) disengages both the autopilot and yaw damper (and control wheel pusher).

Actuation of the button disengages all autopilot servos and, if no lateral modes are selected, cancels all selected vertical modes. The vertical modes may be reselected after the autopilot has disengaged and a lateral mode has been selected.

If lateral and vertical modes are selected, they are retained and may be used in the flight director mode (with no automatic operation of the control surfaces of the aircraft).

AP SYNC Button

Located also on the control column, the autopilot SYNC button (Figure 16-56) is used to manually maneuver the aircraft without completely disengaging the autopilot.

Pressing the button disengages the primary servos without interrupting computations for selected lateral or vertical modes. The aircraft may be maneuvered to any desired pitch attitude while the button is pressed.

When the button is released the primary servos reengage, and the computer synchronizes to the new pitch attitude or vertical mode and maintains it. Lateral control is returned to the previously selected lateral mode. (Return to the lateral mode is restrained to prevent rapid maneuvers when the roll attitude varies from that commanded by the autopilot.)

In APPR mode after glide-slope capture, pressing the SYNC button has no effect on vertical commands (does not cancel the vertical mode). However, the SYNC button still selects the sync mode if the autopilot is engaged. When the SYNC button is released, commands are generated to return the aircraft to the center of the glide slope.

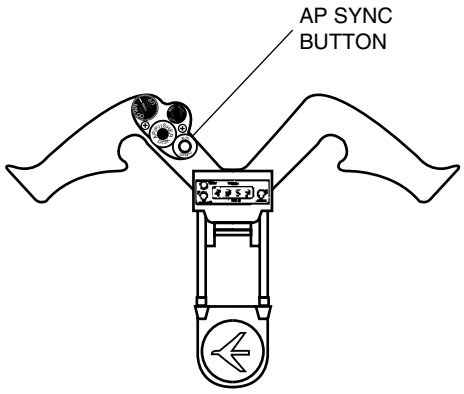


Figure 16-56. Autopilot SYNC Button

GO-AROUND Button

A go-around button is located on the outboard side of each power lever knob. Pressing either button selects the autopilot go-around mode.

AP Engage/YAW Annunciators

The AP engage and YAW annunciators on each glareshield (Figure 16-57) illuminate when the autopilot or the yaw channel are engaged on the opposite side.



Figure 16-57. AP Engage/YAW Annunciators

Autopilot Transfer Switches

Located on the glareshield below the AP/YAW annunciators, the autopilot transfer switches, (Figure 16-58) use the transfer box to select the pilot's or copilot's flight guidance computer. The appropriate arrow illuminates to indicate which computer is selected. Transferring the autopilot causes it to disengage. The AP arrows turn on when the aircraft is energized, defaulting to the pilot's flight guidance computer.

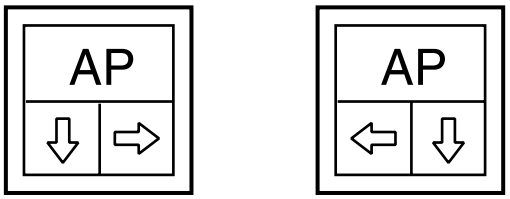


Figure 16-58. Autopilot Transfer Switches

OPERATING MODES

All autopilot functions are controlled by the auto-pilot panel. All operating modes for the autoflight system are selected on the flight control panel. All operating modes may be selected with the autopilot engaged or disengaged. However, engaging or disengaging the autopilot cancels any selected vertical modes.

Flight guidance computer outputs drive the steering display on the EADI. (A lateral mode must be selected to bring the command bars into view.) When the autopilot is engaged, the same guidance commands provide automatic control of the aircraft.

Lateral Modes

Roll hold is the basic mode of the lateral channel. When the system is in the roll hold mode (with no lateral modes selected), the command bars on the EADI are biased out of view.

Roll hold may be selected by deselecting all lateral modes or, if the autopilot is engaged, by rotating the turn control knob on the autopilot panel out of detent.

The bank angle may be changed with the turn control knob or the SYNC button. The turn control knob commands the aircraft to roll smoothly to a roll attitude proportional to knob displacement, with a bank limit of 30°. Actuating the SYNC button allows the aircraft to be flown manually. The system returns to the previously selected lateral mode when the button is released.

Pressing the SYNC button with the turn control knob out of detent inactivates the knob. When the button is released, the aircraft returns to the turn control knob command.

With the turn control knob in detent or the SYNC button released with less than 5° bank, the system switches to a wings-level attitude-hold submode, and the aircraft is rolled out to a wings-level position.

If the turn control knob is in detent and the bank angle is changed, either by using the SYNC button, or by engaging the autopilot while in a bank, the bank angle is maintained until the turn control knob is rotated out of detent, or a lateral mode is selected.

Interlocks are provided to prevent taking command with the turn control knob, or using the roll hold mode, while in the ILS approach mode.

Pressing the SYNC button with a lateral mode selected disconnects the servos while the button is pressed, allowing the pilot to maneuver the airplane. Upon release of the button, the system returns to the lateral guidance commands.

Heading (HDG)

Selection of the HDG mode brings the EADI command bars into view, and commands are provided to fly to the heading selected on the EHSI heading bug.

Navigation (NAV)

Cross-country VOR or terminal localizer are flown using the NAV mode. Tuning the navigation receiver to the desired VOR or localizer frequency and selecting NAV puts the fight control system into the NAV ARM submode. In this submode, the system annun-ciates HDG and NAV ARM, and responds to inputs from the heading knob on the EHSI.

Using the heading knob, the system can be set up to intercept the radial or course at any intercept angle up to 90°. When the flight path situation is such that the computer, using radio deviation, course intercept, and rate information, determines it is time to capture the radio course, the system switches from NAV ARM submode to NAV mode. The HDG and ARM annunciators go out, leaving only NAV, and steering commands are provided to capture and track the radio signal.

Automatic crosswind correction is provided, with crab angles up to 30°.

During VOR operation, command smoothing is provided to facilitate station passage. Thedead reckoning (DR) annunciator is illuminated to indicate the system is in the station passage submode.

Course change capability during station passage is provided, up to 30°. If the situation requires, the system will perform a 45° intercept and recapture the new course.

In NAV mode, the roll steering command is limited, providing a maximum commanded bank angle of 25° during capture and over the cone, and 10° when tracking the NAV course.

Bank commands from other navigation systems, such as RNAV or OMEGA, are also flown in this mode. Logic inputs from the navigation system, not VOR/LOC deviation inputs, are used to generate the bank command inputs. All angle capture capability and bank command limits are the same as for VOR.

Approach (APPR)

The APPR mode is used any time an approach to a runway is desired. It consists of four types of approaches:

- **VOR approach**—For a VOR approach, the navigation receiver is tuned to a VOR frequency and APPR mode is selected. The system configures itself in the NAV

  ARM submode for an all-angle capture similar to the NAV mode.

  The computer sets its gains and filters for a VOR approach, and provides steering commands to acquire the VOR radial through the capture and track submodes.

  When conducting a VOR approach, a station passage submode is provided to allow course changes at the VOR station for a dog-leg approach.

Glide slope operation is locked out.

The system annunciation consists of APPR ARM, APPR, and DR at the appropriate stages of the approach.

- **RNAV approach**—This approach is similar to a VOR approach except the signals are from an RNAV computer.

  APPR mode logic is available to effect a gain change on the deviation output for display and computation, providing improved accuracy and flyability.

  Glide slope operation is locked out.

  The system annunciation consists of APPR ARM and APPR at the appropriate stages of the approach.

- **ILS approach**—For an ILS approach, the navigation receiver is tuned to a localizer frequency and APPR mode is selected.

  Localizer operation consists of all-angle capture and track, similar to NAV mode, with appropriate changes in the capture and tracking computations to accommodate localizer beam geometry.

  When the system reaches the tracking configuration, the GS ARM annunciators illuminate to indicate the system is ready to capture the glide-slope beam.

  Glide-slope capture can be accomplished from any vertical mode, and from above or below the glide-slope beam.

  Capture occurs prior to the EADI glide-slope pointer indicating beam center. Commands are presented on the EADI to capture and track the glide-slope beam.

  At glide-slope capture, the annunciation switches from GS ARM to GS, and other vertical mode annunciators extinguish.

  Glide-slope gains are programmed throughout the approach as a function of radio altitude. If radio altitude information is not available, a step gain change is made at the middle marker.

  The system annunciation consists of APPR ARM, GS ARM, and GS at the appropriate stages of the approach.

- **Back-Course approach**—This mode is similar to the ILS approach mode, except the B/C button is selected and commands are provided to capture and track the localizer back course. In this mode, the B/C annunciator is illuminated.

  Glide-slope operation is locked out.

  Capture and tracking of the back-course localizer beam is the same as an ILS approach, except the system must be manually switched to HDG mode after placing the heading bug under the lubber line, or the pilot must ignore commands as the aircraft approaches and flies over the localizer transmitter.

  The system annunciation consists of B/C, APPR ARM, and APPR at the appropriate stages of the approach.

Go-Around

The go-around mode may be selected at any time, and should be selected at the decision to discontinue the approach.

The go-around mode (GA) is a wings-level, fixed 7° pitch-up mode, selected by pressing the go-around button (on the outboard side of each power lever).

GA mode may be selected from any lateral mode, and is canceled by selection of a lateral mode. The system is internally synchronizing to the pitch attitude of the aircraft while in GA, and maintains the GA pitch-angle command after a lateral mode is selected.

Actuation of the SYNC button cancels the GA mode and synchronizes the vertical command to the aircraft attitude.

Re-engagement of the autopilot cancels GA, synchronizes the autopilot commands to the aircraft pitch angle at the time of engagement, and continues to maintain wings level.

Vertical Modes

With a lateral mode (HDG, NAV, APPR) selected and no vertical mode selected, the EADI command bars display roll commands appropriate to the selected lateral mode, and pitch commands to maintain the pitch attitude at the time of mode selection.

The pitch command reference may be changed in two ways:

- Actuate the SYNC button on the control wheel.

  The computer synchronizes with the current aircraft pitch attitude. This allows the pilot to position the aircraft to any desired pitch angle (within the pitch limits), then momentarily press the SYNC button to resynchronize the computer to the new attitude.

  The EADI command bars provide commands to maintain the new pitch attitude reference.

  With a vertical mode selected (ALT, IAS CLIMB, DSC, or VS), operation of the SYNC button does not cancel the vertical mode and synchronizes the system with the present pitch attitude.

  The SYNC button is interlocked so it does not cancel the vertical mode during glide-slope tracking.

- Actuation of the vertical trim switch on the autopilot control panel.

  When the switch is momentarily actuated, it provides a 0.5° pitch change.

  When it is held down longer then 1/2 second, it provides a fixed slew rate command for the EADI command bars.

  With the autopilot engaged, the air-craft responds to the pitch commands. If a vertical mode is selected, the vertical trim switch provides a fixed incremental change for each actuation.

  Vertical trim switch operation is interlocked so it does not operate during glide-slope tracking.

Altitude Hold (ALT)

The ALT mode works in conjunction with the air data sensor to provide commands to maintain a selected altitude.

As the aircraft approaches the desired altitude, the vertical speed should be reduced to 500 fpm or less, and the ALT button pressed as the altitude is reached.

The computer reads the pressure altitude output of the air data sensor and stores the information in digital memory. Commands are generated for display on the EADI to maintain the altitude present at the time of mode selection.

Small changes to the reference altitude may be made by momentarily actuating the vertical trim switch. A ±25 foot altitude change is commanded for each actuation of the switch.

The altitude hold mode is canceled by:

- Selection of IAS or VS modes
- Pressing the ALT mode button
- Capturing the glide slope

Indicated Airspeed (IAS)

The IAS mode provides commands to maintain the indicated airspeed at the time of mode selection.

As with ALT hold, the computer reads and stores data from the air data sensor at the time of mode selection. It compares new data from the air data sensor with the stored data for calculation of the IAS hold command.

The vertical trim switch may be used to change the IAS reference at the rate of 1 knot for each momentary actuation of the switch.

The IAS mode is canceled by:

- Selection of ALT or VS modes
- Pressing the IAS mode button
- Capturing the glide-slope

Vertical Speed (VS)

The VS mode is also a hold mode that provides commands to maintain the vertical speed present at the time of mode selection.

As with the other air data modes, the computer reads and stores data from the air data sensor at the time of mode selection and provides commands from this reference.

Operation of the vertical trims switch provides a 200 fpm change for each momentary actuation of the switch.

The VS mode may be canceled by:

- Selection of ALT or IAS modes
- Pressing the VS mode button
- Capturing the glide slope

Altitude Preselect Mode (ALT SEL)

The altitude preselect mode (ALT SEL) works in conjunction with the altitude alerter.

The desired altitude is dialed into the altitude alerter, and the ALT SEL mode is selected on the flight control panel. This arms the altitude preselect mode to capture the alert altitude when it is reached.

Any vertical mode may be selected in conjunction with ALT SEL, to establish a climb or descent to the preselected altitude.

As the preselect altitude is approached, commands are provided to capture the altitude. When the altitude is captured, the system automatically engages the altitude hold mode and the system then functions the same as in altitude hold.

Descent Mode (DSC)

Prior to selecting DSC, the desired altitude should be selected on the altitude alerter.

When descent mode is selected, the autopilot begins a gradual descent stabilizing at an average rate of 2,000 fpm. The pilot may vary the rate of descent with the vertical trim switch.

The DSC and ALT ARM annunciators illuminate when DSC is selected.

When the altitude is captured, the system automatically engages the altitude hold mode (ALT) and the descent mode is canceled.

Climb Mode (CLIMB)

Prior to selecting CLIMB, the desired altitude should be selected on the altitude alerter.

When climb mode is selected, the autopilot begins a gradual climb stabilizing at an indicated airspeed defined by the climb profile. The profile is a constant 170 KIAS to 20,000 feet where it is decreased approximately 2 knots/1,000 feet to 155 KIAS at 32,000 feet.

The CLM and ALT ARM annunciators illuminate when CLIMB is selected.

When the altitude is captured, the system automatically engages the altitude hold mode (ALT) and the climb mode is canceled.

If CLIMB mode is selected at a speed below the climb profile, the autopilot decreases the rate of climb to 50 fpm to increase the airspeed to the climb profile speed.

The pilot is responsible for maintaining proper power to ensure the climb profile speed.

POINTS TO REMEMBER

- The vertical trim switch (located on the autopilot panel) is used for vertical control of the autopilot. Operating the switch without a vertical mode selected (basic pitch mode) results in a 0.5° pitch command followed by a continuous slew of 1°/second.

  The vertical trim switch is interlocked to prevent operation during glide-slope tracking.

- In VS mode, the synchronized vertical speed is changed in ±200 fpm increments with each actuation of the vertical trim switch. The maximum vertical speed change without reselecting the VS mode is 1,000 fpm.

- In IAS mode, the synchronized vertical speed is changed in ±1 knot-increments with each actuation of the vertical trim switch. The maximum airspeed change without reselecting the IAS mode is 10 knots.

- In ALT hold mode, the synchronized altitude is changed by ±25 feet with each actuation of the vertical trim switch. The maximum altitude change without reselecting the ALT mode is 1,000 feet. The engage vertical speed limit is ±500 fpm.

- Operating the SYNC button with a vertical mode selected (ALT, IAS, CLIMB, DSC, or VS) does not cancel the vertical mode. It synchronizes the system to present attitude, airspeed, altitude, or vertical speed. After glideslope capture in APPR mode, pressing the SYNC button has no effect on vertical commands (i.e., does not cancel the vertical mode). However, the SYNC button still selects the SYNC mode if the autopilot is engaged. When the SYNC button is released, commands are generated to return the aircraft to the center of the glide slope.



- Operation of the SYNC button cancels GA.
- Bank command limits are ±30° with the autopilot panel turn control, ±25° in lateral modes, ±15° in approach, and ±10° in VOR tracking. Pitch command limits are +20 and –10°. In the VS, IAS, ALT, or GS modes, an incremental limit of +6° is provided.
- Vertical hold mode information is displayed on the flight director by pushing the desired vertical mode button on the flight control panel. If a vertical mode is selected during manual flight director operation, it is automatically canceled when the autopilot is engaged for coupled operation.
- In flight director function, vertical commands are synchronized to the aircraft attitude by actuating the SYNC button after any lateral mode has been selected.
- During a coupled approach, the primary flight instruments must be monitored by the pilot. Presence of a VOR/LOC/GS flag must be considered as a warning of system or signal failure. Any flight director presentation with a flagged system may be unreliable.
- When flying a coupled VOR (RNAV system operation) with a valid DME (not on localizer frequency), the lateral deviation bar on the EHSI displays a course width of 10 nm in the NAV mode and 2 nm in APPR mode. If exceeded, the LIN DEV annunciator (if installed) illuminates. LIN DEV may be disabled by selecting HOLD on the DME.
- When either the DME is inoperative, or the DME is operated from a navigation radio other than the one in use by the autopilot, or on a LOC frequency, the HSI lateral deviation bar displays standard angular deviation. In this condition, the DEV annunciator is extinguished.
- When flying inbound toward a VOR station, do not set up a VOR capture problem within approximately 5 nm of the VOR (depending upon ground speed). The system does not have time to solve the problem and stabilize on course before passing over the station.
- Radio course may be changed over a VOR station when operating in NAV mode as long as the course change is not more than 30°. If the course change is more than 30°, the HDG mode should be selected to establish a new intercept, then NAV mode reselected to set up a new capture. For optimum NAV mode operation, select the intercept angle so the system maintains straight and level flight in the NAV mode a minimum of 30 seconds prior to capture.
- When making a station passage in NAV mode, the computer may automatically switch into heading memory submode to allow a smooth station passage. During this time, the dead reckoning (DR) annunciator illuminates.
- Closely monitor outbound tracking after station passage to ensure the system is maintaining the desired track, especially after course changes or during low-altitude operations.
- Best capture operation is achieved when intercept angles are less then 60°.
- Always check the autopilot annunciator display to verify that proper switching has taken place.
- Any autopilot system failure disengages the autopilot, or prevents it from engaging. This is for flight safety reasons to prevent the autopilot from controlling the aircraft when there has been a system failure.

GROUND PROXIMITY WARNING SYSTEM

GENERAL

The ground proximity warning system (GPWS) monitors radio altitude, vertical speed, airspeed, gear and flap position, and glide-slope deviation. It provides audio and visual warnings to the flight crew whenever the flight path condition could result in terrain impact.

The GPWS warns of dangerous terrain proximity in six flight modes. There are voice messages associated with each mode. The messages for Modes 1 through 4 continue until a new safe flight condition has been established. The Mode 5 message may be canceled by pressing the GS INHBT switch/indicator on the glareshield and the Mode 6 message is given only once.

GPWS MODES

Mode 1—Excessive Sink Rate

Mode 1 has two unique boundaries (Figure 16-59). The outer boundary advises the pilot that the rate of descent for a given altitude is excessive and the condition should be adjusted. When it is penetrated, the voice message “Sink Rate” is given.

If the rate of descent is not corrected and the secondary boundary is penetrated, a second voice message “Whoop-Whoop Pull Up” is given.

Mode 1 is independent of aircraft configuration. It also provides additional protection at lower altitude for windshear and visual transition.

