FlightSafety international

EMB-120 Brasilia PILOT TRAINING MANUAL

VOLUME 2
AIRCRAFT SYSTEMS

FlightSafety International, Inc.
Marine Air Terminal, LaGuardia Airport
Flushing, New York 11371
(718) 565-4100
www.flightsafety.com

Courses for the EMB-120 Brasilia are taught at the following FlightSafety Learning Centers:

Long Beach Learning Center
Long Beach Municipal Airport
4330 Donald Douglas Drive
Long Beach, California 90808
Phone: (562) 938-0100
Toll-Free: (800) 487-7670
Fax: (562) 938-0110

Atlanta Learning Center
1010 Toffie Terrace
Atlanta, Georgia 30354
Phone: (678) 365-2700
Fax: (678) 365-2699

Paris Learning Center
BP 25, Zone d Aviation d Affaires
Bldg. 404 Aeroport duBourget
93352 Le Bourget, CEDEX
France
Phone: (+33) (1) 49-92-1919
Fax: (33) (1) 49-92-1892

NOTICE

The material contained in this training manual is based on information obtained from the aircraft manufacturer's Pilot Manuals and Maintenance Manuals. It is to be used for familiarization and training purposes only.

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

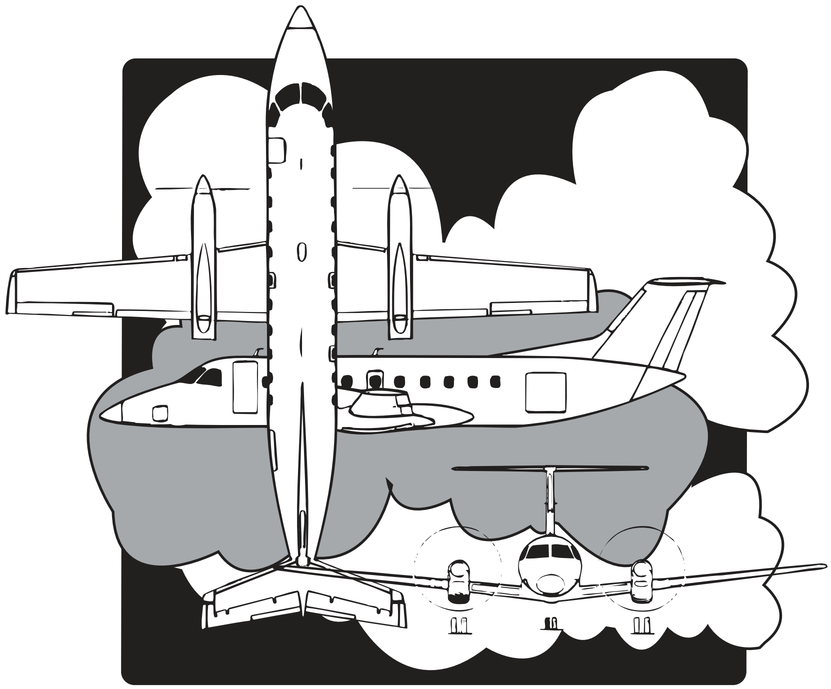
CONTENTS

CHAPTER 1
AIRCRAFT GENERAL

CONTENTS

ILLUSTRATIONS

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.

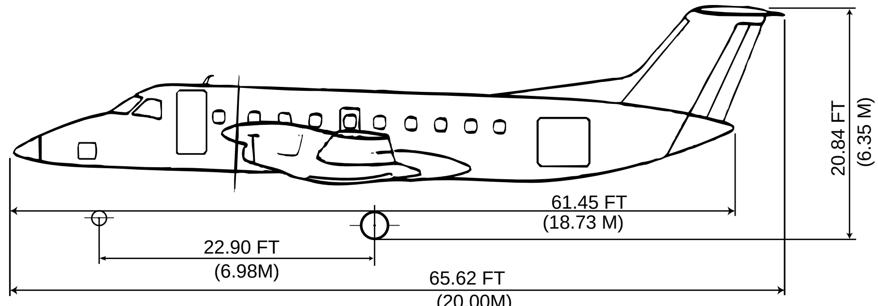
STRUCTURES

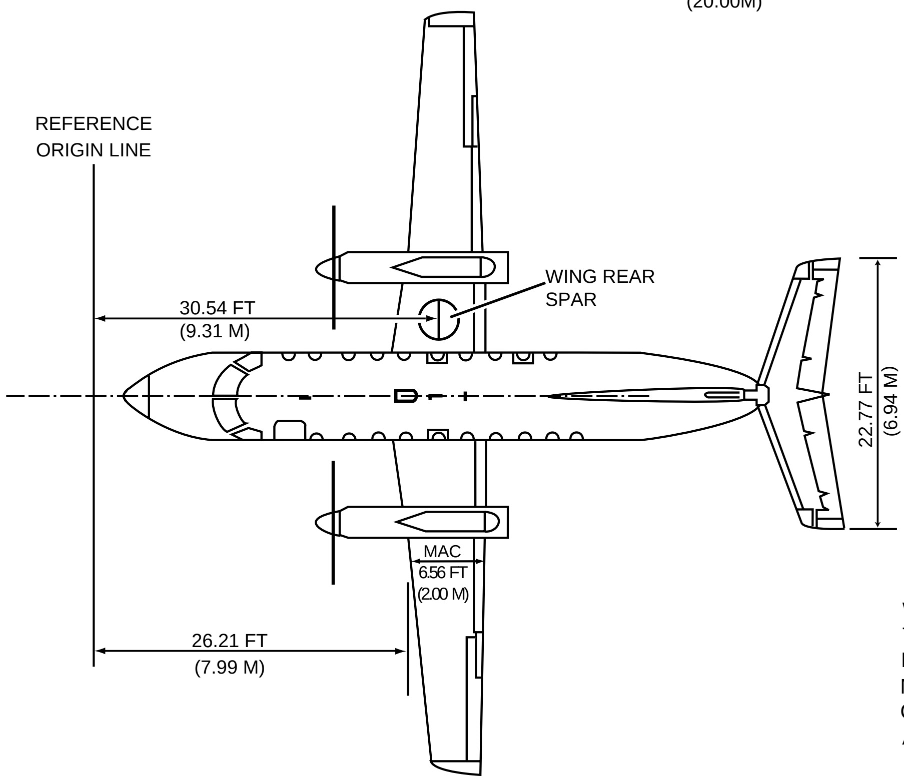
GENERAL

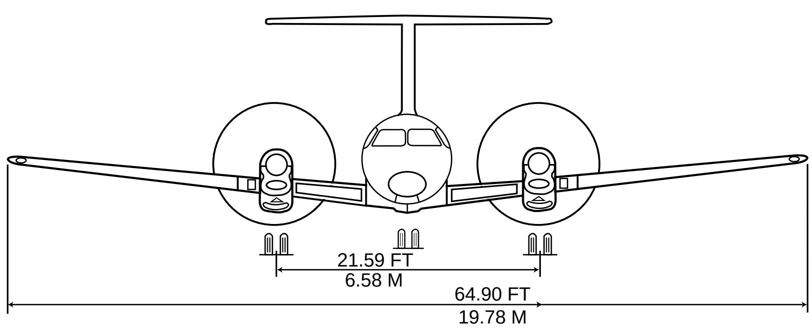
The EMB-120 Brasilia (Figure 1-1) is an all-metal construction, pressurized, low-wing T-tail, monoplane. Two Pratt and Whitney PW118 engines are mounted on the wing, supported by a semimonocoque/tubular nacelle structure. The landing gear is the retractable, twin-wheel type.

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 RATIO............................ 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

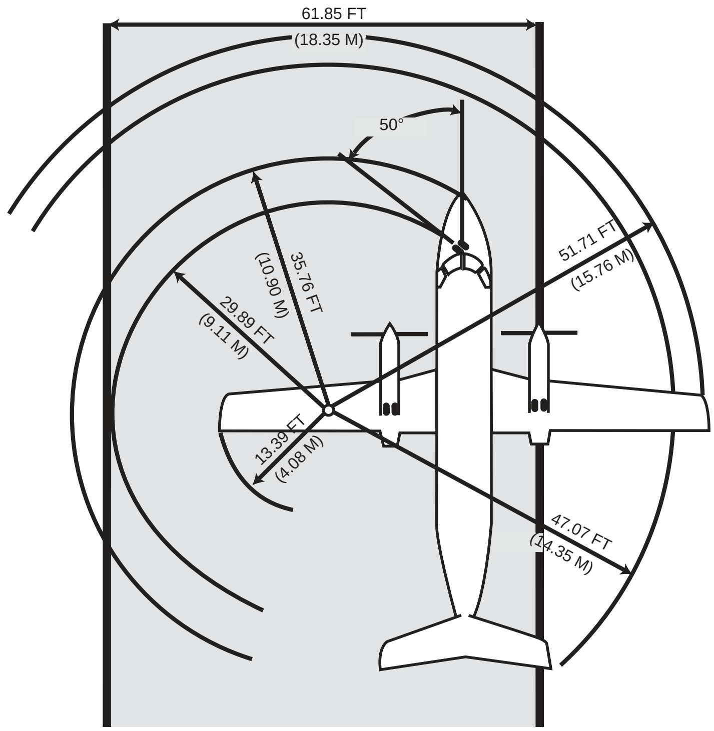


Figure 1-3. Turning Distance

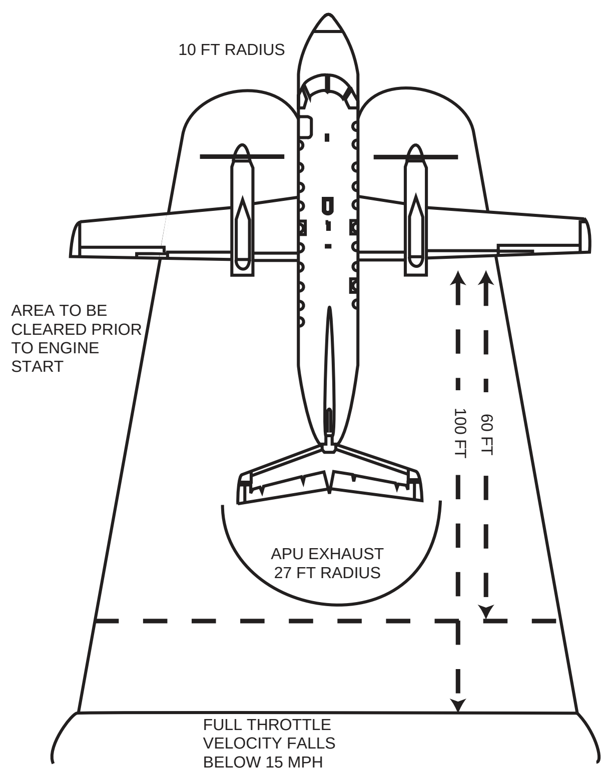


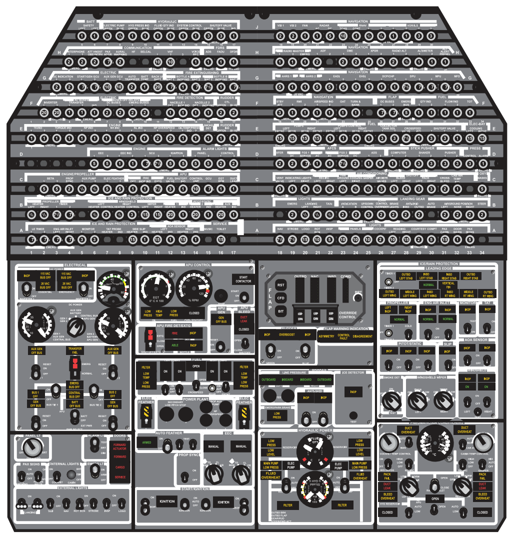
Figure 1-4. Danger Zones

COCKPIT

General

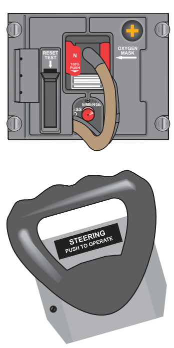
The general layout of the cockpit is shown in Figure 1-5. The overhead panel is shown in more detail in Figure 1-6.

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

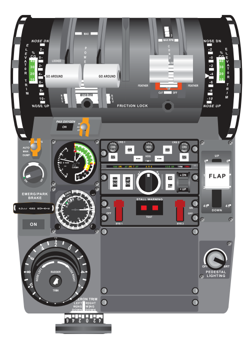


OVERHEAD PANEL

FORWARD PANEL



PILOT'S CONSOLE



CENTER CONSOLE

Figure 1-5. Cockpit Layout

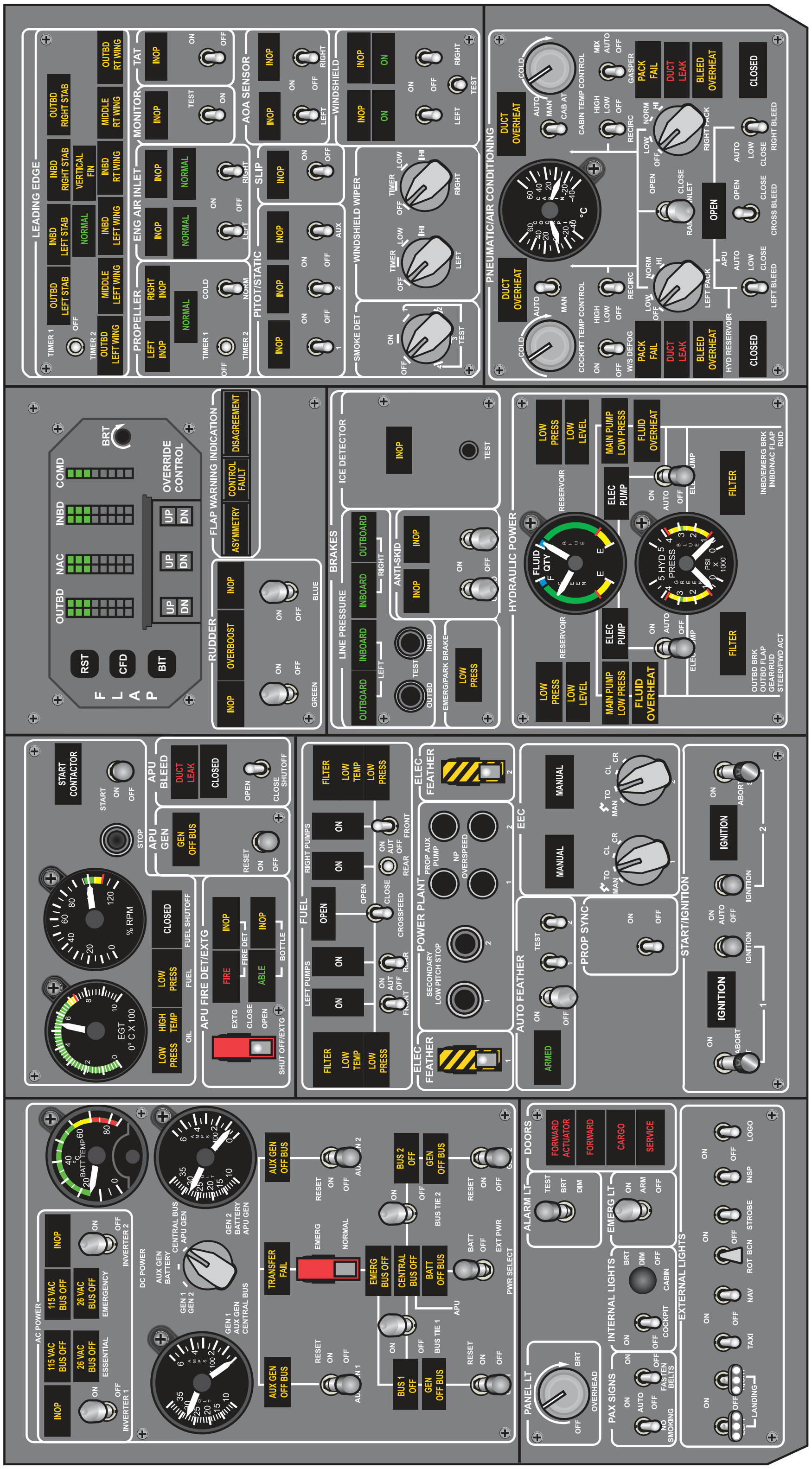


Figure 1-6. Overhead Panel

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

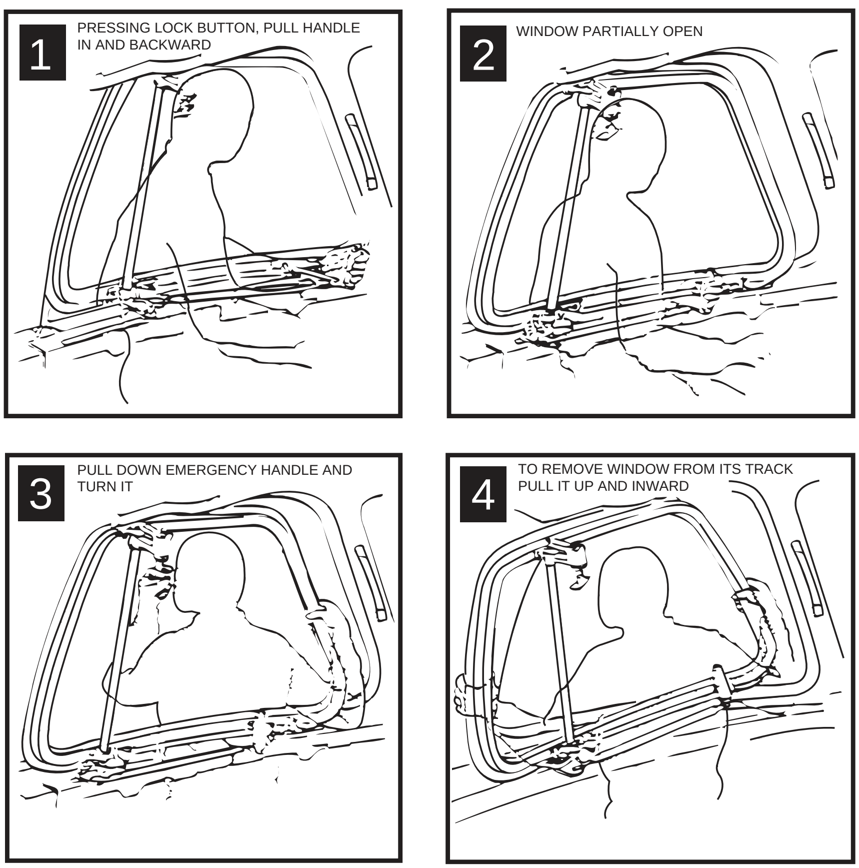


Figure 1-8. Normal and Emergency Window Operation

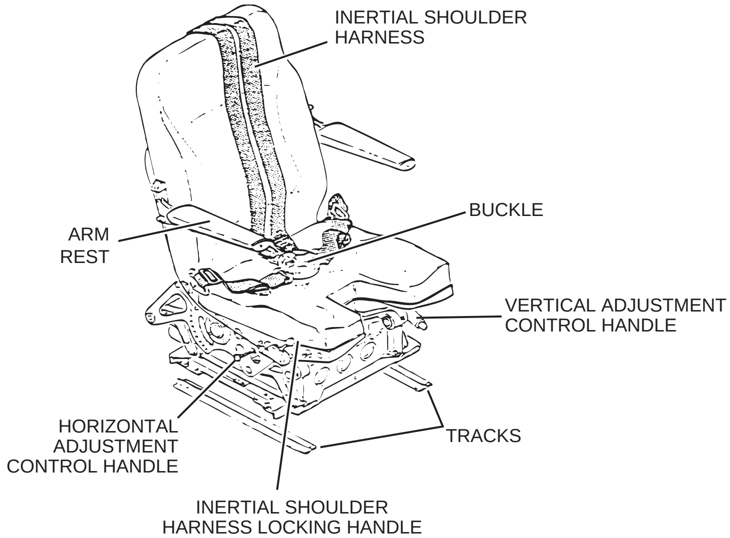


Figure 1-9. Pilot Seats

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.

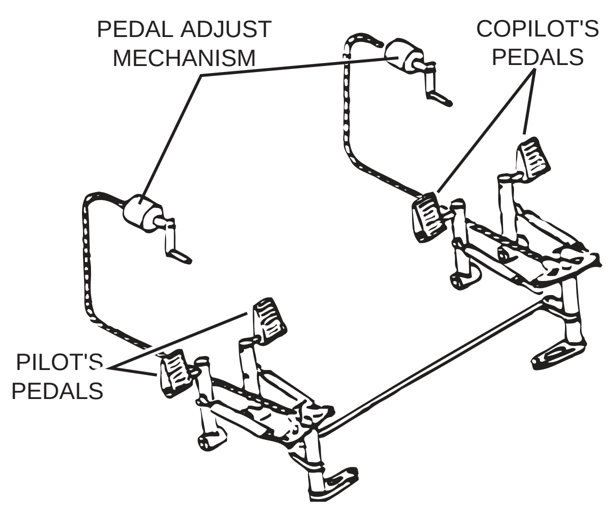


Figure 1-11. Pedal Adjust Mechanism

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.

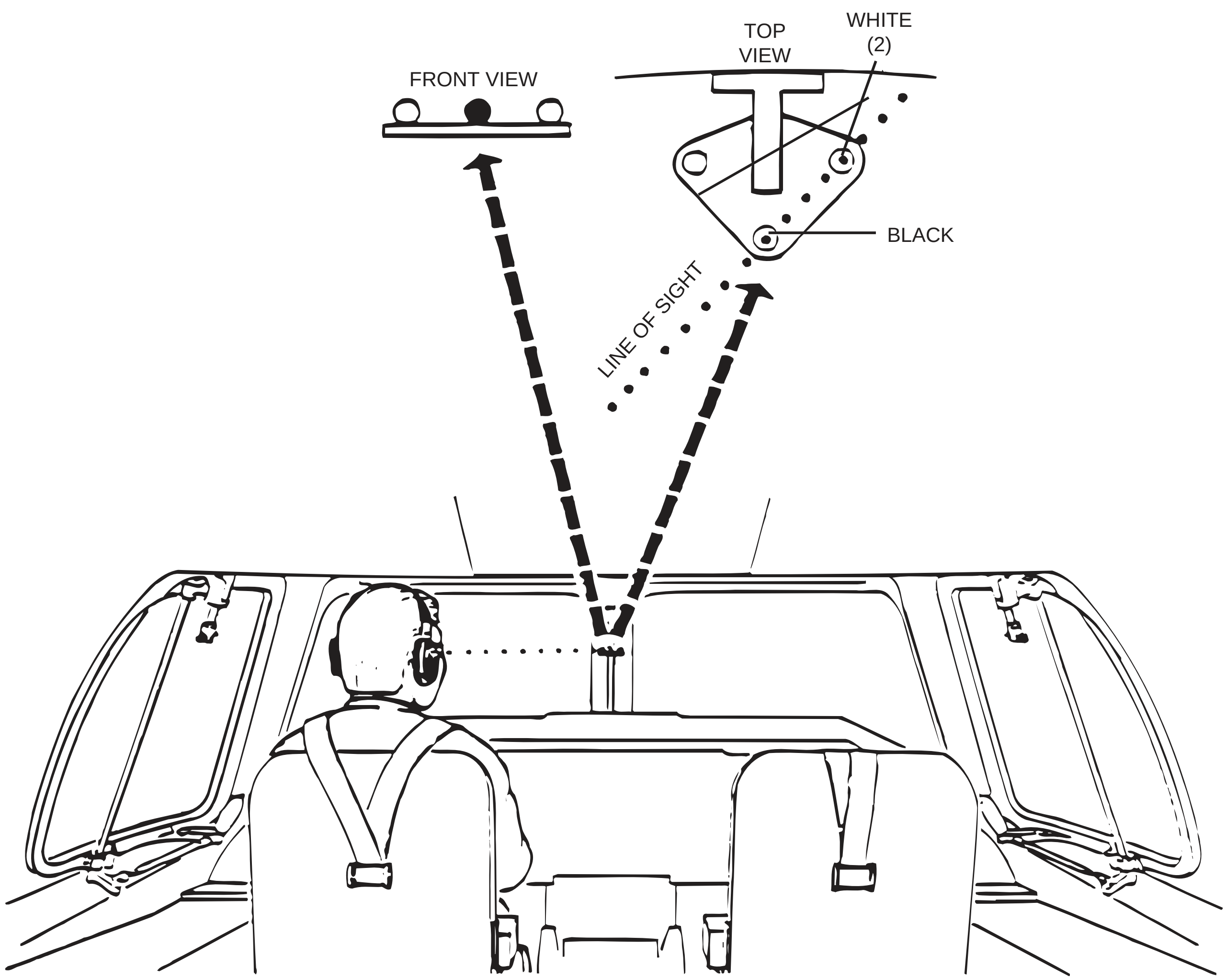
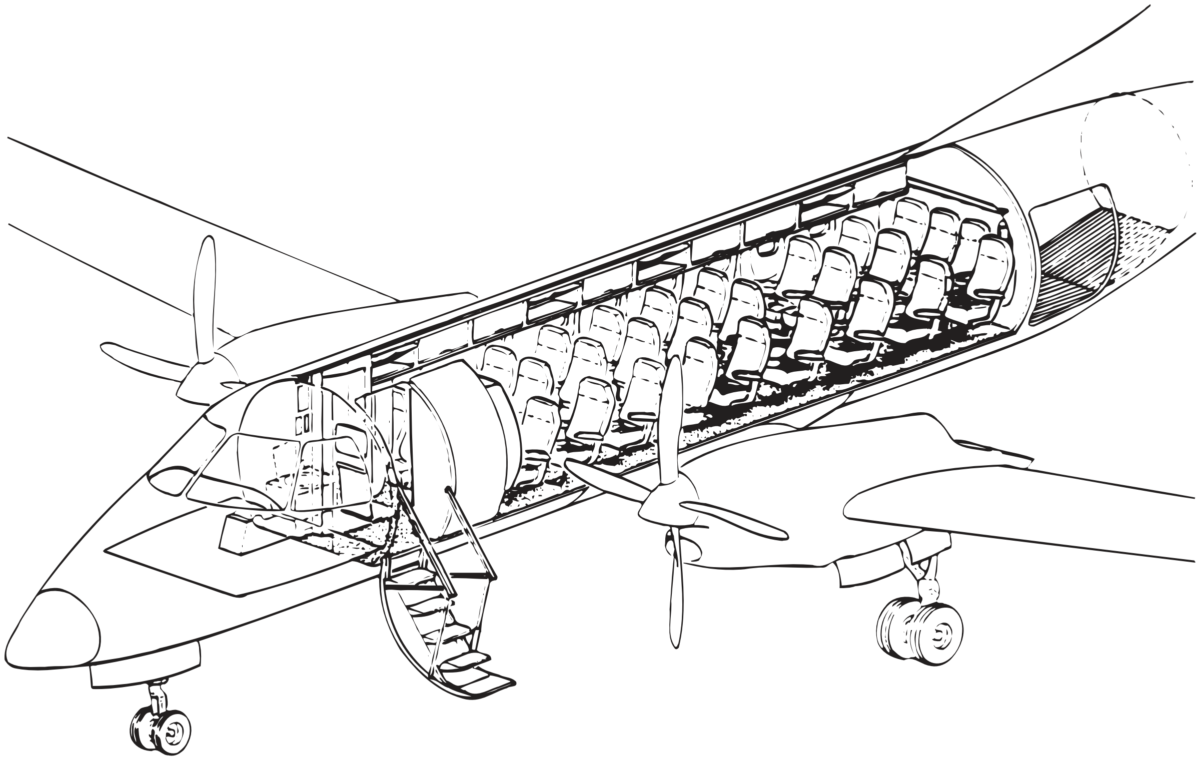


Figure 1-12. Pilot Seat Adjustment



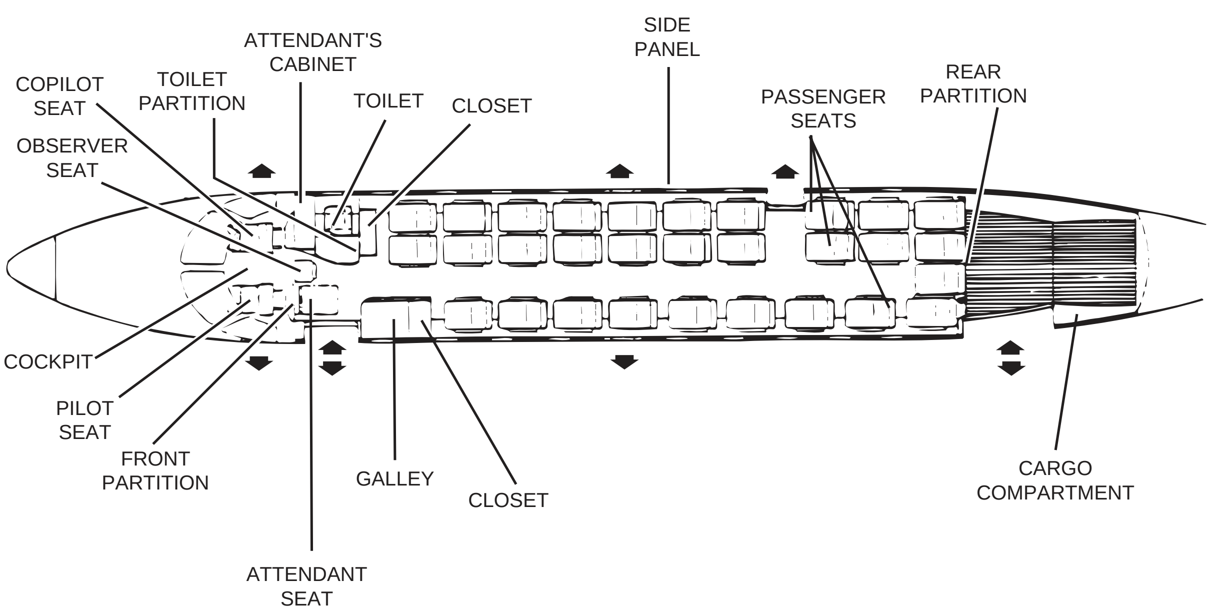


Figure 1-13. Passenger Configuration/Interior Layout (Typical)

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.

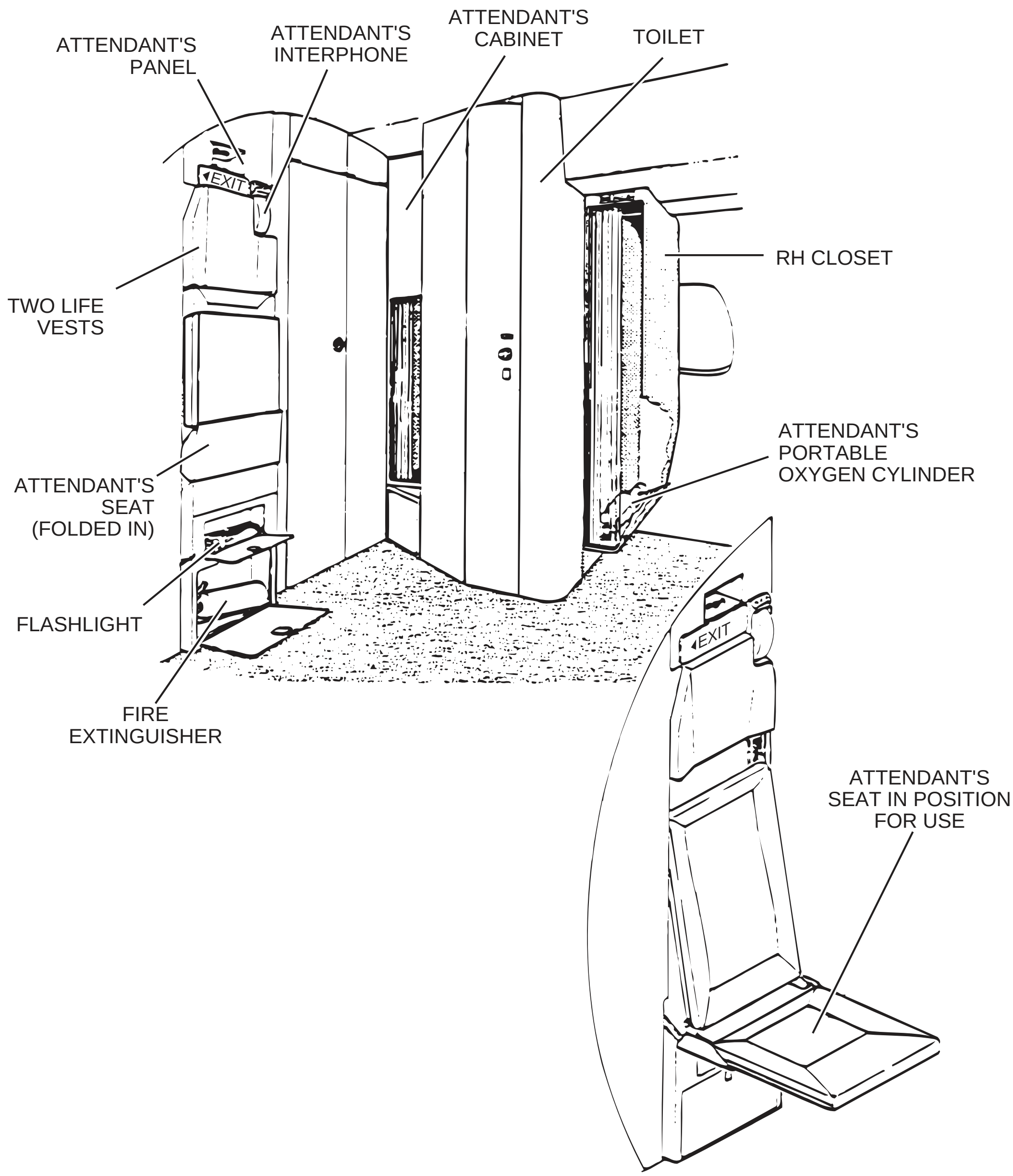


Figure 1-14. Attendant's Station (Typical)

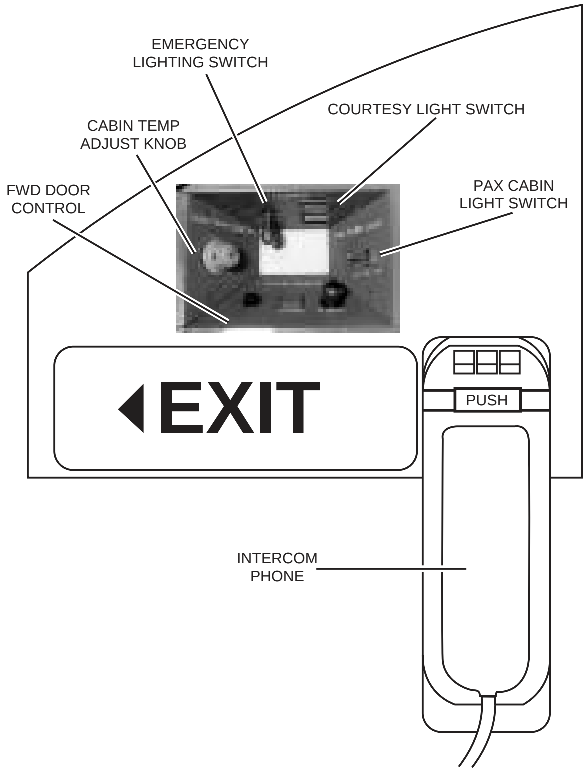


Figure 1-15. Attendant's Panel

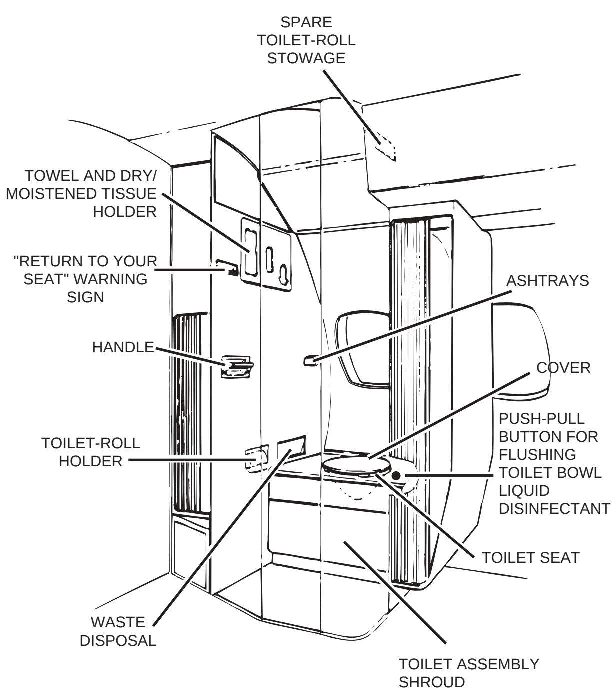


Figure 1-16. Toilet

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.