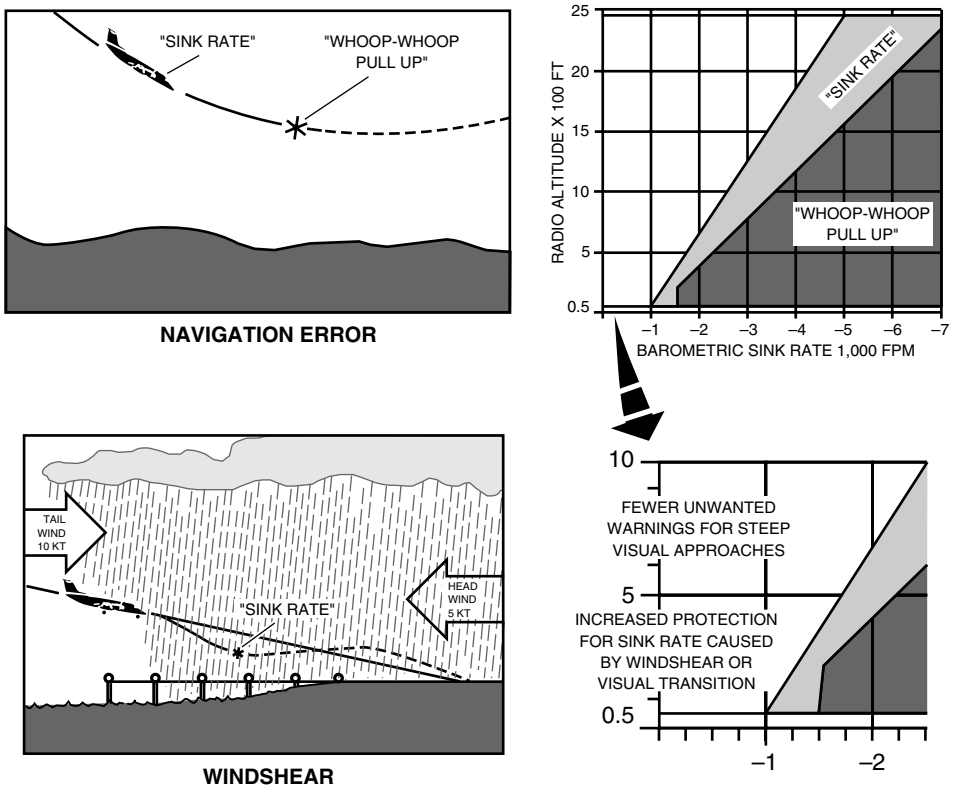


Figure 16-59. GPWS Mode 1

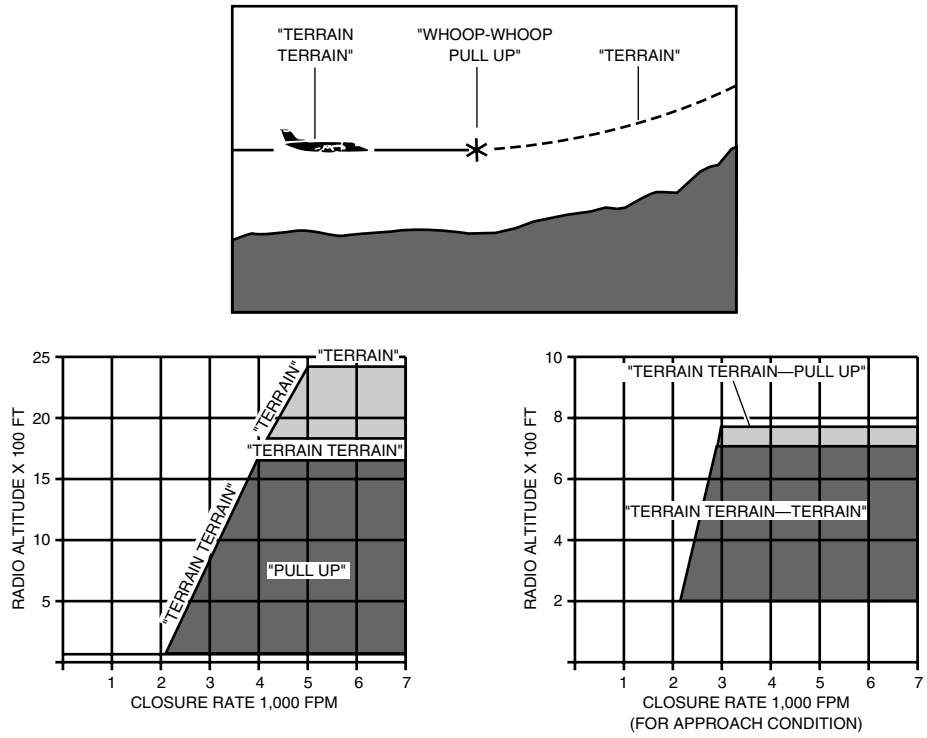


Figure 16-60. GPWS Mode 2

Mode 2—Excessive Terrain Closure Rate

Mode 2 (Figure 16-60) uses a complex function involving airspeed, radio altitude, vertical speed, barometric altitude, and aircraft configuration logic.

Upon penetrating the outer warning area the voice message "Terrain Terrain" is given. If the closure condition is not corrected, the "Whoop-Whoop Pull Up" voice message is given.

After the closure condition has been corrected and the "Pull Up" voice message stops, an altitude gain function is initiated. The gain function requires a barometric altitude gain of 300 feet, with the voice message "Terrain" given until the required altitude gain is achieved.

During an approach with gear or flaps extended, the altitude gain function is inhibited.

Below 700 feet with gear and flaps extended, the "Pull Up" voice message is replaced by "Terrain".

NOTE

This system does not protect against extremely rising terrain.

Mode 3—Altitude Loss After Takeoff

Mode 3 (Figure 16-61) provides a "Don't Sink, Don't Sink" advisory if the airplane loses approximately 10% of its barometric altitude after takeoff.

The reference altitude used to monitor the altitude loss, is the altitude from which the initial descent occurs.

Once triggered, the advisory may be canceled only by achieving a positive rate of climb. If the reference altitude is not reached again, a subsequent descent (exceeding the advisory threshold value) immediately retriggers the advisory.

After the reference altitude is reached again, the system resets to a new altitude reference for an eventual altitude loss.

Mode 3 operates only between 50 and 700 feet radio altitude during takeoff; or when either flaps or gear is retracted during a missed approach initiated below 200 feet.

Above 700 feet radio altitude, the GPWS switches to Mode 4.

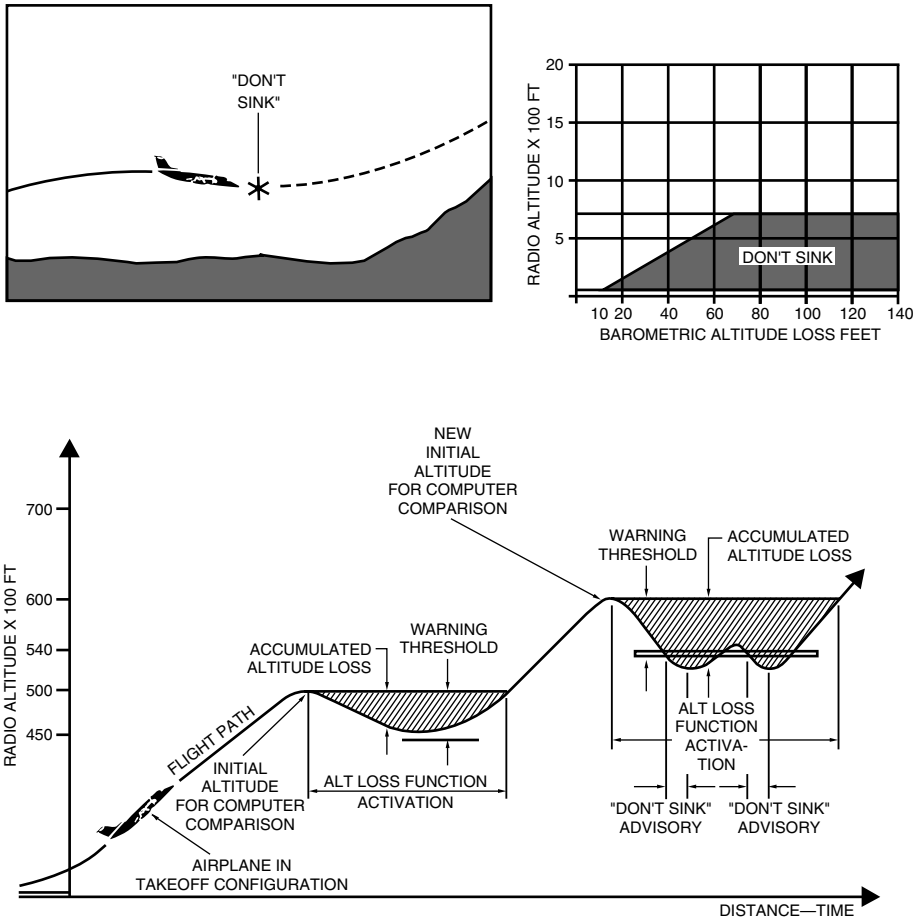


Figure 16-61. GPWS Mode 3

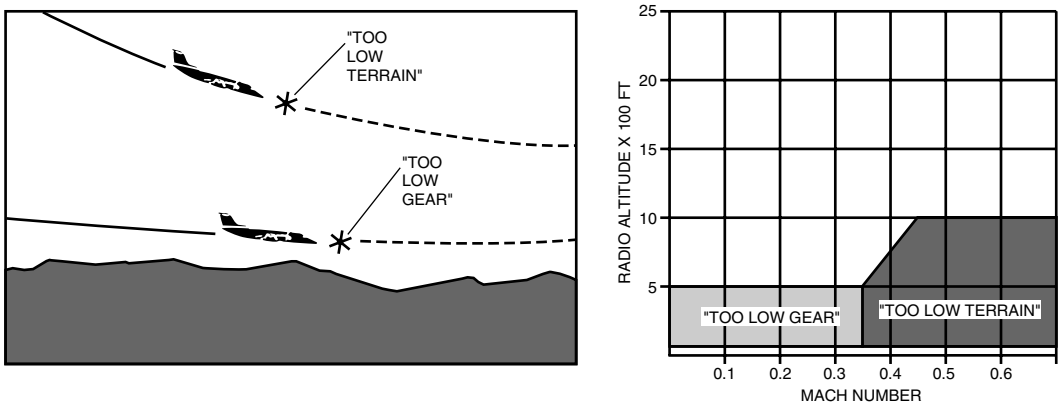


Figure 16-62. GPWS Mode 4A

Mode 4A—Terrain Clearance (Gear Up)

Mode 4A (Figure 16-62) is activated upon passing 700 feet AGL after takeoff with gear up.

Below 1,000 feet AGL in close proximity to flat or gently sloped terrain, the system generates a "Too Low, Terrain" voice message.

Descending below 500 feet AGL with the gear not down and locked, a "Too Low, Gear" voice message is given.

Mode 4B—Terrain Clearance (Flaps Up)

Mode 4B (Figure 16-63) provides protection when the gear is down, but the flaps are less than 17°. It is similar to Mode 4A in that flight below 1,000 feet generates the "Too Low, Terrain" voice message.

Descending below 200 feet AGL with the flaps less than 17°, a "Too Low, Flap" voice message is given.

In either case, the warning is repeated until the condition is corrected.

Both Mode 4A and 4B are inhibited below 50 feet for any gear or flap position.

The computer automatically switches from Mode 4B to Mode 3 when the 4B boundary is passed in full landing configuration; or when the aircraft is below 50 feet with gear down.

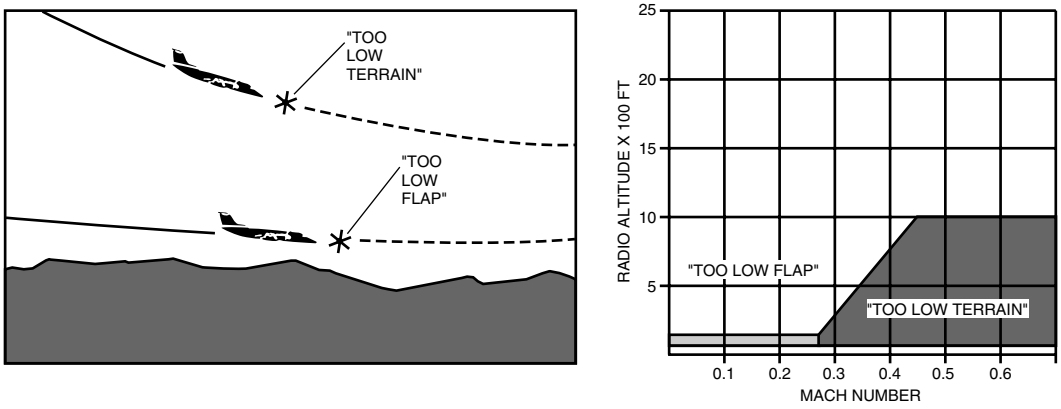


Figure 16-63. GPWS Mode 4B

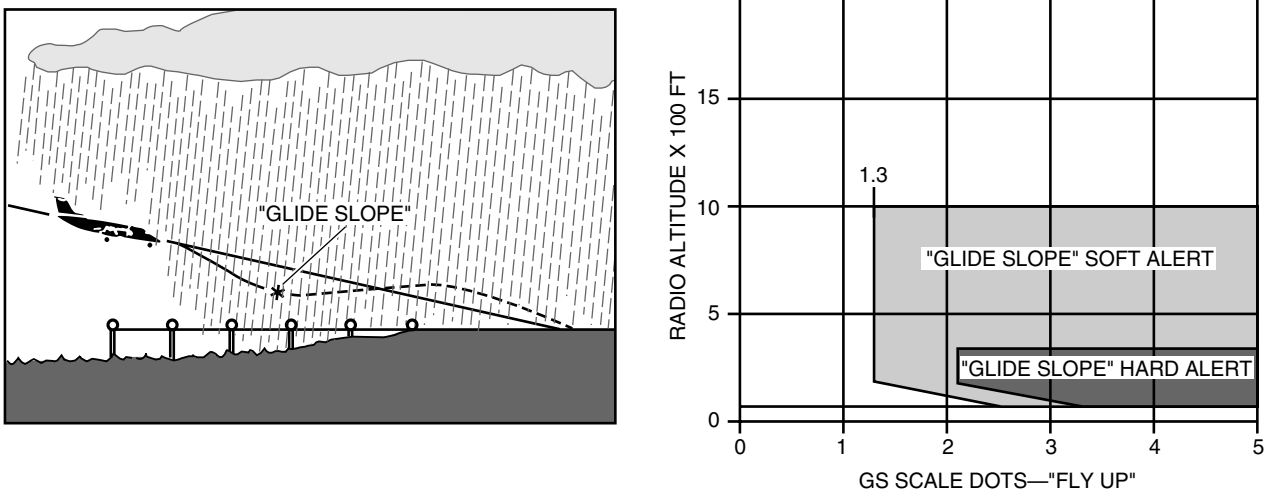


Figure 16-64. GPWS Mode 5

Mode 5—Glide Slope

Mode 5 advises of descent below the glide path during an ILS approach (Figure 16-64).

When the airplane penetrates the outer envelope, a "Glide Slope" voice warning is given softly. If the inner envelope is penetrated the voice warning is repeated, faster and at a higher volume. The BELOW G/S light (Figure 16-65) is continuously activated.

Below 1,000 feet AGL, the glide-slope advisory alert may be canceled by pressing the BELOW G/S light, or the alternate glide-slope cancel switch. Mode 5 automatically rearms above 1,000 feet AGL.

Mode 6—Descent Below Decision Height (DH)

Mode 6 operates in conjunction with the altitude, set with the DH SET knob on the display control panel. The message "Minimums Minimums" is heard once when the airplane passes through the altitude with the landing gear down.

No warning lamps are turned on. Mode 6 will not function again until either 1,000 feet or 50 feet is transitioned, or the landing gear is cycled. A 50 feet minimum cutoff prevents unwanted messages on the ground if the DH setting is changed.

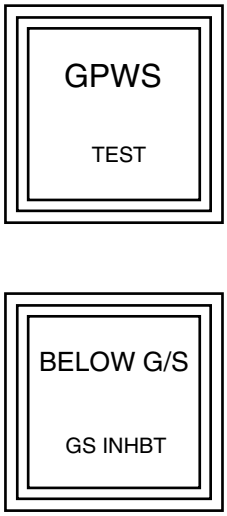


Figure 16-65. GPWS Controls and Indicators

GPWS AURAL WARNING MESSAGE PRIORITIES

Due to the possibility of activating more than one warning at a time, a warning priority is incorporated. A higher priority message immediately overrides any lower priority message in progress. A lower priority message is announced only after a higher priority message stops.

The message priority, listed in descending order, is as follows:

Priority	Message	Mode
1.	"Whoop Whoop Pull Up"	1 and 2
2.	"Terrain"	2
3.	"Too Low, Terrain"	4
4.	"Too Low, Gear"	4
5.	"Too Low, Flaps"	4
6.	"Minimums"	6
7.	"Sink Rate"	1
8.	"Don't Sink"	3
9.	"Glide Slope"	5

GPWS CONTROLS AND INDICATORS

The two primary controls and indicators for the GPWS are the GPWS annunciator/TEST button and the BELOW G/S annunciator/GS INHBT button (Figure 16-65). There are two sets of buttons, normally located on the instrument panel in front of both the pilot and copilot. Other controls and indicators include a GPWS FLAP OVRD button and a GPWS INOP annunciator.

GPWS Annunciator/TEST Button

The GPWS annunciator blinks anytime the system creates an audio output associated with Modes 1 through 4.

When pressed, the GPWS annunciator functions as a GPWS TEST button and initiates the GPWS self-test. Self-test can be initiated on the ground, or above 1,000 feet AGL, by pressing and holding either button.

BELOW G/S Annunciator/ GS INHBT Button

As previously discussed, the BELOW G/S annunciator illuminates in conjunction with Mode 5 audio outputs, and also functions as the GS INHBT button.

GPWS FLAP OVRD Button

The GPWS FLAP OVRD button (Figure 16-66) is used in the event of a no flap landing to prevent false warnings from the GPWS.

GPWS INOP Annunciator

The GPWS INOP annunciator (Figure 16-66) illuminates anytime a failure is detected in the system.

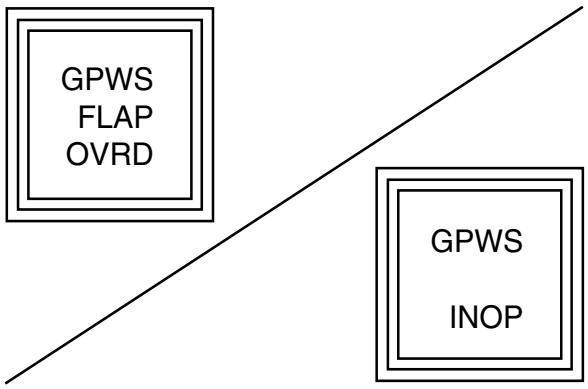


Figure 16-66. GPWS FLAP OVRD Button/ INOP Annunciator

TRAFFIC ALERT AND COLLISION AVOIDANCE SYSTEM (TCAS I)—OPTIONAL

GENERAL

The EMB-120 may be equipped with a Traffic Alert and Collision Avoidance System (TCAS I) that provides the crew with an indication of possible traffic conflict when in flight. The system is based upon transponder signals and provides visual and aural warning (Figure 16-67).

SYSTEM DESCRIPTION

The TCAS consists of a transmitter-receiver computer (TRC), a directional antenna, an L-band omni-directional antenna, and a control display unit (CDU) (Figure 16-68).

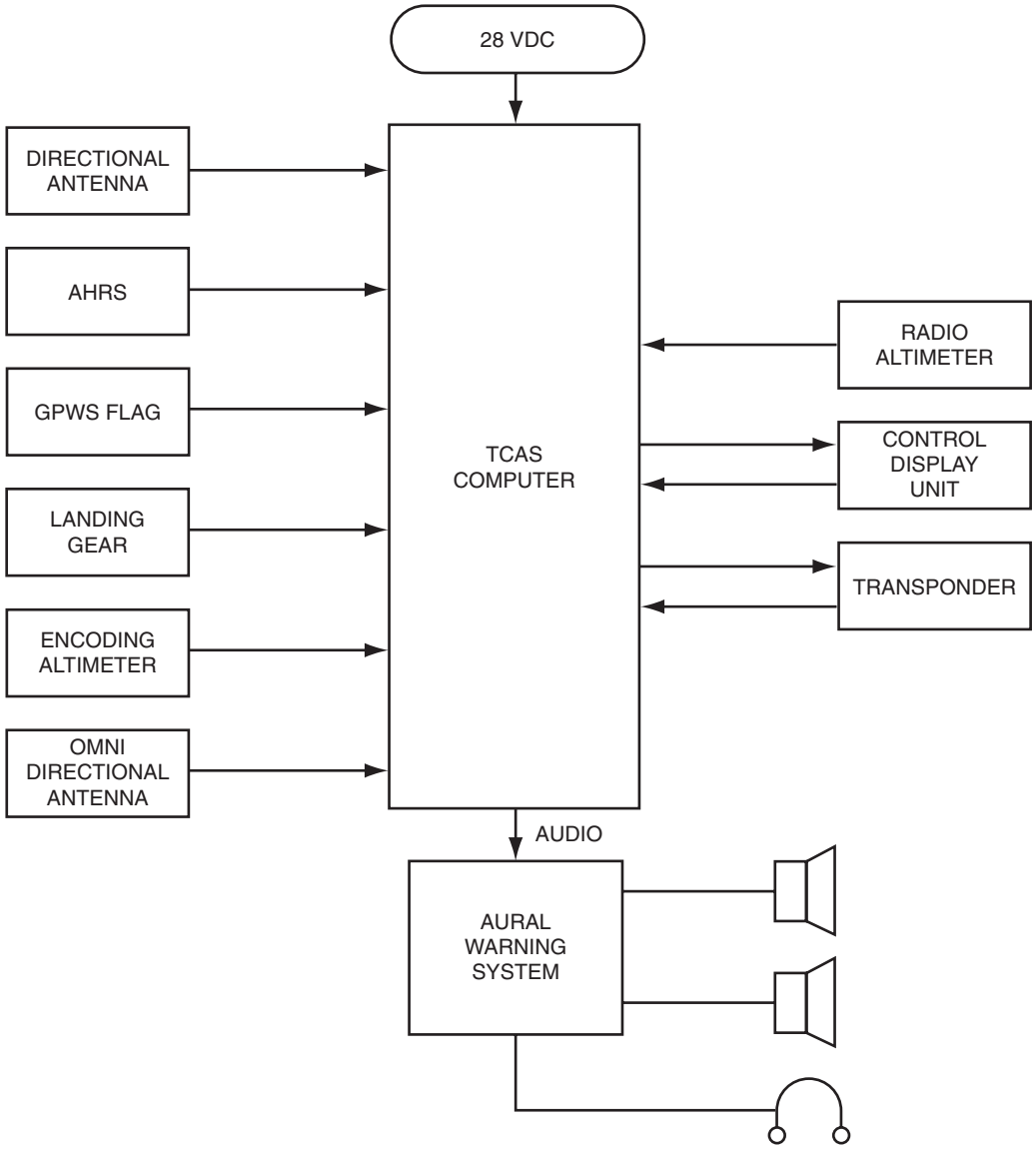


Figure 16-67. TCAS Schematic

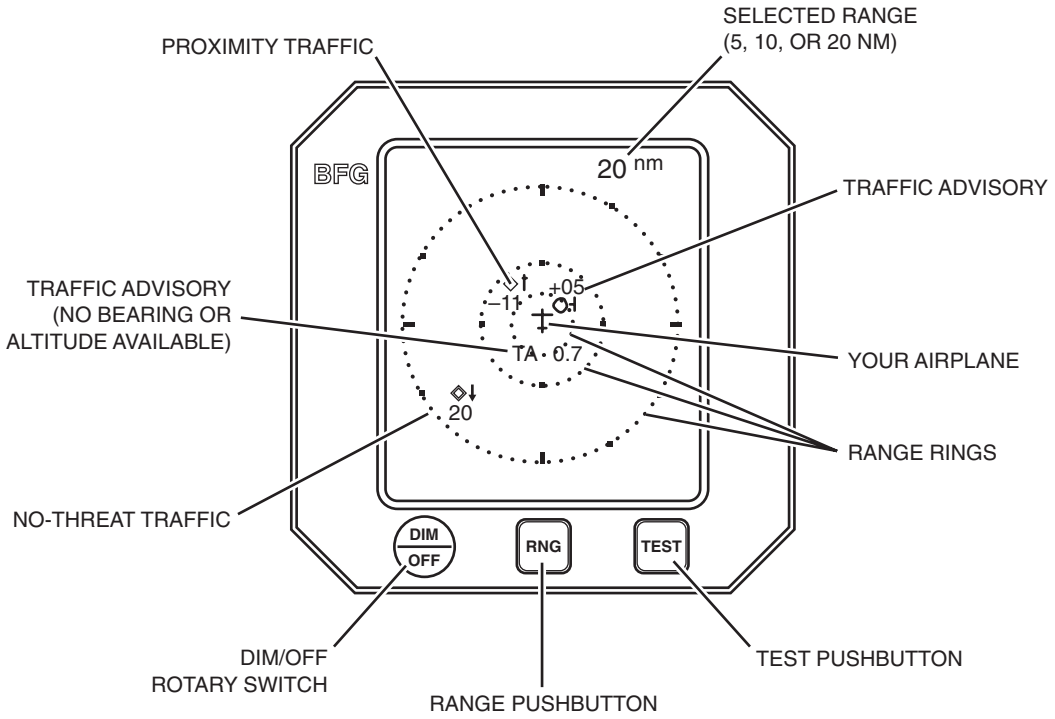


Figure 16-68. TCAS Control Display Unit (CDU)

The control and test functions of the TCAS are performed through the CDU that is used to present traffic information to the pilot.

TCAS computer receives data from the installed transponder, radio altimeter, encoding altimeter, antenna signals, and landing gear air/ground switches. AHRS provides magnetic heading information to TCAS.

By interrogating other airplane transponders, TCAS determines relative altitude, range, and bearing of any airplane equipped with a mode "C" or "S" transponder.

CAUTION

TCAS provides no protection against airplanes that do not have an operating transponder or airplanes equipped with a mode "A" transponder.

The system is capable of tracking up to 35 intruders. However, to reduce display clutter, only those eight targets determined to present the greatest threat will be displayed on the CDU.

Airplane intruders are displayed as symbols identified by color and shape followed by an up-or-down trend arrow, and a number. The number is a (+) or minus (–) followed by two digits that represent the vertical distance in hundreds of feet between the airplane and the intruder plane. It is positive when the intruder plane is higher and negative when lower. If the intruder is at the same altitude as the airplane, the digits "00" will be displayed. If the intruder altitude is not reported, the altitude field will be blank.

The up-or-down trend arrow indicates that the intruder is either climbing or descending at a rate greater than 500 feet per minute. If the intruder is nonaltitude reporting, no trend arrow will appear and the system will assume the intruder is at the same altitude as the airplane.

Selectable range scale is available through the RNG pushbutton on the CDU. The selected range (5, 10, or 20 nm) will be indicated in the upper right corner of the display.

TCAS SELF-TEST

The TCAS self-test is activated through the TEST pushbutton on the CDU. Before initiating the TCAS self-test, set pilot's barometric altimeter to 29.92 mmHg and observe the indicated pressure altitude. Then turn TCAS on and momentarily press the TEST pushbutton on the CDU. A normal 5-nm display with all intruder symbols will be presented. In addition, the following messages will be displayed: "TEST" on the upper left, "5 nm" on the upper right, and the airplane altitude at the lower middle of the display. The altitude displayed should be within ±250 feet of the altitude indicated on the pilot's altimeter. If the displayed altitude is out of range, it indicates a failure of the equipment.

At the completion of the self-test, the display will return to normal, and a voice message either "TCAS TEST PASSED" or "TCAS TEST FAILED" will be heard.

NOTE

TCAS self-test is inhibited in flight.

TCAS CONTROL DISPLAY UNIT

DIM/OFF ROTARY Switch—Controls display brightness and power to both the transmitter-receiver computer (TRC) and the control display unit (CDU). Power is applied by rotating the knob clockwise past the detent. Continued clockwise rotation increases display brightness.

RANGE PUSHBUTTON—Press the RNG pushbutton to select among the 5-, 10-, or 20-nautical mile ranges. The selected range will be indicated in the upper right hand corner of the display. Whenever the airplane is on the ground, range selection is inhibited and the system defaults to 5 nm.

TEST PUSHBUTTON—Initiates a system self-test (push to start), while the airplane is on the ground. Test function is inhibited, whenever the aiplane is airborne.

RANGE RINGS—Three or two range rings of blue dots around the blue miniature airplane represent distances surrounding your airplane. Depending on which range is selected, the range rings will have different values as shown in Table 16-1.

Table 16-1. RANGE RINGS

RANGE RINGS		
SELECTED RANGE	**RING**	**DISTANCE FROM YOUR AIRPLANE**
5 nm	INNER OUTER	2 nm 5 nm
10 nm	INNER MIDDLE OUTER	2 nm 5 nm 10 nm
20 nm	INNER MIDDLE OUTER	5 nm 10 nm 20 nm

DISPLAY SYMBOLS

The symbols represent several degrees of threat. Intruders are classified as nonthreat, proximity, or traffic advisory (TA) as described in Table 16-2.

FLIGHT RECORDERS

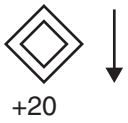
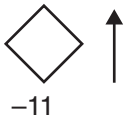
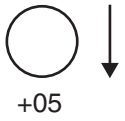
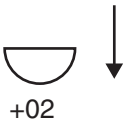
COCKPIT VOICE RECORDER (OPTIONAL)

General

The aircraft may be equipped with a voice recorder system to record all flight crew conversations and communications. The system preserves a continuing record of the latest 30 minutes of flight.

The voice recording system has four separate tracks for recording signals transmitted and received from the pilot and copilot, the cockpit area, and the passenger address system.

Table 16-2. DISPLAY SYMBOLS

1. An open white diamond symbol is used for nonthreat traffic, indicating an intruder is more than ±1,200 feet vertically or more than a distance of 6 nm;	
2. A solid white diamond symbol is used for proximity traffic, indicating an intruder is within ±1,200 feet vertically and within 6 nm, but is not yet considered a threat.	
3. A solid yellow circle is a traffic advisory (TA). It indicates that an intruder is now considered a threat and there is a risk of potential collision within 15 to 30 seconds. At the moment the threat airplane is first detected, TCAS will display it as a yellow circle and the voice message "TRAFFIC, TRAFFIC" will be heard in the cockpit. If the threat airplane bearing is unknown, instead of the yellow circle, TCAS will display it as a "TA" followed by a number that represents the horizontal distance in nm from the airplane. NOTE: • The voice message is inhibited below 400 feet AGL. • The pilot should not initiate evasive maneuvers using information from the traffic display or traffic voice message, without visually noticing the intruder. Safe visual separation should be maintained in accordance with regulation requirements and good operating practice.	
4. A solid yellow half-circle symbol is applicable only to 5 nm range selection and indicates a threat airplane located outside the surveillance range (between 6 and 5 nm). It will be displayed on the edge of the outer range ring, until the threat airplane comes within range and can be completely displayed.	

Components

The cockpit voice recording system consists of the following major components:

- Voice recording unit
- Control unit
- Impact switch

Voice Recording Unit

The voice recording unit contains an endless magnetic tape that records all inputs transmitted or received by the flight crew members for a maximum of 30 minutes. After 30 minutes new data is recorded over the old.

The voice recorder receives audio inputs from four sources:

- Pilot
- Copilot
- Cockpit area microphone
- Passenger warning/communications system

Each of the inputs corresponds to one trace on the four-track magnetic recording tape.

The recording unit is located in the tail cone of the aircraft.

The voice recording unit also contains an underwater acoustic beacon, powered by a self-contained battery. The battery is activated when immersed in fresh or salt water, and has a 30-day operating life. The beacon transmits on a frequency of 37.5 kHz and can be heard in a radius of approximately two miles.

Control Unit

The voice recorder control unit (Figure 16-69) is located on the forward part of the center console. It contains the following controls and indicators:

- Microphone
- Meter
- HEADSET jack
- ERASE button
- TEST button

Microphone—The cockpit area microphone is used to pick up all conversation in the cockpit.

Meter—The meter gives a GO or NO-GO indication when the TEST button is pressed.

HEADSET jack—A headset jack is provided to monitor the 600 Hz test tone when the TEST button is pressed.

ERASE button—The ERASE button permits the entire tape to be erased if the aircraft is on the ground and the parking brake is applied. To erase the tape, the button must be pressed for at least two seconds.

TEST button—The TEST button is used to test the voice recorder system.

When the button is pressed, a 600-Hz tone is heard in a headset plugged into the HEADSET jack. At the same time a GO or NO-GO indication is presented on the meter.

In the event of a system failure, the meter needle returns to zero for a definite NO-GO indication.

The testing of the system does not interrupt voice recording.

Impact Switch

An impact switch is located in the tail cone of the aircraft. It activates the acoustic beacon and cuts off electrical power to the recorder.

FLIGHT DATA RECORDER SYSTEM (OPTIONAL)

General

The flight data recorder system receives and records information for use in accident investigation or maintenance monitoring.

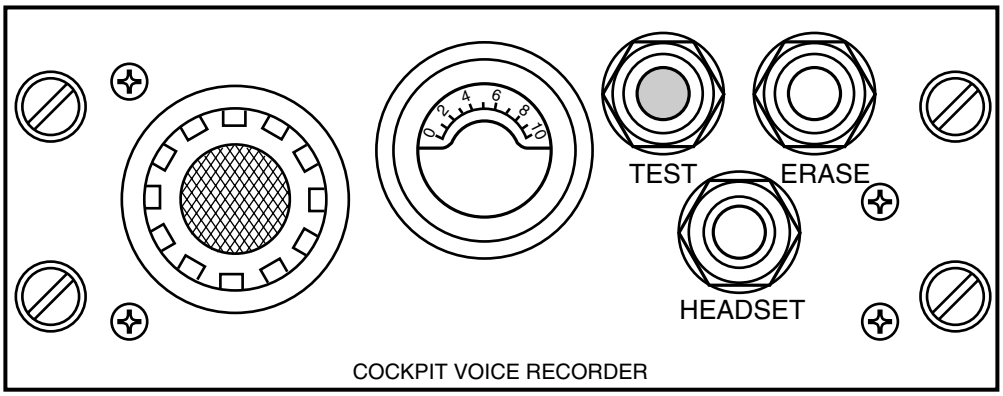


Figure 16-69. Cockpit Voice Recorder

Components

The flight data recorder system consists of the following major components:

- Flight data acquisition unit
- Flight data entry panel
- Digital flight data recorder

Flight Data Acquisition Unit

The flight data acquisition unit receives information from the aircraft systems and transducers. It transforms the data and sends the mandatory parameters to the flight data recorder where it is stored on magnetic tape and records the maintenance parameters internally.

Flight Data Entry Panel

Located on the control console forward panel, the flight data entry panel (Figure 16-70) is used by the flight crew to enter data into the flight data recorder.

The panel contains the following controls and indicators:

- Flight number select switches
- EVENT pushbutton
- FDAU fault indicator
- DFDR fault indicator
- MEMory condition annunciator

Flight number select switches—The flight number select switches are four thumb wheel switches used to enter the flight number into the flight data system.

EVENT pushbutton—The EVENT pushbutton is used to register the maintenance parameters at any phase of the flight.

FDAU fault indicator—The amber FDAU fault indicator monitors the operation of the flight data acquisition unit.

DFDR fault indicator—The amber DFDR fault indicator monitors the operation of the digital flight data recorder.

MEMory condition annunciator—The white memory condition annunciator (MEM) illuminates when the computer storage memory is 80% full.

Digital Flight Data Recorder

The digital flight data recorder, located in the tail cone of the aircraft, receives and stores data from the flight data acquisition unit. It records on six tracks and has a total storage time of 25 hours.

The recorder is impact and fire protected. It has an underwater acoustic beacon externally attached to its case to track the box in the event the aircraft goes down in the water. The operating frequency of the beacon is 37 kHz.

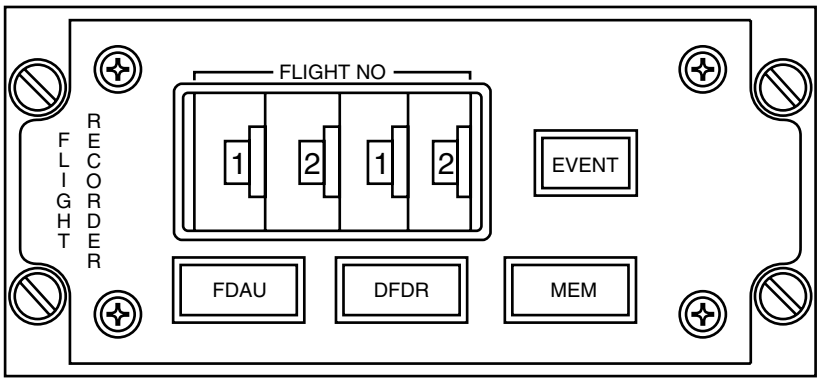


Figure 16-70. Flight Data Entry Panel

Mandatory Parameters

Certain mandatory parameters (data) are collected and stored in the digital flight data recorder from the moment the flight data acquisition unit is energized.

Those parameters are:

- Time
- Altitude
- Airspeed
- Vertical acceleration
- Lateral acceleration
- Longitudinal acceleration
- Heading
- Pitch attitude
- Roll attitude
- Pitch trim control
- Control column
- Control wheel
- Rudder pedal
- N_P
- Torque
- Beta mode
- Flap angle
- PTT action

Maintenance Parameters

The flight data acquisition unit stores the maintenance parameters (data) if the condition lever is out of SHUTOFF and the rotating beacon switch is ON, or when the aircraft is airborne.

The maintenance parameters are:

- T_6
- N_H
- Static air temperature
- Bleed condition
- Fuel flow
- Propeller imbalance
- GND/AIR condition
- Condition lever position

ELECTRONICS BAY VENTILATION SYSTEM

The electronics bay is located in the aircraft nose compartment. To keep the electronic equipment operating properly, a ventilation system is installed to maintain the bay temperature within safe limits (Figure 16-71).