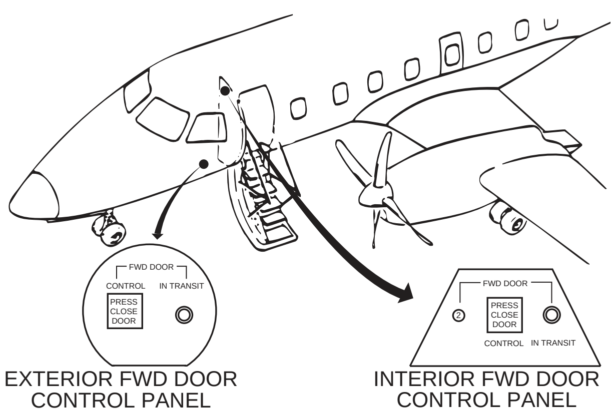


Figure 1-18. Forward Door Controls

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.

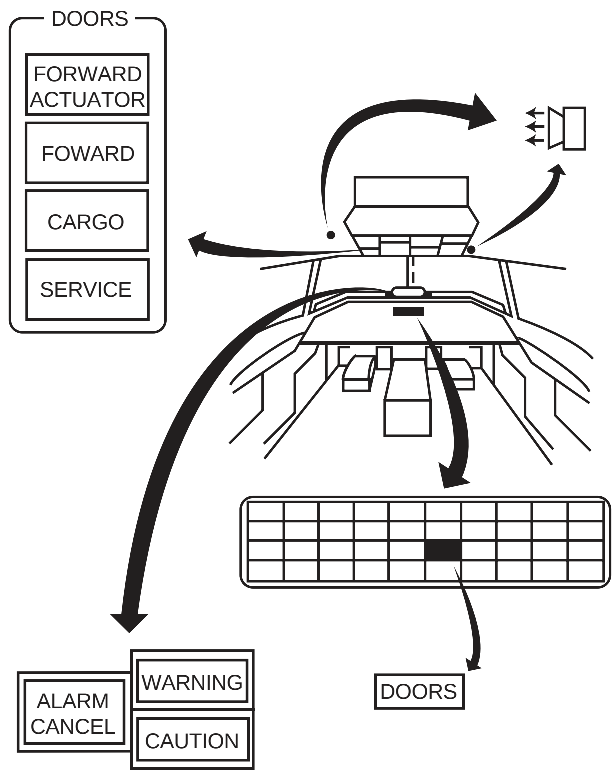


Figure 1-19. Door Warning Lights

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.

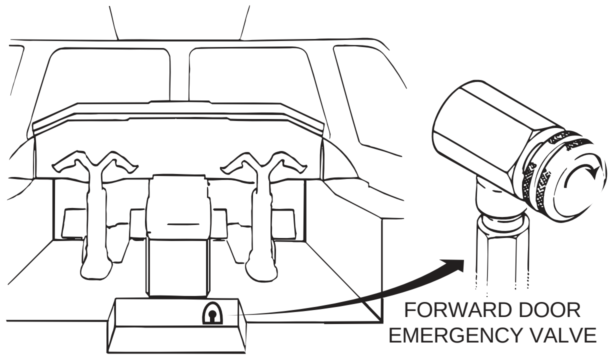


Figure 1-20. Forward Door Emergency Valve

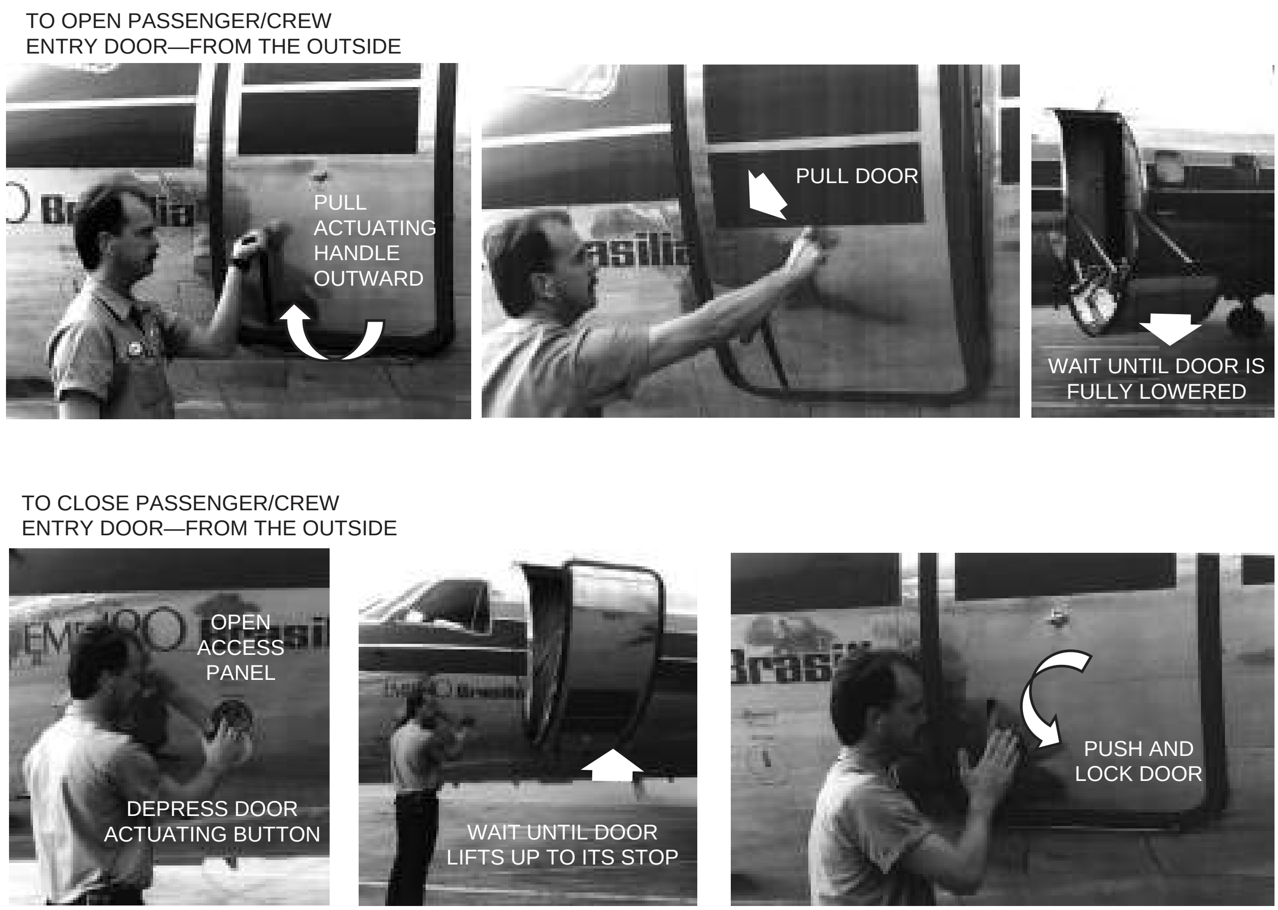


Figure 1-21. Forward Entry Door Operation

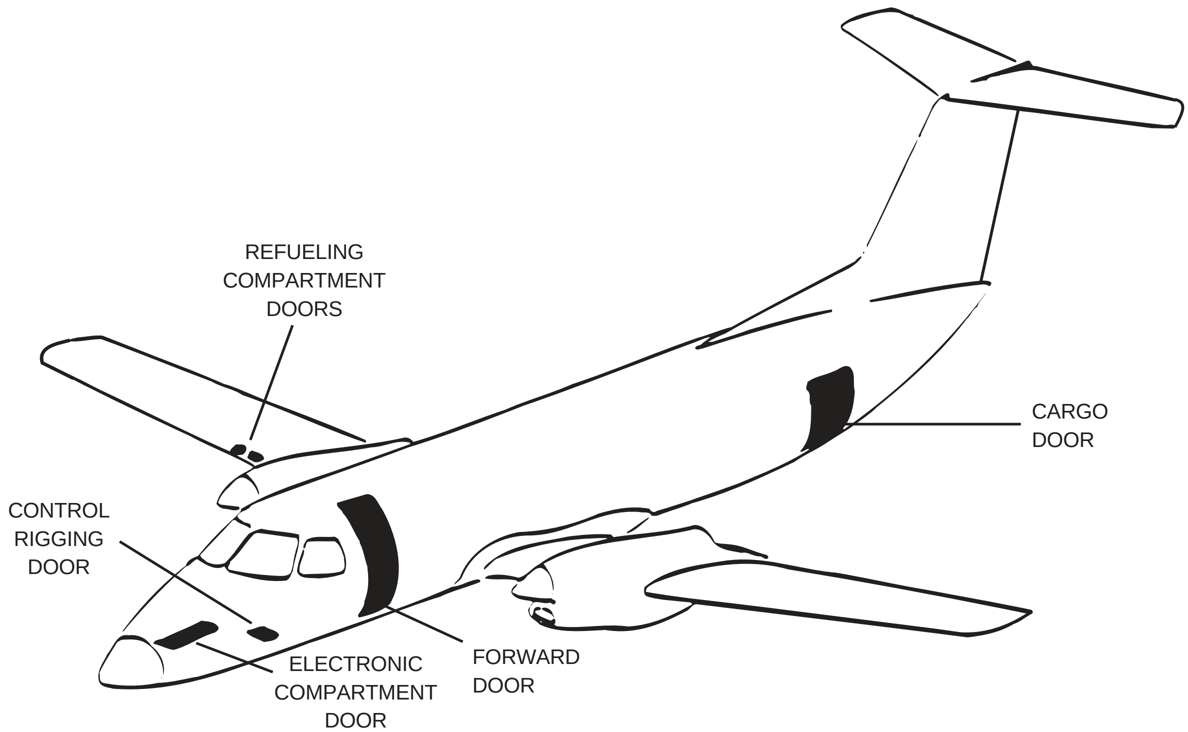


Figure 1-22. Cargo/Service Door Location

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.

TO OPEN:

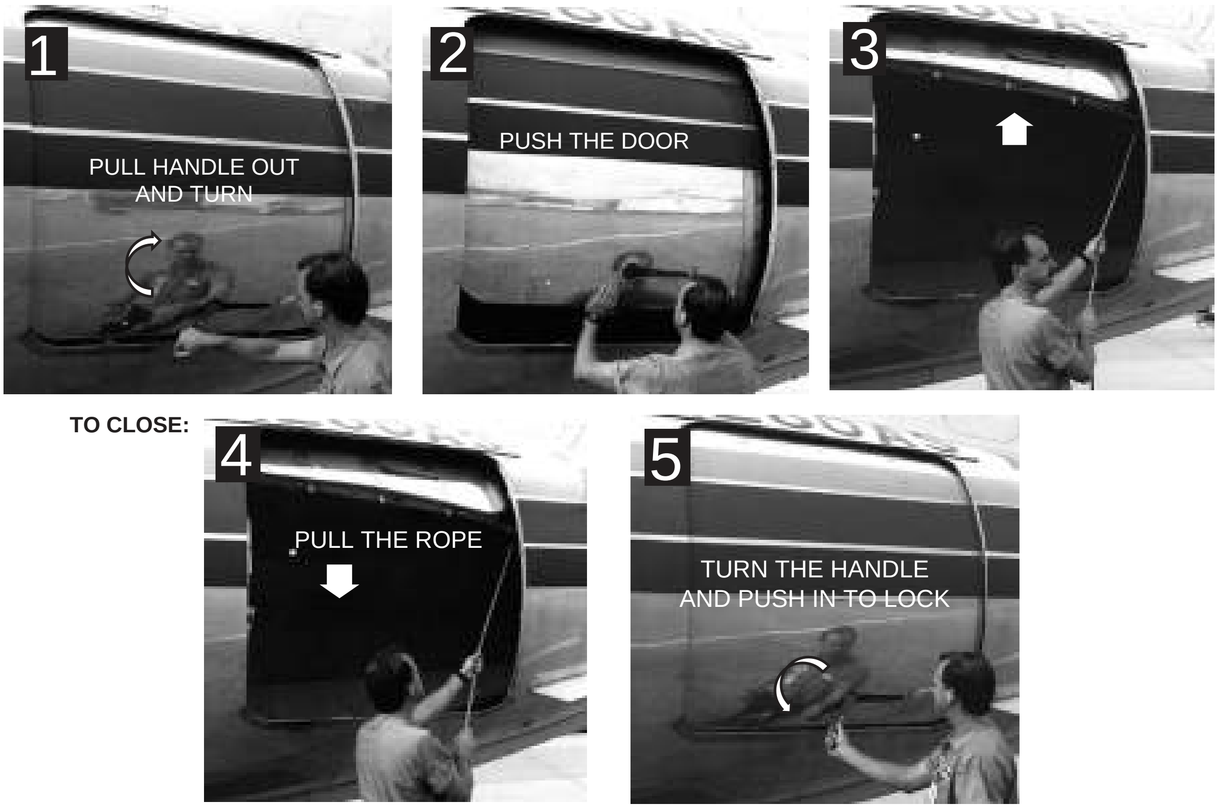


Figure 1-23. Cargo Door Operation

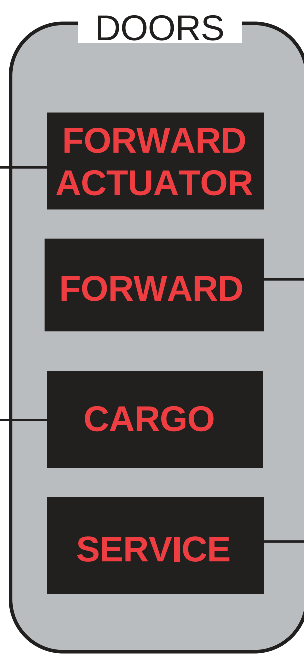
FORWARD ACTUATOR LIGHT (RED)

ILLUMINATED - WHEN THE FORWARD DOOR ACTUATORS HYDRAULIC LINE REMAINS PRESSURIZED AFTER DOOR CLOSING. FOWARD ENTRY DOOR IS HYDRAULICALLY BLOCKED.

CARGO DOOR LIGHT (RED)

ILLUMINATED - WHEN THE CARGO DOOR IS NOT CLOSED AND LATCHED

NOTE:
FOR AIRPLANES POST-MOD. SB 120-031-0008 OR SN 120.046, 120.050, AND SUBSEQUENT, THE SERVICE LIGHT IS ILLUMINATED WHENEVER ANY REFUELING/DEFUELING SYSTEM DOOR IS OPEN.



FORWARD DOOR LIGHT (RED)

ILLUMINATED - WHEN THE FORWARD DOOR IS NOT CLOSED AND LATCHED.

SERVICE DOOR LIGHT (RED)

ILLUMINATED - WHEN THE CONTROL RIGGING DOOR OR ELECTRONIC COMPARTMENT DOOR IS NOT CLOSED AND LATCHED.

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.

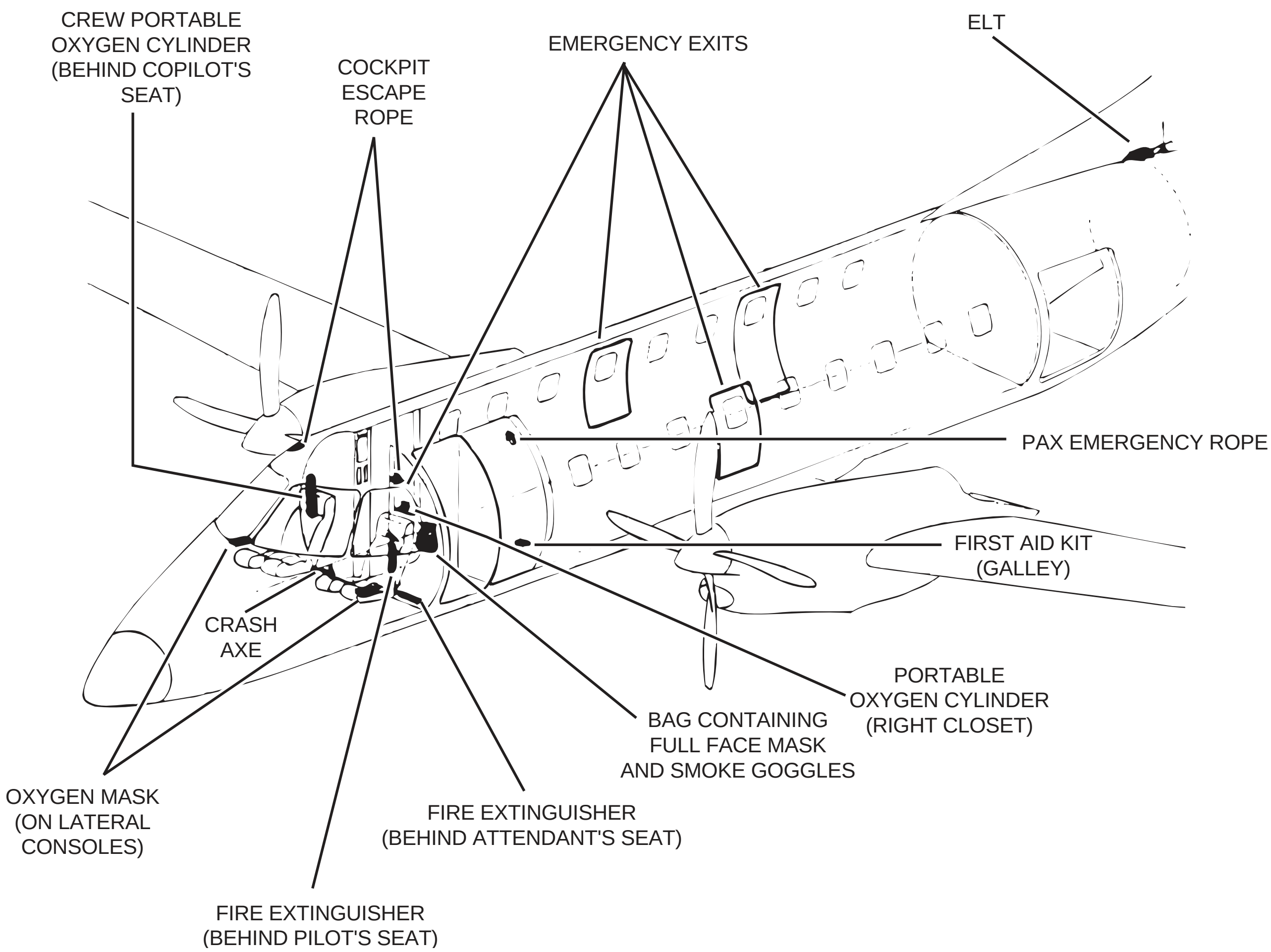


Figure 1-25. Standard Emergency Equipment Location

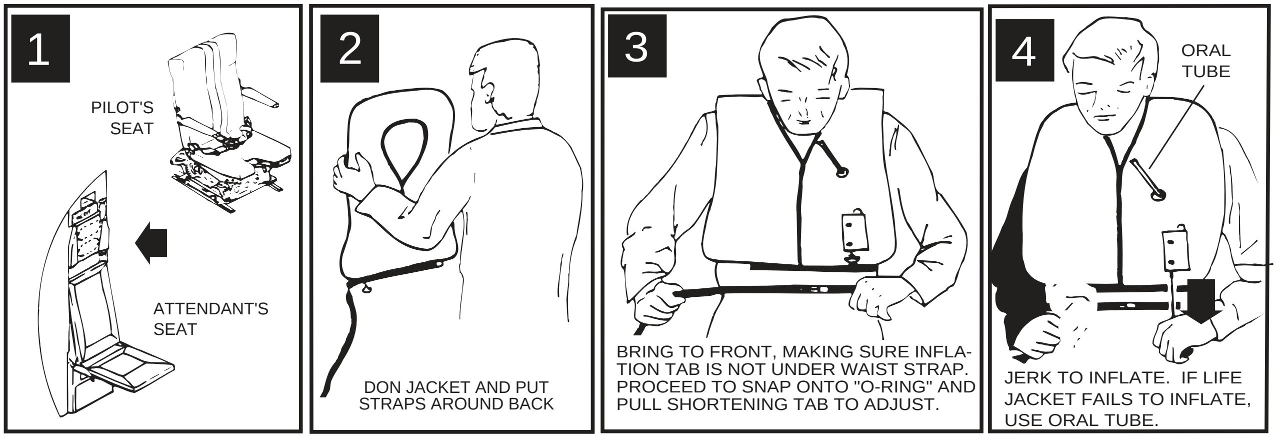


Figure 1-26. Life Vest Location and Operation

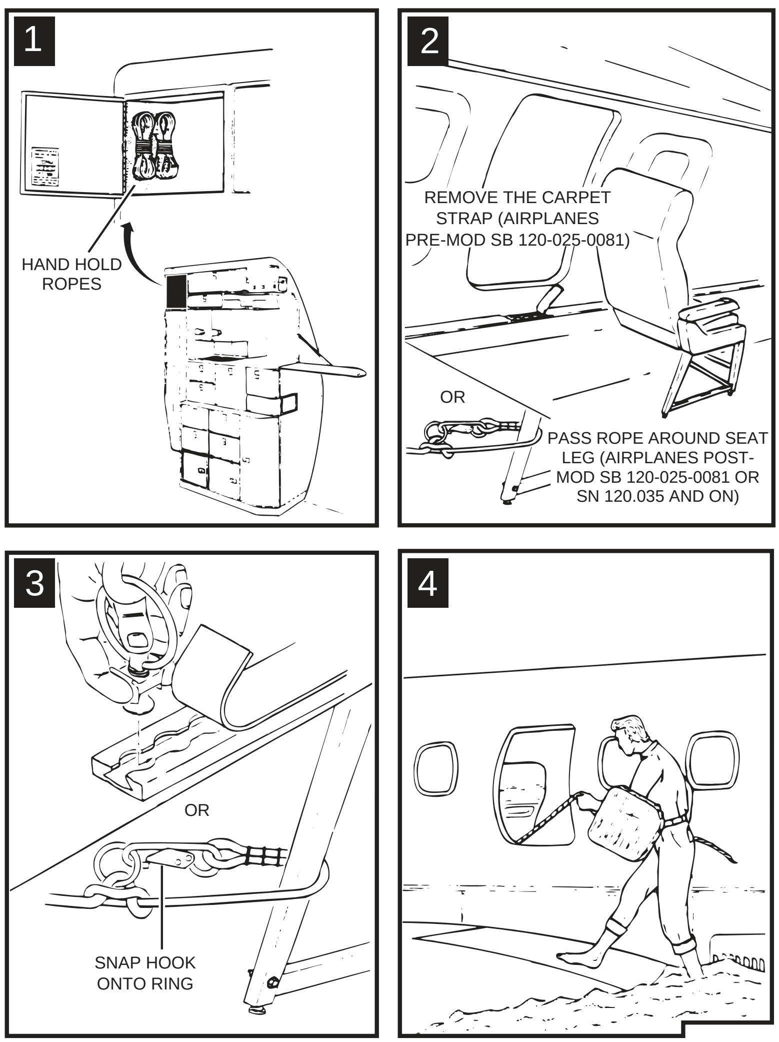


Figure 1-27. Hand Hold Rope Location and Use

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.

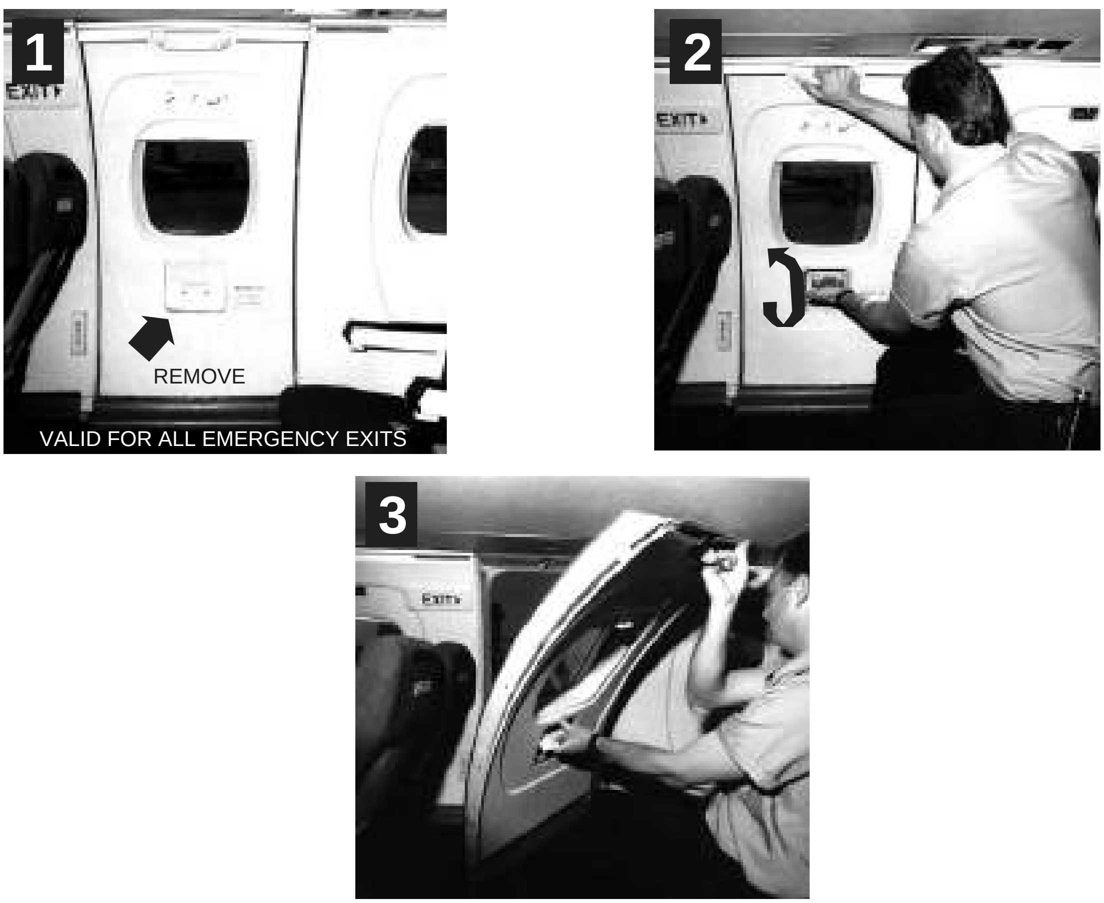


Figure 1-28. Emergency Exit Operation

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

ILLUSTRATIONS

TABLE

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.

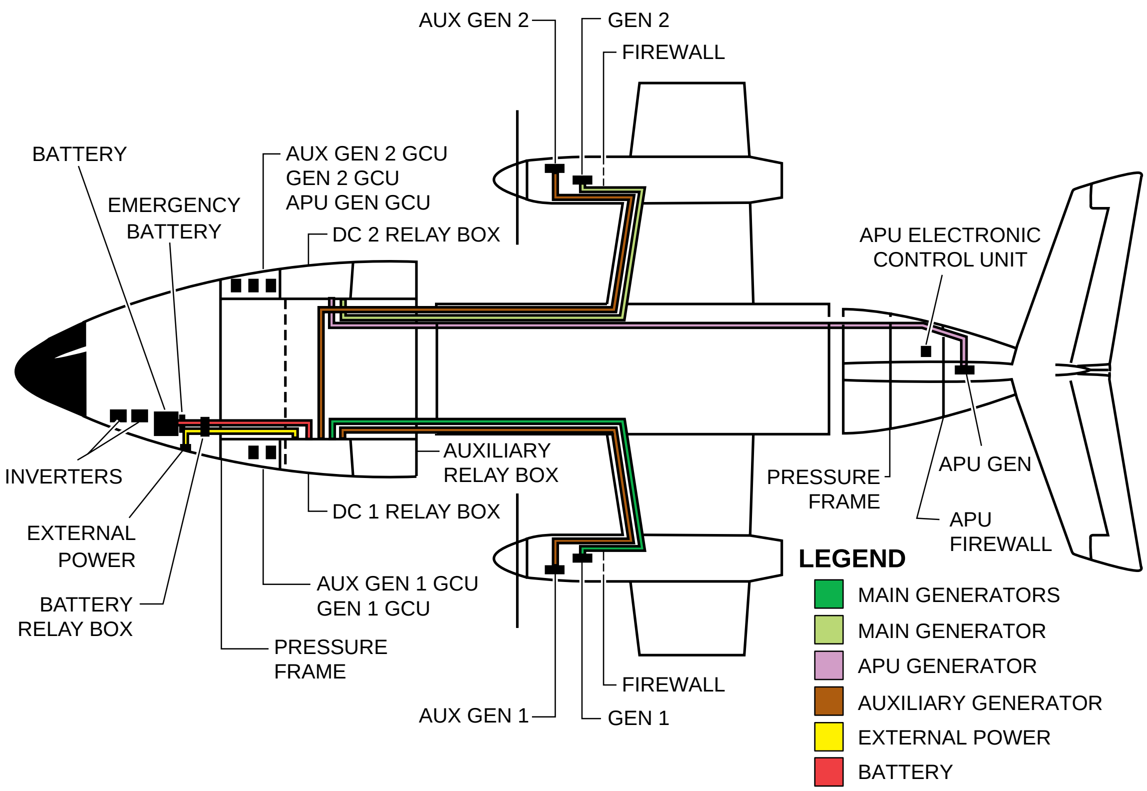


Figure 2-1. Electrical Component Location

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).

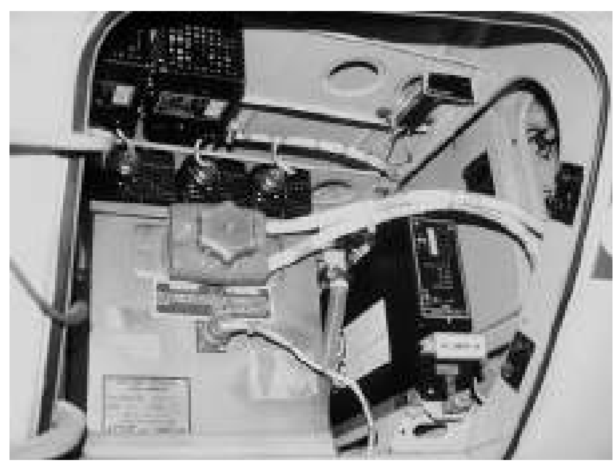


Figure 2-2. Battery Location

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.

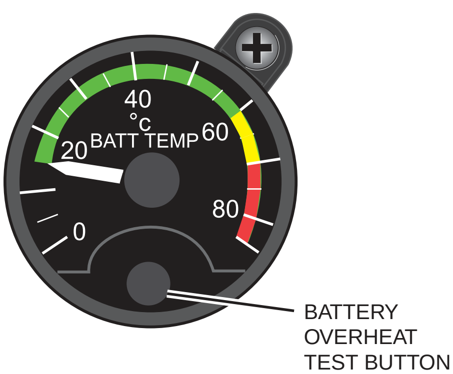


Figure 2-3. Battery Temperature Gage

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.