The components are thermally insulated and system operation is automatic.

COMPONENTS

The electronics bay ventilation system is made up of the following components:

- NACA air intakes
- Shutoff valves
- Internal air recirculation fans
- Exhaust fans
- Check valves
- Control thermostats
- Indicator thermostats

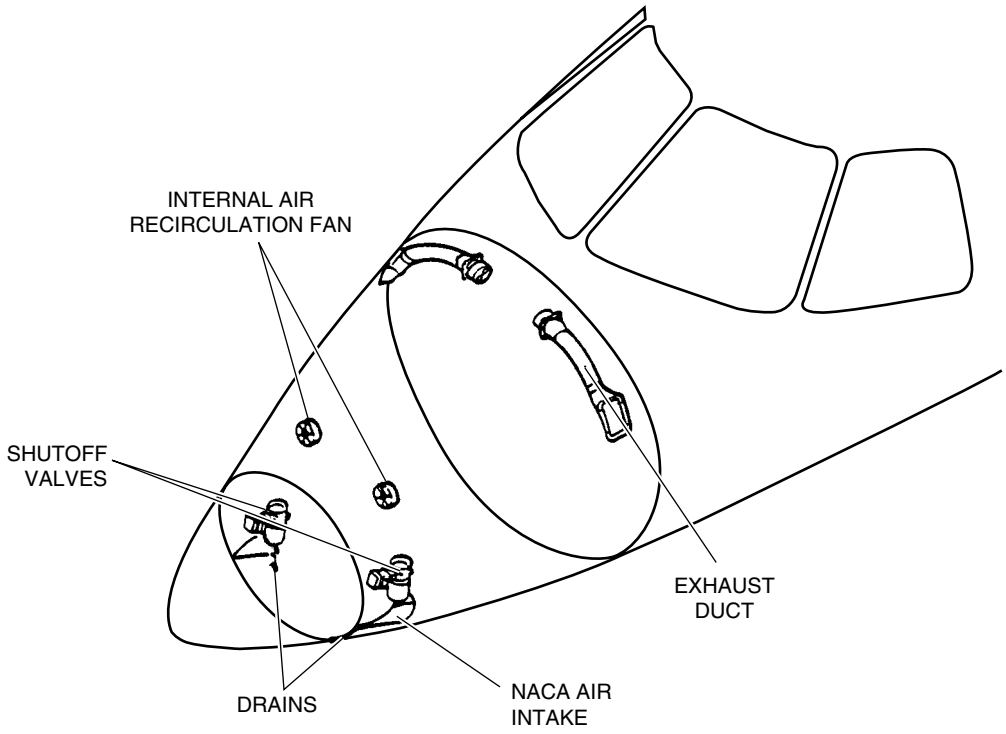


Figure 16-71. Electronics Bay Ventilation System

NACA Air Intakes

The air enters the electronics bay through two nose mounted NACA air intakes. Excessive moisture is drained overboard through drains installed in each intake.

Shutoff Valves

A 3-inch, motor-controlled butterfly valve is installed downstream of each NACA air intake. The valve has a droplet baffle/collecting plate to prevent moisture from reaching the electronic equipment.

Internal Air Recirculation Fans

Two recirculating fans with protective grills are installed in the ceiling of the forward portion of the equipment bay.

Exhaust Fans

Two exhaust fans are installed in the upper rear portion of the equipment bay. They form the intakes for the exhaust ducts.

Check Valves

A check valve is installed on the outlet side of each exhaust fan. The valves prevent the reverse flow of air into the electronic equipment bay.

Control Thermostats

Four control thermostats, two on each side of the equipment bay in the air flow from the recirculation fans, control the exhaust fans and the shutoff valves.

Indicator Thermostats

These two thermostats are installed on the left and right sides of the upper rear portion of the equipment bay. This is the hottest region. Should the temperature exceed 160°F/70°C, either thermostat may trigger the amber ELEC BAY OVERTEMP light on the MAP.

OPERATION

The components of the electronic bay ventilation system are subdivided into two groups, right and left, powered by DC Bus 1 and DC Bus 3 respectively.

Each group is composed of one shutoff valve, one internal air recirculation fan, one exhaust fan, two control thermostats, and one indicator thermostat.

The internal air recirculation fans operate continuously.

When the temperature in the equipment bay exceeds 80°F/25°C, the control thermostat opens the shutoff valve and energizes the exhaust fan. When the temperature in the equipment bay drops below 70°F/20°C, the control thermostat closes the shutoff valve and deenergizes the exhaust fan.

When temperature in the equipment bay exceeds 160°F/70°C, either of the indicator thermostats may trigger the following indications and warnings:

- A single-chime aural alert sounds.
- The amber ELEC BAY OVERTEMP light on the MAP illuminates.
- The master CAUTION lights flash.

QUESTIONS

1. The EMB 120 has _____ independent pitot-static systems.

A. One
B. Two
C. Three
D. Four

2. The mode C input for the ATC transponder comes from:

A. No. 1 air data sensor
B. Pilot's encoding altimeter
C. Copilot's encoding altimeter
D. A as primary, B as secondary

3. The altitude alerter system visual and aural warnings trigger when:

A. The aircraft is 400 feet from the selected altitude.
B. The aircraft is 200 feet from the selected altitude.
C. The aircraft altitude deviates approximately 200 feet from the selected altitude.
D. Both A and C

4. In the event of a complete electrical failure, the standby horizon provides valid attitude information for:

A. 5 minutes
B. 9 minutes
C. 30 minutes
D. Standby horizon tumbles if power is lost

5. The EFIS uses four identical cathode ray tubes as the primary attitude and navigation displays for the pilot and copilot.

A. True
B. False

6. The EADI has the following two display formats:

A. Normal and composite
B. Normal and ARC
C. Composite and MAP
D. Composite and ARC

7. The EHSI has the following four display formats:

A. HSI, composite, ARC and MFD
B. Normal, ARC, MAP and DML
C. HSI, ARC, MAP and COMP
D. Composite, ARC, MAP and SAT

8. The MFD is the primary weather radar display.

A. True
B. False

9. The EMB-120 has two identical, and independent, AHRS.

A. True
B. False

10. The EMB-120 navigation system incorporates the EFIS and AHRS with the RMIs, various navigation radios, and weather radar.

A. True
B. False

11. The VHF NAV radio frequency select knobs:

A. Normally change the frequency in the upper (active) window
B. Normally change the frequency in the lower (inactive) window
C. May be used to change the frequency in the active window
D. Both B and C

12. The pilot's RMI No. 1 is connected to the copilot's AHRS No. 2.

A. True
B. False

13. The maximum range of the DME is:

A. 150 NM
B. 200 NM
C. 250 NM
D. 300 NM

14. Weather radar information:

A. Is not available on the EHSI
B. May be displayed on the EHSI in place of the normal displays
C. Is displayed on the EHSI with either ARC or MAP selected
D. Is superimposed on an expanded compass sector in the MAP format on the EHSI

15. The two VHF COM radios:

A. Are 720 channel radios, from 118.000 to 135.975 MHz, spaced 0.025 MHz apart
B. Display only five of the six digits of the selected frequency
C. Can store six preset frequencies
D. All of the above

16. The autoflight system consists of:

A. Two flight director systems and two autopilots
B. Two flight director systems and one autopilot
C. One flight director system and two autopilots
D. One flight director system and one autopilot

17. Each flight guidance computer is common to both the flight director and the autopilot.

A. True
B. False

18. The electric pitch trim switch on the control wheel will disengage the autopilot when actuated.

A. True
B. False

19. The vertical trim switch on the autopilot control panel will disengage the autopilot when actuated.

A. True
B. False

20. The GPWS:

A. Monitors only radio altitude, vertical speed, and airspeed
B. Has voice messages associated with all six modes; only the mode 5 ("Glide Slope") may be canceled
C. Warns of all potential terrain conflicts
D. Mode 6 message "Minimums" given as the aircraft passes through the DH with the gear up or down

21. The electronics bay ventilation system:

A. Is controlled by a switch in the cockpit
B. Has two internal recirculation fans that come on when the temperature exceeds 80°F
C. Uses bypass air from the air-conditioning system
D. Provides a cockpit warning if an overheat condition exists



CHAPTER 17
OXYGEN SYSTEMS

CONTENTS

Page

INTRODUCTION 17-1

OXYGEN SYSTEM 17-1

General 17-1

Flight Crew Oxygen System 17-3

Passenger Oxygen System 17-8

Portable Oxygen System 17-12

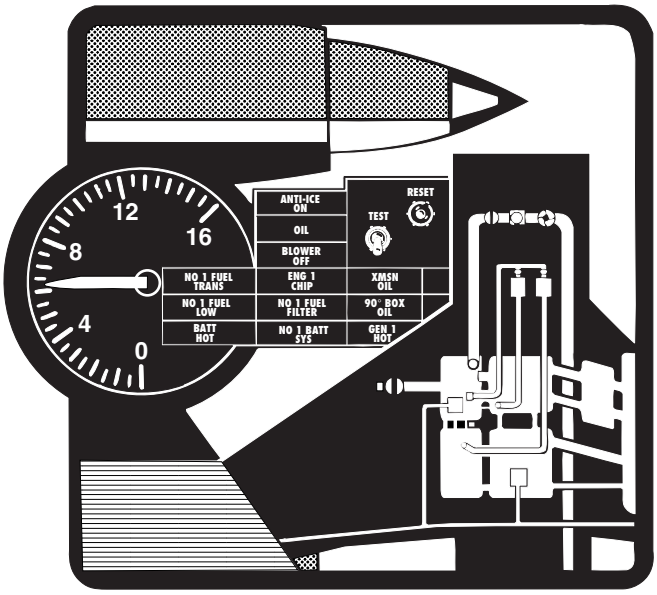
ILLUSTRATIONS

Figure	Title	Page
17-1	General Oxygen System Arrangement	17-2
17-2	Oxygen Cylinder Service Panel and Blowout Disc	17-3
17-3	Oxygen Pressure Gage	17-4
17-4	Oxygen Pressure Correction Chart	17-5
17-5	Crew Oxygen Mask	17-6
17-6	Mask Stowage Box	17-6
17-7	Full Face Mask	17-7
17-8	Smoke Goggles	17-7
17-9	Passenger Oxygen Mask Location and Use	17-9
17-10	Oxygen System	17-10
17-11	PAX OXYGEN Panel	17-11
17-12	Copilot's Oxygen Panel	17-11
17-13	Portable Oxygen Bottle	17-12
17-14	Oxygen System Controls and Indicators	17-13

TABLE

Table	Title	Page
17-1	Minimum Oxygen Pressure for Dispatch	17-4

CHAPTER 17
OXYGEN SYSTEMS



INTRODUCTION

This chapter discusses miscellaneous systems not covered in other chapters of this *EMB-120 Pilot Training Manual*.

OXYGEN SYSTEM

GENERAL

The EMB-120 oxygen system provides supplemental oxygen for both the crew and passengers.

The aircraft may be equipped with one of two different passenger oxygen systems:

Version I, installed on most aircraft, and discussed in this chapter, uses the same storage bottle for both the crew and passenger subsystems. It has the capacity to provide oxygen for 2 hours to the crew and 30 minutes to all passengers.

Version II, installed on some early aircraft, and not discussed in this manual, has a chemical oxygen generation system for the passengers. Its capacity is 2 hours for the crew and 12 minutes for passengers. For information on this system, consult the appropriate manufacturers' publications.

As a matter of interest, it takes more than 5 minutes to execute an emergency descent from 32,000 feet to 14,000 feet.

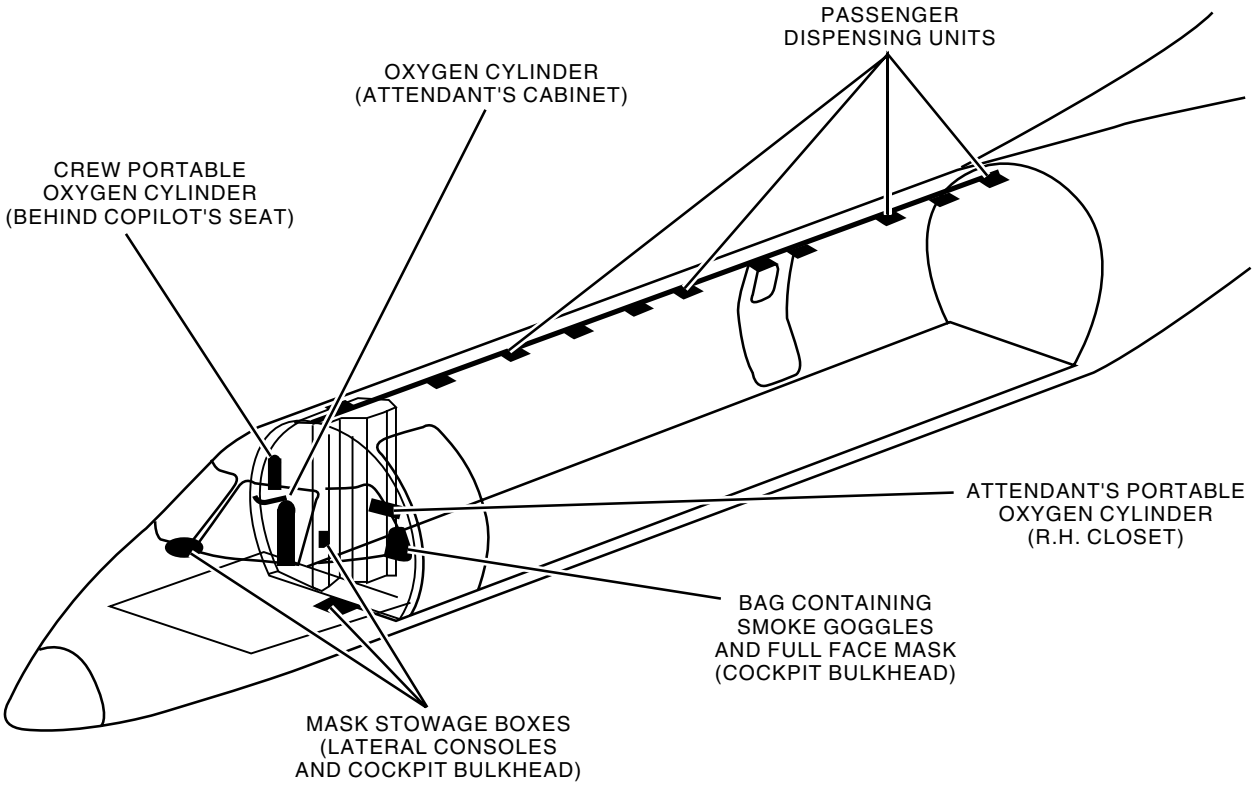


Figure 17-1. General Oxygen System Arrangement

The oxygen system (Figure 17-1) is a conventional high-pressure gaseous system. The system uses a single high-pressure bottle for storage, and distributes the oxygen at low pressure to the crew and passenger masks.

The oxygen system is divided into the following subsystems:

- Flight crew oxygen
- Passenger oxygen
- Portable oxygen for supplemental, smoke combat, and first aid

FLIGHT CREW OXYGEN SYSTEM

General

The crew oxygen system supplies oxygen to the pilot, copilot, and observer.

Oxygen is supplied, through a low-pressure line from the oxygen cylinder assembly, to the crew masks. The masks are stored in stowage boxes located on the pilot's and copilot's lateral consoles. An optional mask is stored near the observer's seat.

Components

The crew oxygen system's major components are:

- Oxygen cylinder assembly
- Crew oxygen equipment
  - Oxygen masks
  - Smoke goggles

Oxygen Cylinder Assembly

The crew and passenger oxygen systems use a common 115-cubic-foot oxygen cylinder, installed in the attendant cabinet (Figure 17-1). The cylinder has a capacity of 2,950 liters at 1,850 psi, with a 200-psi residual pressure.

Cylinder recharging is accomplished through an external service panel on the right front side of the fuselage (Figure 17-2).

The high-pressure oxygen in the cylinder is reduced to a usable level by an integrated shutoff/regulator valve in the cylinder top. The valve controls the oxygen outlet pressure (70 ±10 psi) at a minimum flow rate of 5 LPM to the flight crew and passenger distribution lines.

A relief valve opens should the outlet pressure exceed 90 psi.

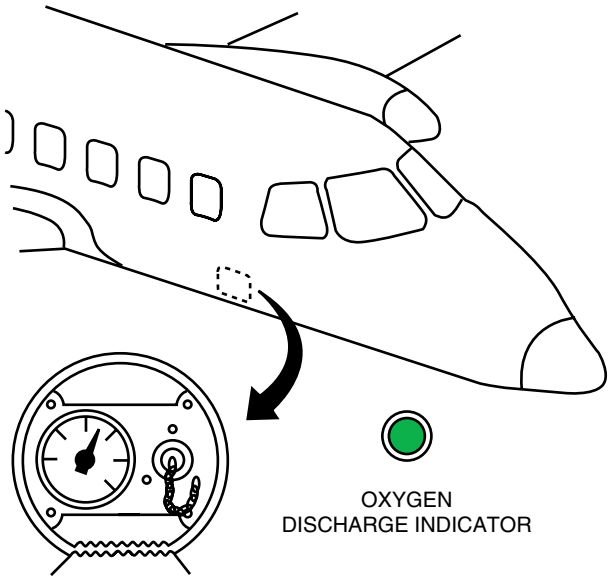


Figure 17-2. Oxygen Cylinder Service Panel and Blowout Disc

Overboard Discharge Indication

The oxygen cylinder also is protected against overpressure by a safety disc installed in the shutoff/regulator valve. The disc is designed to blow out if the cylinder pressure increases to approximately 2,450 psi at 71°C (2,770 psi at 21°C). Once the safety disc blows out, the oxygen cylinder completely discharges.

The overpressure discharge is externally indicated by a green blowout disc mounted in the fuselage skin near the oxygen service panel (Figure 17-2).

Pressure Gages

Oxygen quantity is monitored on two pressure gages, one installed on the external oxygen service panel (Figure 17-2), and the other on the copilot's panel.

The gage located on the external recharging panel is used by maintenance personnel during recharging.

The cockpit gage, located on the right side of the copilot's instrument panel (Figure 17-3), enables the flight crew to monitor the oxygen cylinder pressure.

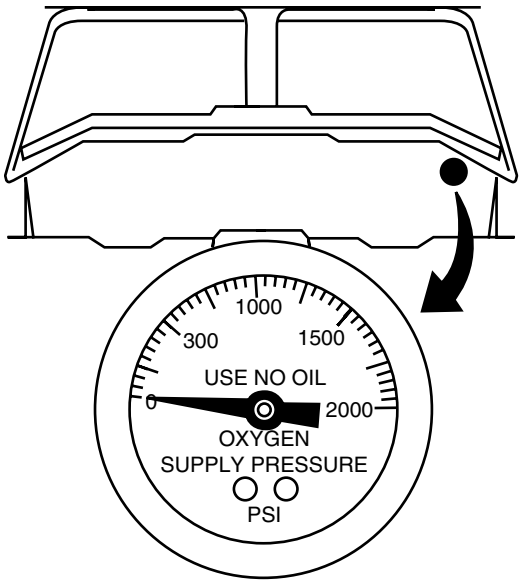


Figure 17-3. Oxygen Pressure Gage

Low-Pressure Switch

When the oxygen cylinder pressure drops below 375 ±25 psi, a low-pressure switch illuminates the CREW OXY light on the MAP. This indicates that approximately 35 minutes of oxygen remains for the flight crew (pilot, copilot, and observer), provided the passenger oxygen supply is shut off.

Minimum Oxygen Cylinder Pressure for Dispatch

Table 17-1 shows the minimum oxygen system pressure, according to the number of passengers, to meet consumption requirements for dispatch.

NOTE

The minimum oxygen pressure for dispatch are for an ambient temperature of 21° C (70° F). For other temperatures, refer to the oxygen pressure correction chart (Figure 17-4).

Table 17-1. MINIMUM OXYGEN PRESSURE FOR DISPATCH

WITH OBSERVER

NUMBER OF PAX	INDICATED PRESSURE (PSI)	NUMBER OF PAX	INDICATED PRESSURE (PSI)
01	780	16	1140
02	800	17	1160
03	820	18	1180
04	850	19	1210
05	870	20	1230
06	900	21	1260
07	920	22	1280
08	940	23	1300
09	970	24	1330
10	990	25	1350
11	1020	26	1380
12	1040	27	1400
13	1060	28	1420
14	1090	29	1450
15	1110	30	1470

WITHOUT OBSERVER

NUMBER OF PAX	INDICATED PRESSURE (PSI)	NUMBER OF PAX	INDICATED PRESSURE (PSI)
01	600	16	960
02	620	17	990
03	650	18	1010
04	670	19	1030
05	700	20	1060
06	720	21	1080
07	740	22	1110
08	770	23	1130
09	790	24	1150
10	820	25	1180
11	840	26	1200
12	870	27	1230
13	890	28	1250
14	910	29	1280
15	940	30	1300

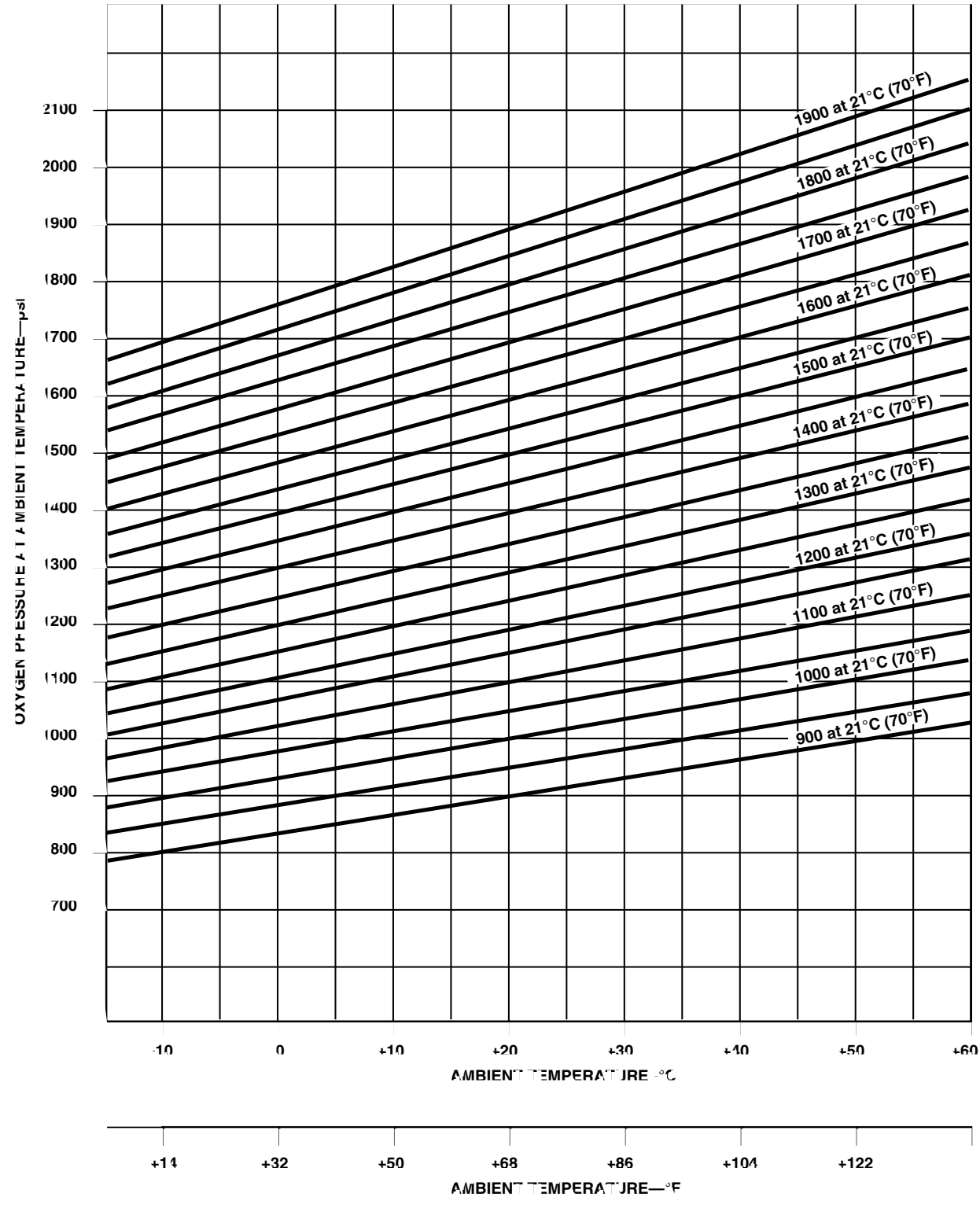


Figure 17-4. Oxygen Pressure Correction Chart

Crew Oxygen Equipment

Crew Oxygen Masks

The crew oxygen mask (Figure 17-5) is a quick-donning oro-nasal mask. It provides oxygen flow either by demand or under constant pressure.

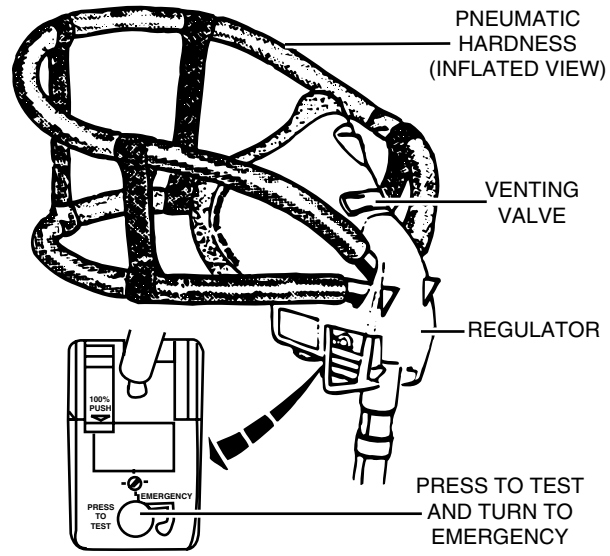


Figure 17-5. Crew Oxygen Mask

The mask supply hose has a quick-disconnect fitting that permits it to be connected to an adapter on the portable oxygen cylinder.

A microphone is built into the mask assembly. To use the microphone with either the control yoke PTT or hot mike switch, the audio panel microphone selector switch must be on MASK.

A manually operated vent valve on the mask shell is for use in conjunction with smoke goggles.

The integral mask regulator is controlled by a sliding switch and a rotary knob located on the mask's underside. The switch selects either N (normal, i.e., diluter demand) or 100% operation. The knob is rotated to select EMERGENCY (constant flow) operation and is pressed to test the regulator.

Observer Oxygen Mask

The observer mask is similar to the crew masks, except the flow indicator is installed in the supply hose. It has no vent valve for smoke goggles.

The flow indicator turns green when adequate oxygen pressure is supplied to the mask.

Mask Stowage Box

The mask stowage box (Figure 17-6) stores the crew oxygen masks, and controls and monitors the oxygen flow at each station.

Each box integrated in the cockpit lateral console is directly connected to both the oxygen distribution line and the aircraft communication system.

An automatic ON/OFF valve incorporated in each stowage box keeps the mask regulator unpressurized when the mask is in the stowed position.

A blinker assembly, located in the top right corner of the box, indicates oxygen flow. When oxygen is flowing, the blinker is yellow.

Also incorporated in the stowage box is a sliding RESET–TEST switch. It is used to reset the shut-off valve and to test the oxygen mask without removing it from the box.

An additional box for the observer's mask is

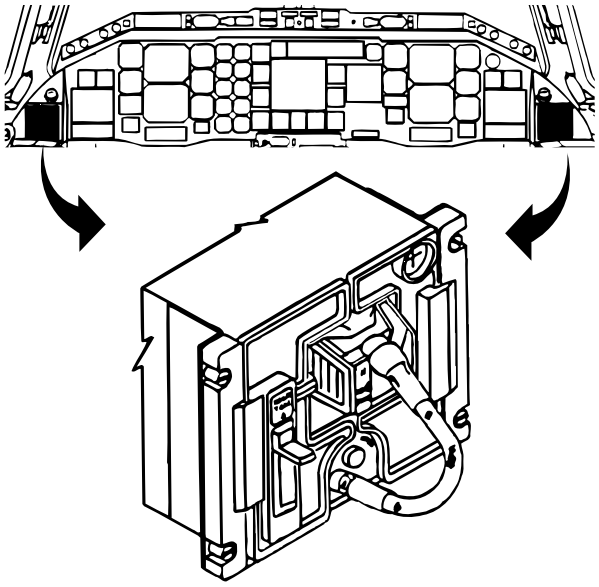


Figure 17-6. Mask Stowage Box

installed on the cockpit bulkhead near the observer's seat.

Full Face Mask

The full face mask (Figure 17-7) is designed for use in toxic atmospheres. It is superior in protection to the crew oxygen mask with smoke goggles.

As with the crew oxygen mask, the full face mask is equipped with a microphone.

A 100% demand regulator with a purge button is attached directly to the mask face piece. Pressing the button creates a positive pressure in the face piece to force out smoke or contaminants.

The full face mask can be fitted to a portable oxygen cylinder to fight fires or work in a toxic gas environment.

There may be up to three full face masks in the aircraft, depending upon customer requirement. One mask can be packed with the smoke goggles behind the pilot's seat.

Optionally, two masks can be hung on quick disconnects on the bulkhead behind the pilot seats. These two masks are directly connected to the crew oxygen system supply lines and communication systems.

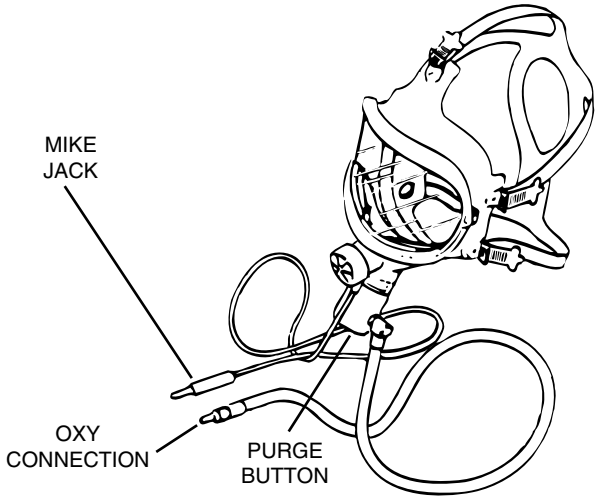


Figure 17-7. Full Face Mask

Additional masks can be stored in the cabin forward left closet or in the miscellaneous compartment.

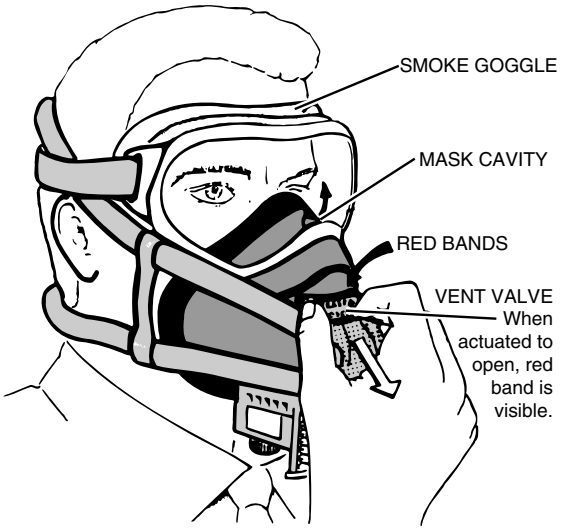


Figure 17-8. Smoke Goggles

Smoke Goggles

The smoke goggles (Figure 17-8) are designed to be used with the crew oxygen mask. The goggles are stored in a bag behind the pilot.

The mask vent valve transmits oxygen between the mask and the smoke goggles. To purge the smoke goggles, the valve must be manually actuated by the wearer.

If the oxygen mask regulator is selected to 100% or EMERGENCY, a metered oxygen flow is directed to the smoke goggles. The goggles can be quickly purged by pressing the EMERGENCY knob on the mask regulator.

Mask Test

Crew Mask Test Procedure

The crew oxygen masks are tested by using the RESET/TEST switch on the mask stowage box.

Prior to testing the mask, configure the pilot-to-pilot communications setup for mask use as follows:

1. Turn the radio master on
2. Set audio control panel selector to MASK and adjust the volume
3. Turn speaker on
4. Select HOT MIC

Once configured, the test is performed as follows:

1. Depress the RESET/TEST switch on the mask stowage box in the direction of the arrow.
2. Check for:
   - White band on the door face
   - Blinker turns yellow, then black
   - Sound of oxygen flow on the speaker, verifying the integrity of the mask mic
3. While holding switch depressed, press regulator EMERGENCY knob; blinker turns yellow until button released.
4. Release switch; white band disappears.
5. Check diluter lever in the 100% position.
6. Return audio panel selector to BOOM.

This check will be unsuccessful if the main oxygen valve in the closet is closed. Leave this valve ON (full clockwise).

Check oxygen pressure gage for minimum dispatch pressure.

Observer Mask Test Procedure

If the observer's seat is occupied, test the observer's mask as follows:

1. Check oxygen/interphone plugs installed.
2. Press the red lever on the right side and check that the harness expands within its container (flow indicator turns white).
3. Set the diluter lever to 100% and rotate the EMERGENCY control knob. Verify harness expands, then release the knob.

Operation

To use the crew oxygen mask, pull the mask from the stowage box and hold the red ears firmly between the forefinger and thumb. The mask harness rapidly inflates to a large rigid shape, allowing the user to quickly put it on. After the mask is on and the ears released, the harness deflates, securing the mask to the face.

The crew oxygen system is always ready for use. Oxygen flow is controlled by the diluter-demand regulator in the mask face piece.

With the diluter lever in the N (normal) position, oxygen is supplied in the diluter-demand mode up to cabin altitudes of 32,000 feet. In the 100% position, 100% oxygen is supplied, as demanded, regardless of altitude.

For normal operations, the diluter lever is positioned to 100%.

If the regulator knob is placed in the EMERGENCY position, 100% oxygen is supplied under pressure, regardless of altitude.

PASSENGER OXYGEN SYSTEM

General

In the event of cabin depressurization, the passenger oxygen system supplies supplemental oxygen to the passenger.