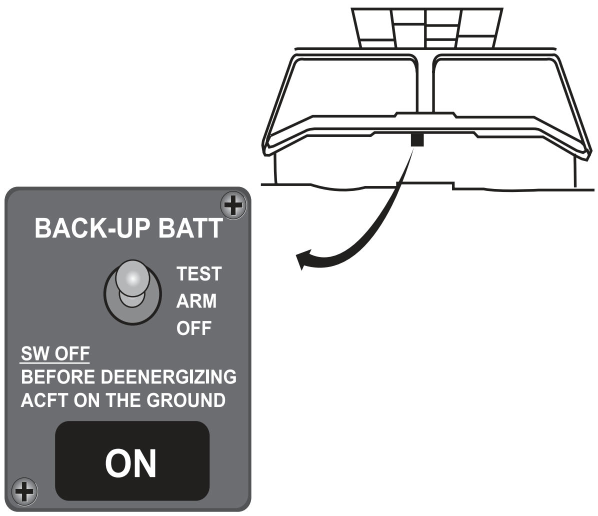


Figure 2-4. Backup Battery Switch

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.

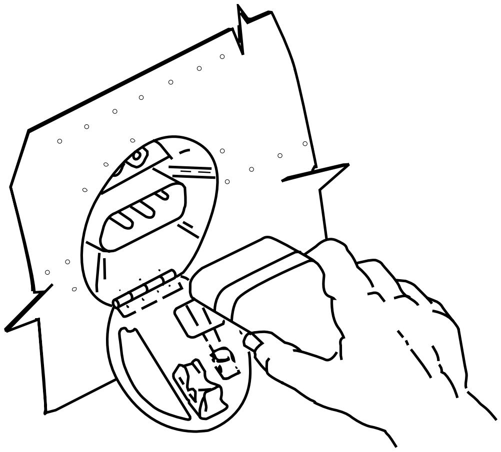


Figure 2-5. External Power Receptacle

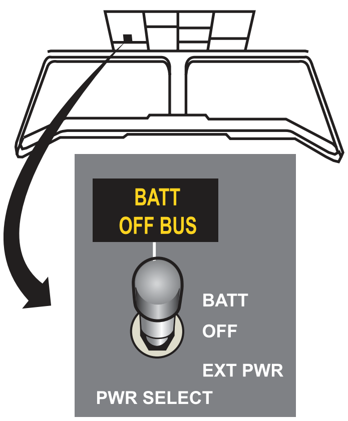


Figure 2-6. Power Select Switch

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

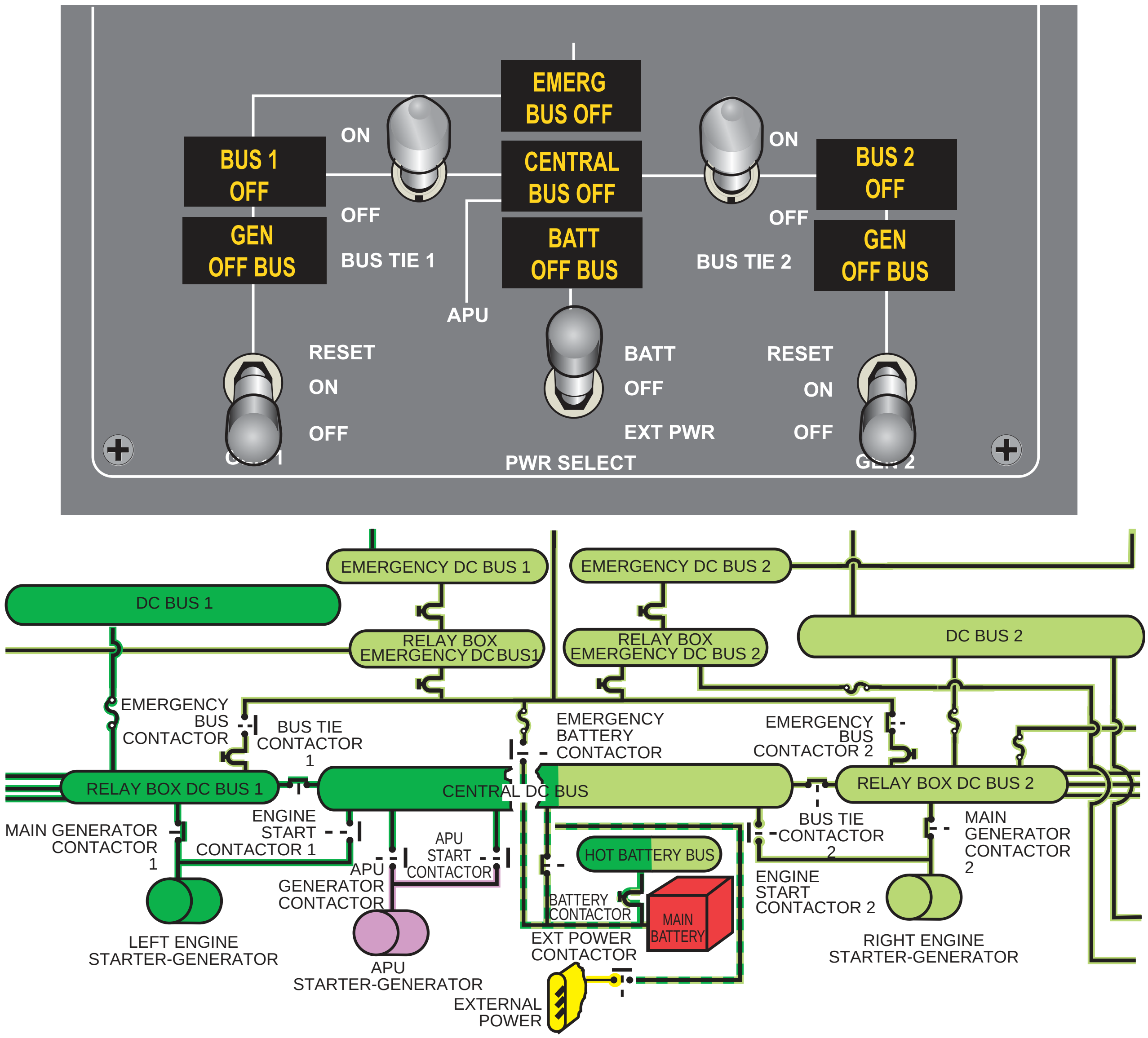


Figure 2-7. Central DC Power Distribution

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

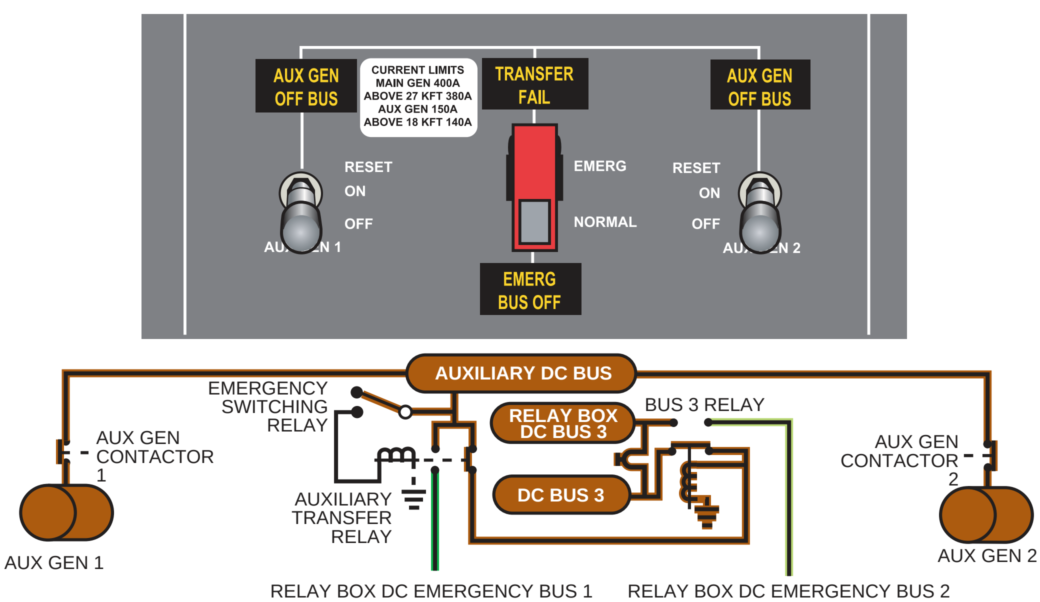


Figure 2-8. Auxiliary DC Power System

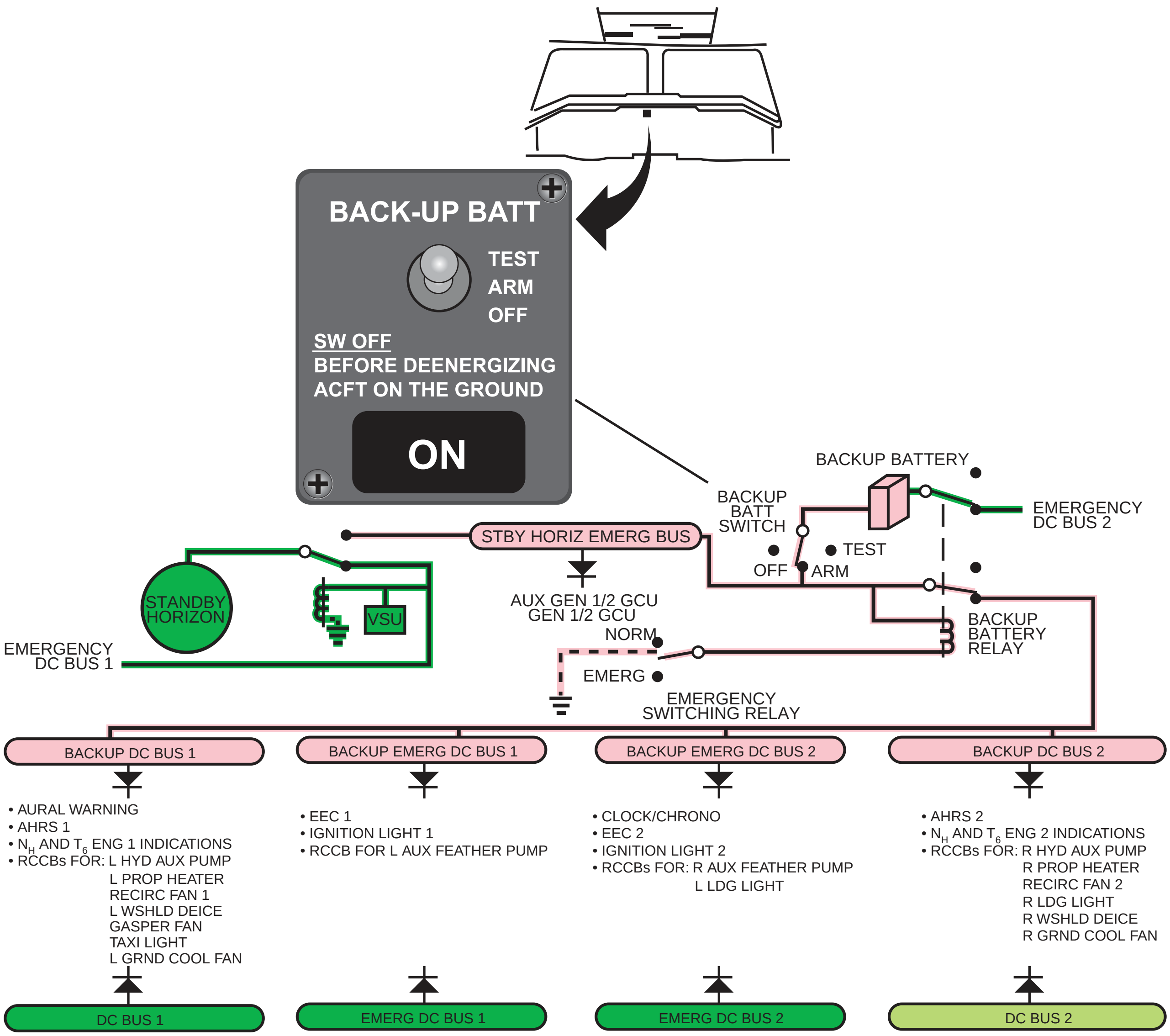


Figure 2-9. Backup DC Power System

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.

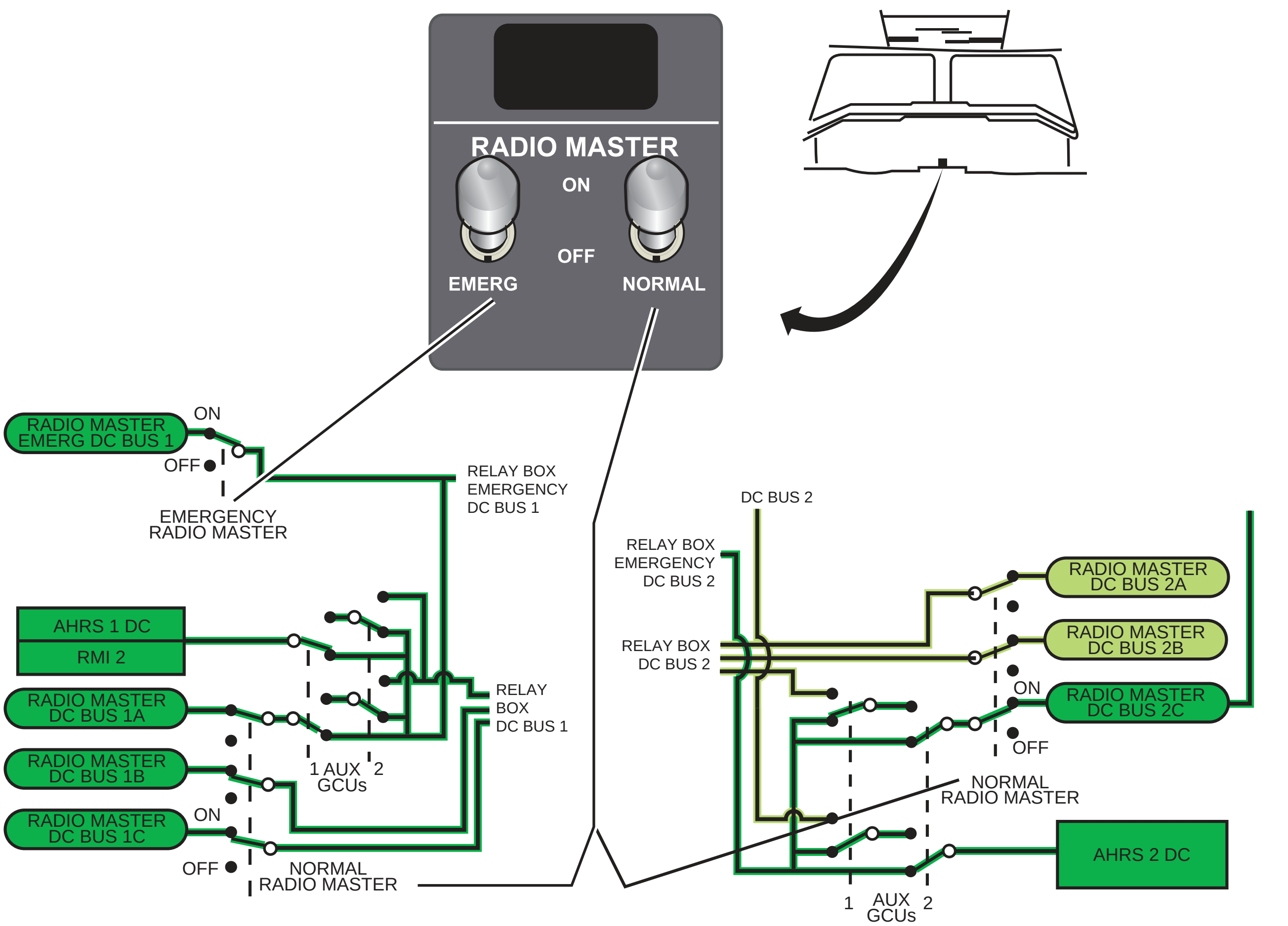


Figure 2-10. Radio Master System

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.

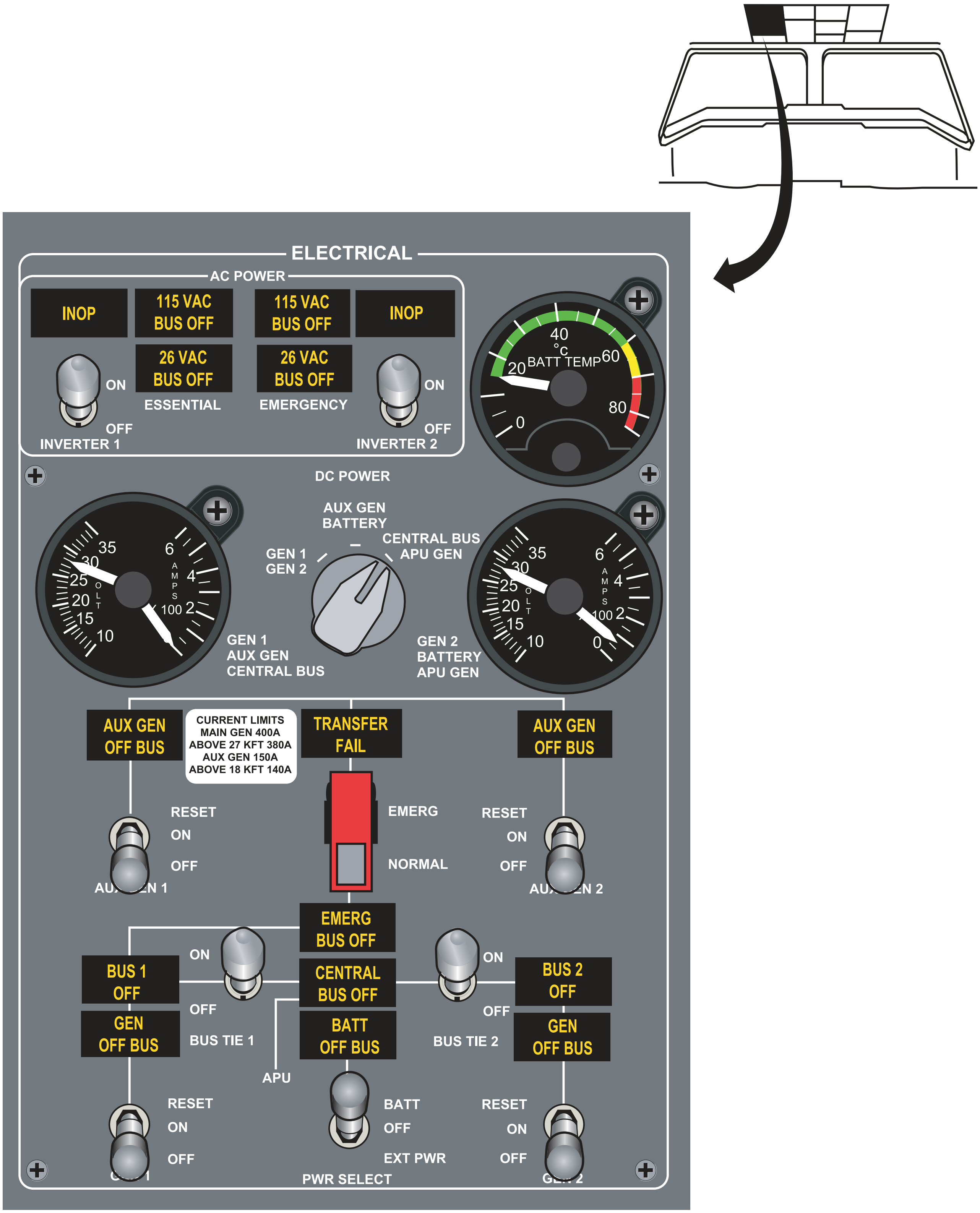


Figure 2-11. Electrical Control Panel—DC System

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.

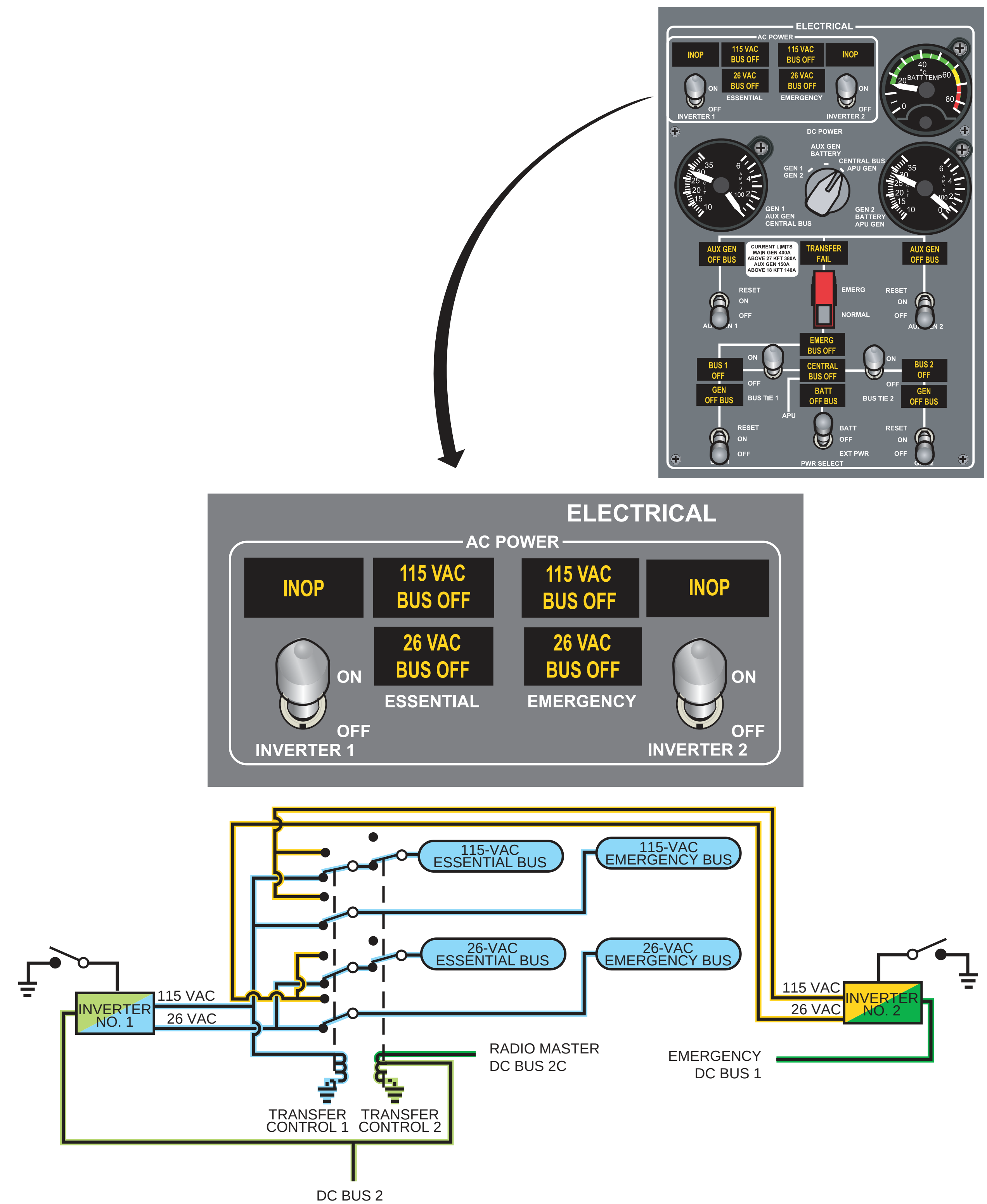


Figure 2-12. AC Power System

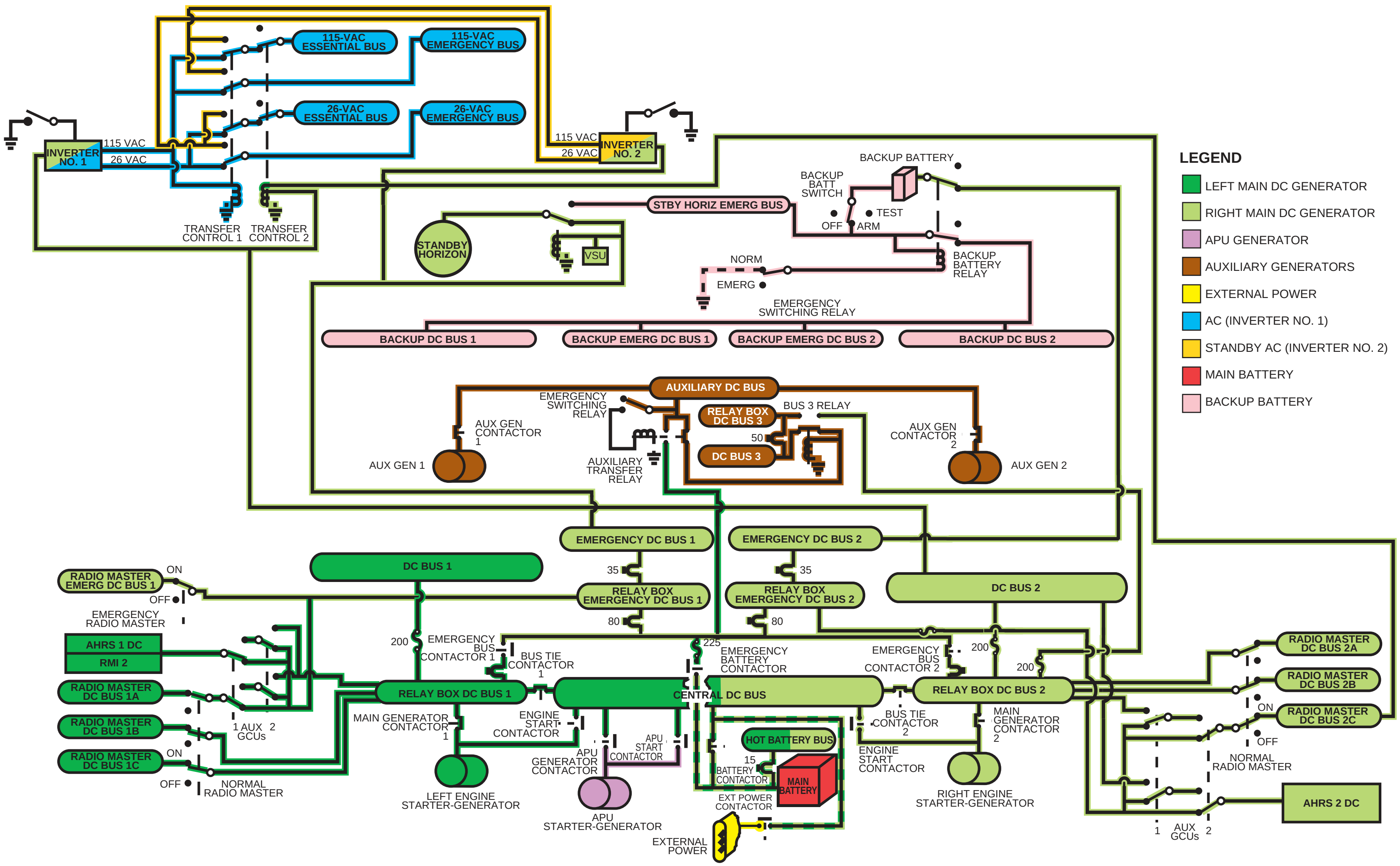


Figure 2-13. Electrical Power System

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%)

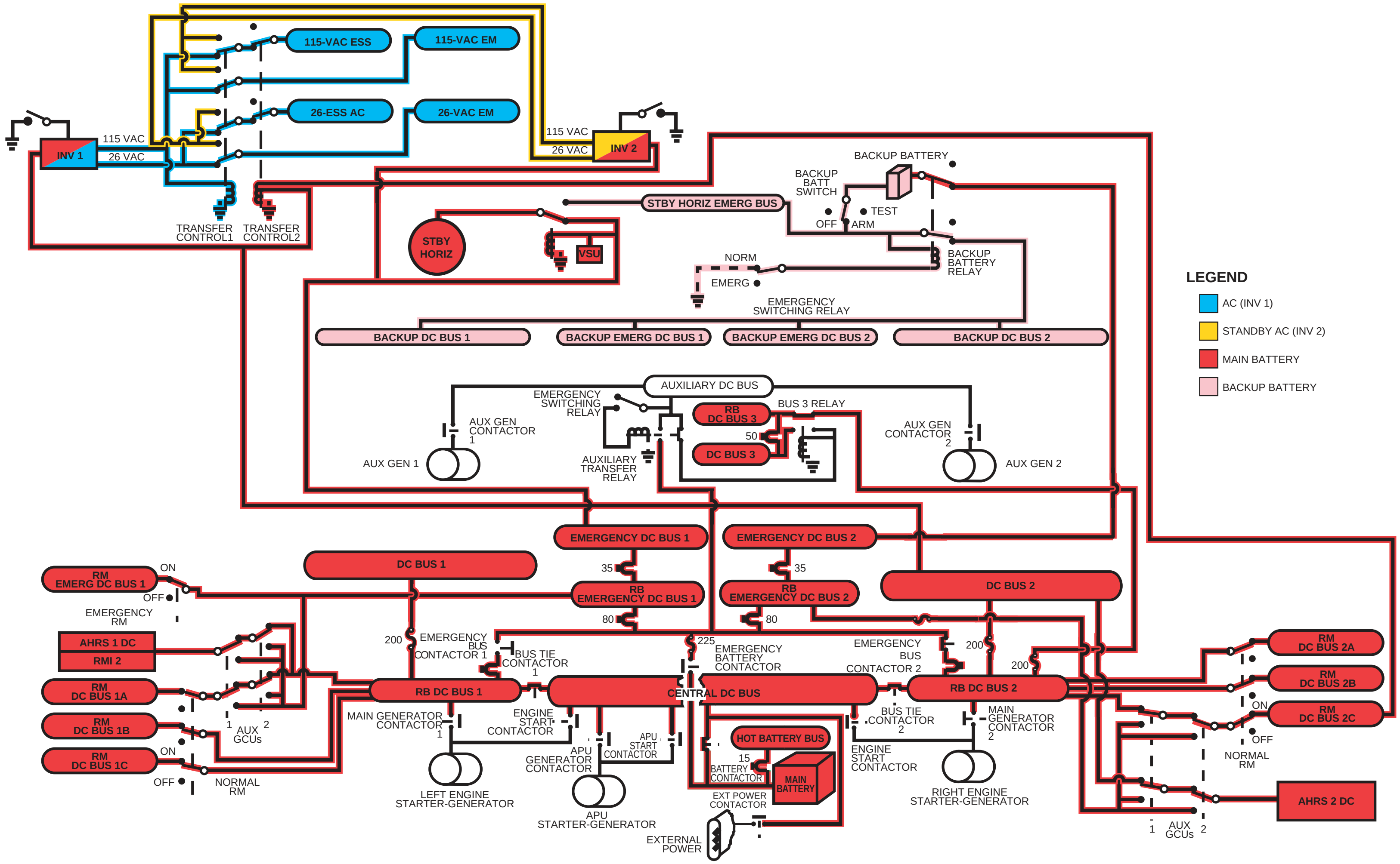


Figure 2-14. Electrical System Configuration—Battery Only

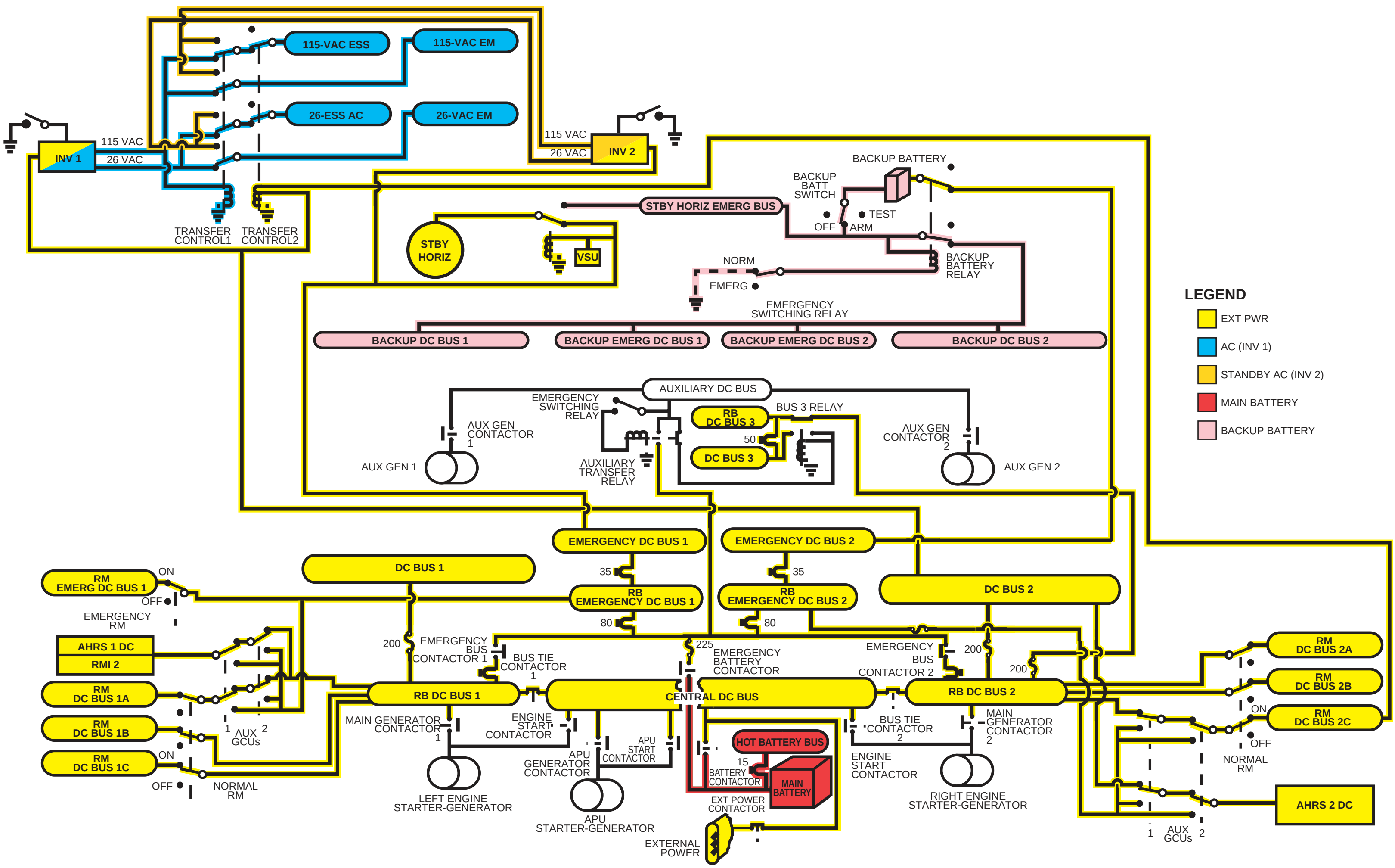


Figure 2-15. Electrical System Configuration—External Power

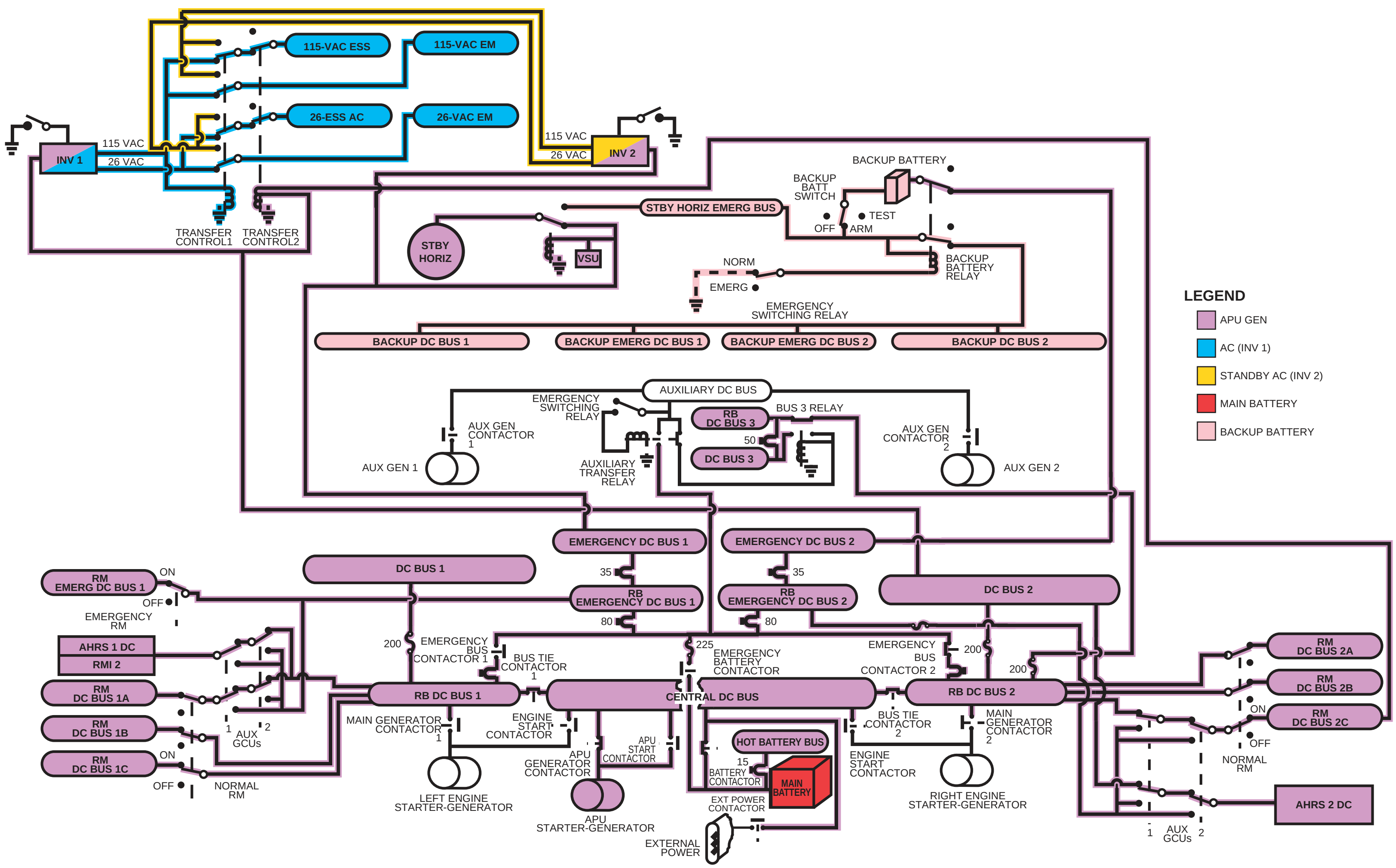


Figure 2-16. Electrical System Configuration—APU Generator

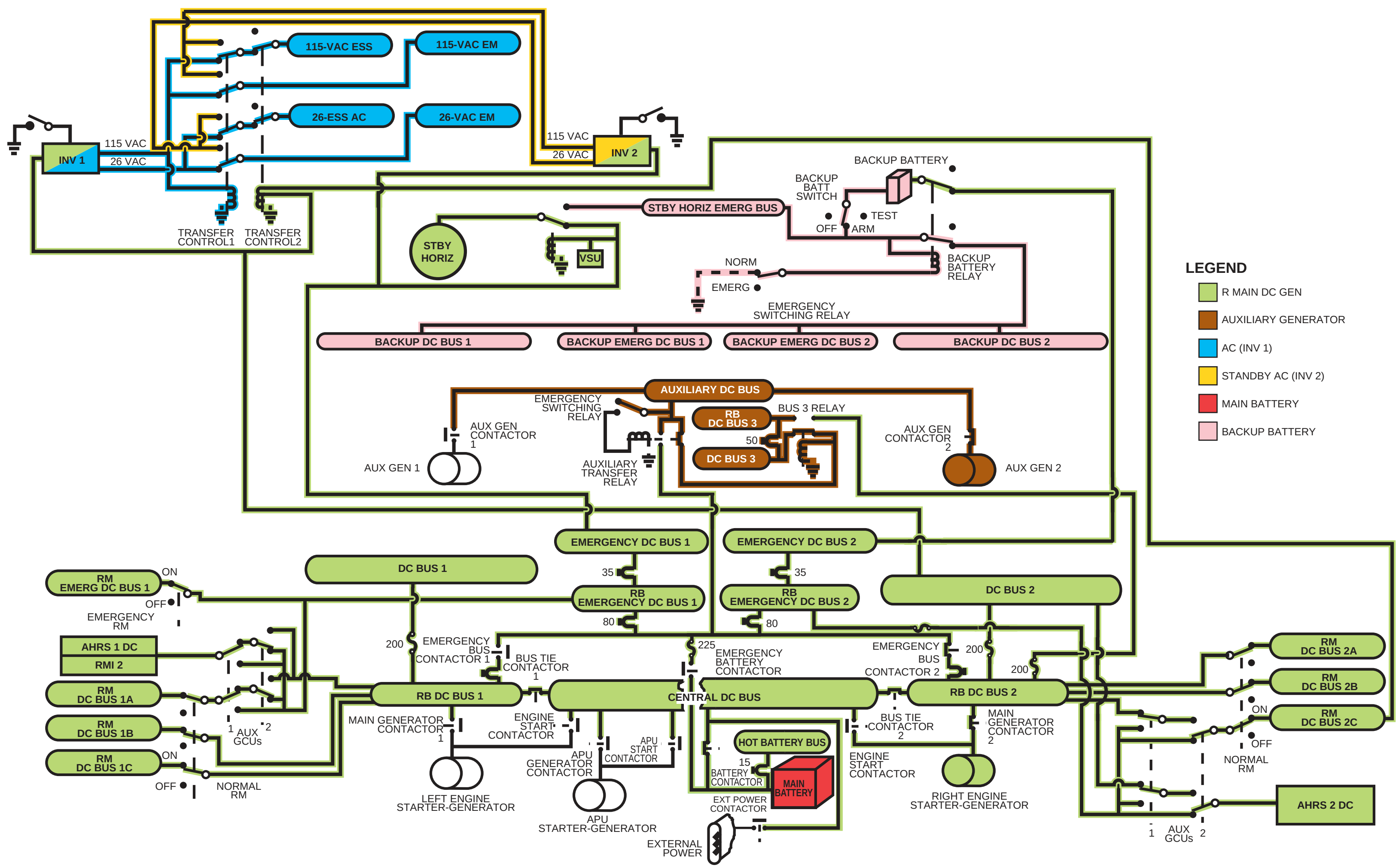


Figure 2-17. Electrical System Configuration—Single Engine

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.

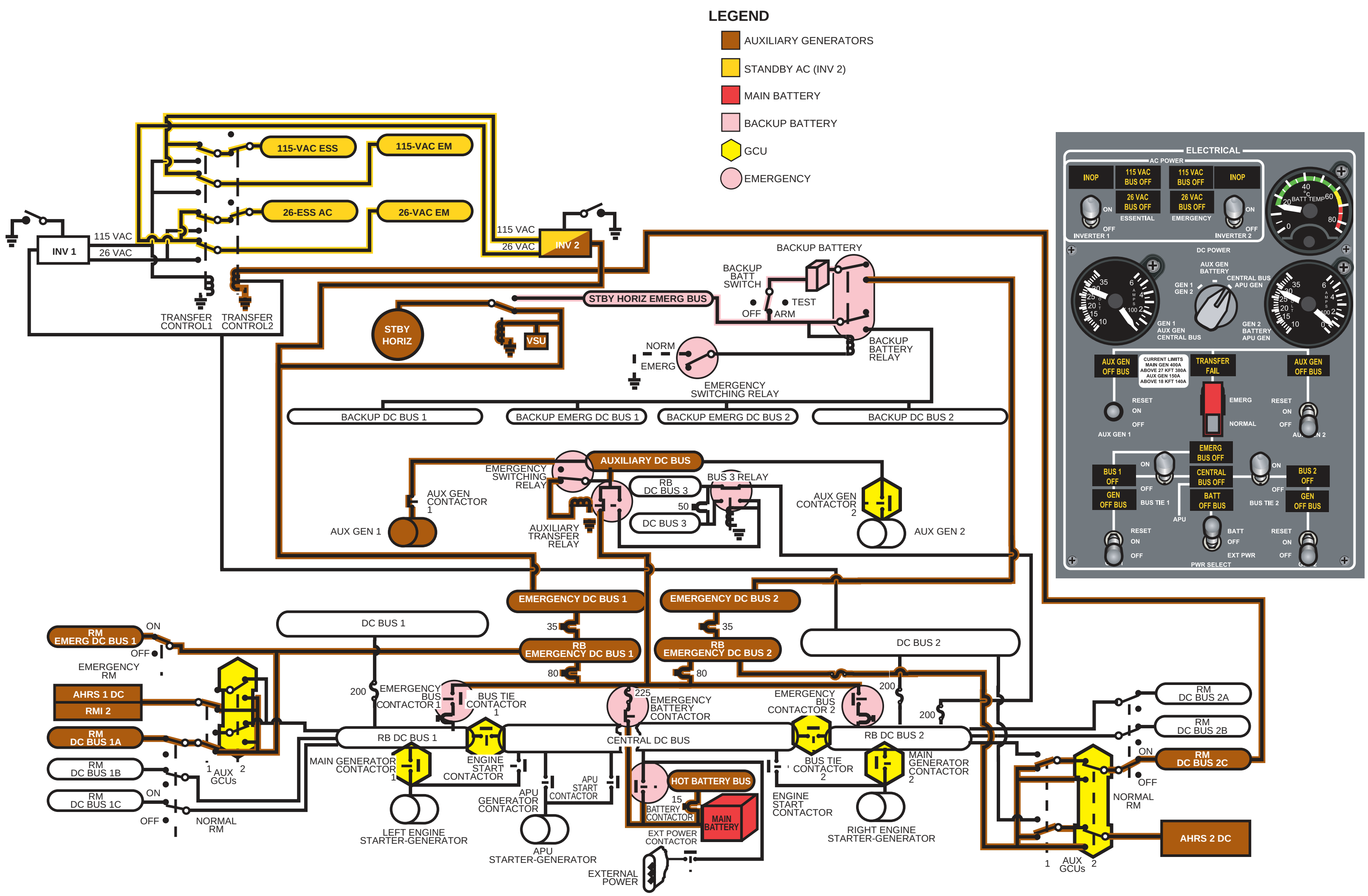


Figure 2-18. Emergency Mode—Both Main Generators and One Auxiliary Generator Failure

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.

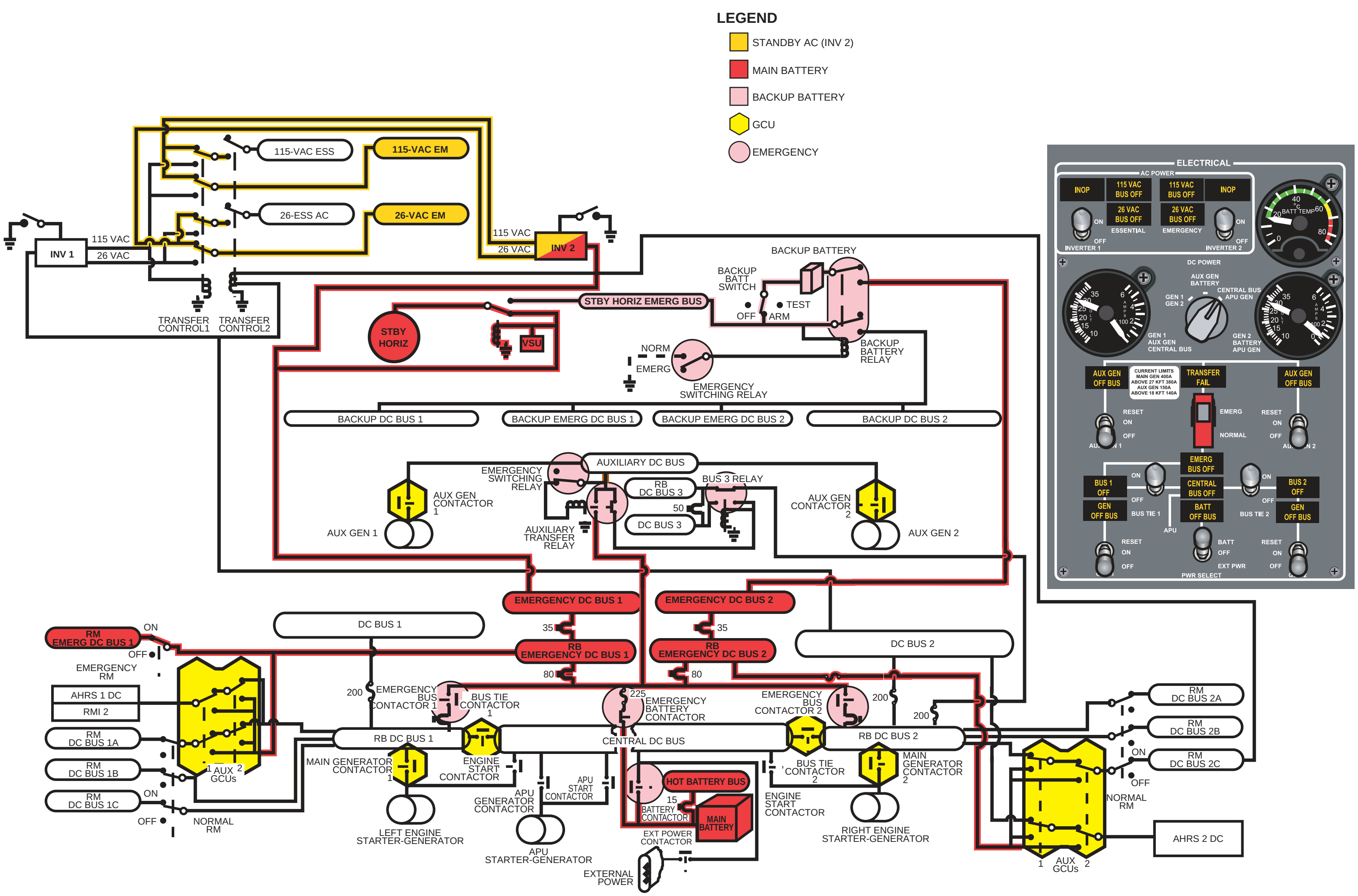


Figure 2-19. Emergency Mode—Total Generator Failure

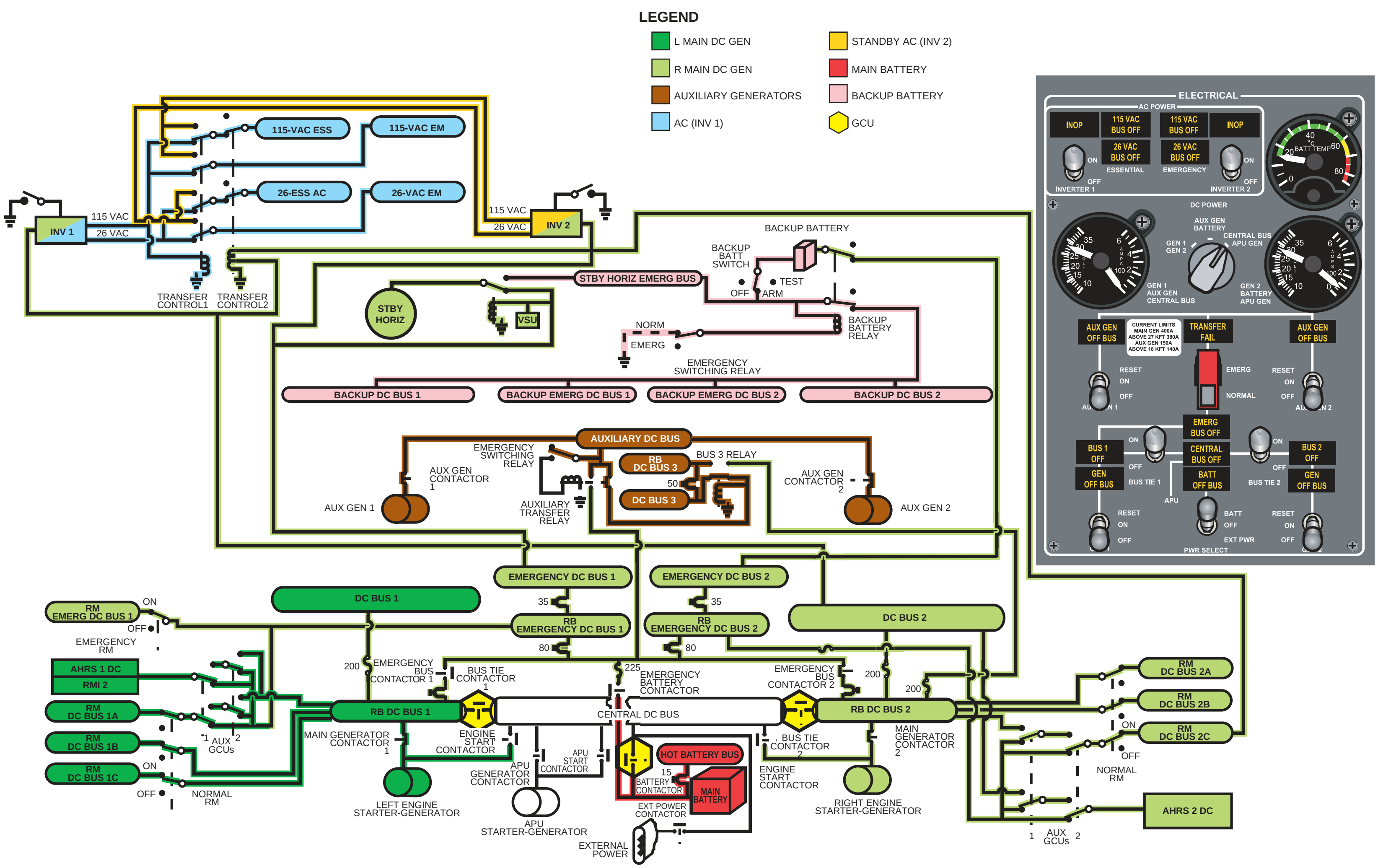


Figure 2-20. Overcurrent Case 1: Short Circuit, Central DC Bus

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.

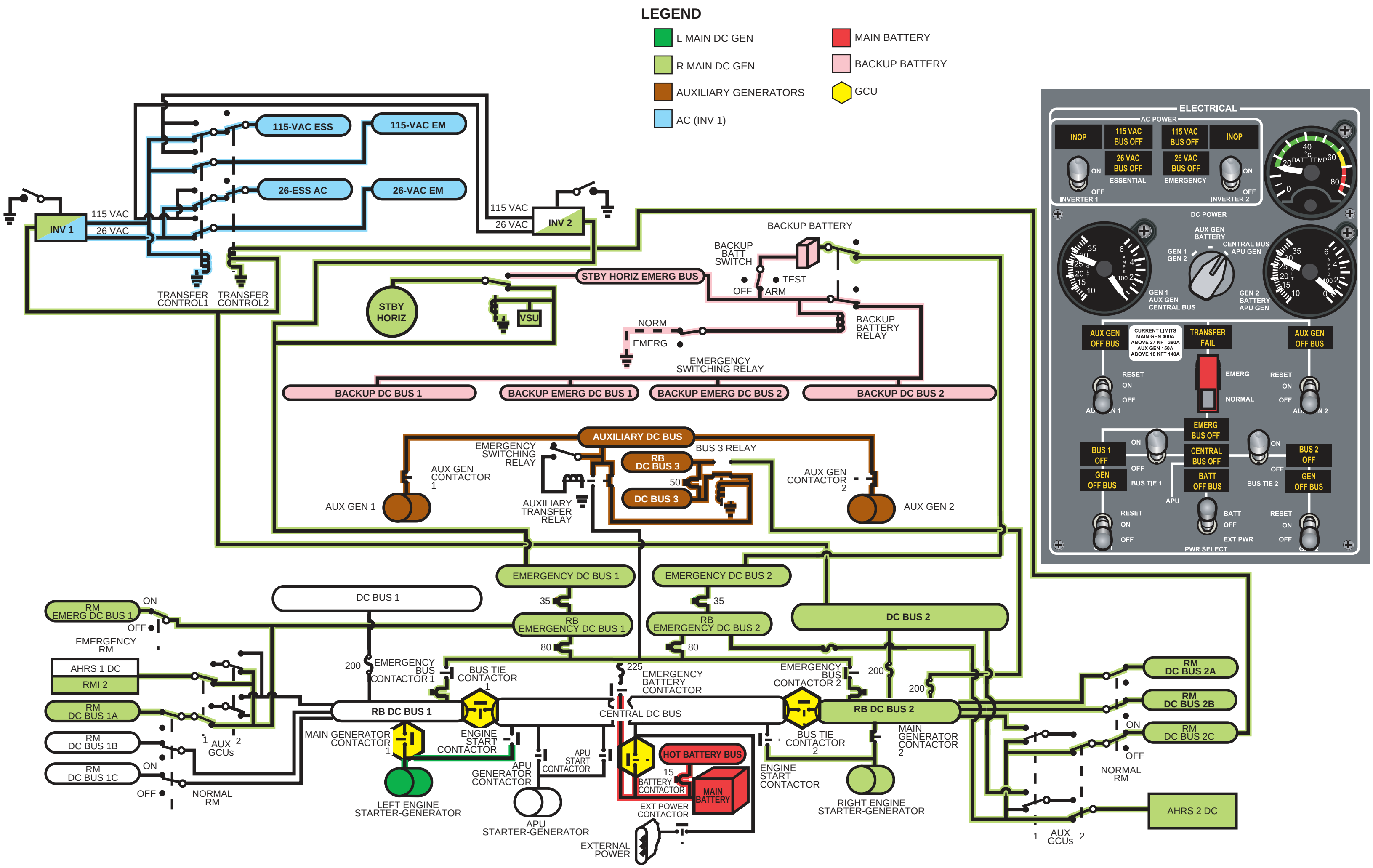


Figure 2-21. Overcurrent Case 2: Short Circuit, Relay Box DC Bus 1

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

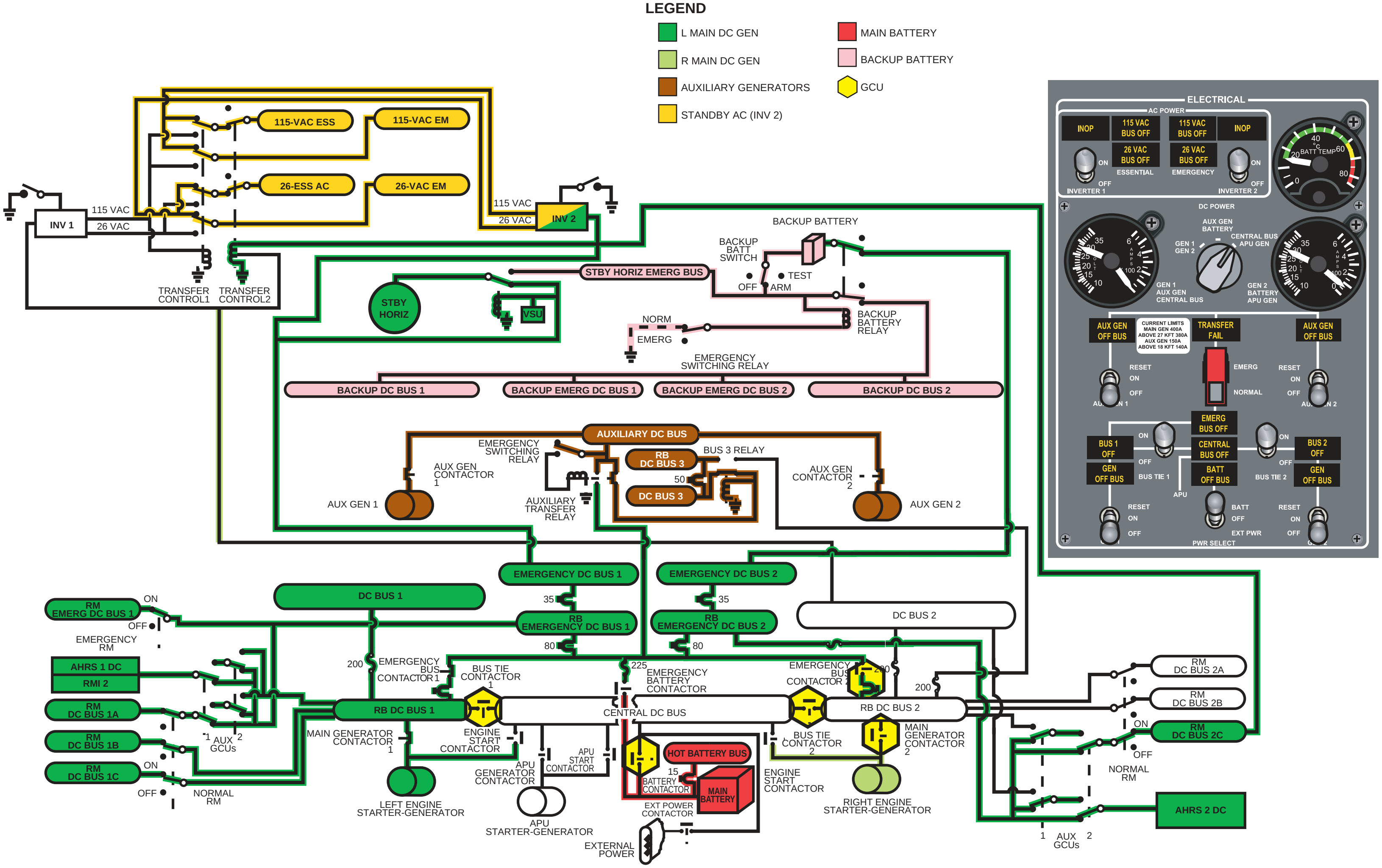


Figure 2-22. Overcurrent Case 3: Short Circuit, Relay Box DC Bus 2

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

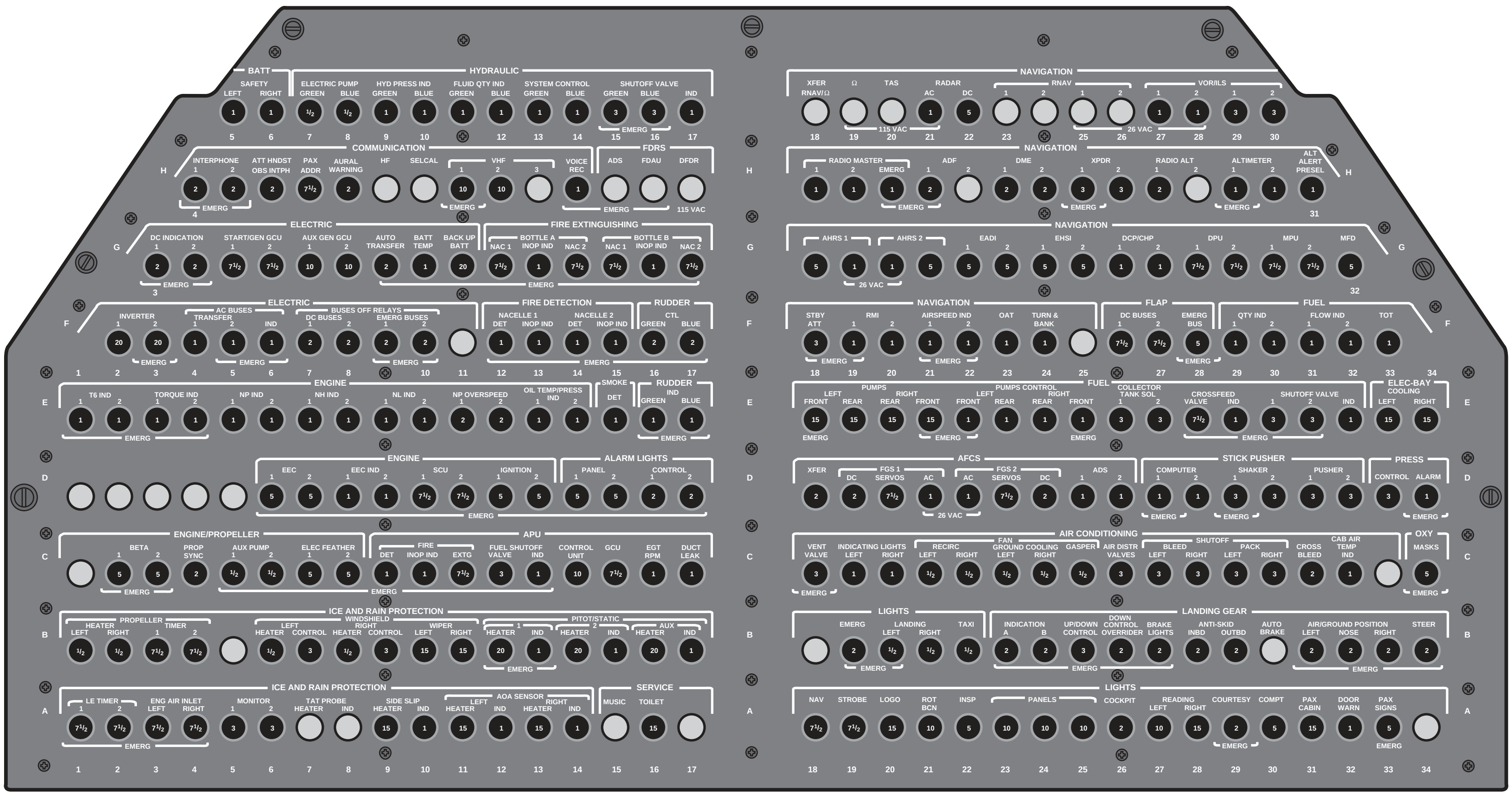


Figure 2-23. Circuit-Breaker Panel

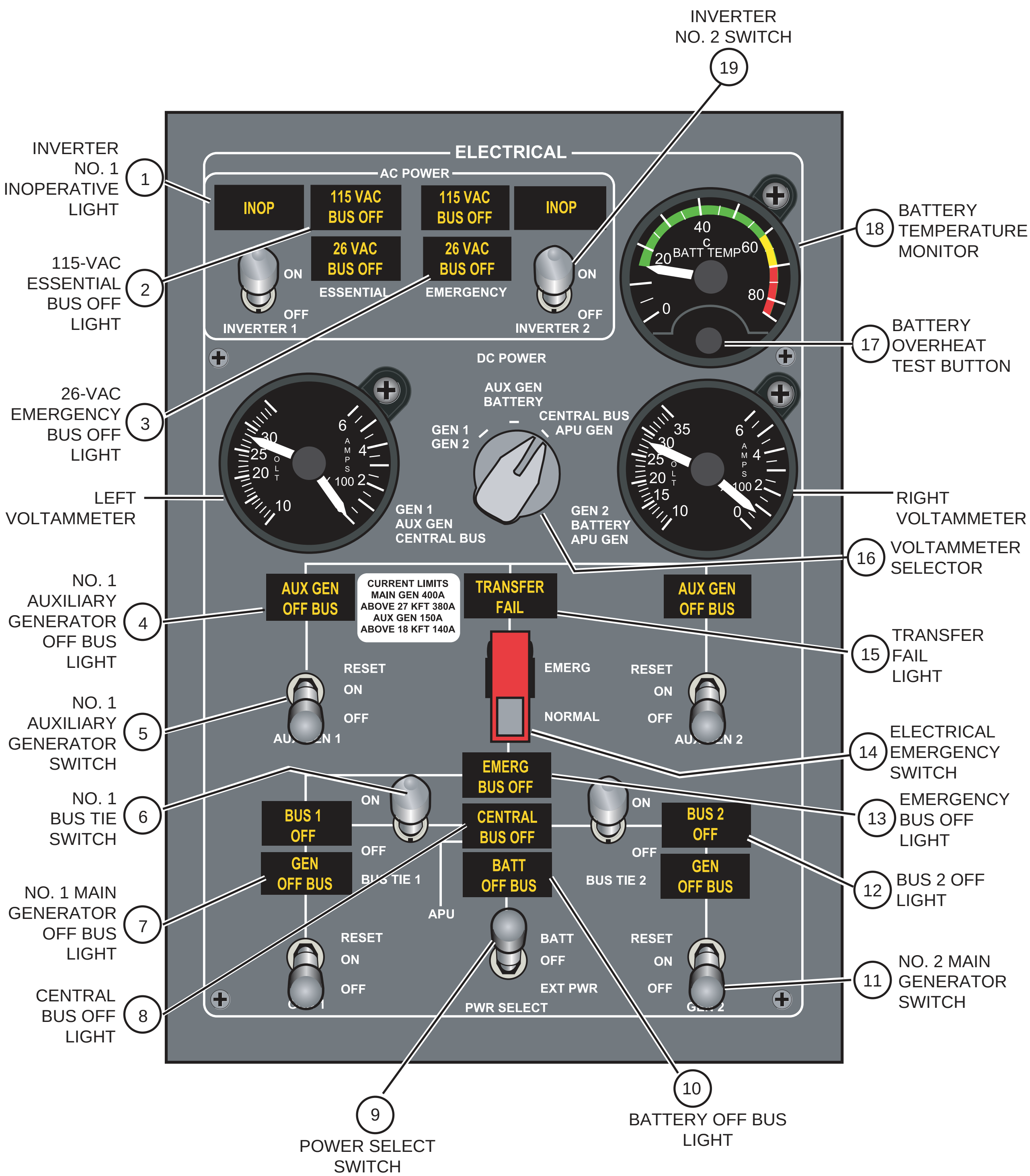


Figure 2-24. Electrical Control Panel

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.