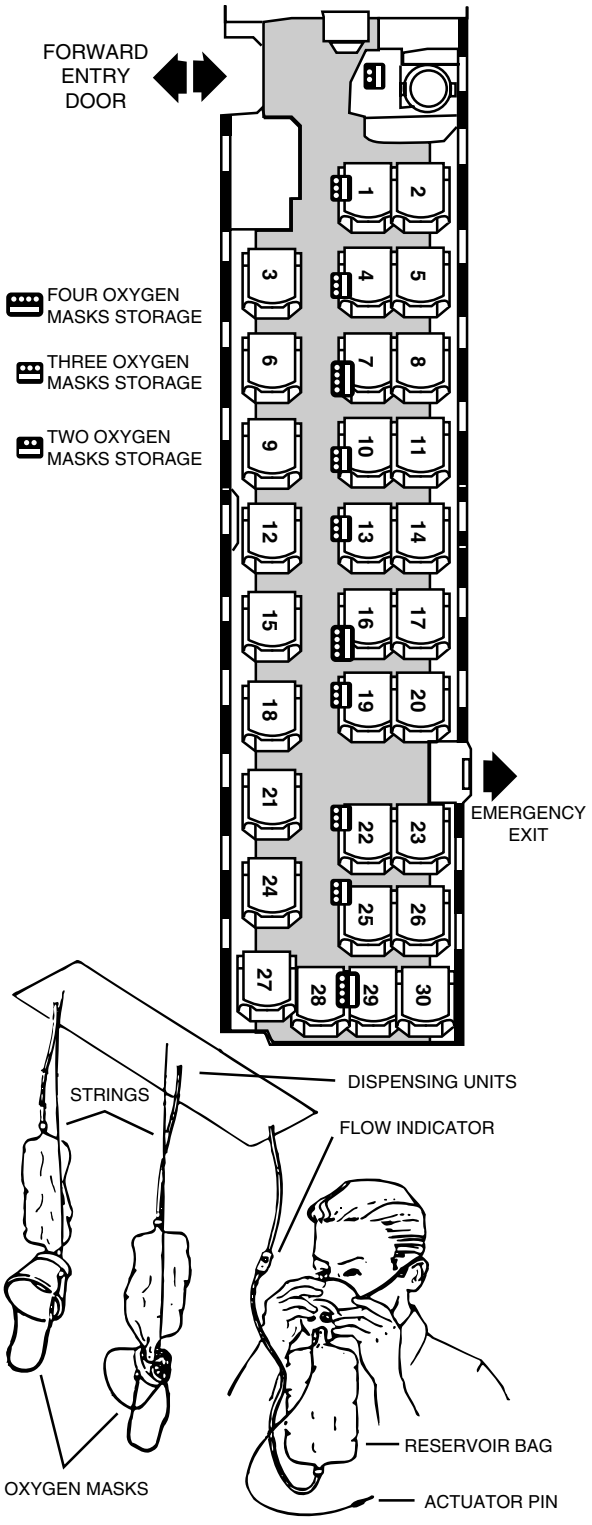


Figure 17-9. Passenger Oxygen Mask Location and Use

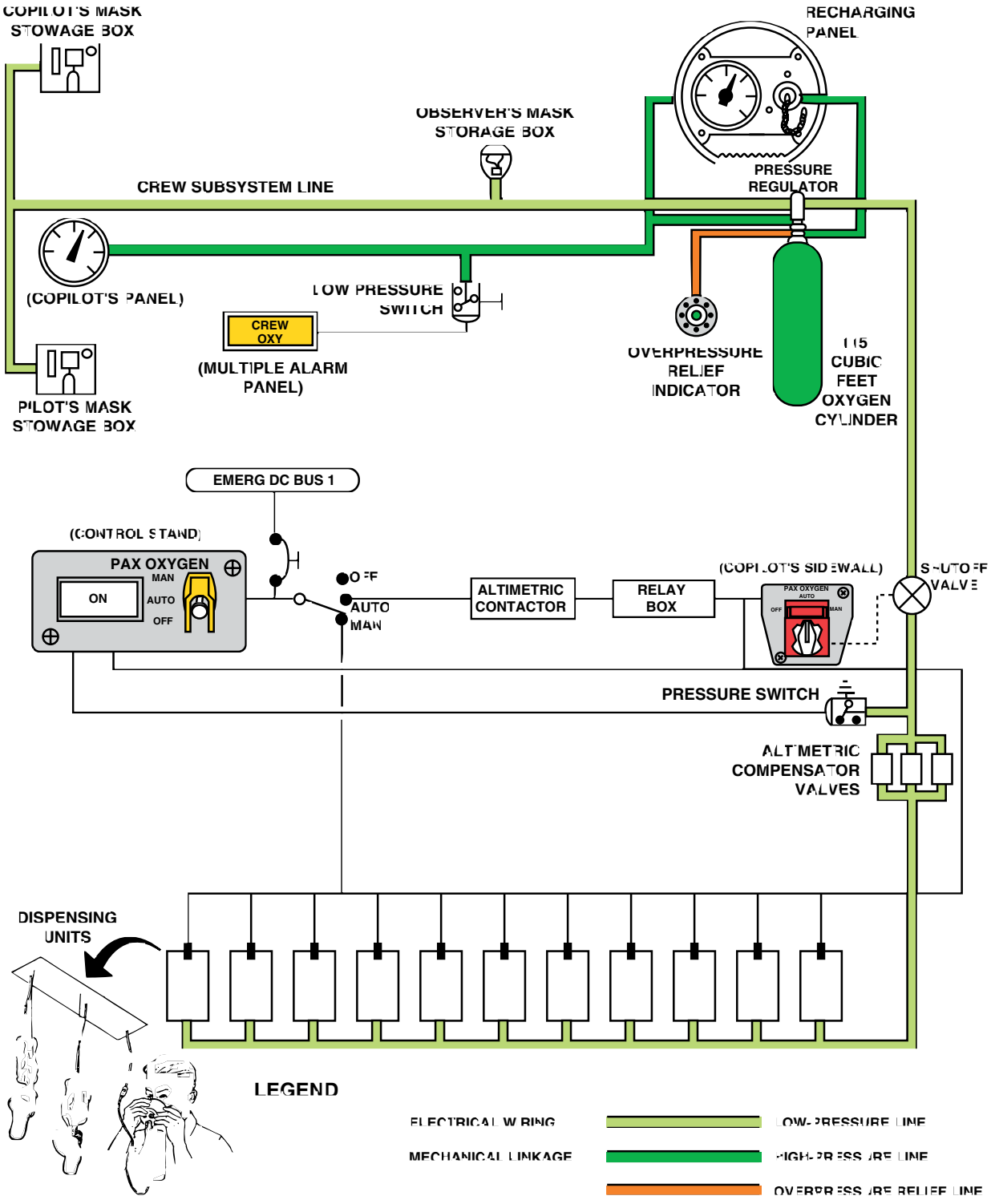


Figure 17-10. Oxygen System

Passenger oxygen is supplied by the same 115-cubic-foot oxygen cylinder, located in the right forward closet, that supplies the cockpit. Oxygen is supplied to the individual passenger oxygen masks through an oxygen manifold assembly.

Passenger Masks

The passenger masks are stored in eleven dispensing units in the cabin ceiling and the lavatory

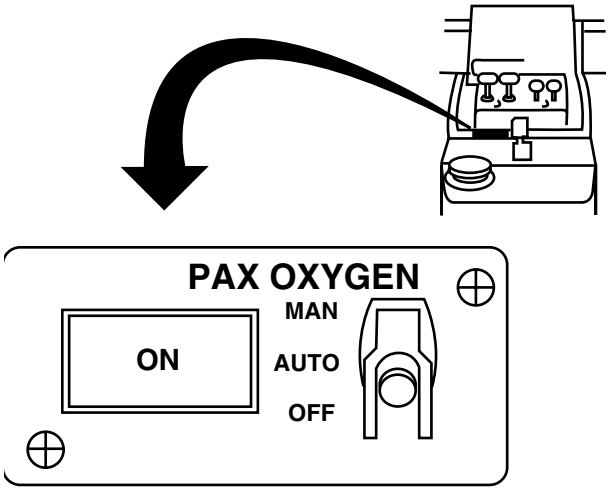


Figure 17-11. PAX OXYGEN Panel

(Figure 17-9).

The mask assembly consists of a continuous-flow oral nasal face mask with inhalation, dilution, and exhalation valves, and an economizer bag. A tube with a flow indicator connects the mask to the oxygen manifold. The flow indicator turns green when adequate oxygen pressure is being supplied. An elastic strap holds the mask to the face.

To use the mask, the passenger must pull it against the face, activating a continuous supply of oxygen to the mask reservoir bag.

The masks are automatically deployed in the event cabin altitude exceeds 14,000 feet MSL, or manually by activation of the cockpit PAX OXYGEN SWITCH.

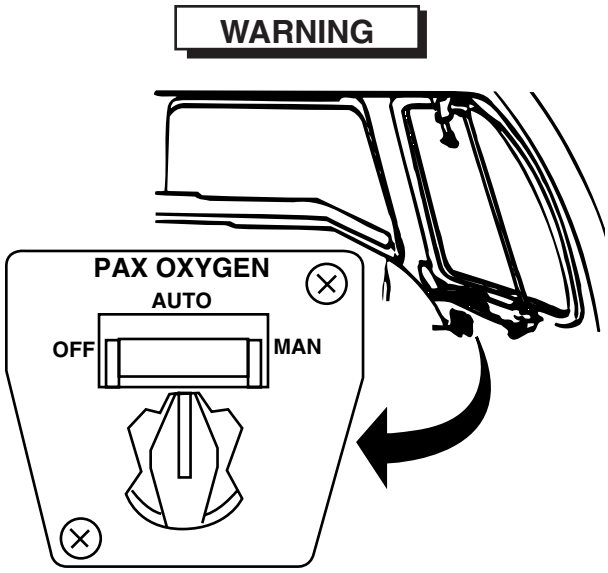


Figure 17-12. Copilot's Oxygen Panel

During oxygen use, the NO SMOKING sign must be on and strictly observed.

Aircraft with a forward lavatory must comply with an Airworthiness Directive issued to ensure an immediate source of oxygen to the flight attendant in case of cabin decompression.

The following condition must be met:

1. Maximum operating altitude is limited to 25,000 ft unless the RH front row aisle seat is reserved for the flight attendant use during operation above 25,000 ft, or

2. Install under the flight attendant seat an oxygen dispensing unit with an oxygen mask permanently attached. Maximum operating altitude in limited to 30,000 ft.

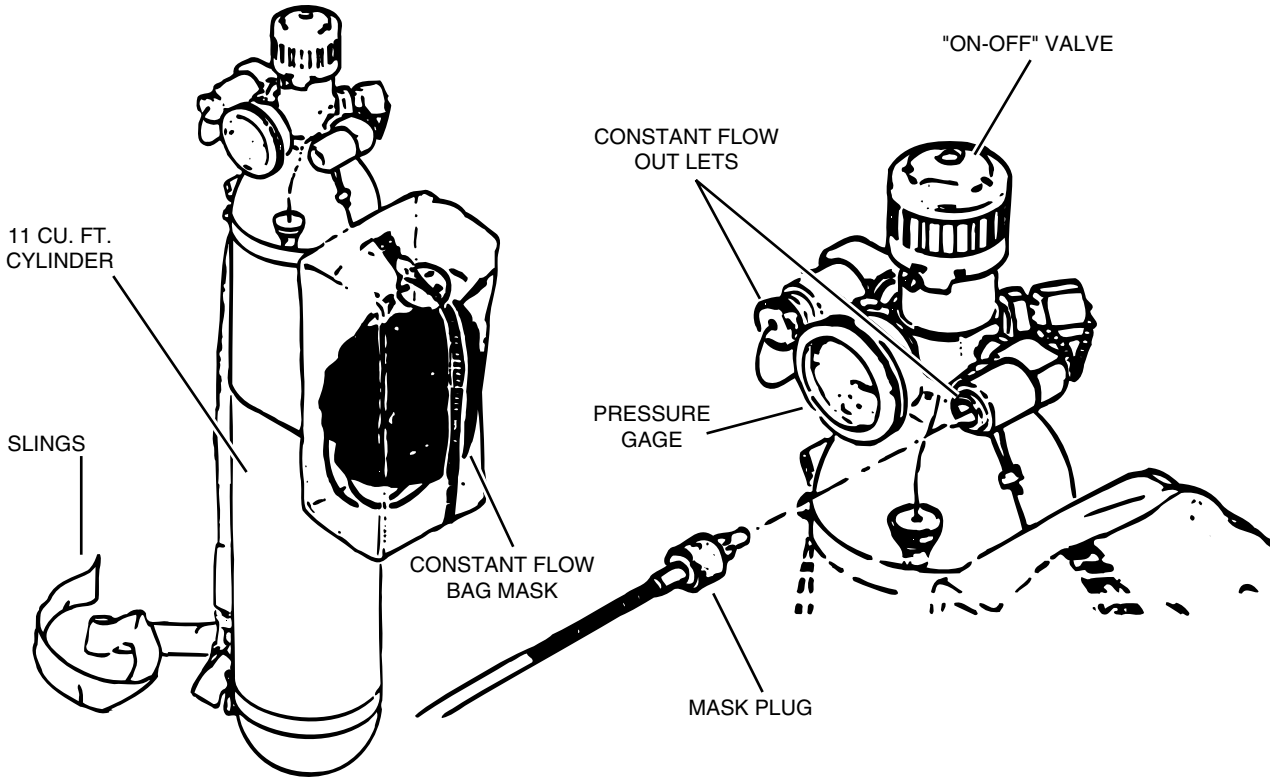


Figure 17-13. Portable Oxygen Bottle

Controls and Indicators

The passenger oxygen system controls are located in two separate places in the cockpit.

The primary oxygen system control is on the PAX OXYGEN panel, located on the center pedestal just aft of the throttles (Figure 17-11). The panel contains a yellow-guarded, three-position (MAN/AUTO/OFF) toggle switch, and a white ON light.

The manual shutoff valve control is located on the copilot's sidewall PAX OXYGEN panel (Figure 17-12). It is a red-guarded, three-position (OFF/AUTO/MAN) rotary switch.

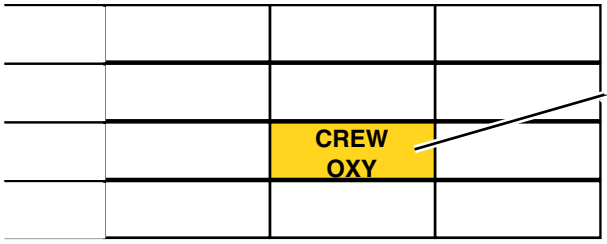
The OFF position deenergizes the passenger oxygen system. It is used for training flights or when configured for cargo operation.

Operation

For automatic operation of the passenger oxygen system, both the primary yellow-guarded toggle switch and the manual red-guarded rotary switch must be in AUTO.

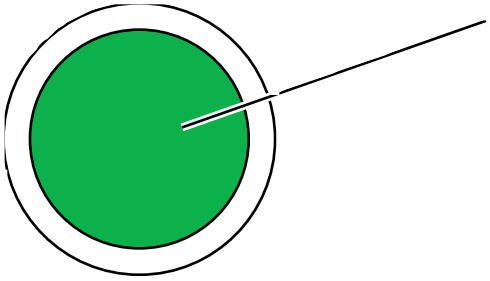
If cabin altitude exceeds 14,000 ±500 feet, an altimetric switch sends an electrical signal to open the passenger oxygen shutoff valve and the dispensing unit doors (Figure 17-10).

The white ON light on the center pedestal PAX OXYGEN panel illuminates when the passenger oxygen line is pressurized. If the light does not illuminate, lift the yellow safety guard and place the toggle switch to MAN. If the light still does not illuminate, lift the red guard on the manual PAX OXYGEN control knob and select MAN.



CREW OXYGEN WARNING LIGHT (MAP)

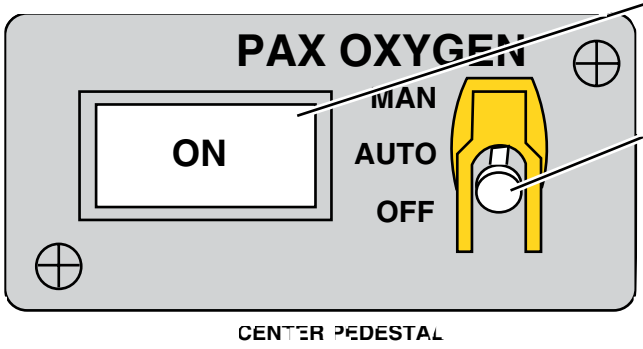
ILLUMINATED — Advises the crew there is approximately 35 minutes of oxygen reserve (375 ±25 psi), provided the passenger oxygen is shut off.



OVERBOARD DISCHARGE INDICATOR (RH front fuselage)

GREEN DISC VISIBLE Pressurized oxygen system is normal.

GREEN DISC NOT VISIBLE — Safety fitting disc is ruptured and oxygen cylinder is depleted.



CENTER PEDESTAL

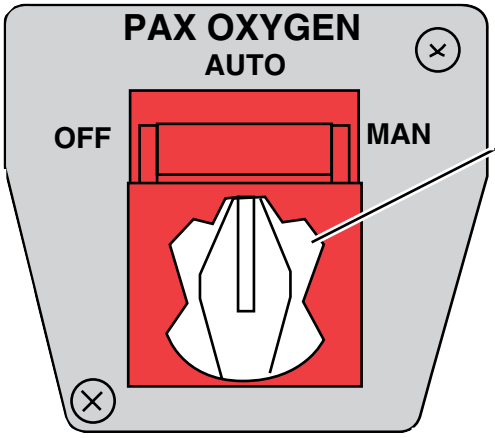
PAX OXYGEN ON LIGHT
ILLUMINATED Passenger oxygen line is pressurized.

PAX OXYGEN TOGGLE SWITCH

MAN The altimetric contactor actuating dispensing units and shutoff valve solenoid are bypassed.

AUTO — Altimetric contactor automatically actuates dispensing units and shutoff valve at 14,000 ±500 feet cabin altitude.

OFF Passenger oxygen system is inoperative.



PAX OXYGEN ROTARY SWITCH

AUTO — Shutoff valve is actuated by altimetric contactor, via PAX OXYGEN toggle switch.

MAN — Shutoff valve is manually opened, pressurizing the passenger oxygen line.

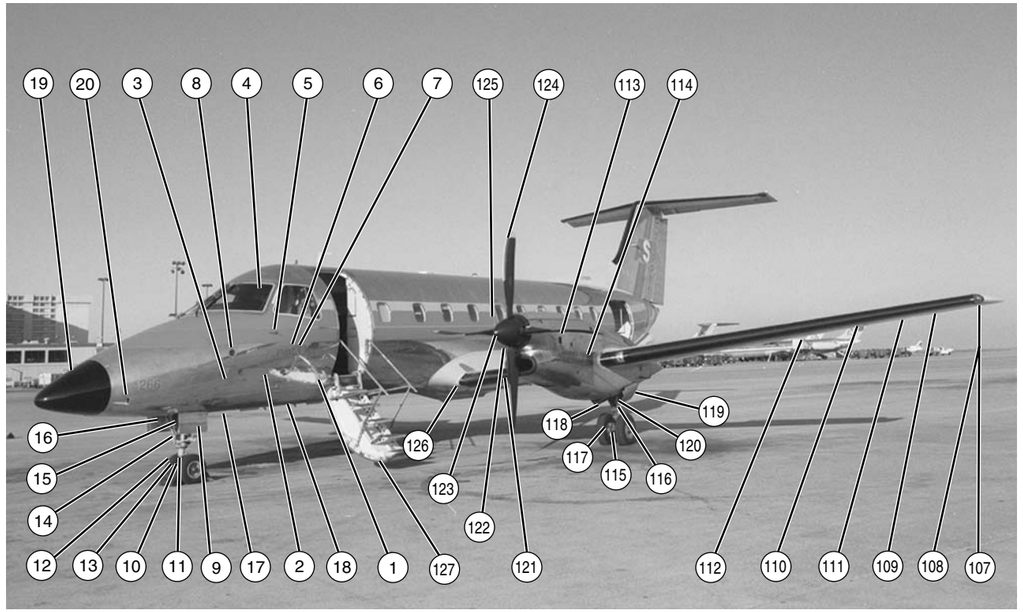
OFF — Shutoff valve is manually closed.

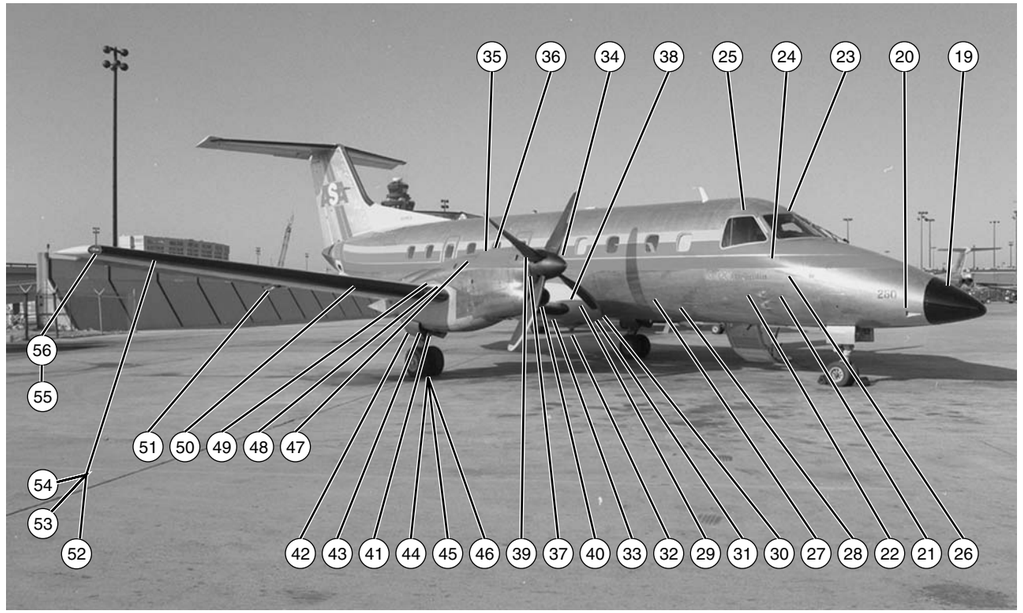
Figure 17-14. Oxygen System Controls and Indicators

WALKAROUND

The following section is a pictorial walkaround on an Embraer-120. It shows each item called out in the exterior power-off preflight inspection. The fold-out pages at the beginning of the walkaround section should be unfolded before starting to read.

The general location photographs do not specify every checklist item. However, each item is portrayed on the large-scale photographs that follow.





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WALKAROUND INSPECTION

LEFT NOSE SECTION



1. **crew/passenger door control panel—**CLOSED



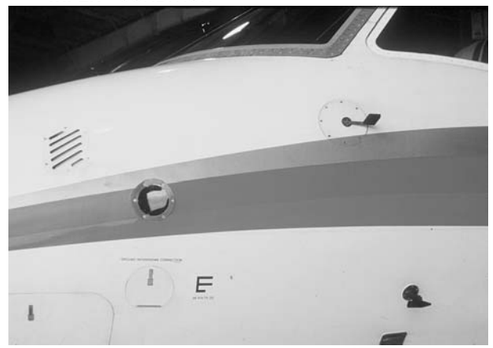
2. **EXTERNAL POWER ACCESS PANEL—**AS REQUIRED



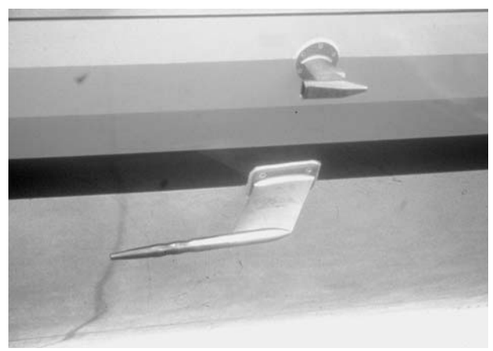
3. **Battery compartment—**CLOSED



4. **LEFT COCKPIT WINDOWS, WINDSHIELD AND WIPER—**CONDITION



5. **ANGLE-OF-ATTACK SENSOR—**CONDITION



6. **TOTAL AIR TEMPERATURE PROBE—**CONDITION
7. **PILOT'S PITOT TUBE—**UNOBSTRUCTED



8. **ICE LIGHT—**CONDITION

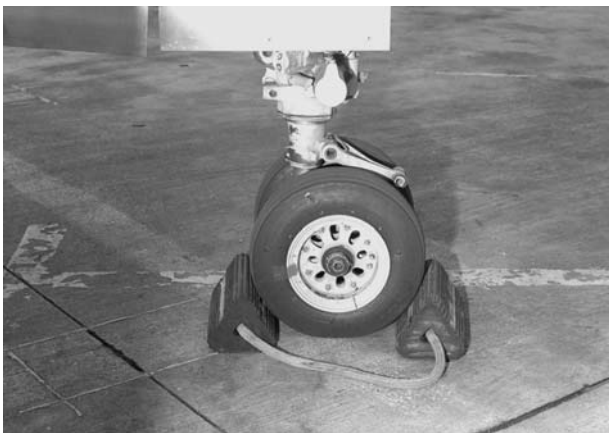
NOSE GEAR



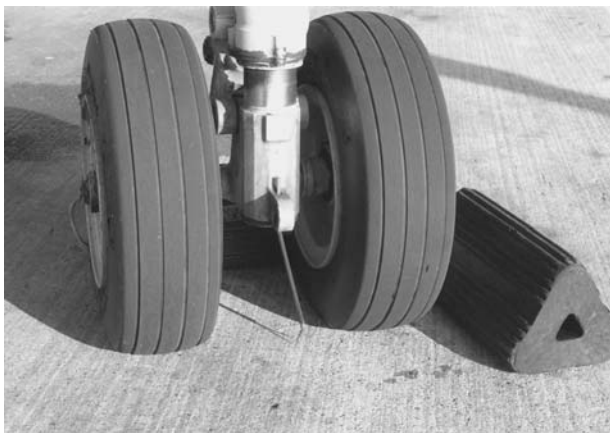
9. **FORWARD DOORS (NOSE GEAR WHEEL WELL)—** LATCHED



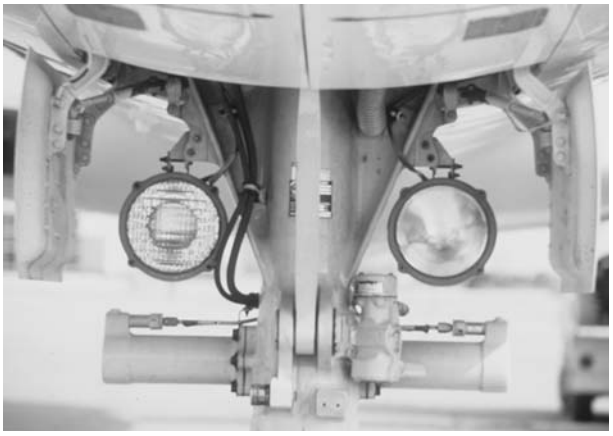
11. **STEERING JACK OVERTRAVEL INDICATOR—**INSIDE HOUSING



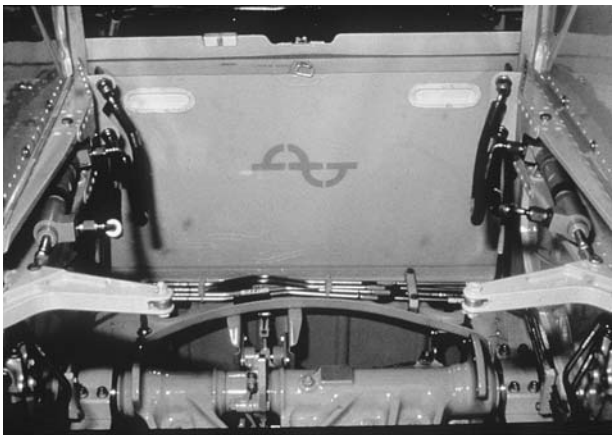
10. **WHEEL CHOCKS—**IN PLACE



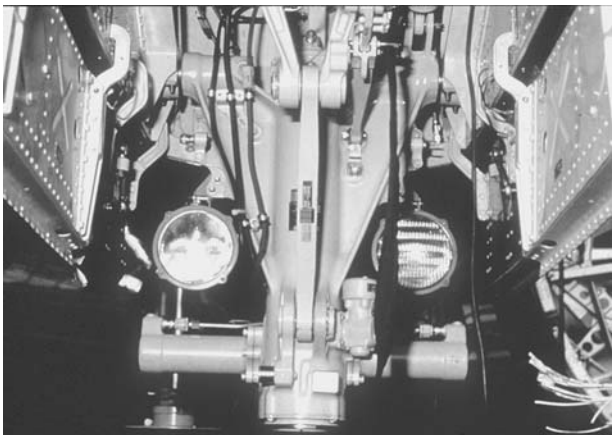
12. **TIRES—**CONDITION/INFLATION
13. **STATIC DISCHARGER—**TOUCHING RAMP SURFACE



14. LIGHTS—CONDITION



16. AVIONICS SERVICE DOOR—CLOSED



15 GROUND LOCKING PIN—REMOVED

UNDERSIDE OF FORWARD FUSELAGE



17. CONTROLS RIGGING DOOR—SECURE
18. ANTENNAS—CONDITION

NOSE CAP



19. **RADOME—**CLOSED, LATCHES SECURE
20. **Air Intakes and VEnts—**UNOBSTRUCTED

RIGHT NOSE SECTION



21. **NOSE HYDRAULIC COMPARTMENT—**CLOSED



22. **OXYGEN DISC (GREEN) AND RECHARGING SERVICE DOOR—**SECURE CLOSED



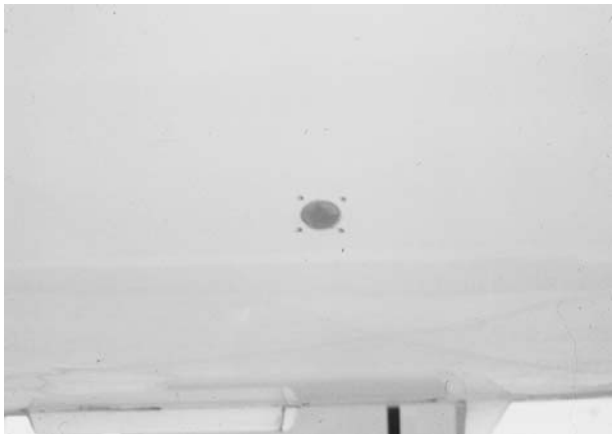
23. **RIGHT COCKPIT WINDOWS, WINDSHIELD AND WIPER—**CONDITION
24. **Angle-of-attack sensor—**CONDITION
25. **Slip sensor—**CHECK



26. **COPILOT AND AUXILIARY PITOT TUBES—**UNOBSTRUCTED



27. TOILET SERVICE DOOR (IF APPLICABLE)—CLOSED



28. OAT SENSOR—CHECK CONDITION

UNDERWING CENTER SECTION

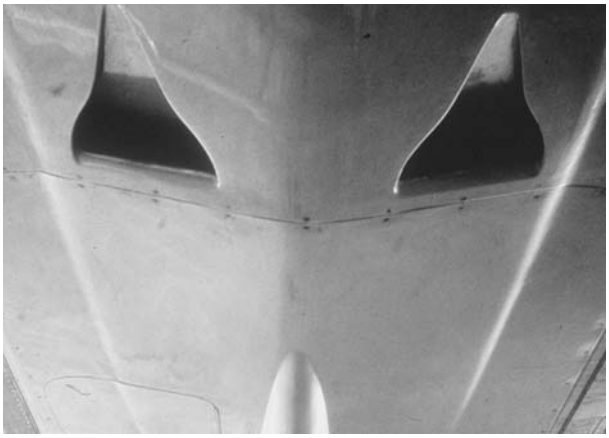


29. ROTATING BEACON—IN PLACE

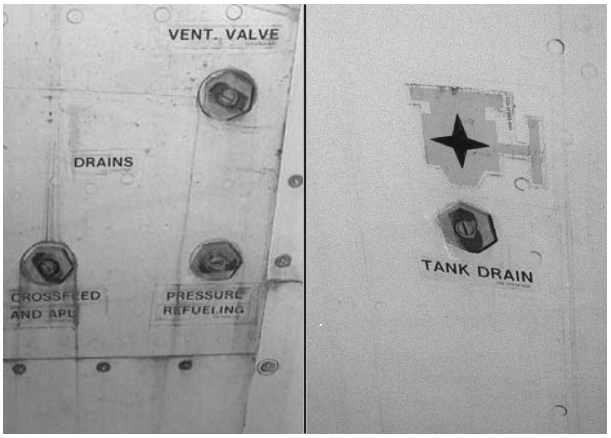


31. ENGINE FIRE EXTINGUISHER RED DISC INDICATORS—INTACT

32. ANTENNAS—CONDITION



30. ENVIRONMENTAL INTAKES—UNOBSTRUCTED



33. DRAINS—NO LEAKS

RIGHT WING



34. INBOARD SIDE OF RIGHT WING—CONDITION



35. INBOARD SIDE OF RIGHT ENGINE NACELLE—CHECK



36. OIL ACCESS PANEL—CLOSED



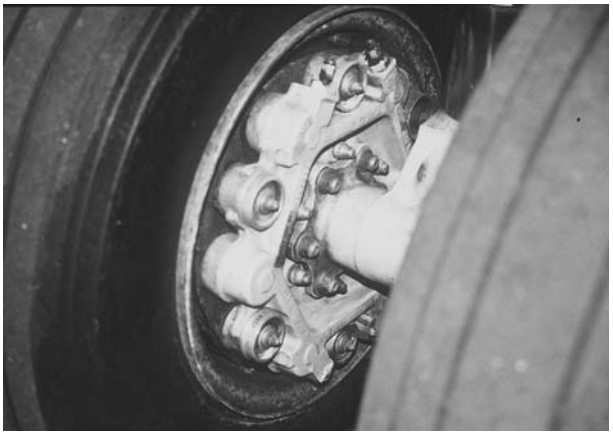
37. RIGHT PROPELLER BLADES—CONDITION NO/LEAKS
38. RIGHT PROPELLER SPINNER—SECURE AND ALIGNED
39. PROPELLER FEATHER POSITION ALIGNMENT MARK—CHECK



40. ENGINE INTAKE—UNOBSRTUCTED



41. FORWARD DOORS (MAIN GEAR WHEEL WELL)— LATCHED



46. brake wear indicators—PROPER INDICATION



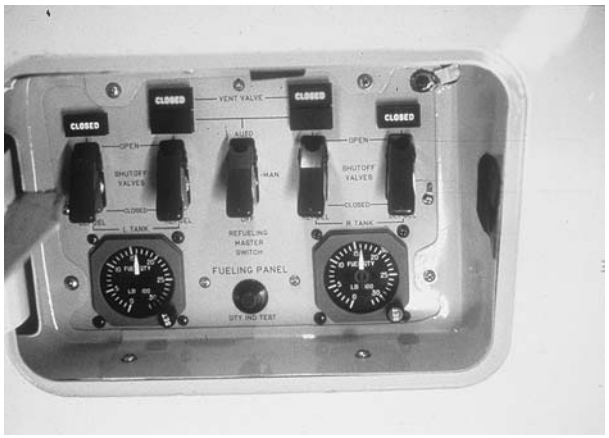
42. MAIN GEAR WHEEL WELL—CONDITION/NO LEAKS
43. GEAR PIN—REMOVED



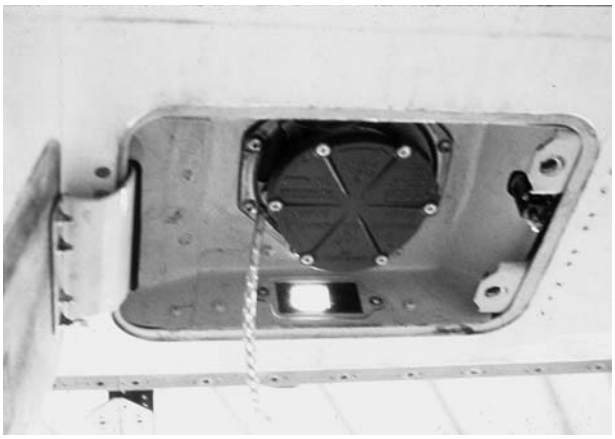
47. OUTBOARD SIDE OF RIGHT ENGINE NACELLE— CHECK



44. RIGHT MAIN GEAR TIRES—INFLATION / CONDITION
45. BRAKE LINES, ELECTRICAL WIRING—CONDITION/ NO LEAKS



48. FUELING PANEL—CHECK



49. FUELING/DEFUELING DOOR—CLOSED



52. FLUX SENSOR PANEL—CHECK
53. REFUELING VENT SCREEN—CONDITION
54. FUEL VENT NACA SCOOP—UNOBSTRUCTED



50. RIGHT WING LEADING EDGE PNEUMATIC DEICER BOOT—CONDITION



55. RIGHT WING NAVIGATION LIGHT CLEAR PLASTIC COVER—CONDITION
56. NAVIGATION LIGHTS—OPERATION/IF REQUIRED



51. DRIPLESS FUEL QUANTITY STICKS (4)—SECURE



57. STATIC DISCHARGE WICKS—CONDITION
58. AILERON AND TRIM TAB—CONDITION
59. VORTEX GENERATORS—SECURE



60. OUTBOARD FLAPS—CONDITION
61. STALL FENCE—CONDITION/SECURE



65. hydraulic service panel—SECURE
66. inboard flap—CONDITION



62. OUTBOARD SIDE OF RIGHT ENGINE NACELLE, (UNDERWING)—CHECK
63. engine tail pipe area—CONDITION
64. nacelle flap—CONDITION

right fuselage



67. TOP ROTATING BEACON—IN PLACE
68. DORSAL FIN RIGHT SIDE—CONDITION
69. FUSELAGE—CONDITION (PANELS AND WINDOWS)
70. EMERGENCY EXITS—SECURE
71. ANTENNAS—CONDITION

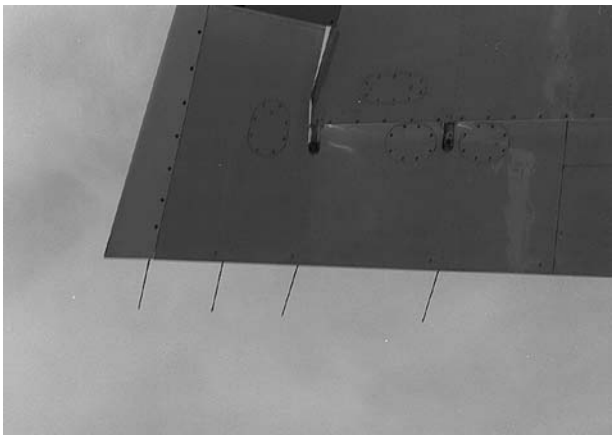


72. TOILET SERVICE DOOR (IF APPLICABLE)—CLOSED

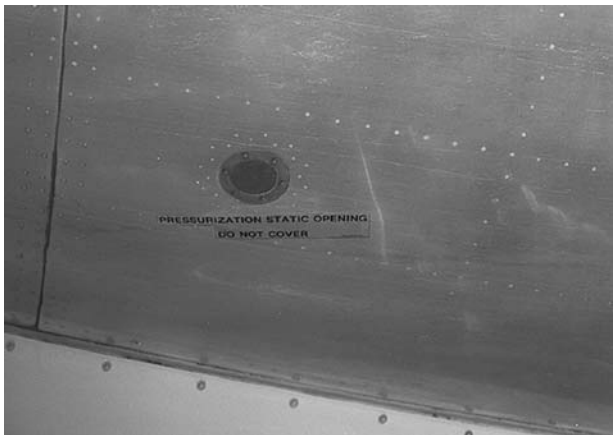
TAIL SECTION



73. **ANTENNAS**—CONDITION
74. **TAIL COMPARTMENT ACCESS**—SECURE
75. **RIGHT VENTRAL STRAKE**—SECURE



80. **RIGHT SIDE HORIZONTAL AND VERTICAL STABILIZERS**—CONDITION
81. **RIGHT LOGO LIGHT**—CONDITION



76. **RIGHT STATIC PORT (ELECTROPNEUMATIC OUTFLOW VALVE)**—UNOBSTRUCTED



82. **VORTEX GENERATORS, RIGHT SIDE OF HORIZONTAL AND VERTICAL STABILIZERS**—SECURE



77. **APU FIRE EXTINGUISHER RED DISC INDICATOR**—SECURE
78. **APU FAIRING INTAKE SCREEN AND DRAINS**—CONDITION/UNOBSTRUCTED
79. **APU TAIL PIPE**—ALIGNMENT/CONDITION



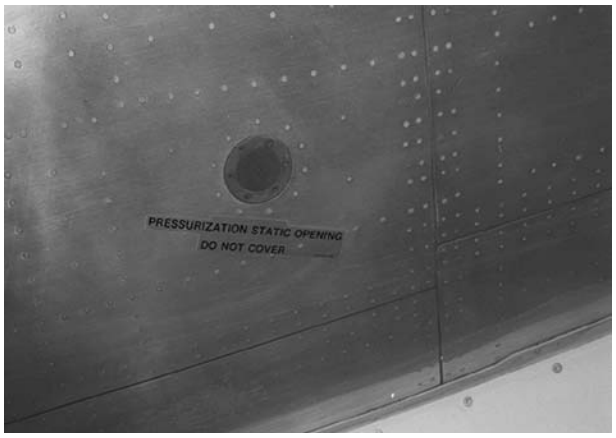
83. **ELEVATOR AND RUDDER CONTROLS**—SECURE/CONDITION
84. **STATIC DISCHARGE WICKS**—IN PLACE/CONDITION
85. **NAVIGATION LIGHT**—IN PLACE/CONDITION



86. LEFT LOGO LIGHT—CONDITION
87. VORTEX GENERATORS, LEFT SIDE OF HORIZONTAL AND VERTICAL STABILIZERS—SECURE
88. VORTEX GENERATORS (14 LEFT SIDE OF RUDDER 1)—SECURE