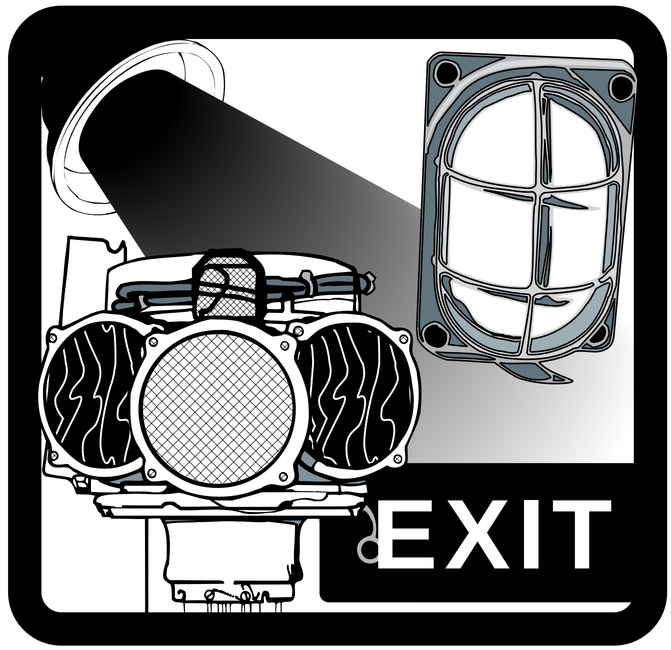
CHAPTER 3
LIGHTING

CONTENTS

Page

ILLUSTRATIONS

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.

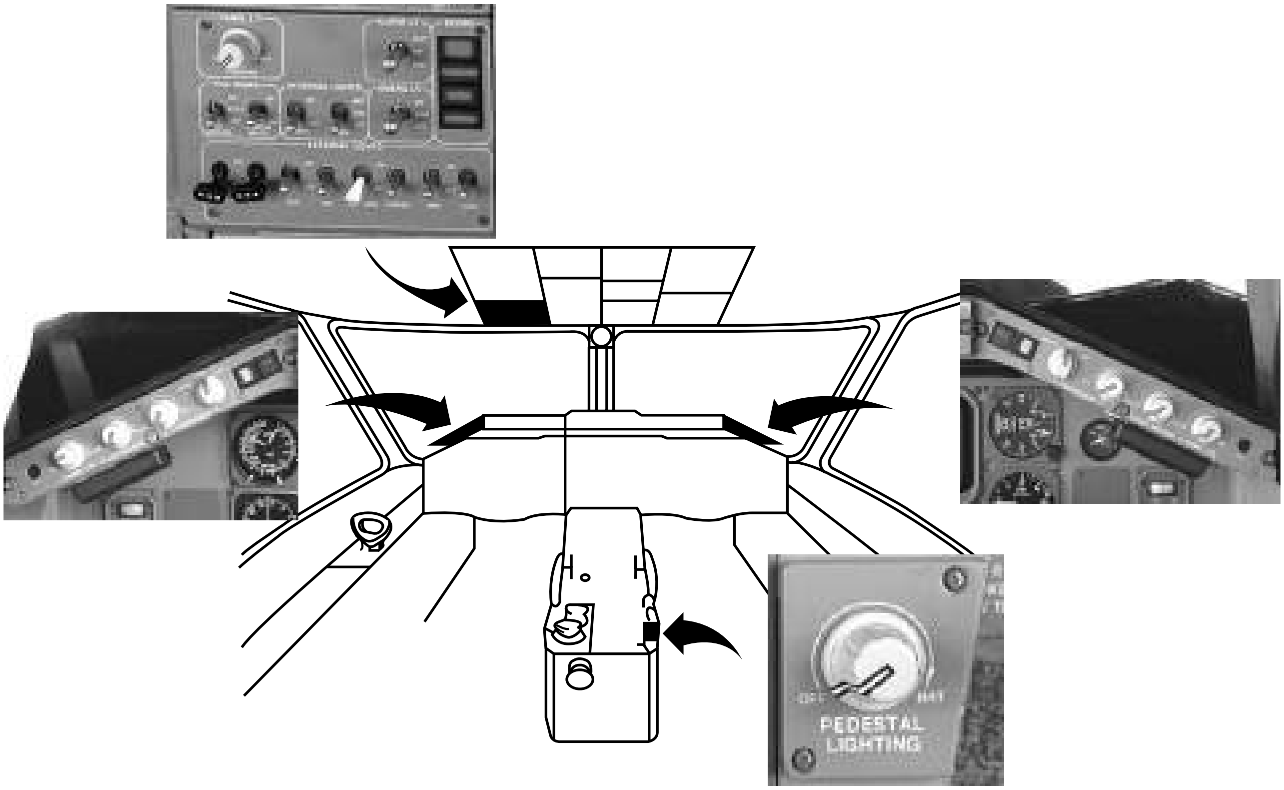


Figure 3-1. Cockpit Lighting Controls

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).

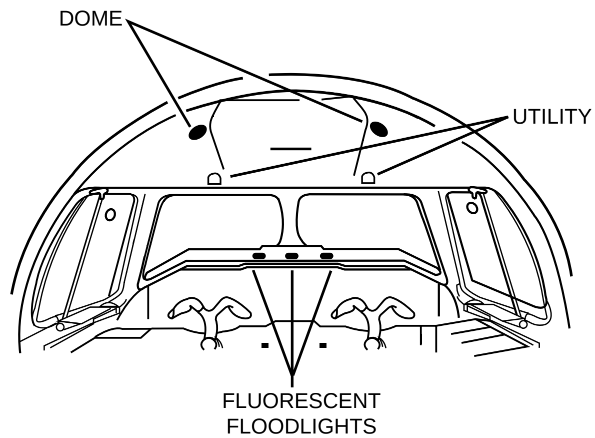


Figure 3-2. Fluorescent, Utility, and Dome Lighting

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.

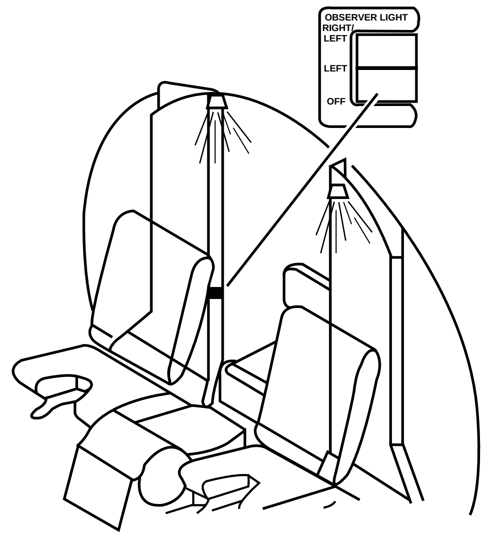


Figure 3-3. Observer Lights

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.

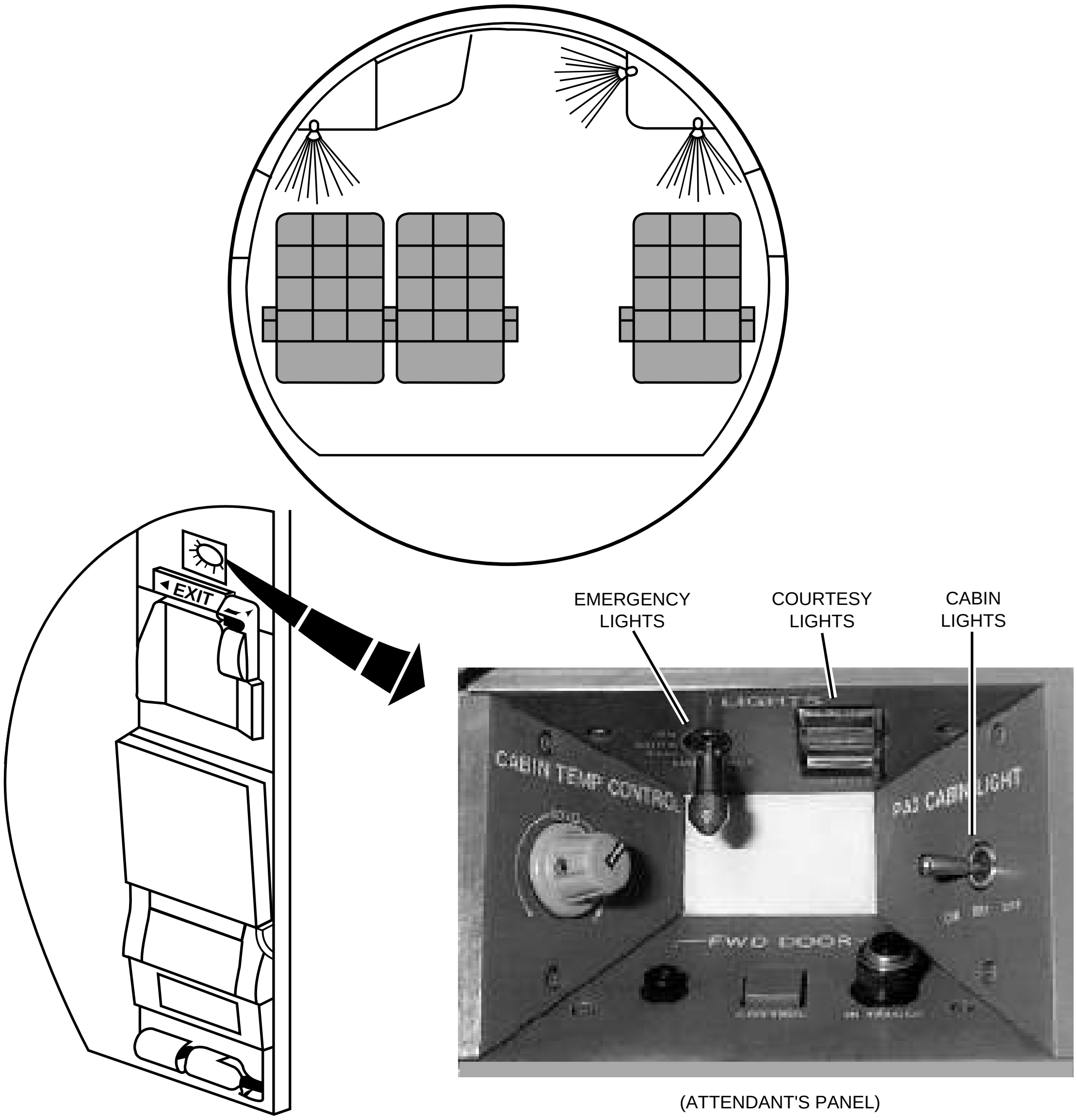


Figure 3-4. Passenger Cabin Lighting

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.

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

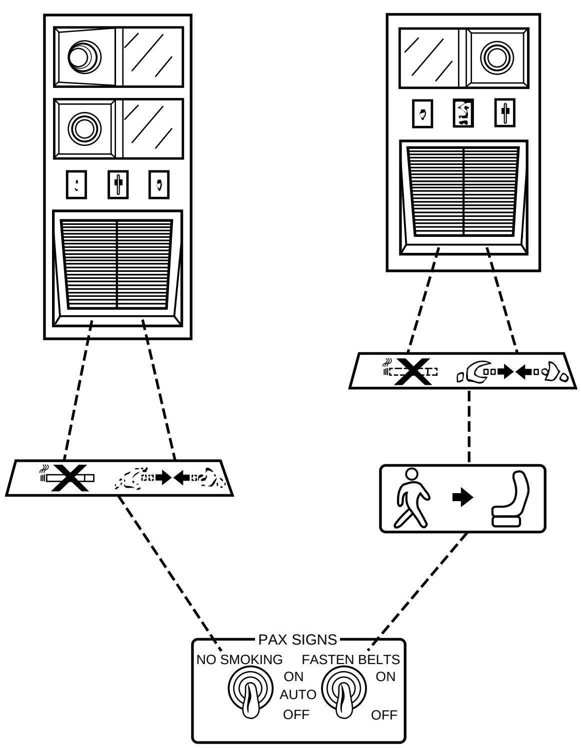


Figure 3-5. Passenger Advisory Lights

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.

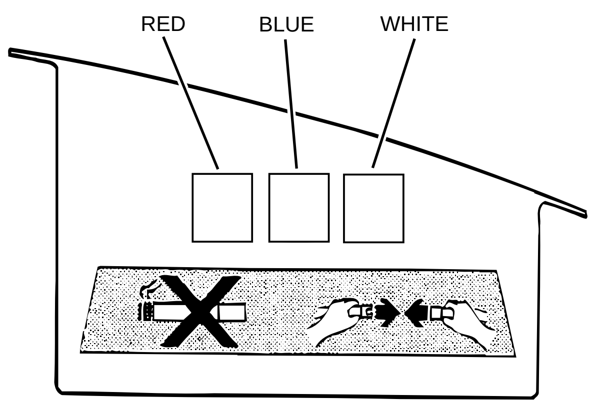


Figure 3-6. Flight Attendant Call Lights

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.

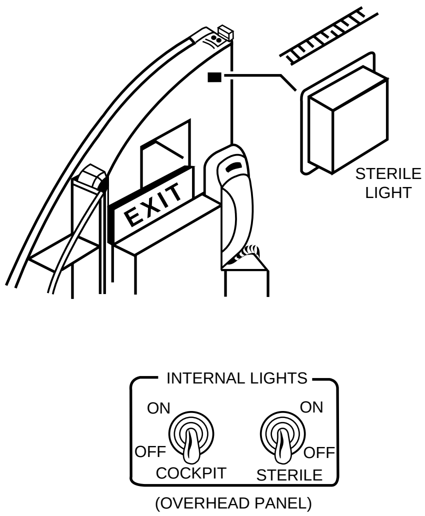


Figure 3-7. Sterile Light

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.

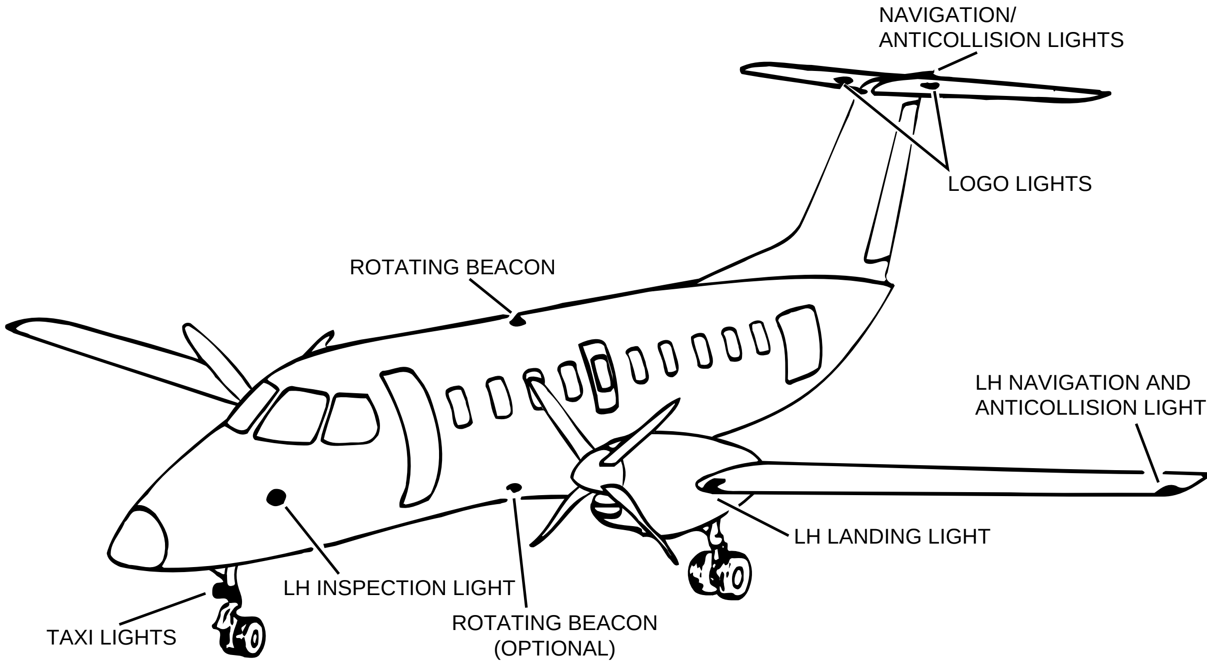


Figure 3-8. External Light

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

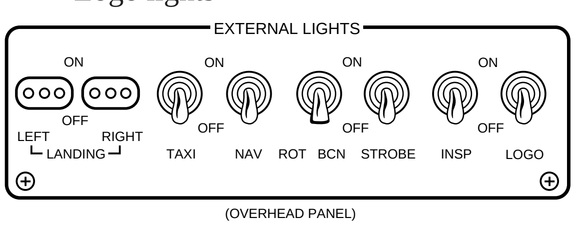


Figure 3-9. External Lights Control Panel

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).

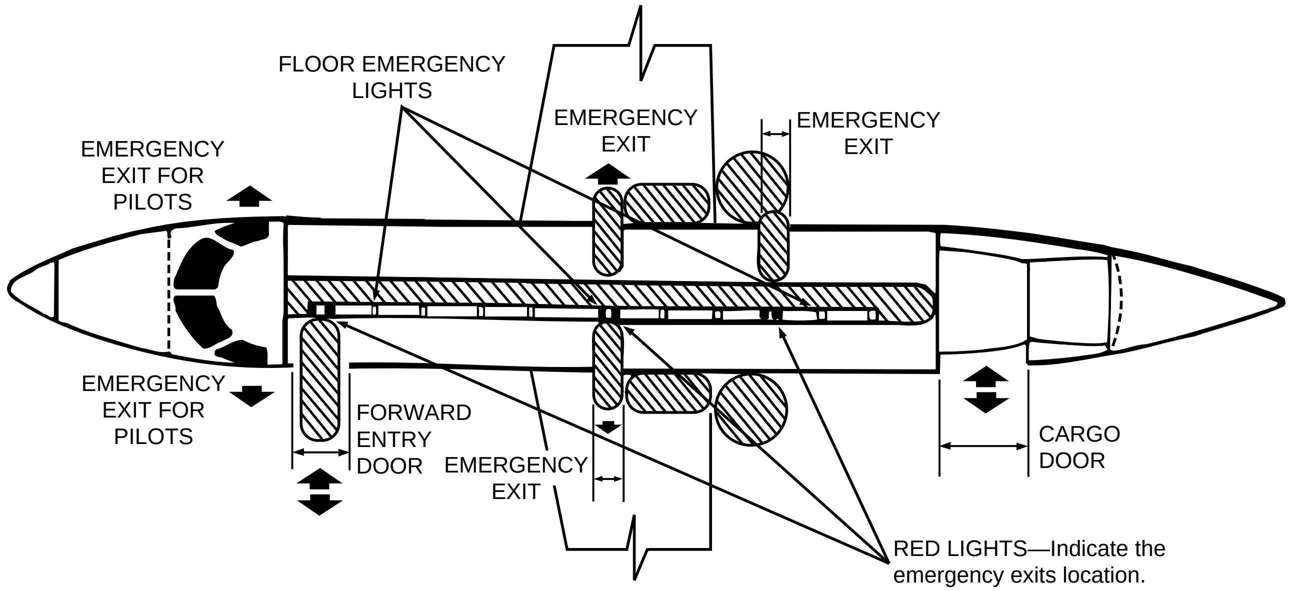


Figure 3-10. Emergency Illuminated Areas

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.

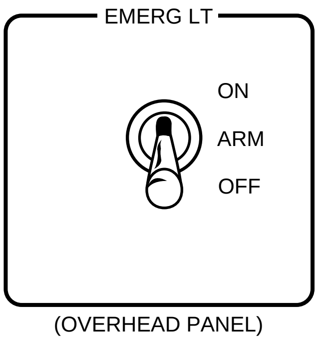


Figure 3-11. Cockpit Emergency Lighting Switch

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).

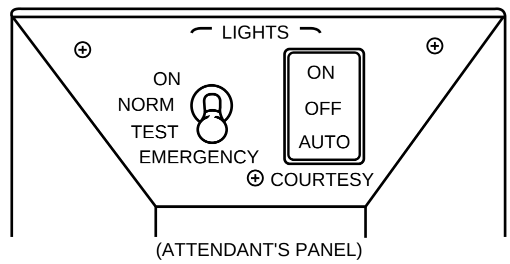


Figure 3-12. Flight Attendant Emergency Lighting Switch

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

ILLUSTRATIONS

TABLE

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.

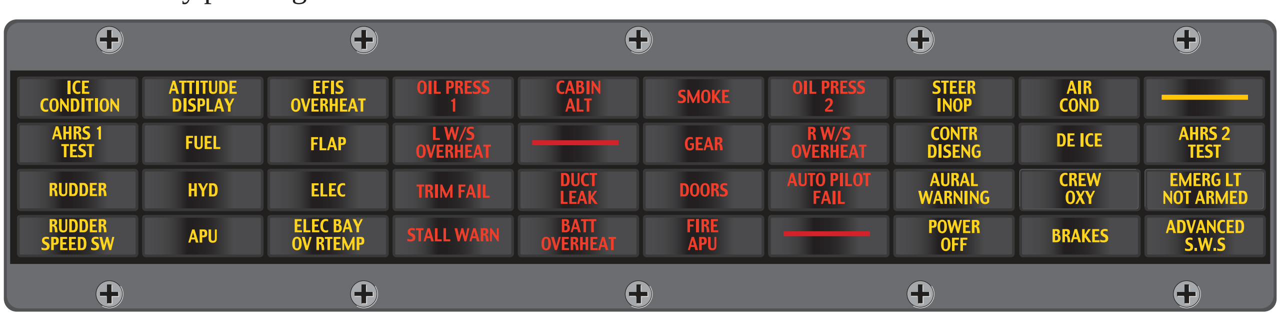


Figure 4-1. Multiple Alarm Panel

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).

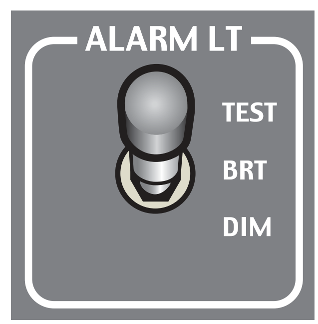


Figure 4-3. Alarm Lights Switch

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.

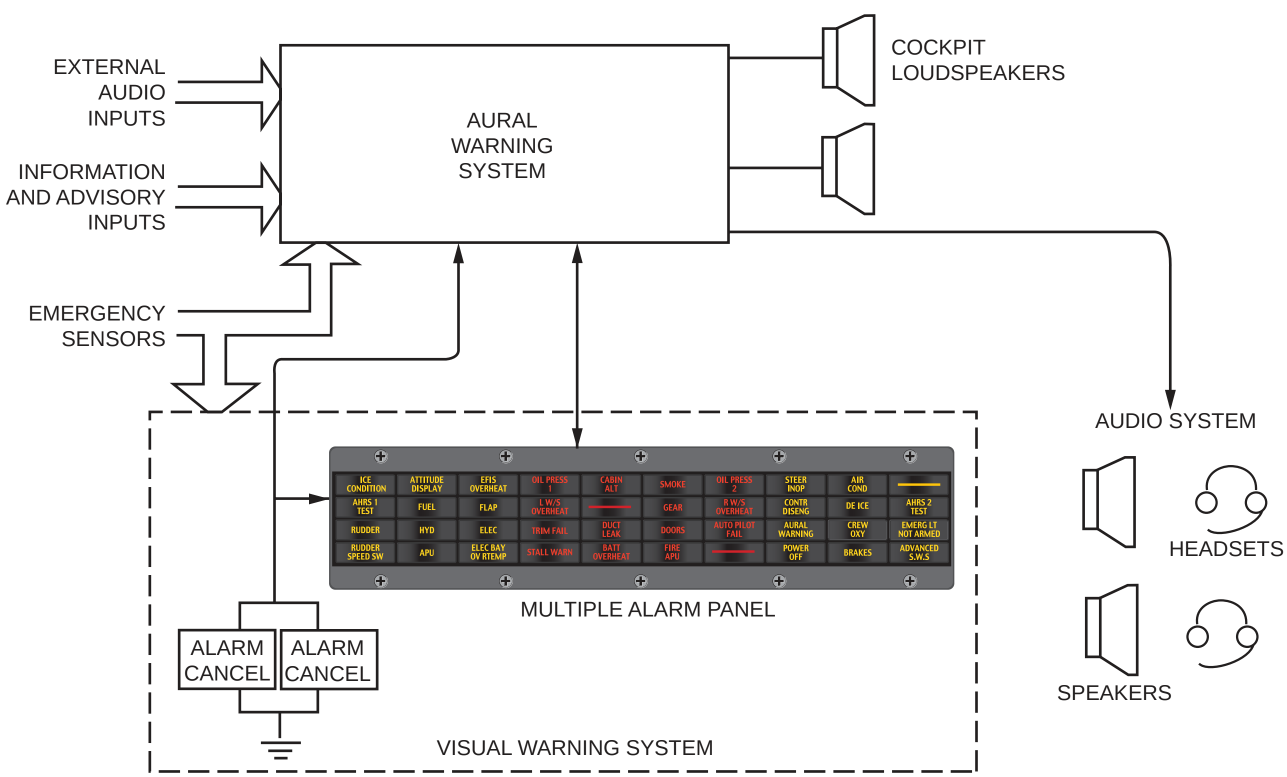


Figure 4-4. Aural Warning System (AWS) Interfaces

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.

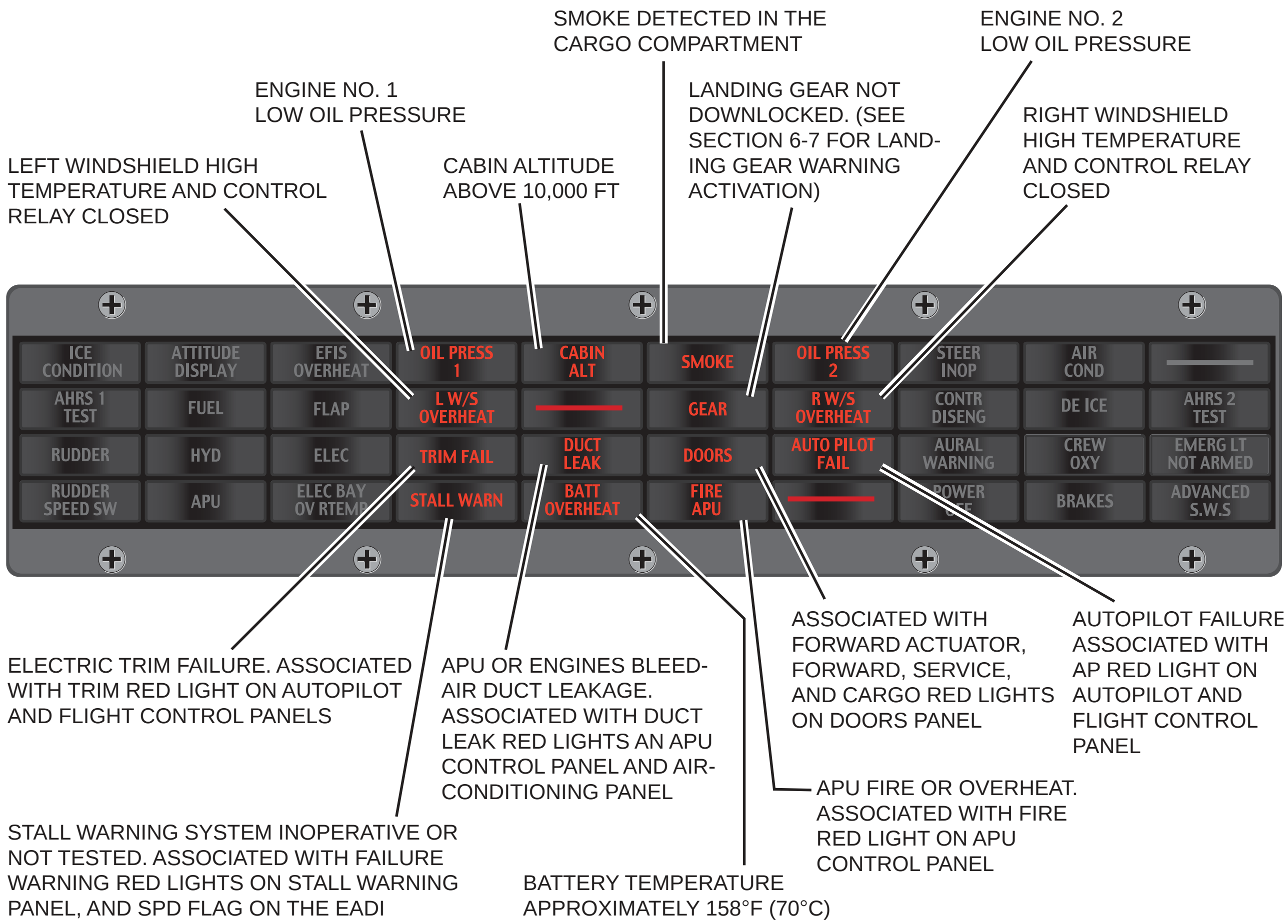


Figure 4-5. Multiple Alarm Panel Emergency Faults

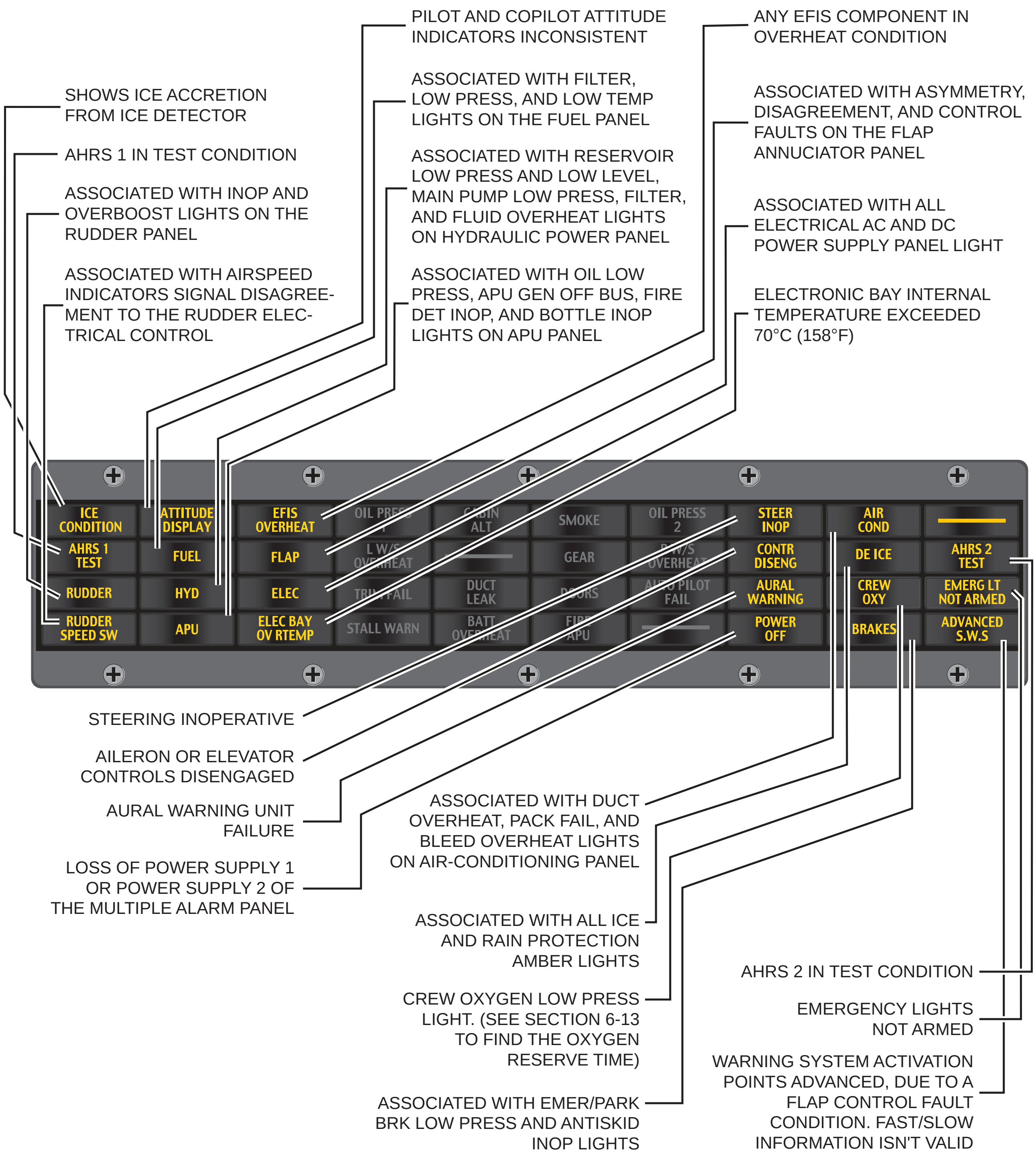


Figure 4-6. Multiple Alarm Panel Abnormal Faults

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:

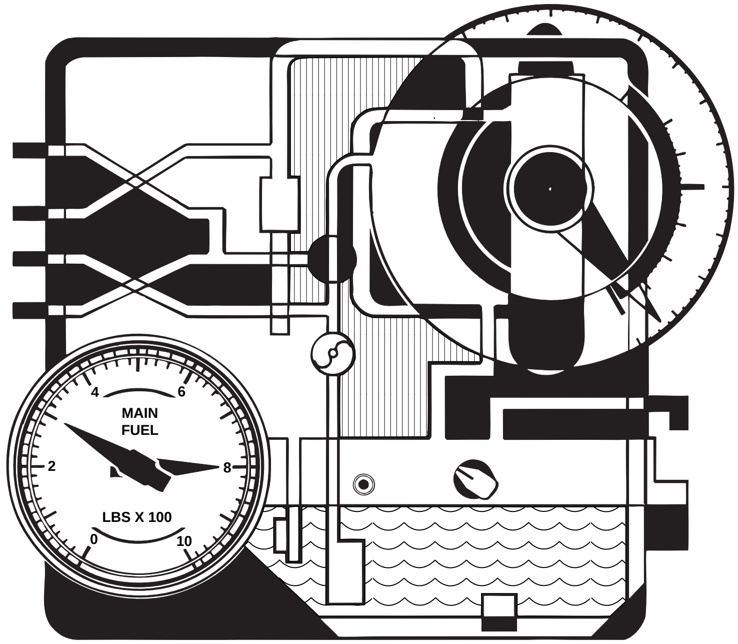
A. A switch on the captain's lower sub panel
B. A switch on the FO's lower subpanel
C. A switch on the top right overhead panel
D. None of the above

CHAPTER 5
FUEL SYSTEM

CONTENTS

ILLUSTRATIONS

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

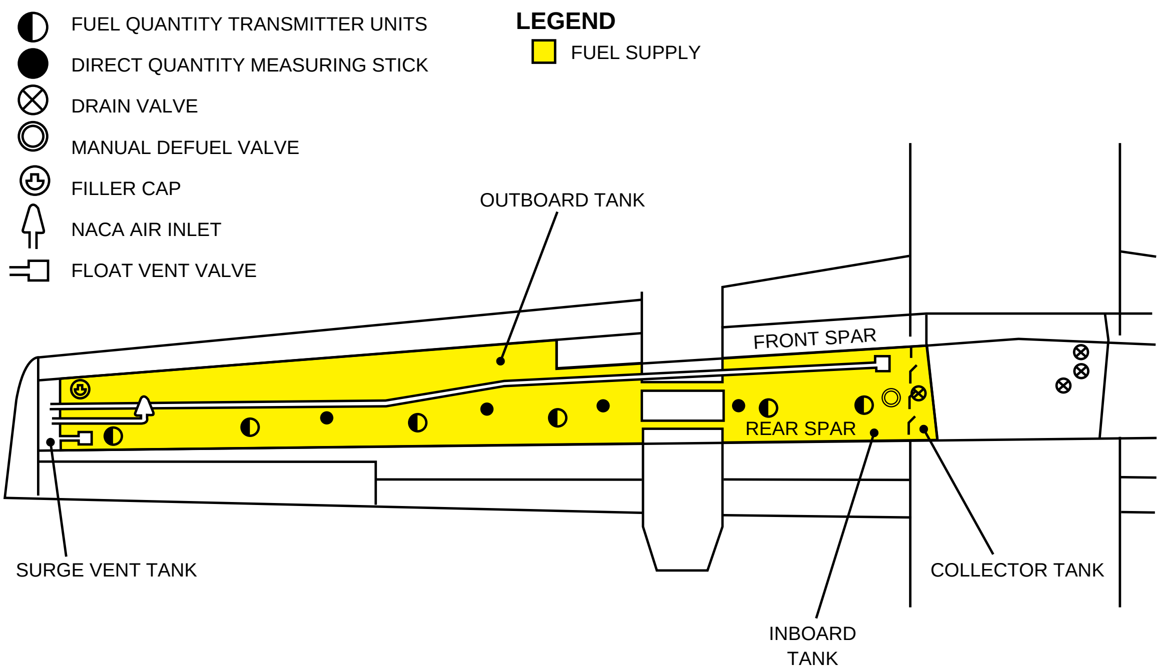


Figure 5-1. Fuel Storage

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

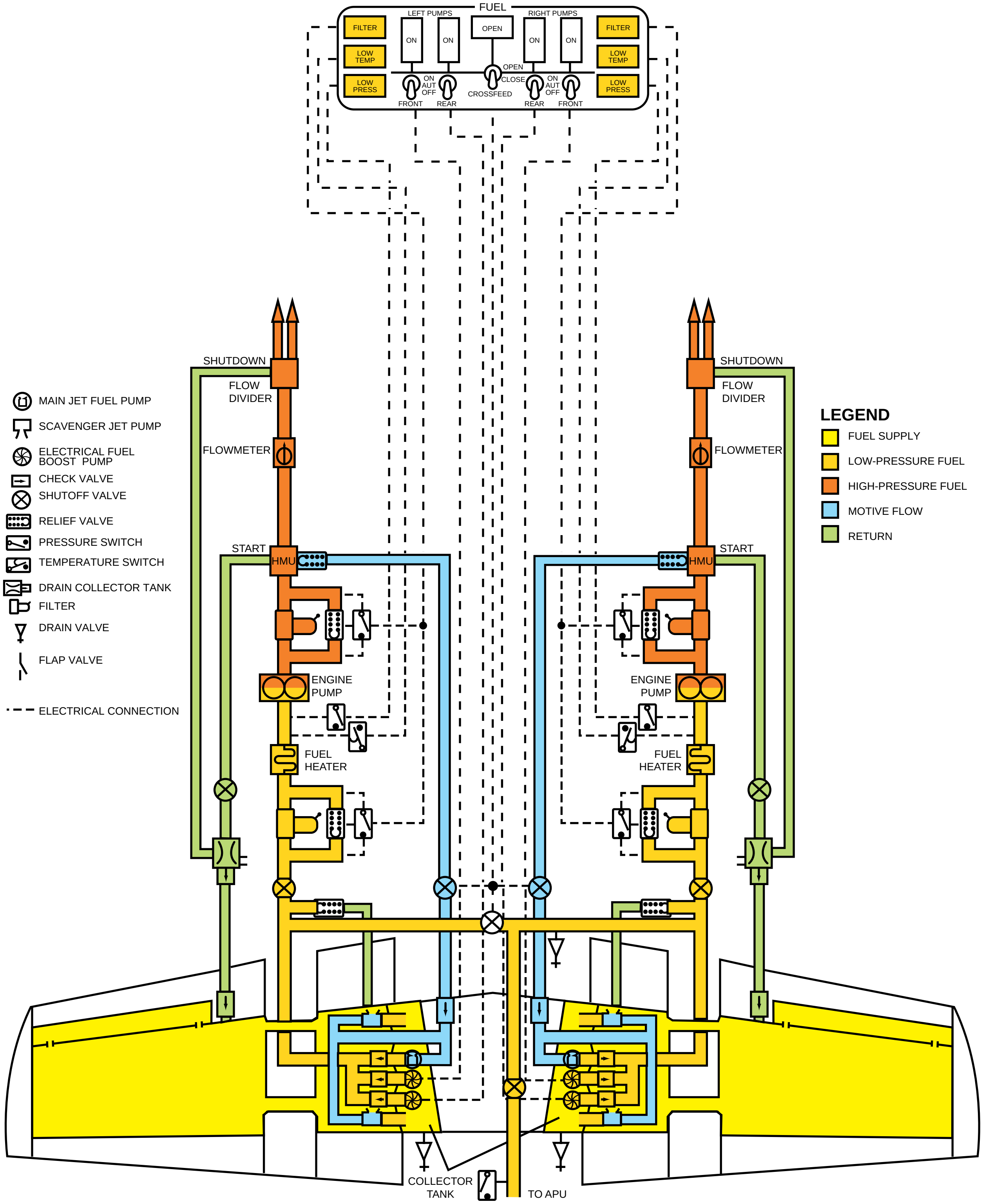


Figure 5-2. Fuel Feed System

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

- Fuel control panel
- Fuel management panel

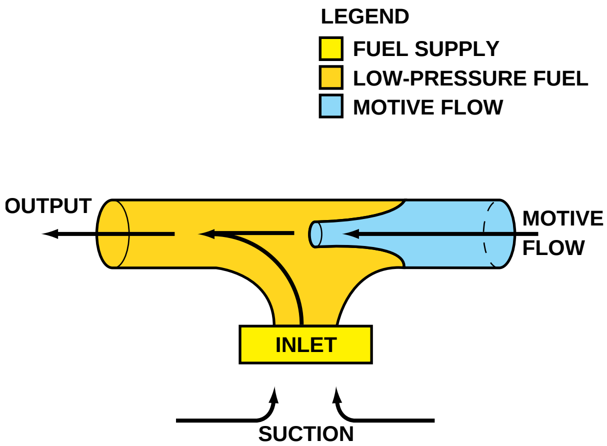


Figure 5-3. Jet Pump

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.

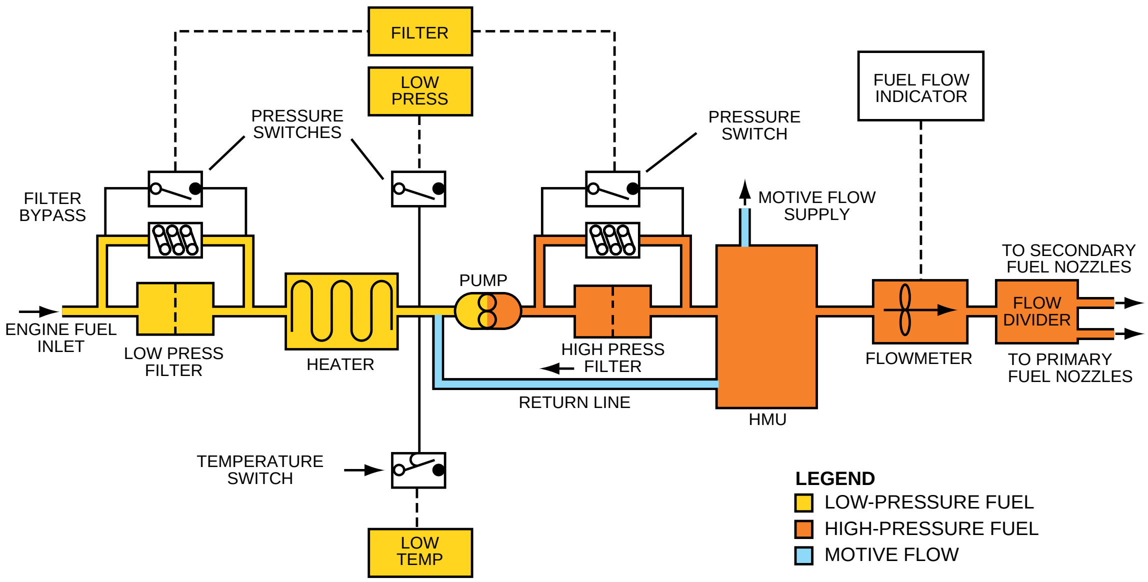


Figure 5-4. Engine Fuel Feed System

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.

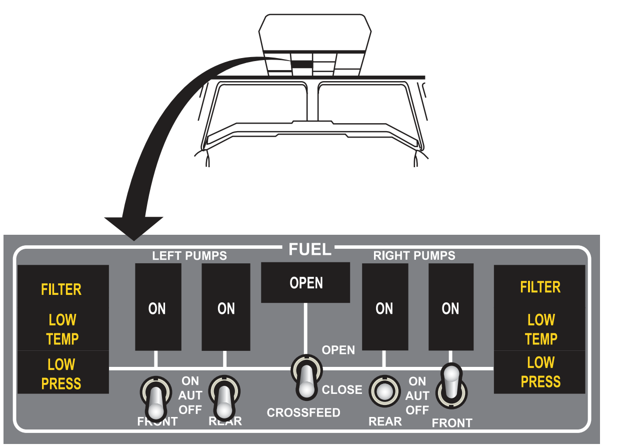


Figure 5-5. Fuel Control Panel

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.

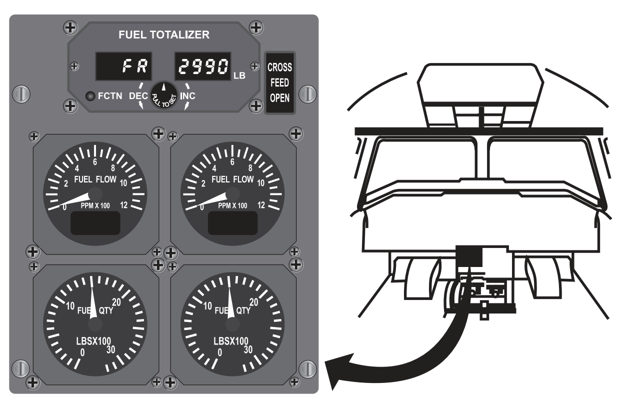


Figure 5-6. Fuel Management Panel

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:

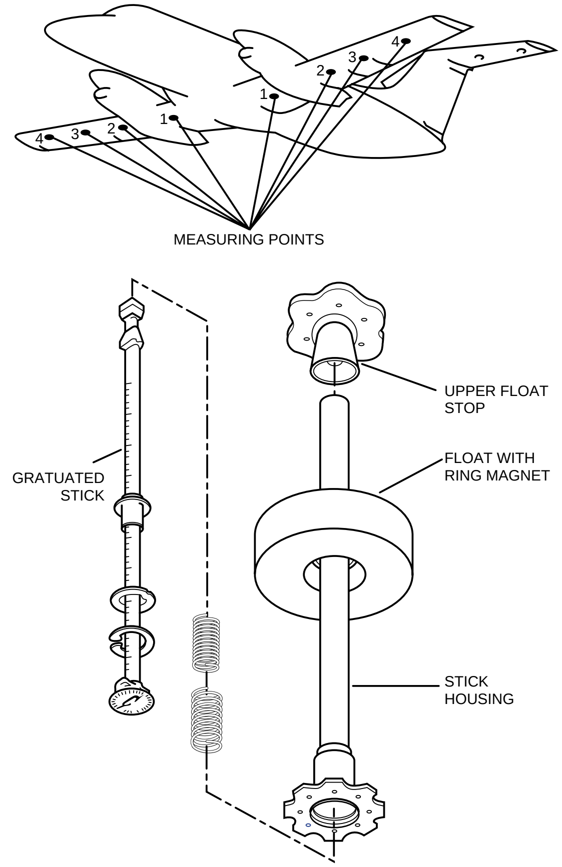


Figure 5-7. Fuel Quantity Measuring Stick

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).

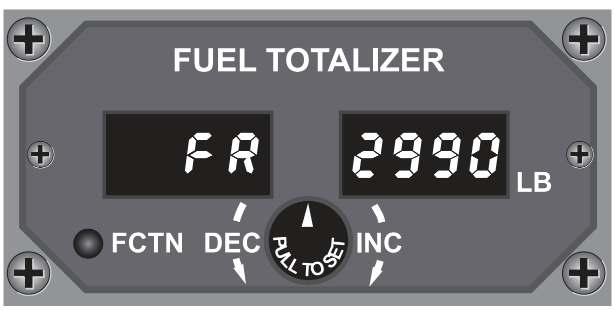


Figure 5-8. Fuel Totalizer

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.

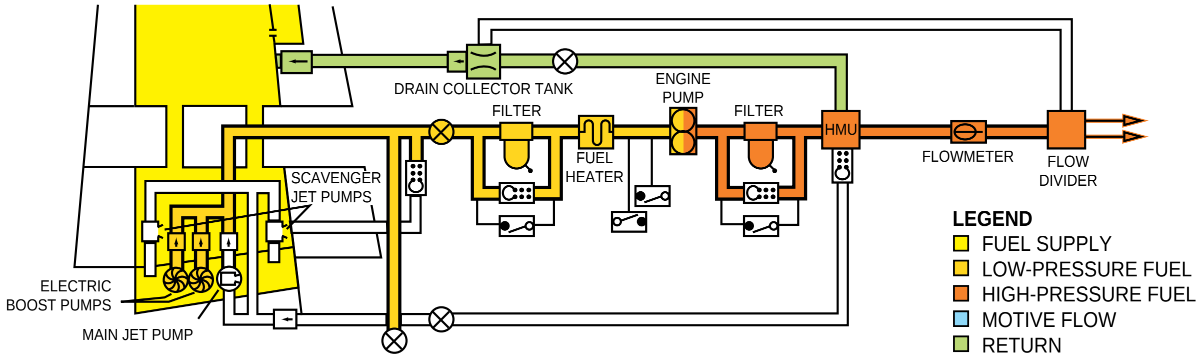


Figure 5-9. Fuel System Operation During Start

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.

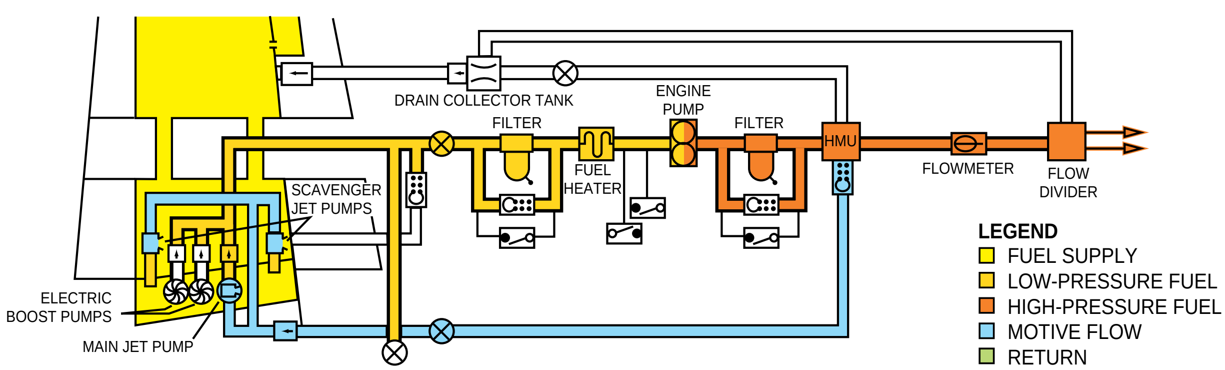


Figure 5-10. Fuel System Normal Operation

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.

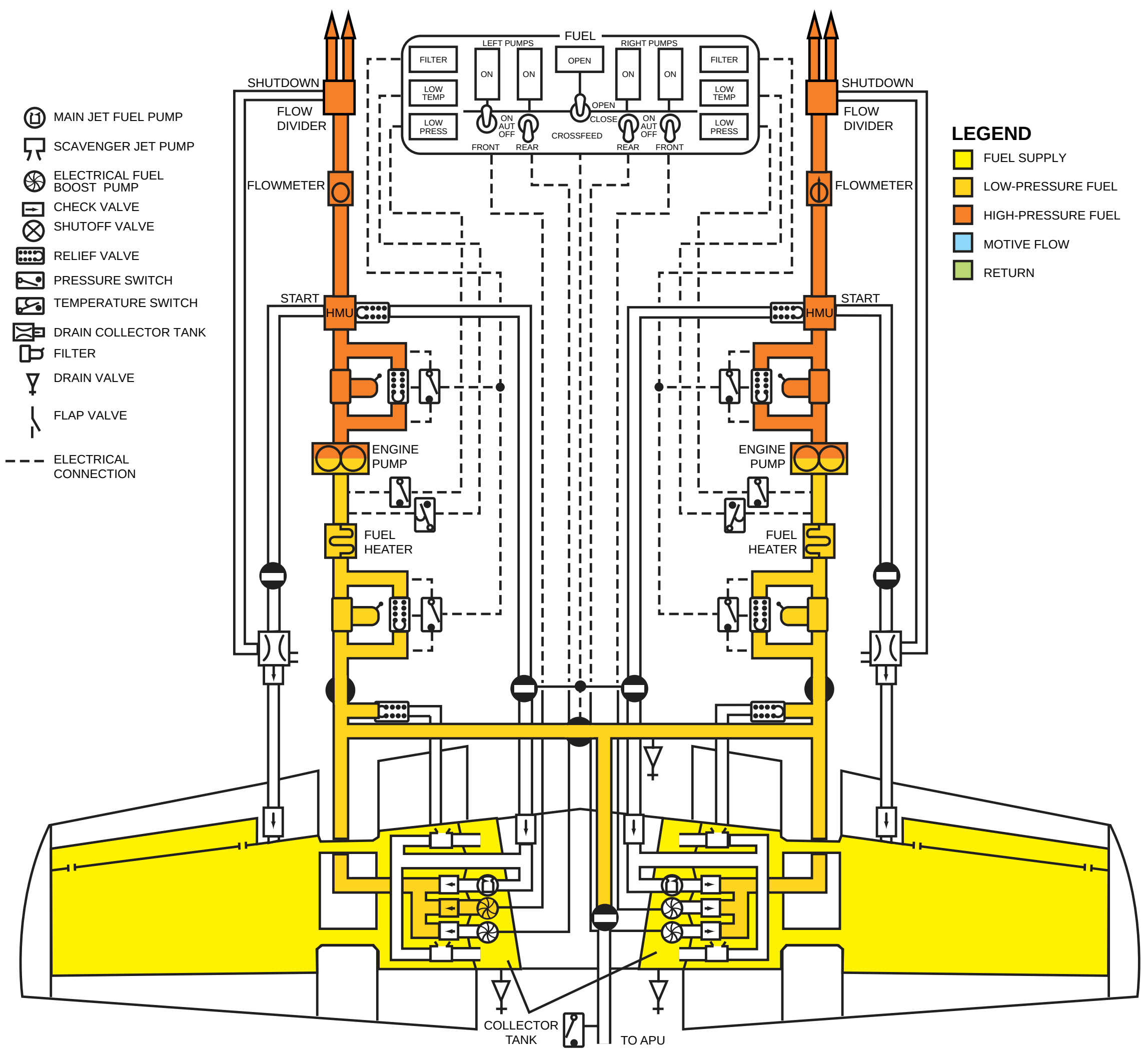


Figure 5-11. Fuel System Crossfeed

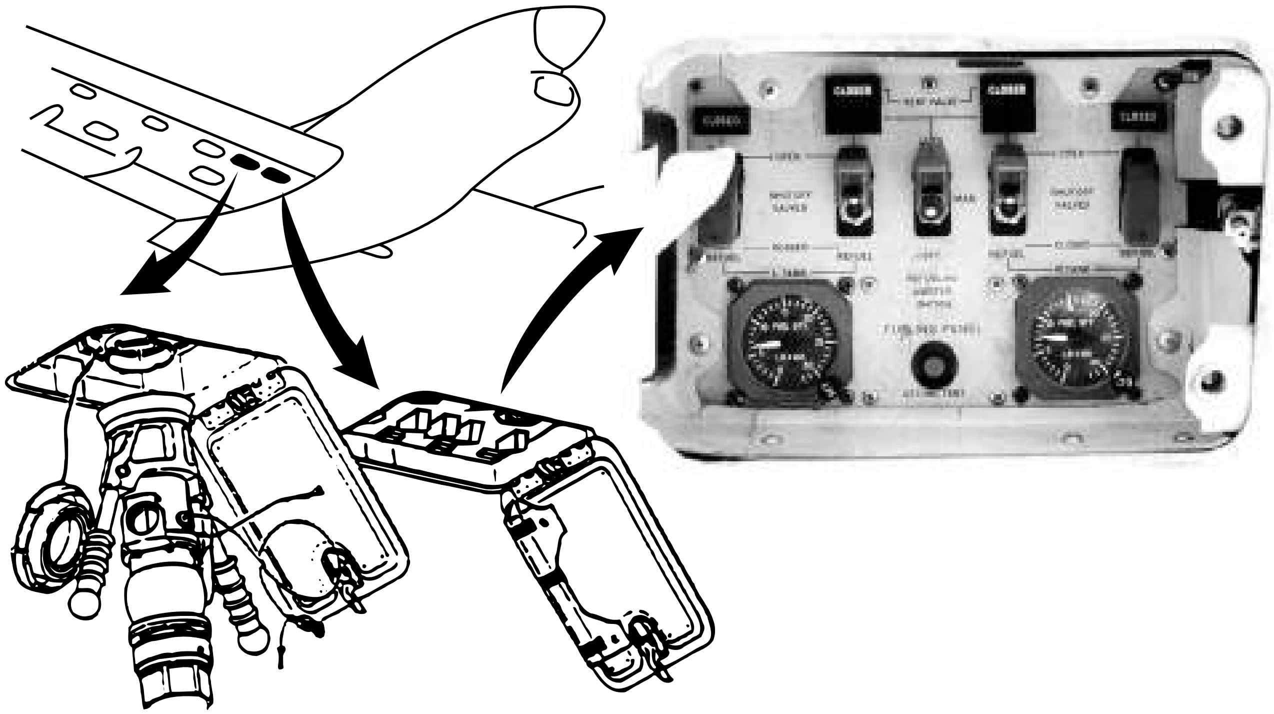


Figure 5-12. Fueling Panel

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.

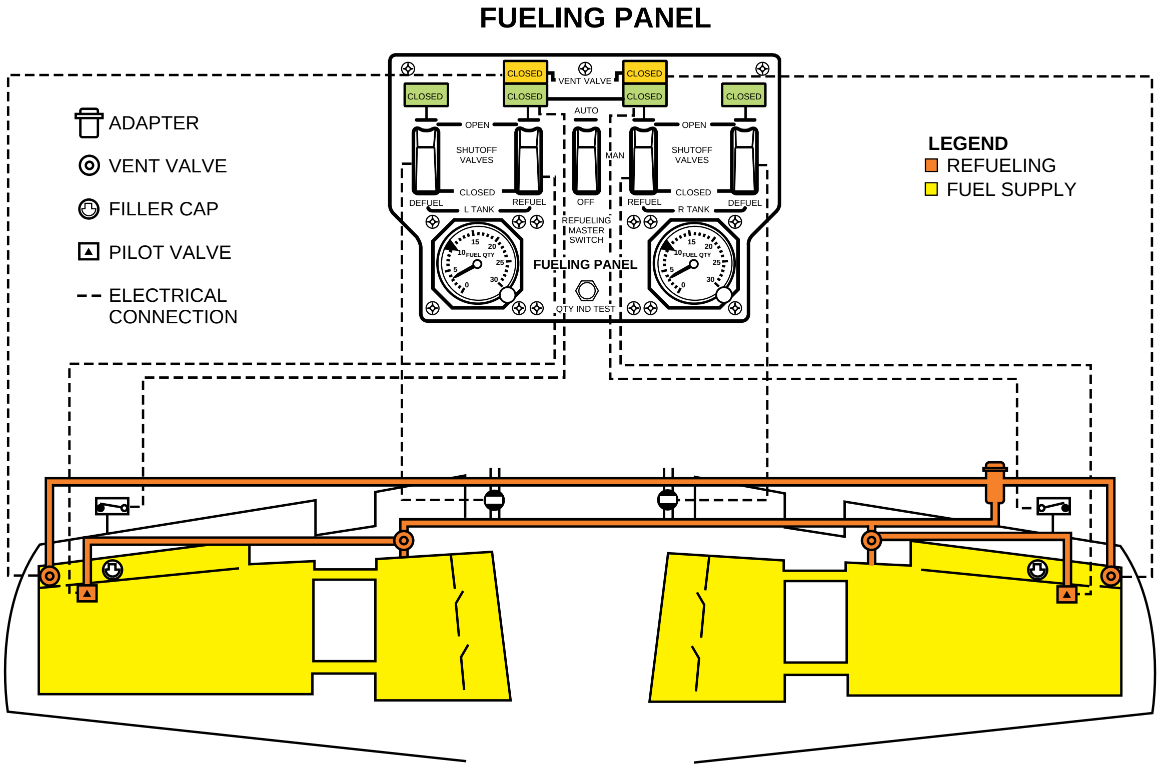


Figure 5-13. Refueling System

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.

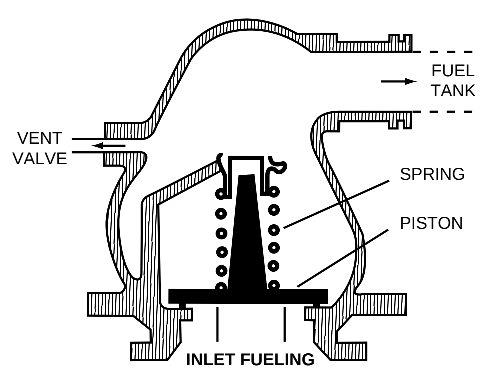


Figure 5-14. Refueling Adapter

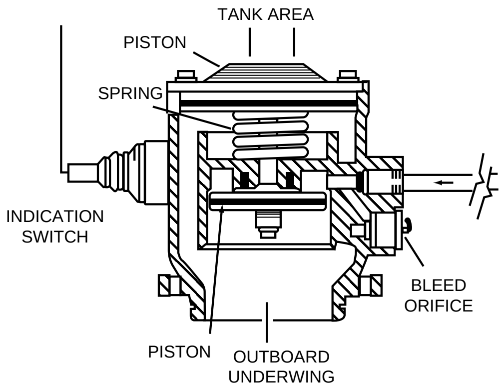


Figure 5-15. Refueling Vent Valve

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.

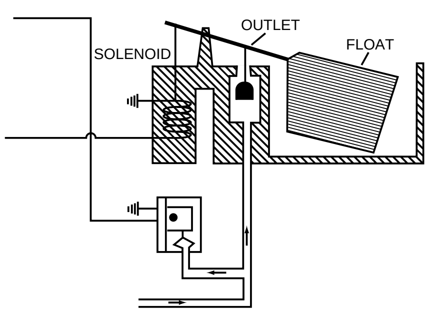


Figure 5-16. Pilot Valve

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.

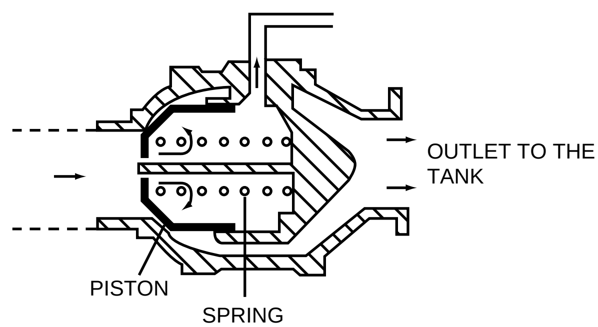


Figure 5-17. Refueling Shutoff Valve

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.

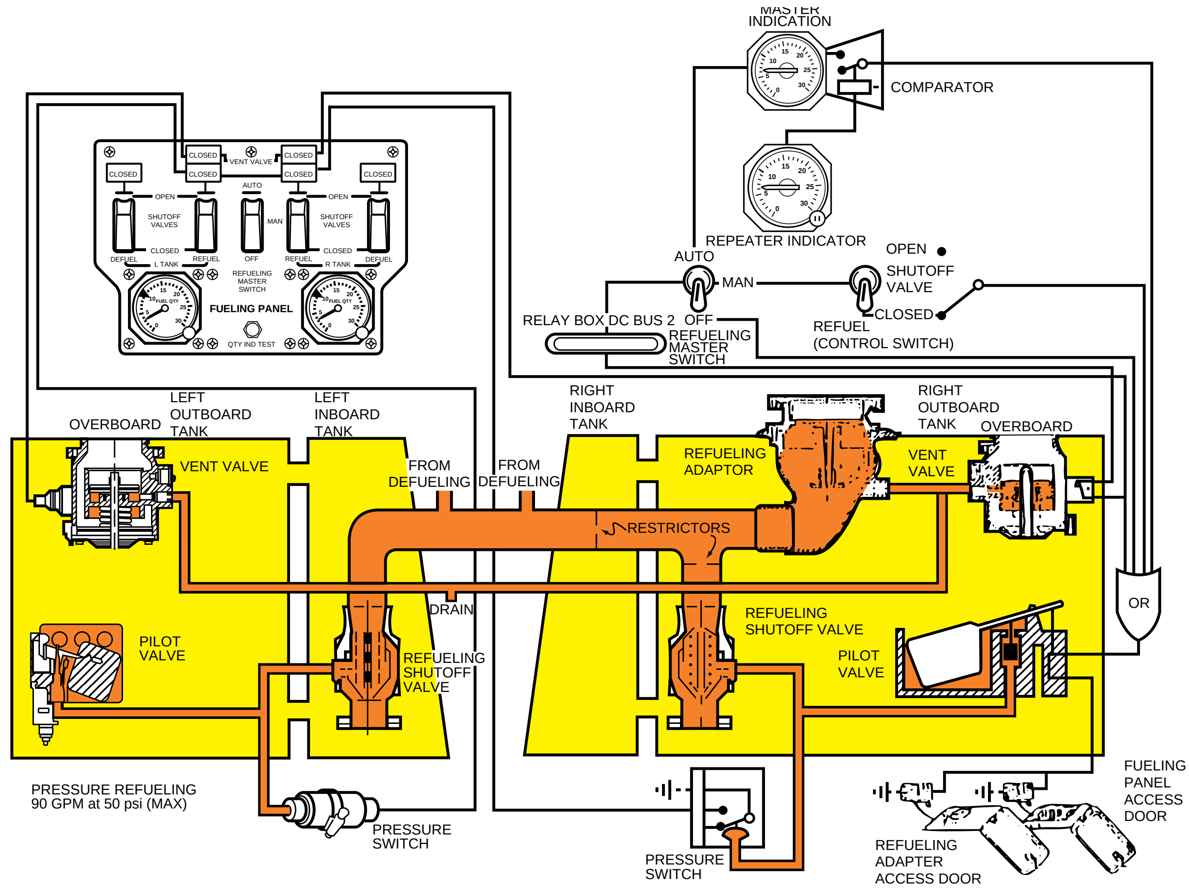


Figure 5-18. Refueling System Operation

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.