89. LEFT SIDE APU SERVICE PANELS—SECURE
90. LEFT VENTRAL STRAKE—SECURE
91. ANTENNAS—CONDITION



92. left static port (PNEUMATIC OUTFLOW VALVE)—UNOBSTRUCTED

LEFT FUSELAGE



93. CARGO COMPARTMENT DOOR—CLOSED/AS REQUIRED



96. EMERGENCY EXIT—SECURE
97. FUSELAGE— CONDITION



94. ANTENNAS—CONDITION
95. DORSAL FIN (LEFT SIDE)—CONDITION

LEFT WING



98. INBOARD FLAP—CONDITION
99. NACELLE FLAP—CONDITION
100. ENGINE TAILPIPE AREA—CONDITION



101. OUTBOARD SIDE OF LEFT ENGINE NACELLE (UNDERWING)—CHECK



102. OUTBOARD FLAPS—CONDITION
103. STALL FENCE—CONDITION/SECURE
104. VORTEX GENERATORS—SECURE



110. DRIPLESS FUEL QUANTITY STICKS (4)—SECURE



105. AILERON AND BALANCE TAB—CONDITION
106. STATIC DISCHARGE WICKS—CONDITION



111. FUEL VENT NACA SCOOP—UNOBSTRUCTED
112. LEFT WING LEADING EDGE PNEUMATIC DEICER BOOT—CONDITION



107. LEFT WING NAVIGATION LIGHT CLEAR PLASTIC COVER—CONDITION
108. NAVIGATION LIGHTS—OPERATION IF REQUIRED
109. FLUX SENSOR PANEL—CHECK



113. OUTBOARD SIDE OF LEFT NACELLE, AIR INLETS, AND VENTS—UNOBSTRUCTED
114. OIL ACCESS PANEL—CLOSED



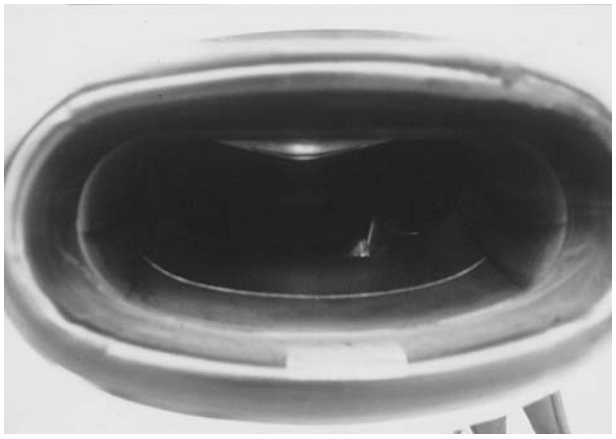
115. BRAKE WEAR INDICATORS—PROPER INDICATION
116. BRAKE LINES, ELECTRICAL WIRING—
NO LEAKS/CONDITION
117. LEFT MAIN GEAR TIRES—INFLATION/CONDITION



118. FORWARD DOORS (MAIN GEAR WHEEL WELL)—
LATCHED



119. GEAR PIN—REMOVED
120. MAIN GEAR WHEEL WELL—CONDITION/NO LEAKS



121. ENGINE INTAKE—UNOBSTRUCTED



122. PROPELLER FEATHER POSITION ALIGNMENT MARK—CHECK
123. LEFT PROPELLER SPINNER—SECURE/ALIGNED
124. LEFT PROPELLER BLADE—CONDITION/NO LEAKS



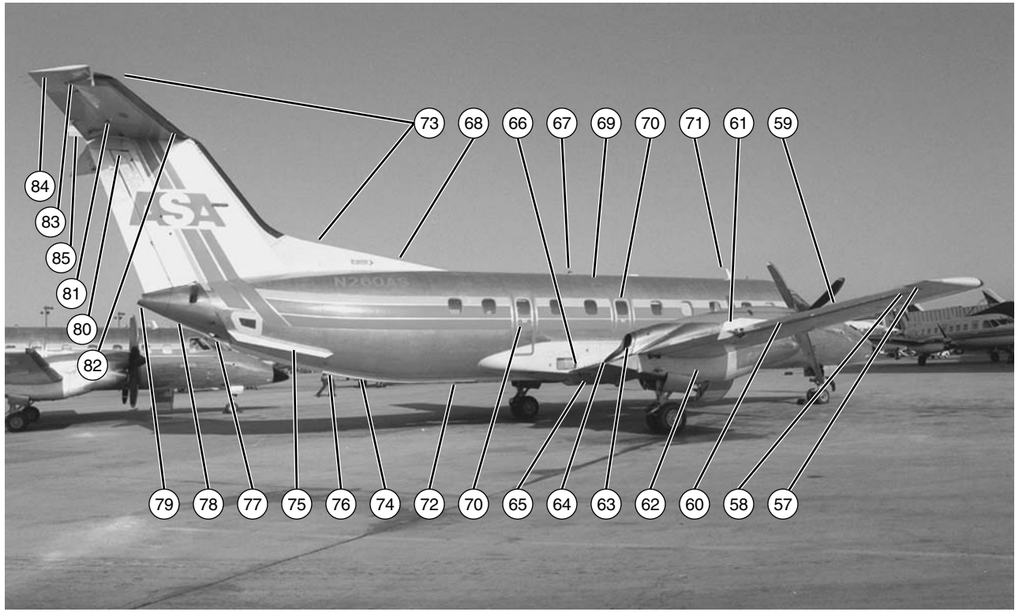
125. INBOARD SIDE OF LEFT ENGINE NACELLE—CHECK

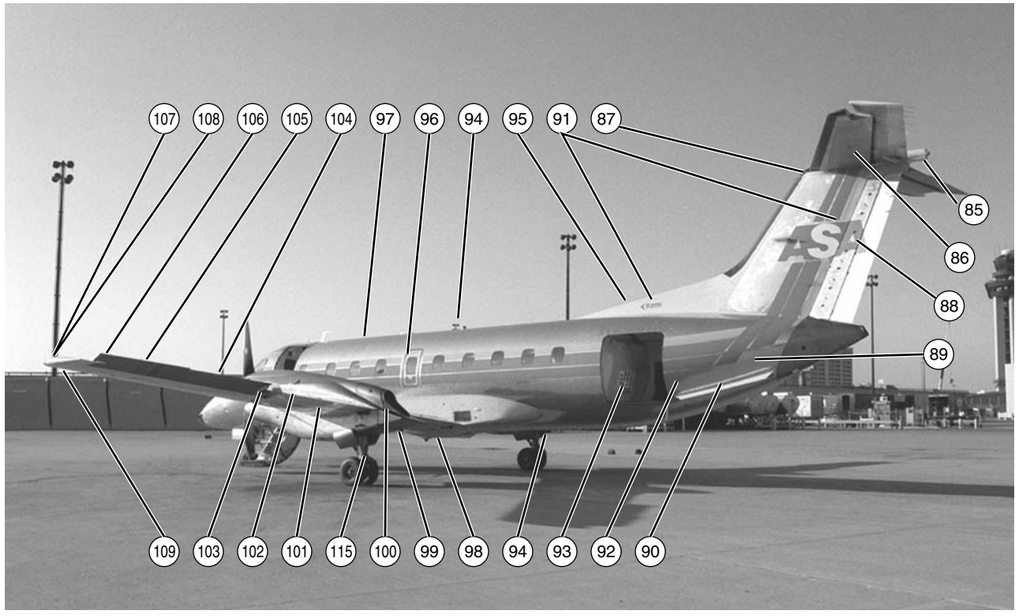


126. INBOARD SIDE OF LEFT WING—CONDITION



127. PASSENGER/CREW ENTRY DOOR—CHECK CONDITION


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Annunciator Panel

The Annunciator Panel section presents a color representations of all the annunciator lights in the airplane.

Please unfold page ANN-3 to the right and leave it open for ready reference as the annunciators are cited in the text.

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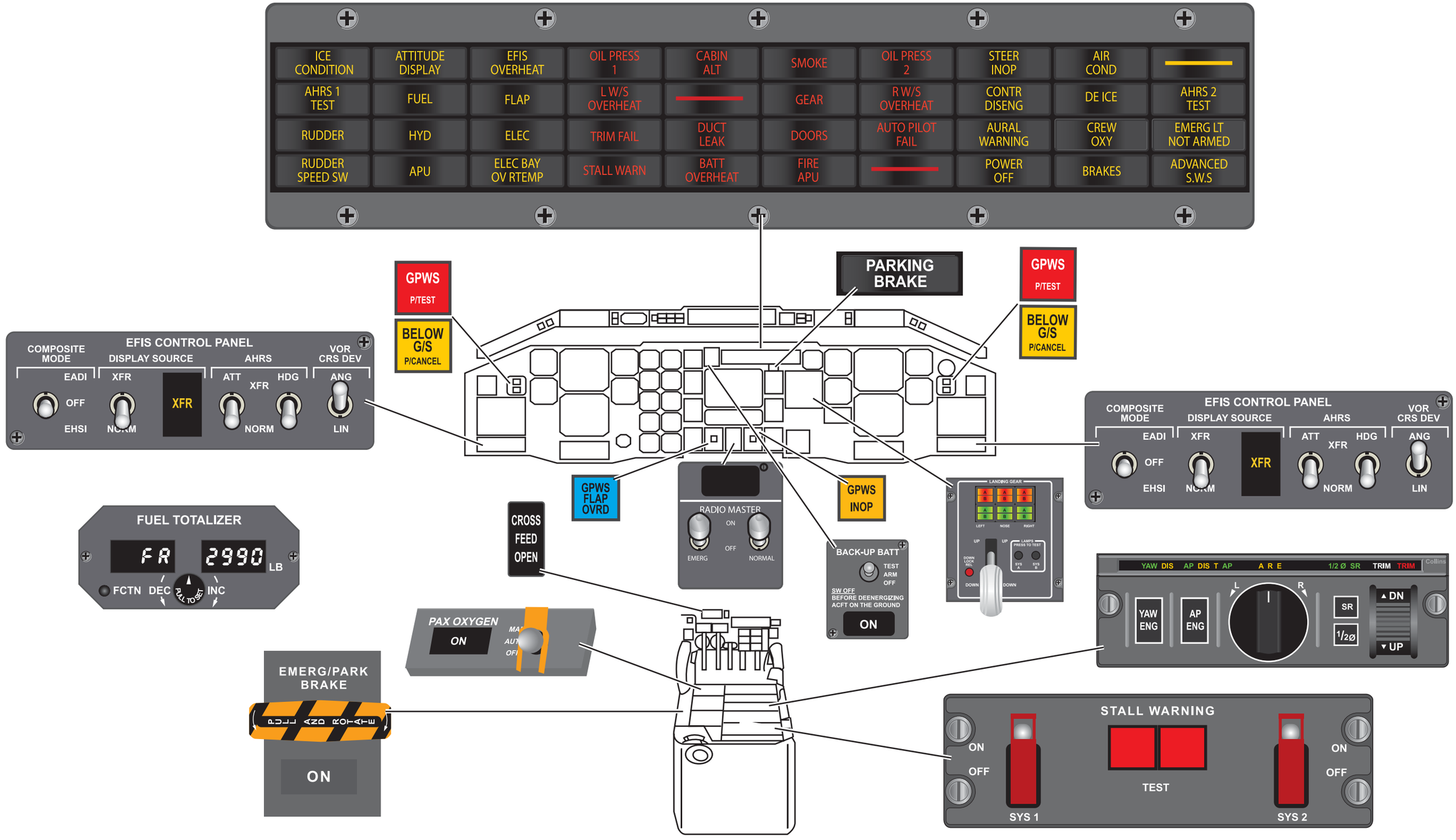


Figure ANN-1. Annunciators—Flight Instrument Panel

FOR TRAINING PURPOSES ONLY

Annunciator Panel

The Annunciator Panel section presents a color representations of all the annunciator lights in the airplane.

Please unfold page ANN-7 to the right and leave it open for ready reference as the annunciators are cited in the text.

FlightSafety international
EMB-120 PILOT TRAINING MANUAL

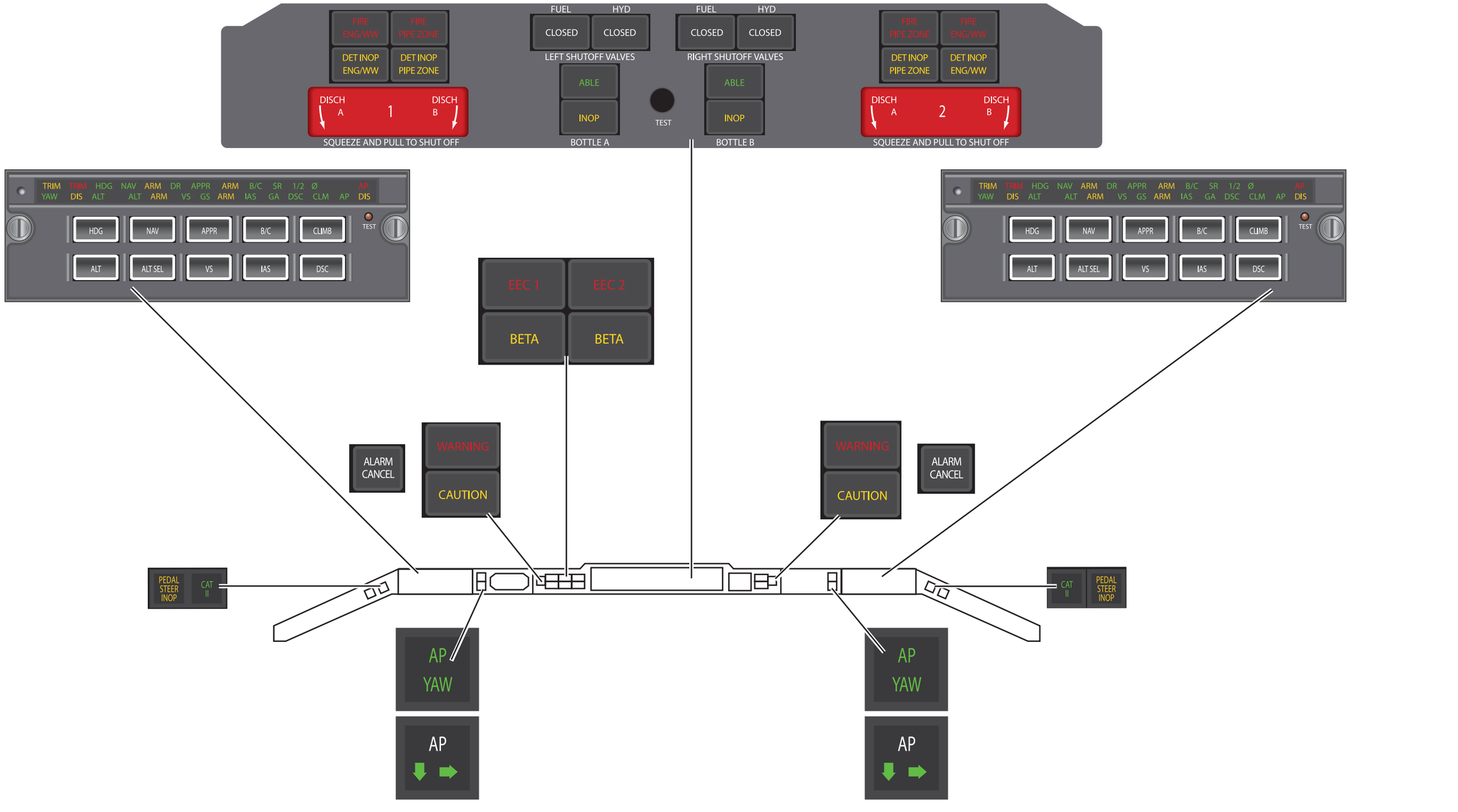


Figure ANN-2. Annunciators—Flight Instrument Panel (Cont.)

Annunciator Panel

The Annunciator Panel section presents a color representations of all the annunciator lights in the airplane.

Please unfold page ANN-11 to the right and leave it open for ready reference as the annunciators are cited in the text.

FlightSafety international
EMB-120 PILOT TRAINING MANUAL

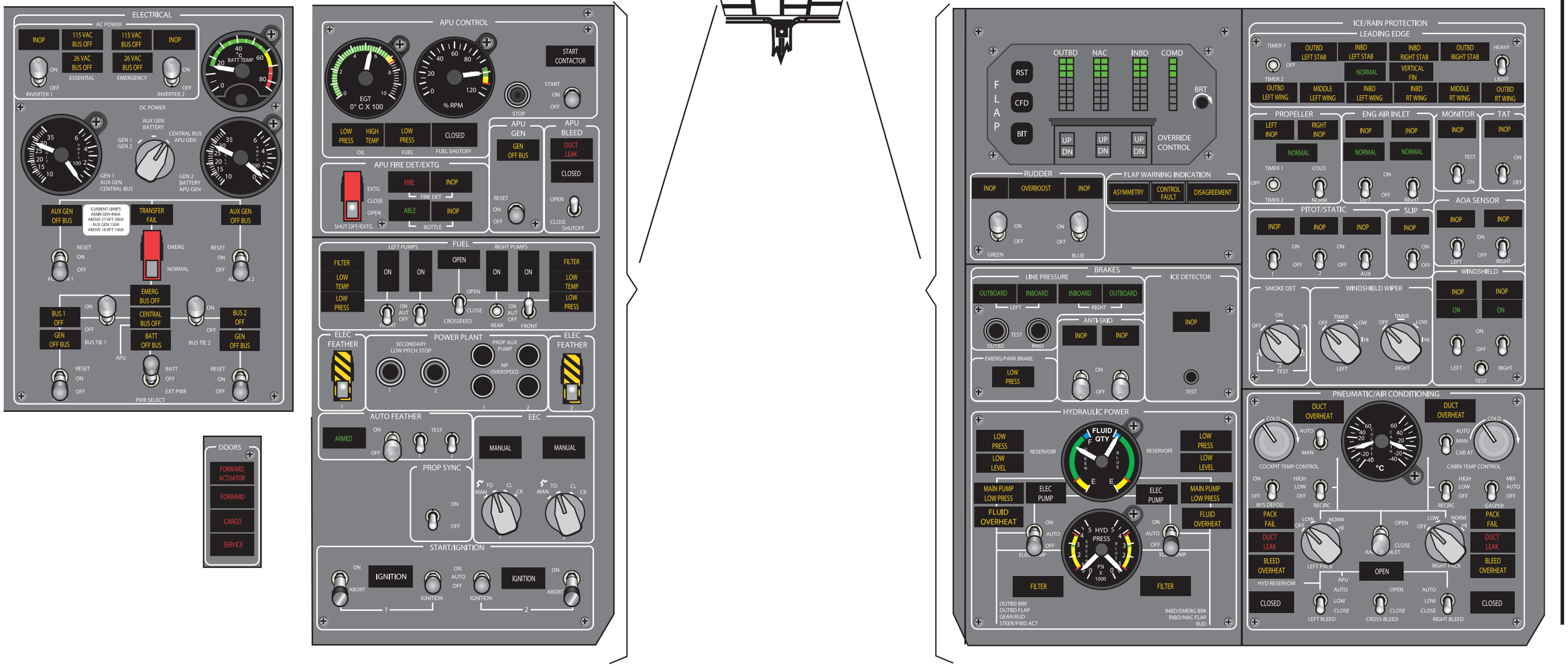


Figure ANN-3. Annunciators—Flight Instrument Panel (Cont.)