FUELING PANEL

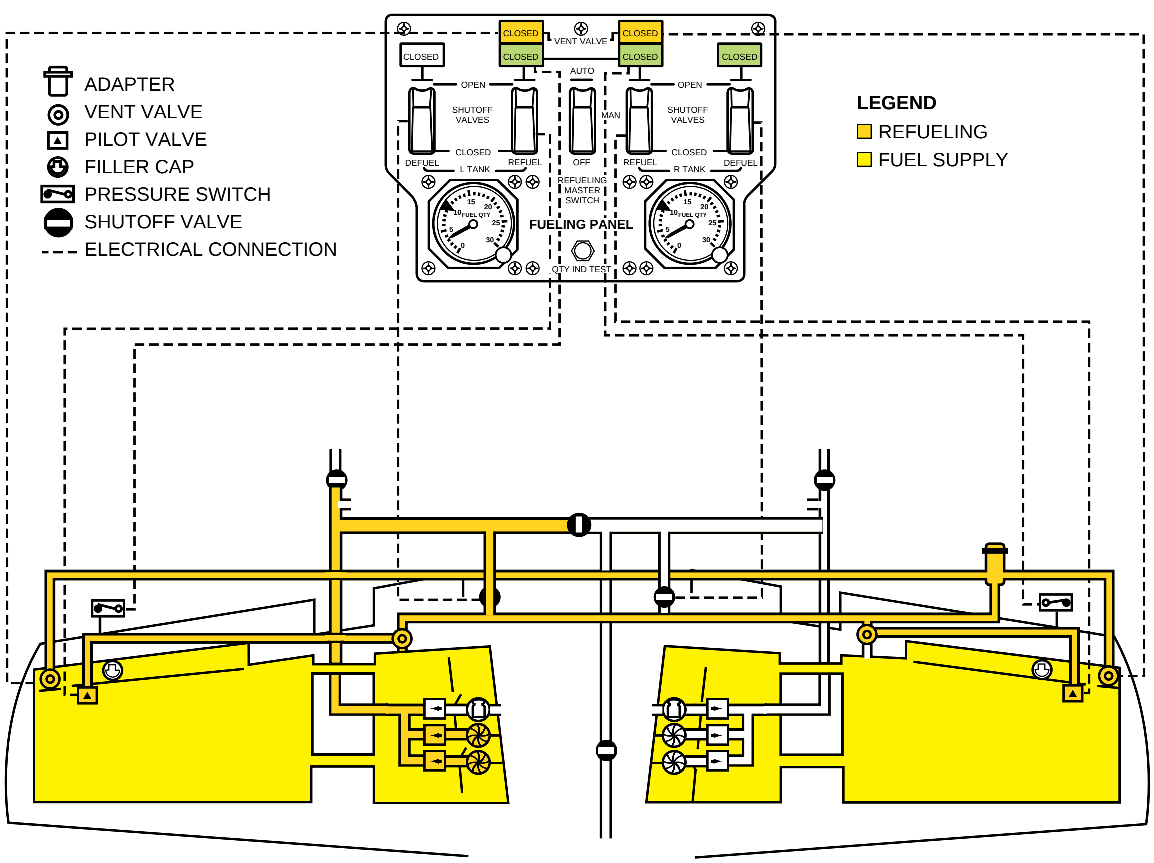


Figure 5-19. Defueling System

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.

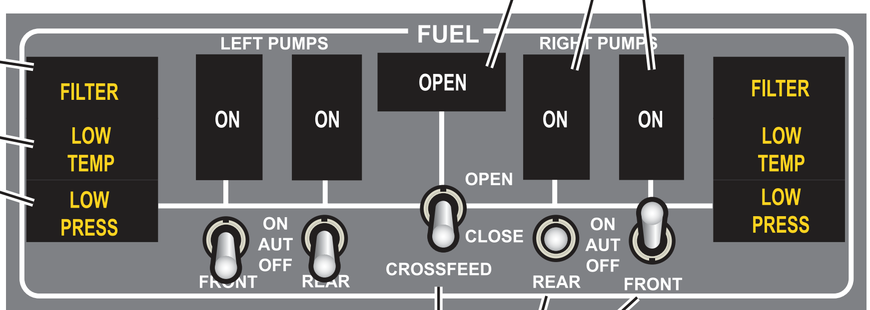
ENG NO.1 FILTER LIGHT (AMBER)
ILLUMINATED—PRESSURE DIFFERENTIAL HAS REACHED 1.5 PSID ACROSS THE LOW-PRESSURE FILTER, OR 50 PSID ACROSS THE HIGH-PRESSURE FILTER. THE FILTERS BYPASS AT 3 PSID AND 50 PSID, RESPECTIVELY. TRIGGERS AMBER FUEL LIGHT ON MAP.

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

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

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

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



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.

NOTE:
THERE ARE NO LOW FUEL LIGHTS OR WARNINGS

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

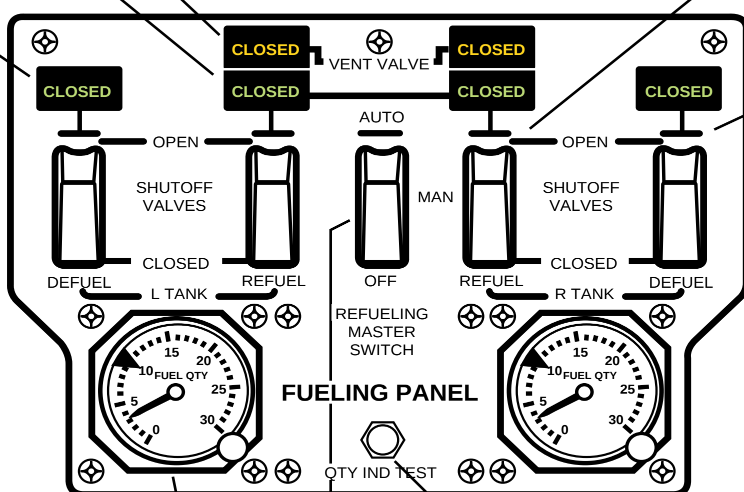
LEFT TANK DEFUELING CLOSED LIGHT (GREEN)
ILLUMINATED – INDICATES DEFUELING SHUTOFF VALVE IS CLOSED.

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

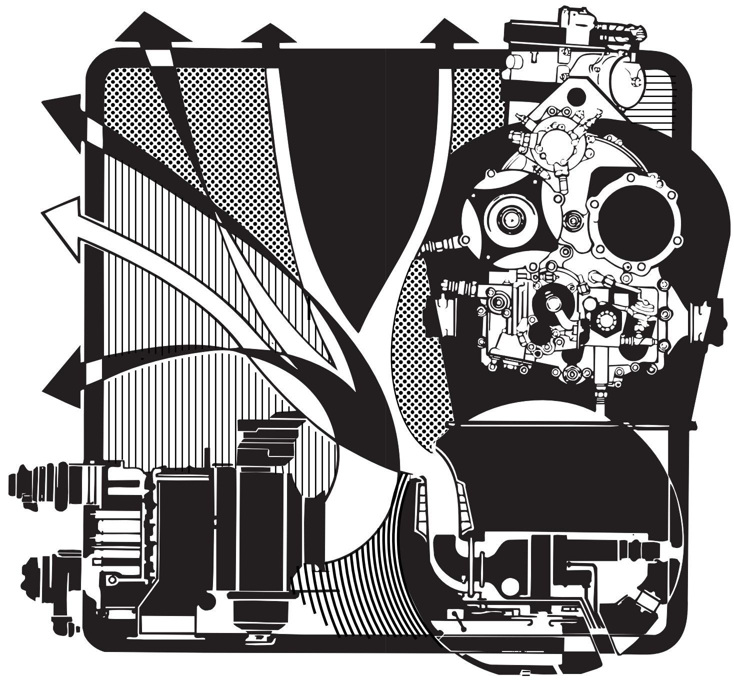
7. In case of a power loss to the fuel flow indicators:
   A. The digital indication goes blank.
   B. The digital indication reads LOW.
   C. The digital indication reads HIGH.
   D. The digital indication reads OFF.

CHAPTER 6
AUXILIARY POWER UNIT

CONTENTS

ILLUSTRATIONS

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.

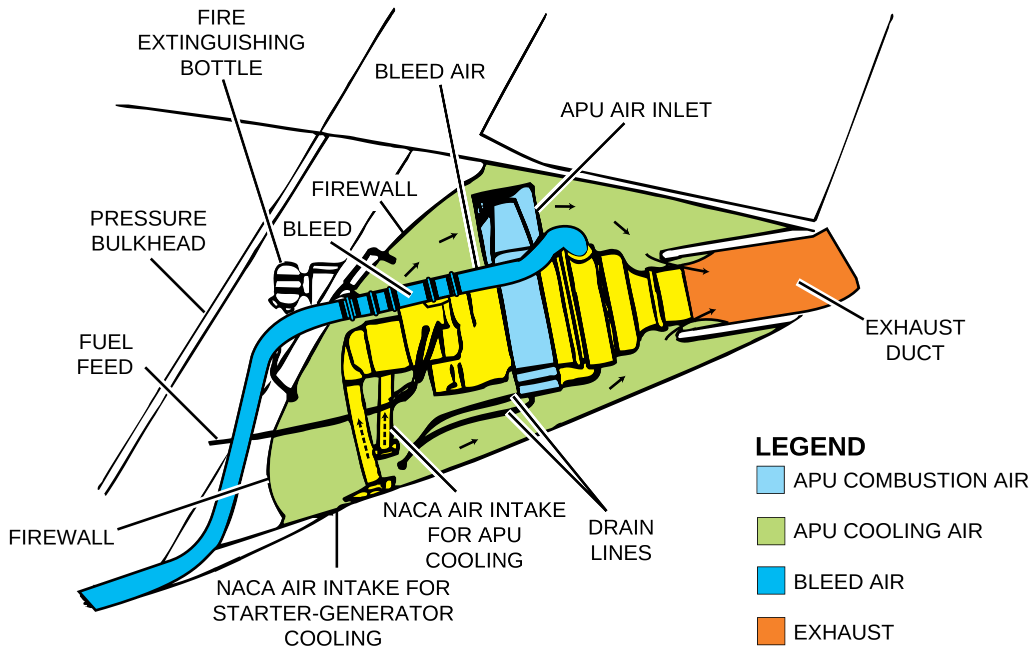


Figure 6-1. APU Layout

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.

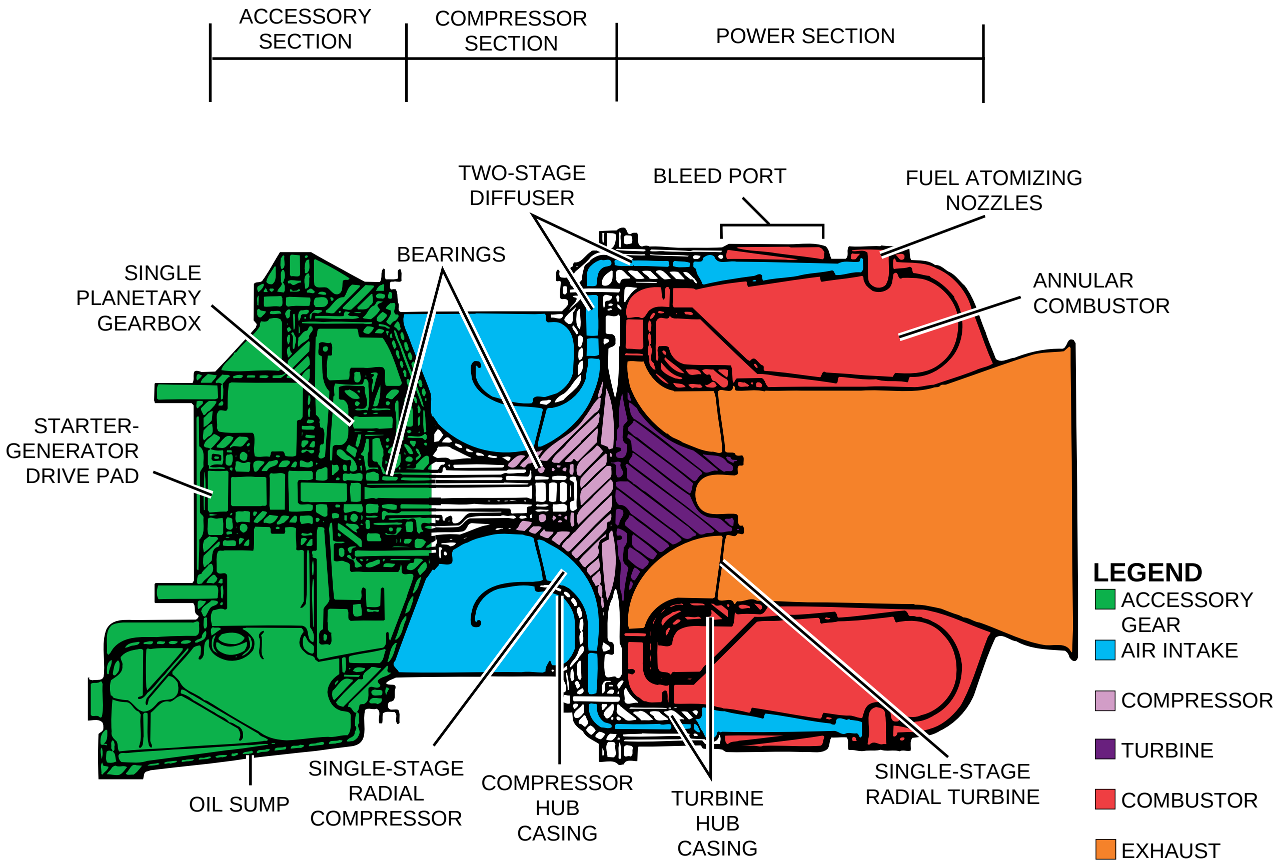


Figure 6-2. Major Sections

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.

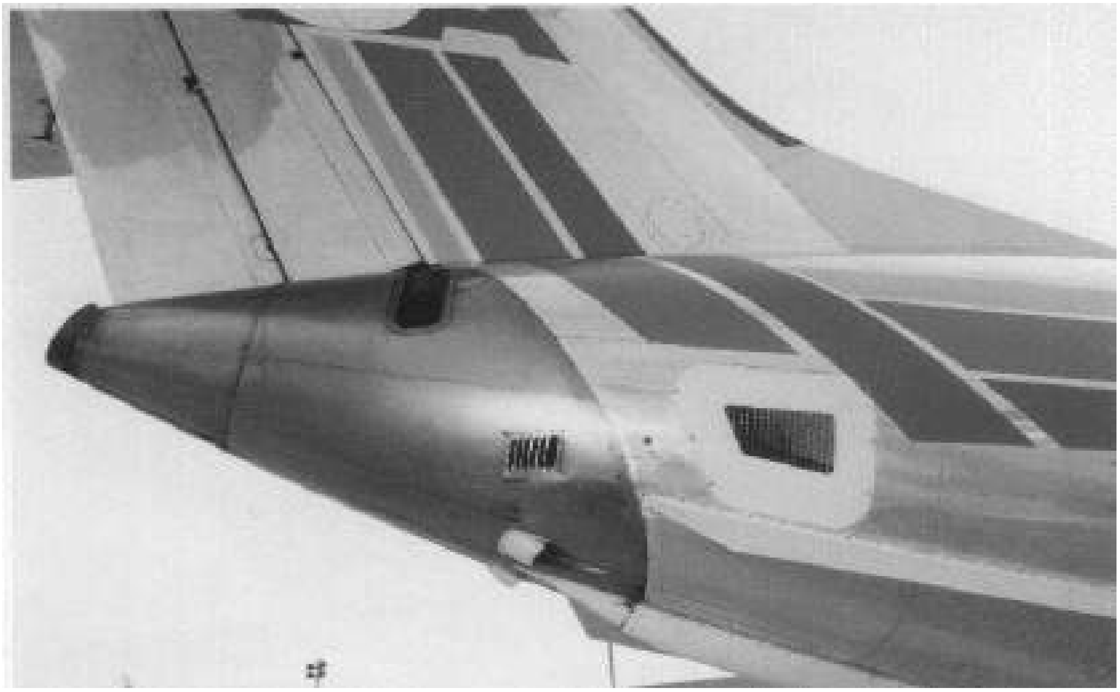


Figure 6-3. APU Installation (External)

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.

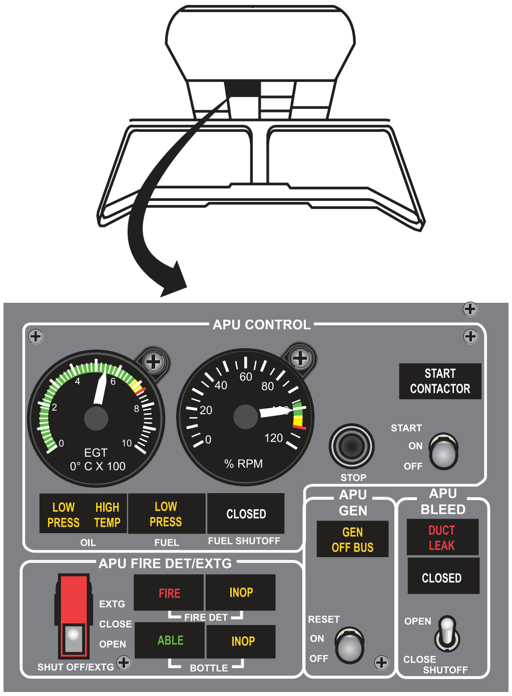


Figure 6-4. APU Panel

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:

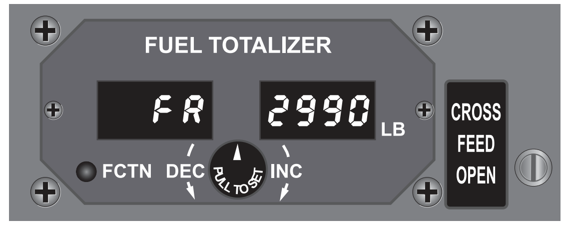


Figure 6-5. Fuel Totalizer

- 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.

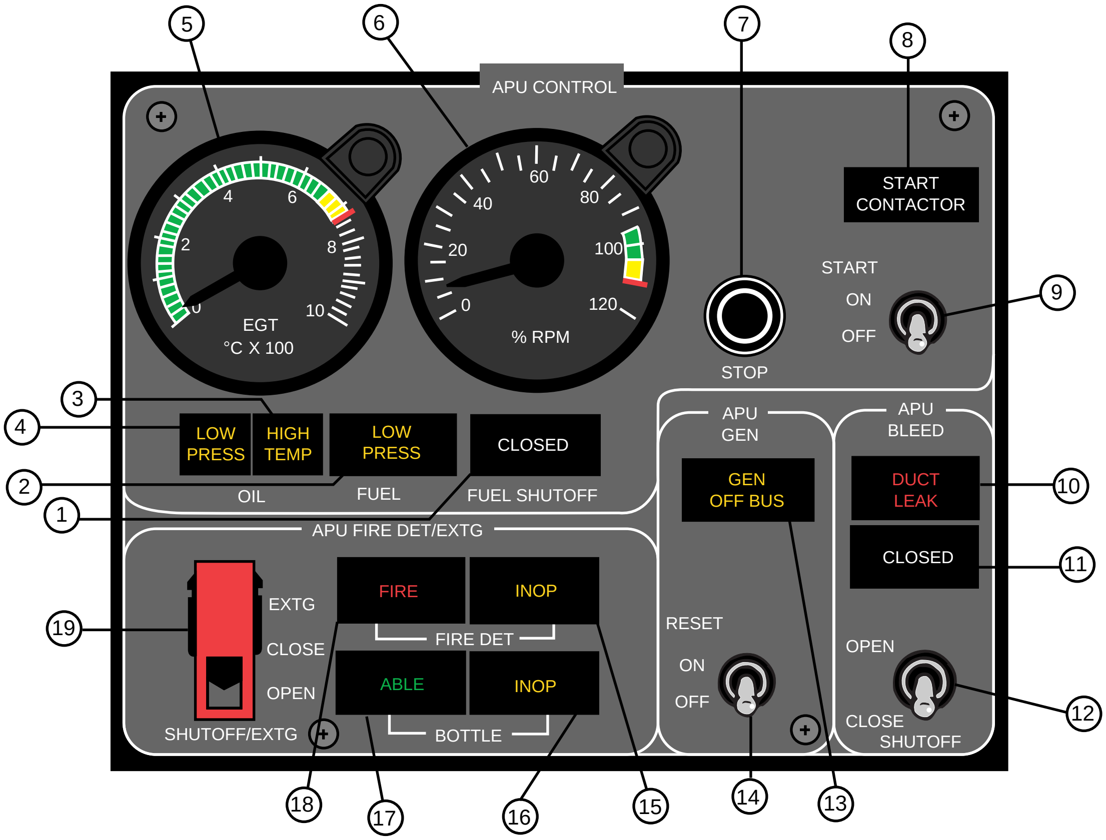
- The APU bleed logic is set up so that the APU cannot supply bleed air to the same air conditioning pack that is being supplied by engine bleed. In this case, priority is given to the engine and the APU bleed closes automatically.

- 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. Illuminates only during fire detection extinguishing test sequence.
18. APU FIRE warning light (red)
    Illuminated—Fire/overheat condition is sensed by the detector. 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

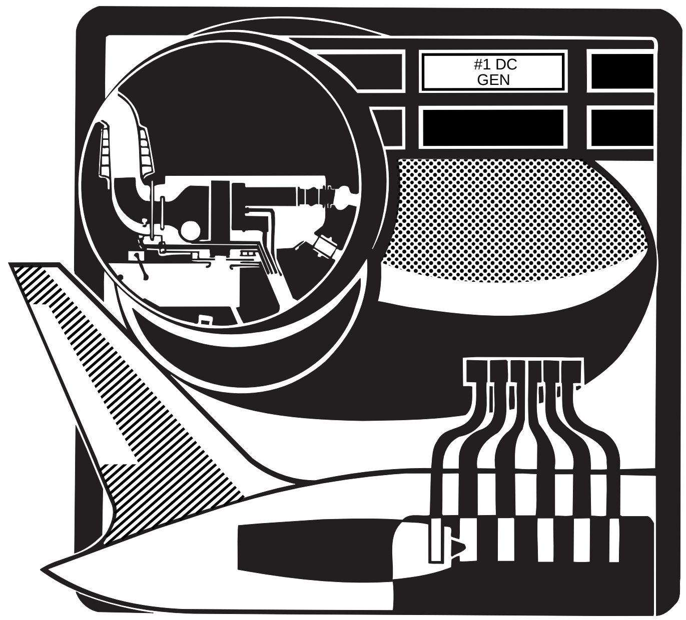
5. The APU can be operated in flight:
   A. True
   B. False

CHAPTER 7
POWERPLANT

CONTENTS

ILLUSTRATIONS

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.

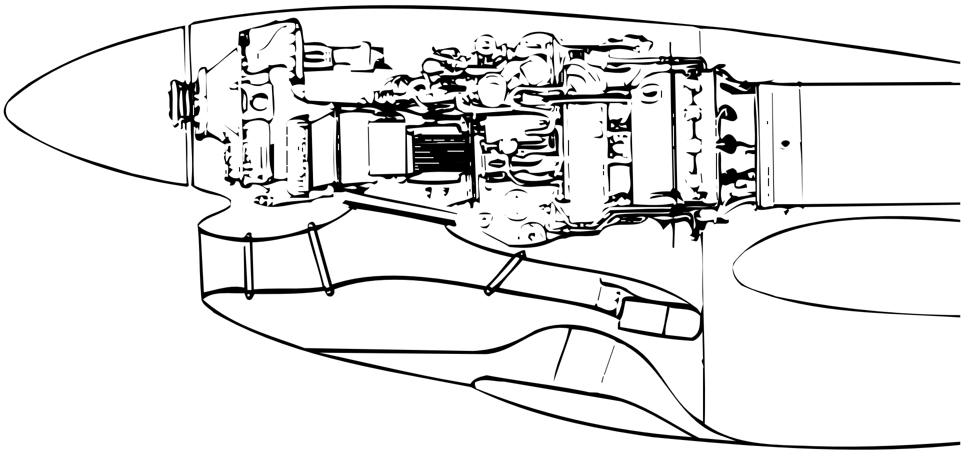


Figure 7-1. Engine Installation

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.

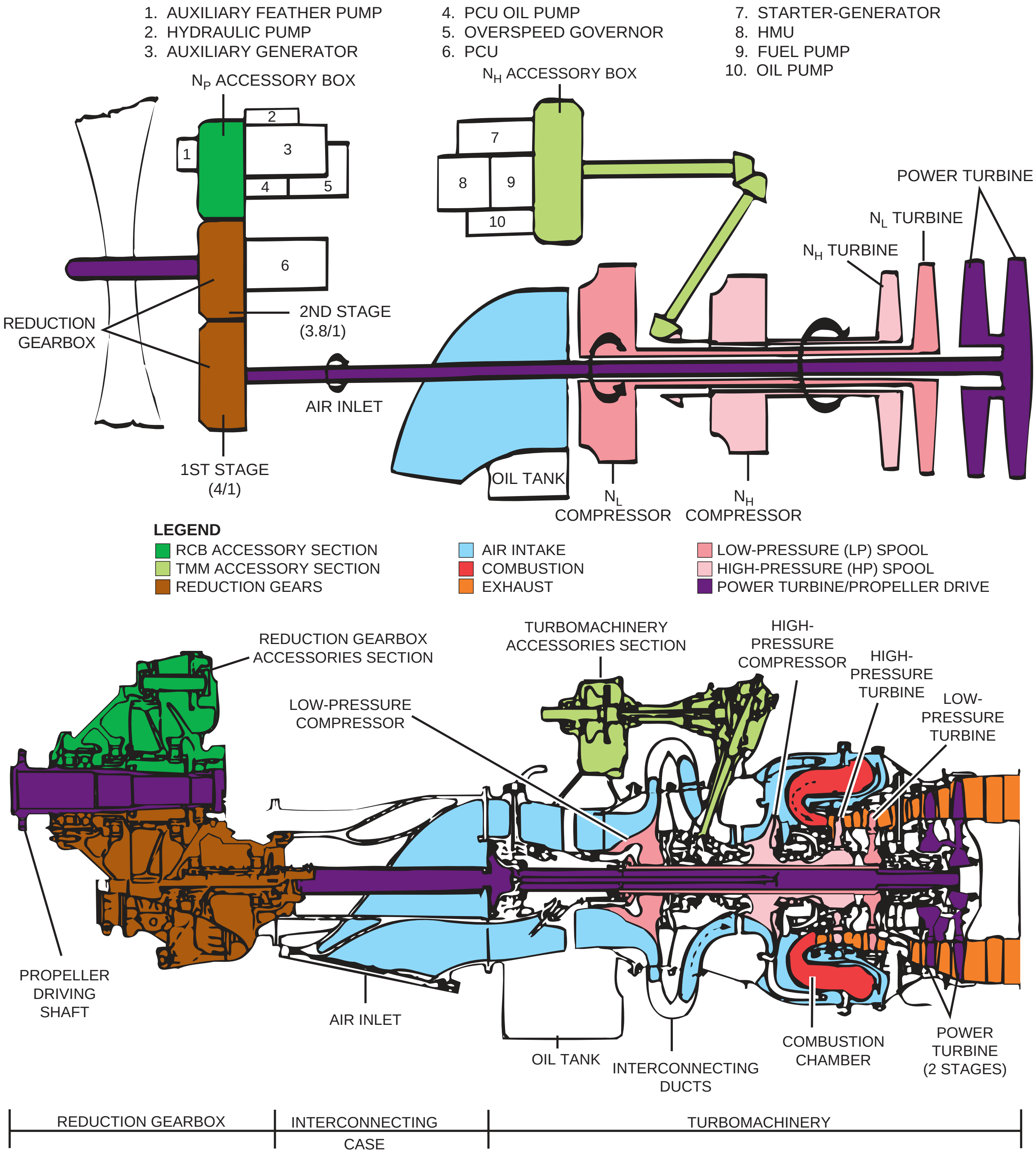


Figure 7-2. Engine Layout

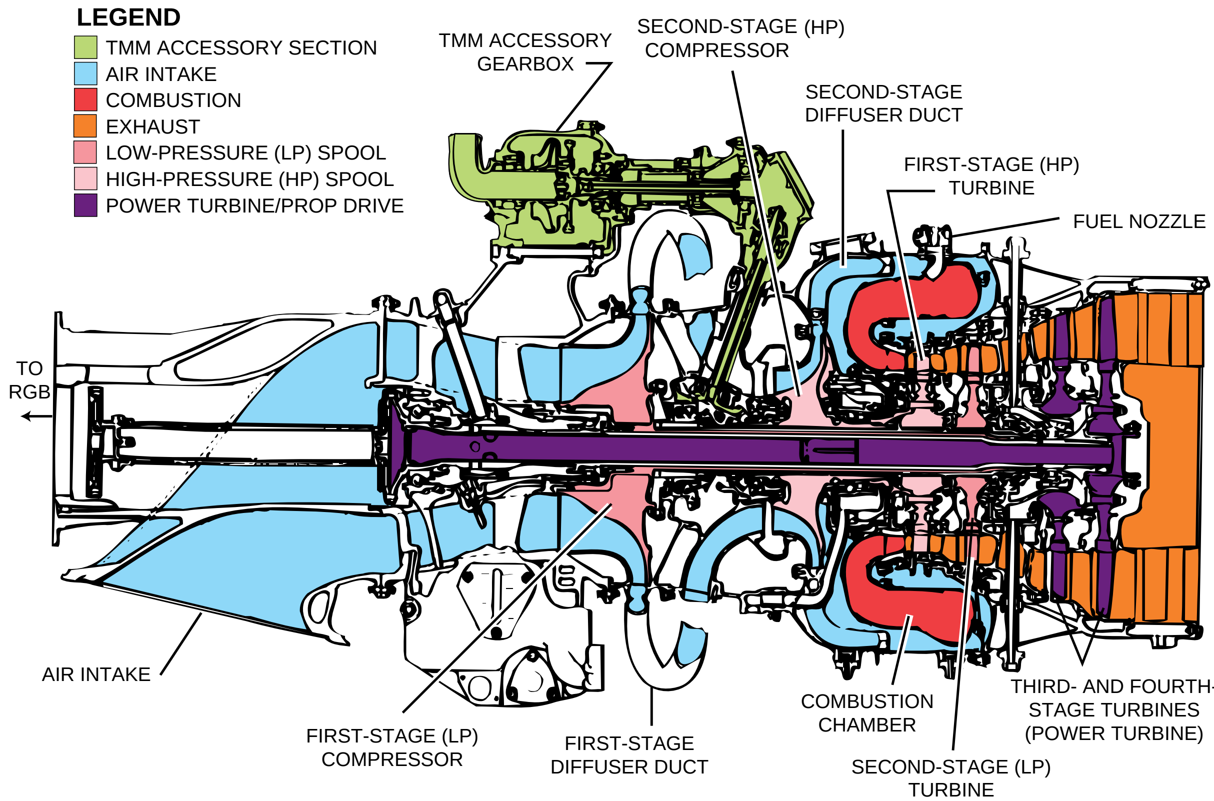


Figure 7-3. Engine Cross Section (Typical)

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.

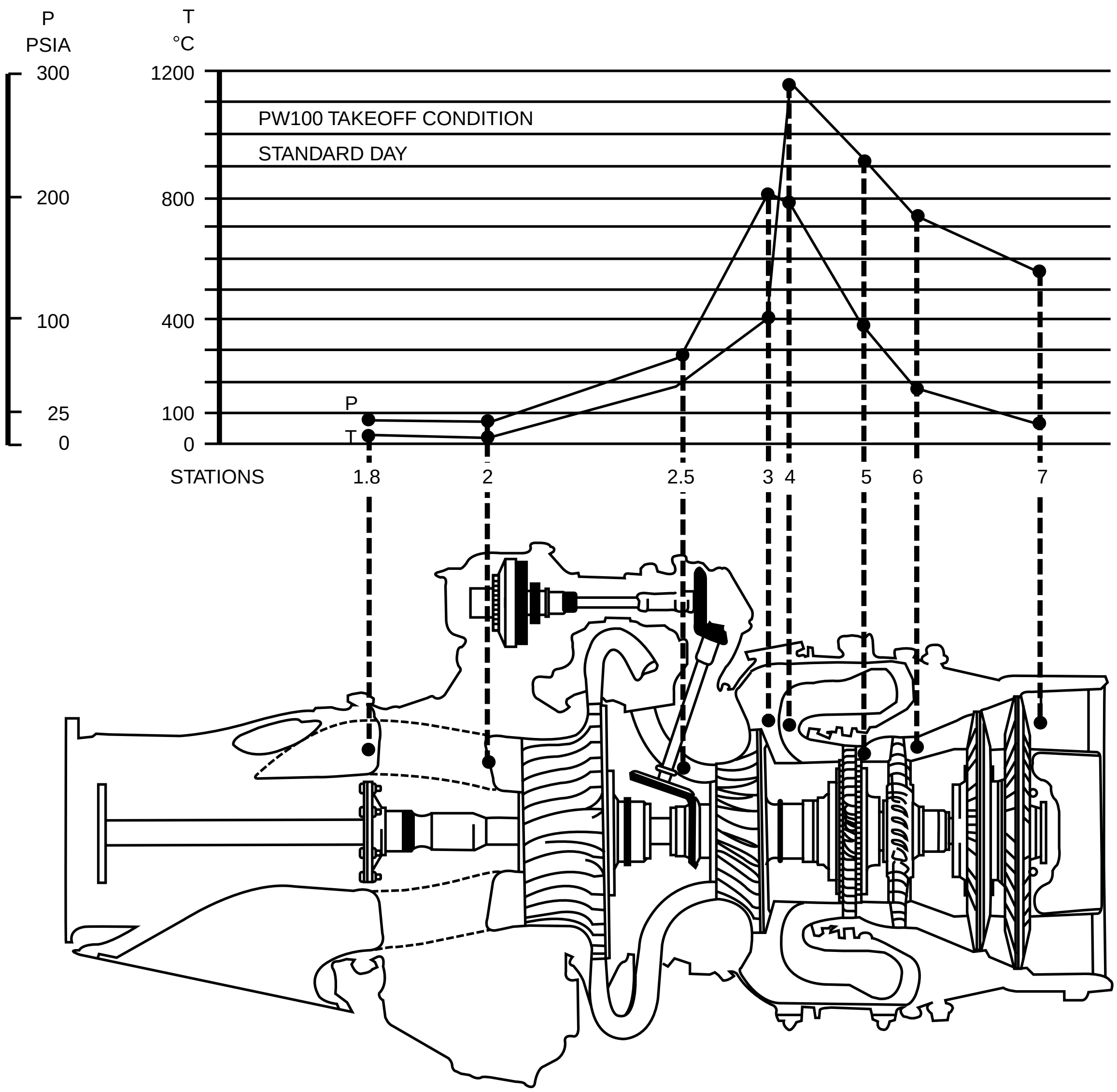


Figure 7-4. Stations

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

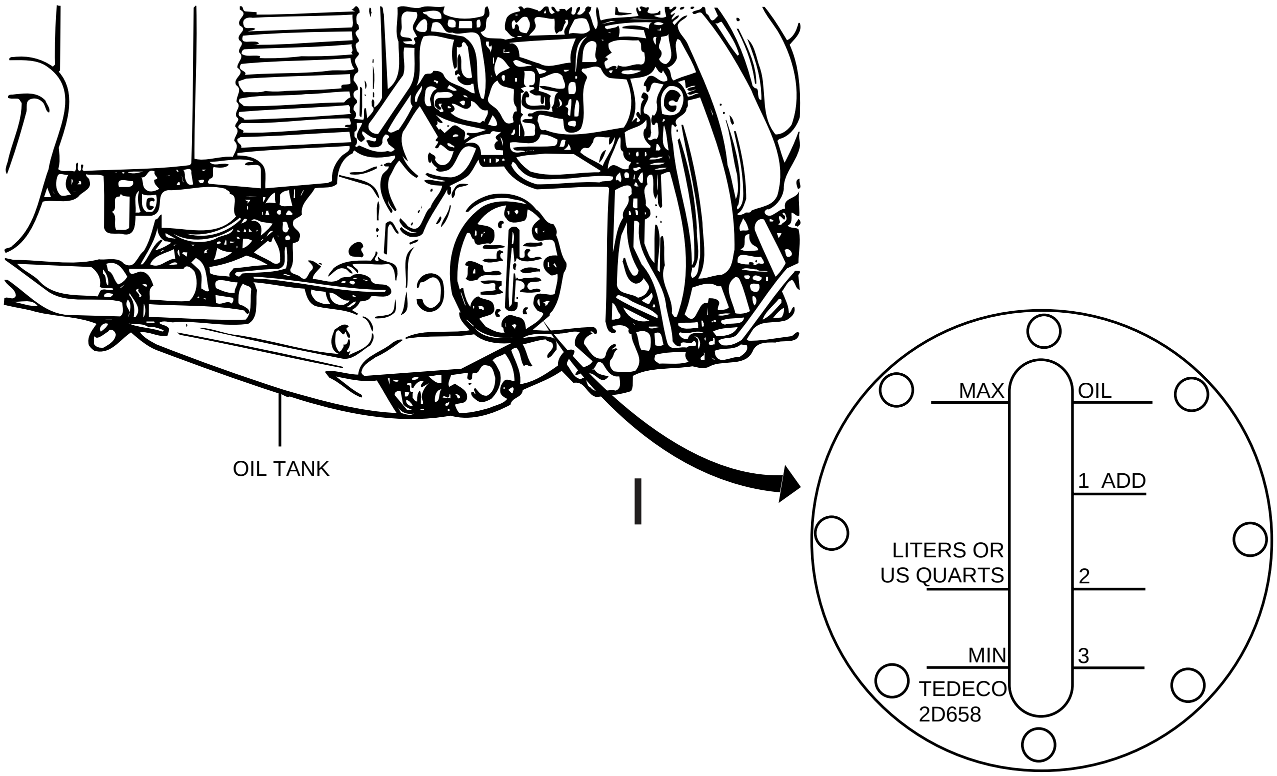


Figure 7-6. Engine Oil Tank/Sight Glass

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.

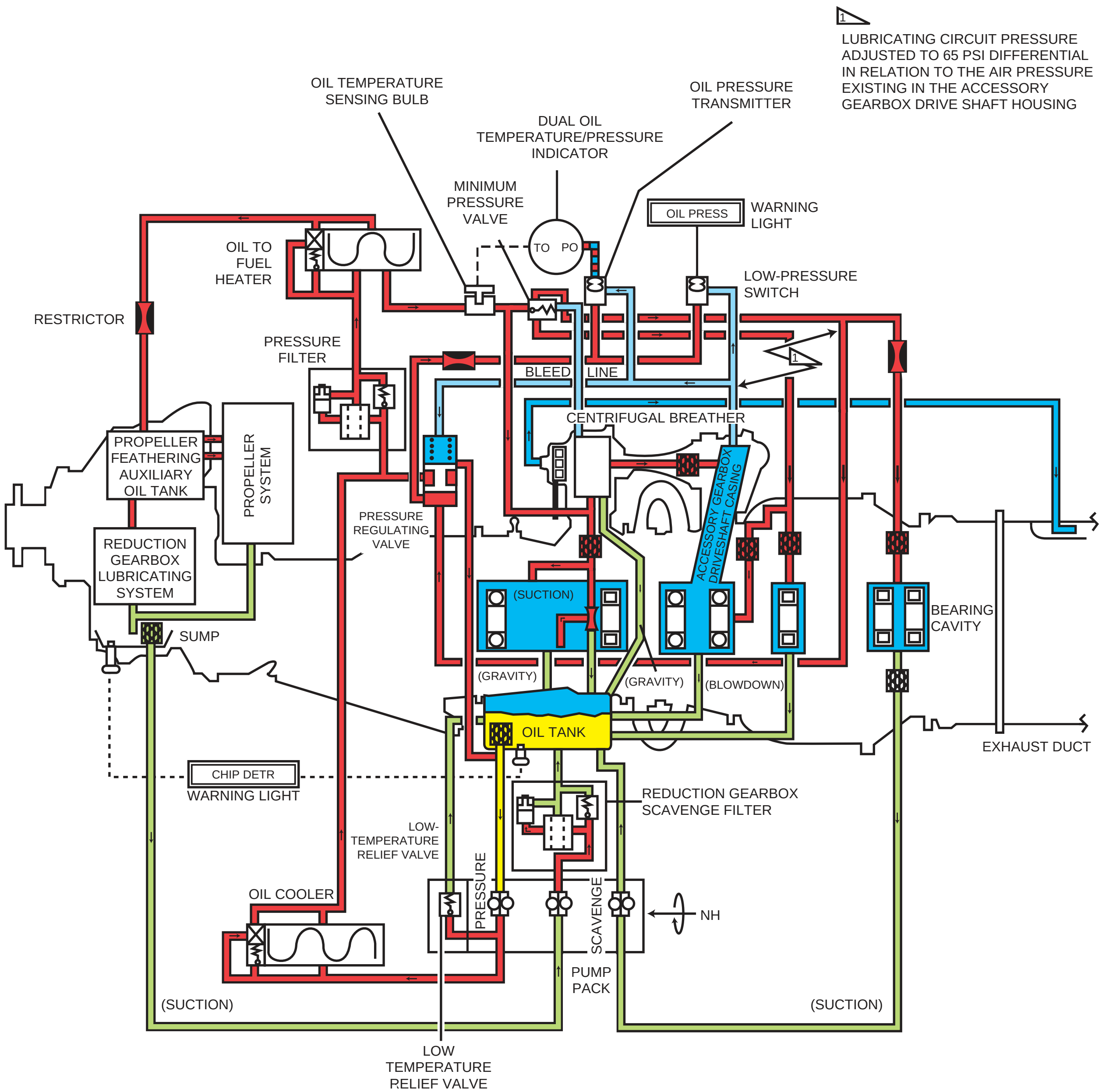
The tank is automatically filled by the engine oil system and when the engine is running or dry motored by the starter-generator.

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.



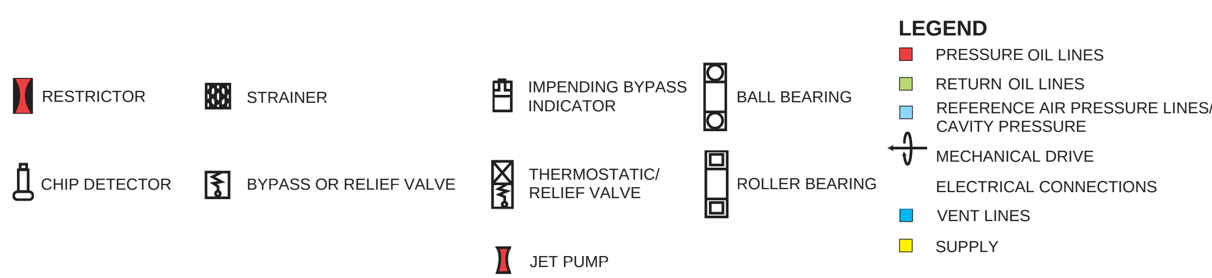


Figure 7-7. Engine Oil System

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

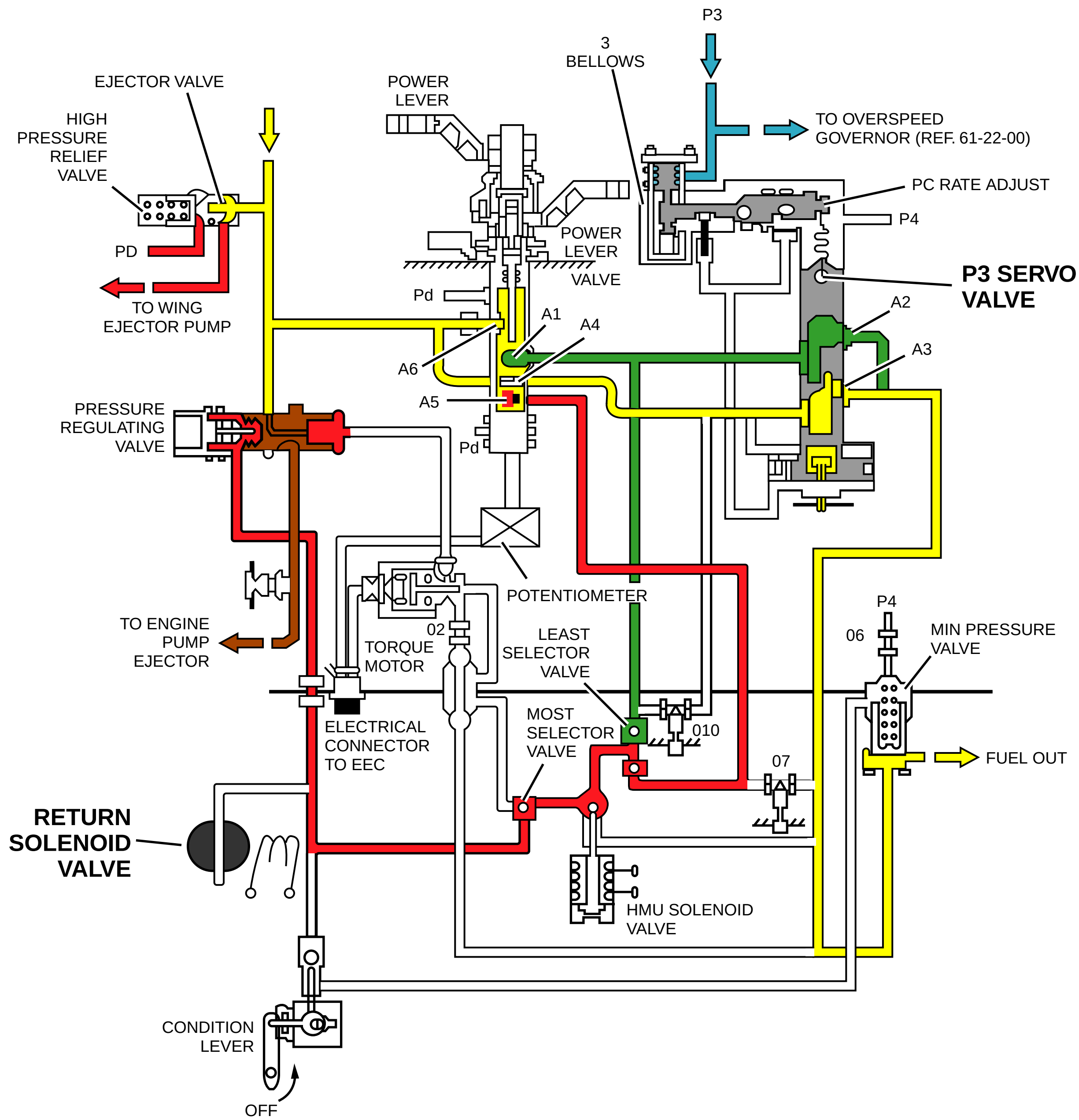


Figure 7-8. Hydromechanical Metering Unit (HMU)

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

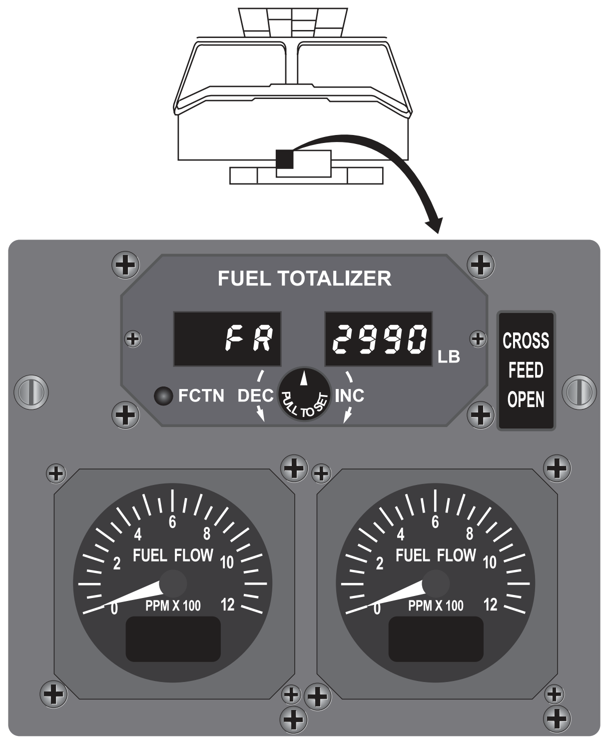


Figure 7-9. Fuel Flow Indicators

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

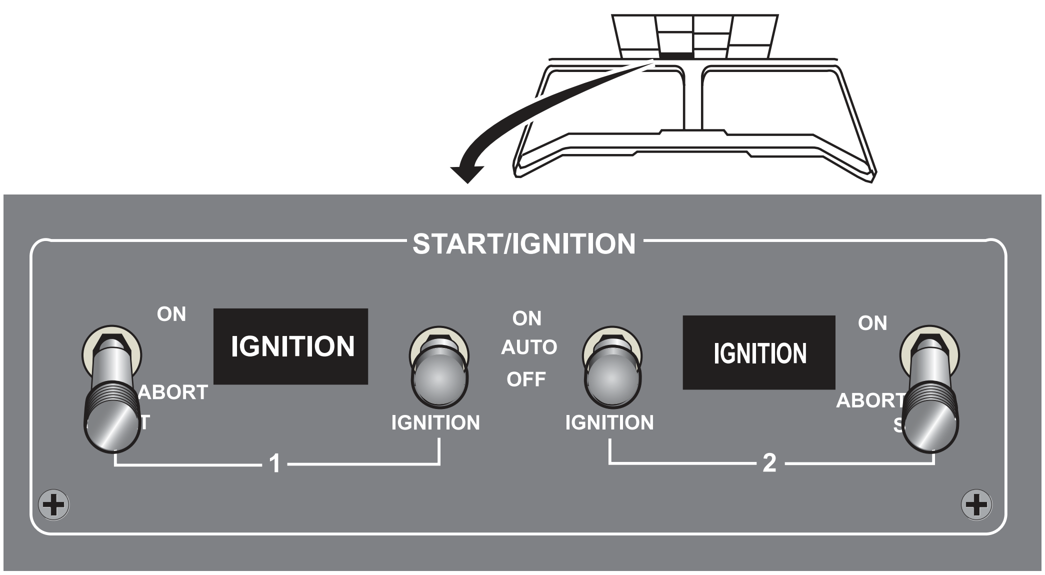


Figure 7-10. Start/Ignition Panel

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

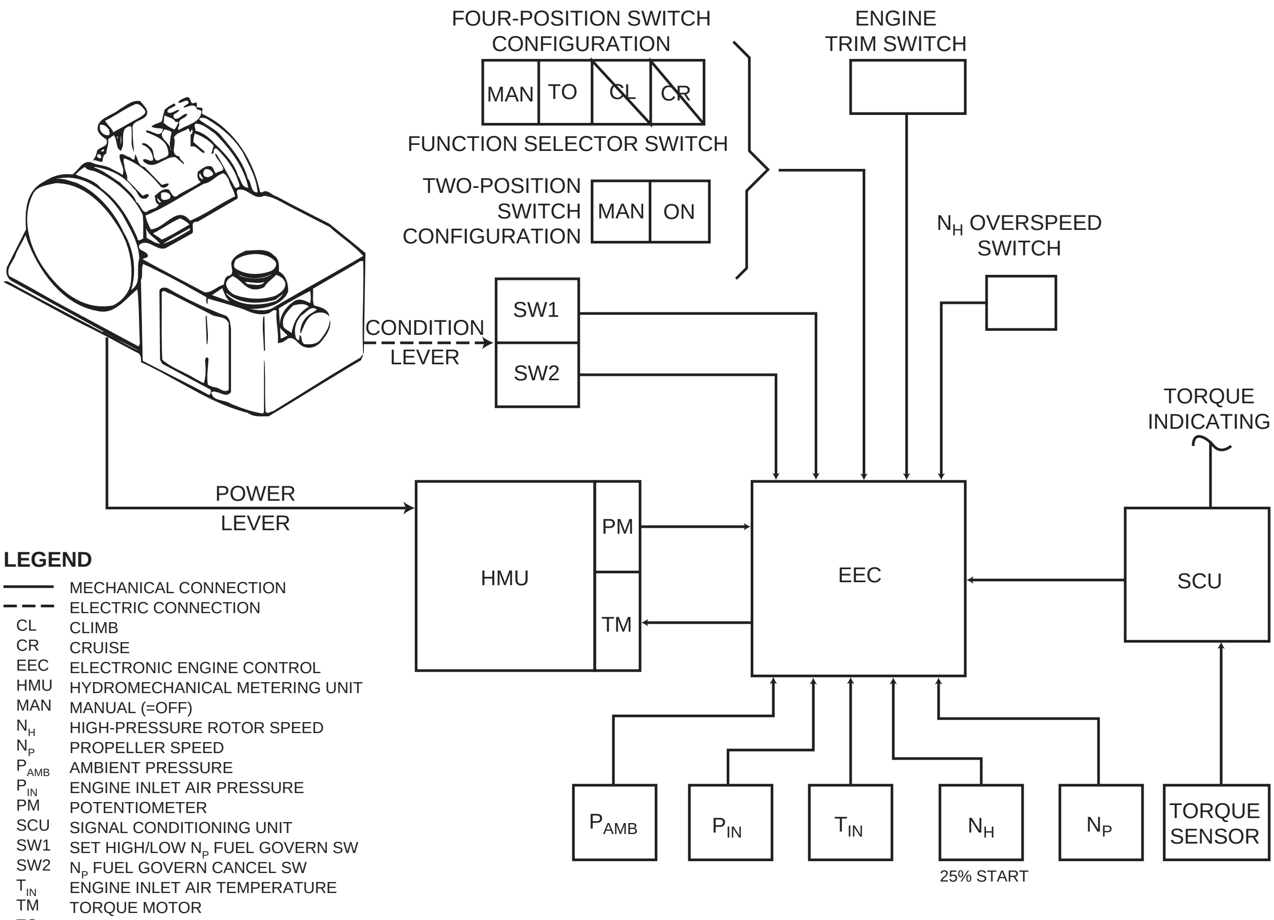
Electronic Engine Control (EEC)

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).



LEGEND

———	MECHANICAL CONNECTION
– – –	ELECTRIC CONNECTION
CL	CLIMB
CR	CRUISE
EEC	ELECTRONIC ENGINE CONTROL
HMU	HYDROMECHANICAL METERING UNIT
MAN	MANUAL (=OFF)
N_H	HIGH-PRESSURE ROTOR SPEED
N_P	PROPELLER SPEED
P_{AMB}	AMBIENT PRESSURE
P_{IN}	ENGINE INLET AIR PRESSURE
PM	POTENTIOMETER
SCU	SIGNAL CONDITIONING UNIT
SW1	SET HIGH/LOW N_P FUEL GOVERN SW
SW2	N_P FUEL GOVERN CANCEL SW
T_{IN}	ENGINE INLET AIR TEMPERATURE
TM	TORQUE MOTOR
TO	TAKEOFF (=ON)

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.

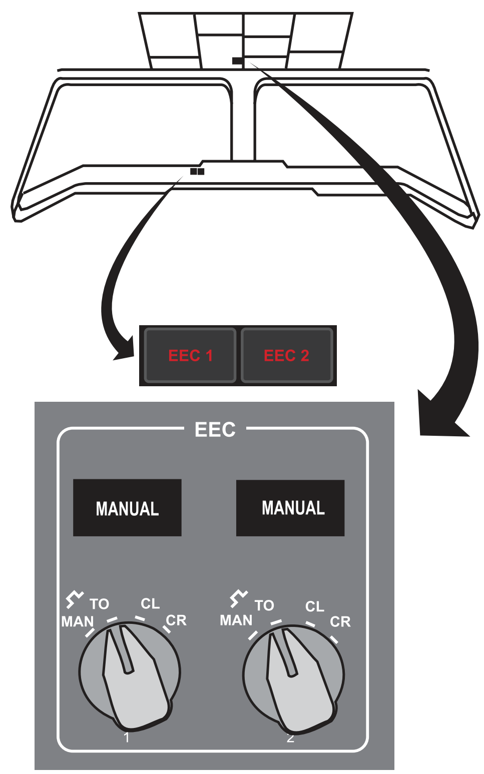


Figure 7-12. EEC Controls and Indicators

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.

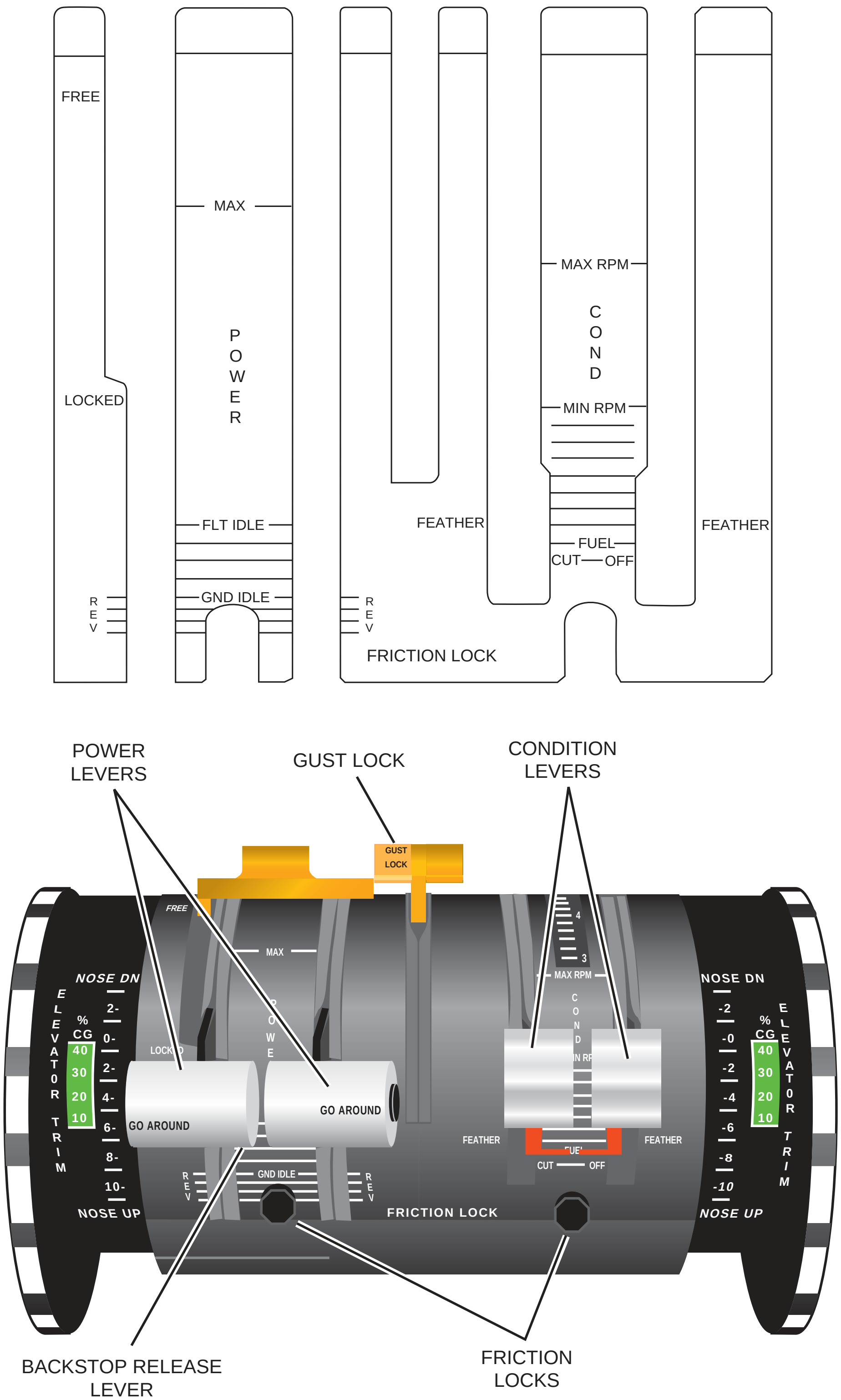


Figure 7-13. Center Pedestal Console

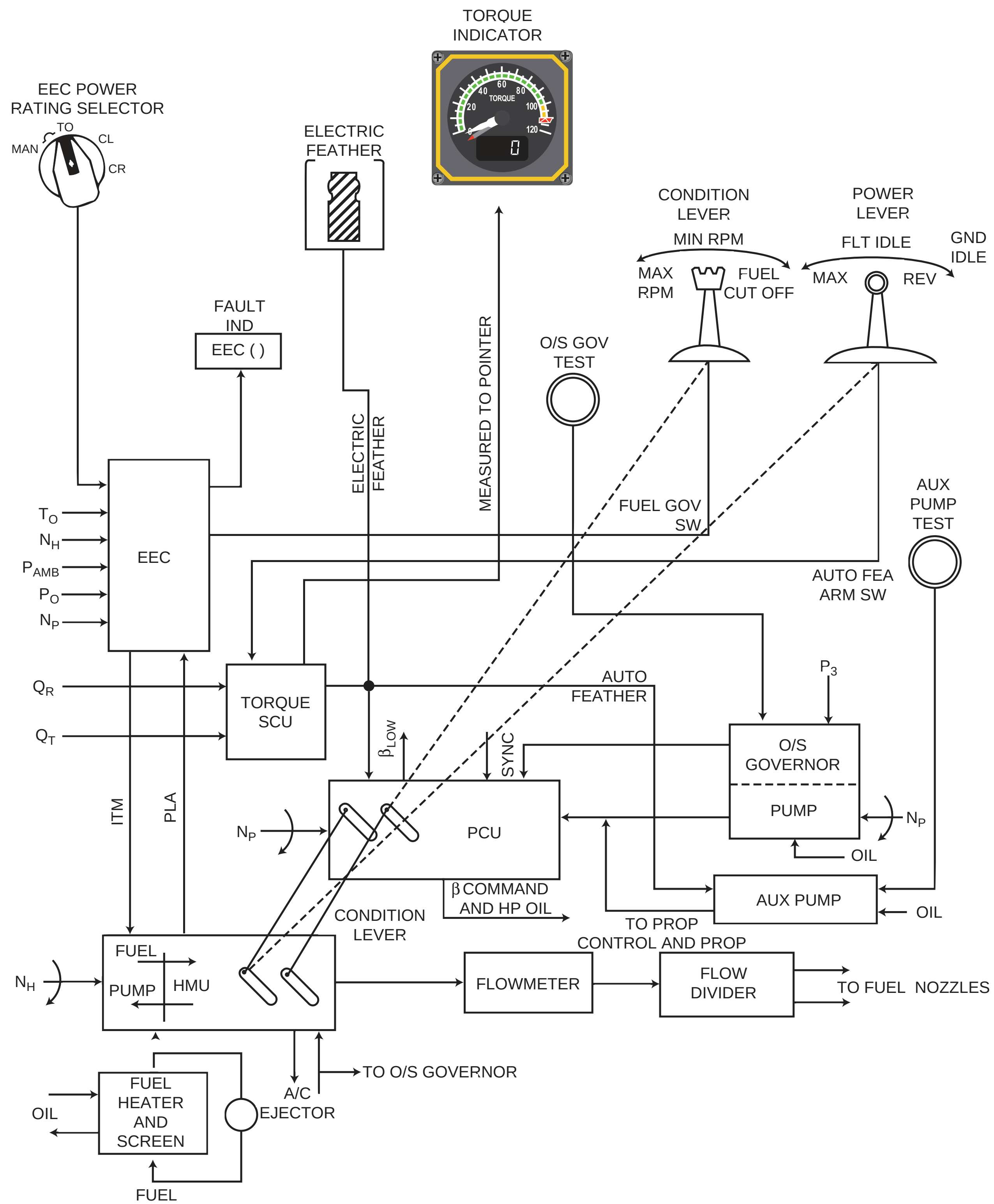


Figure 7-14. Engine Control System Schematic

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).

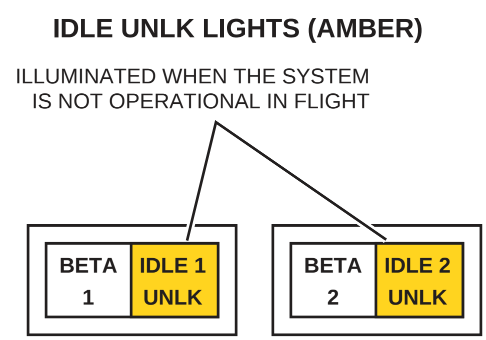


Figure 15. IDLE 1 (2) UNLK Light

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.

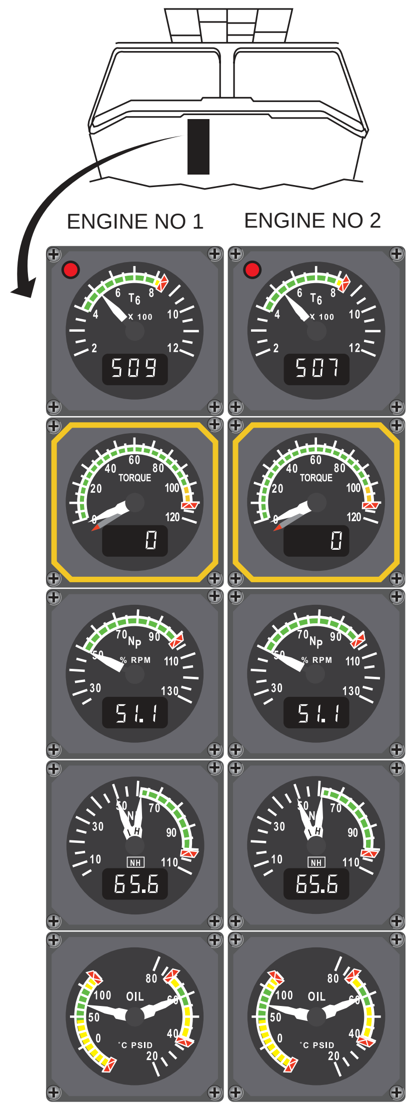


Figure 7-16. Engine Instrument Location

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.

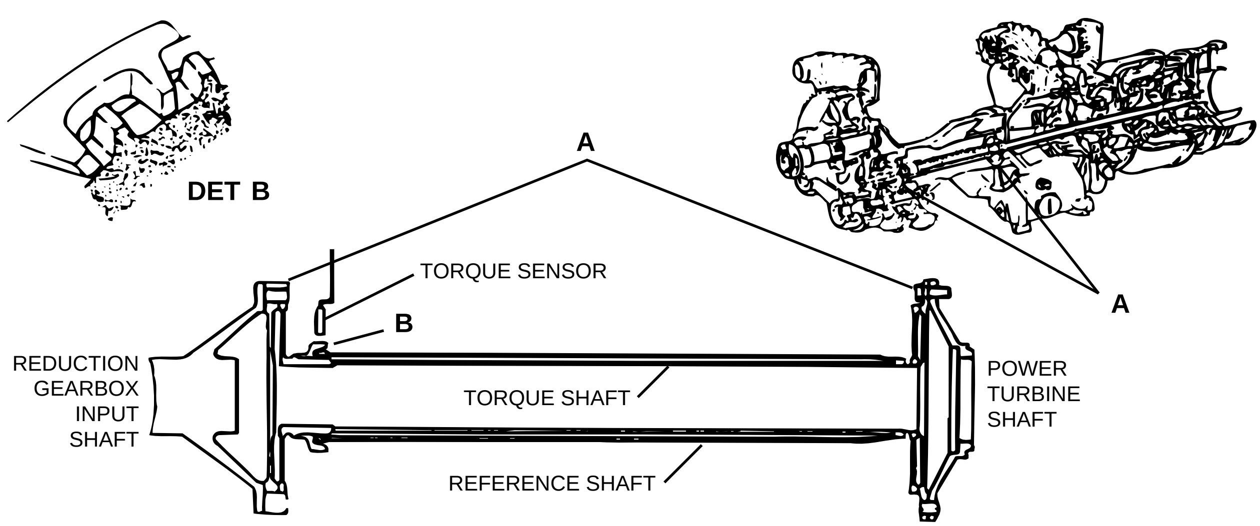


Figure 7-17. Torque Indicating System

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

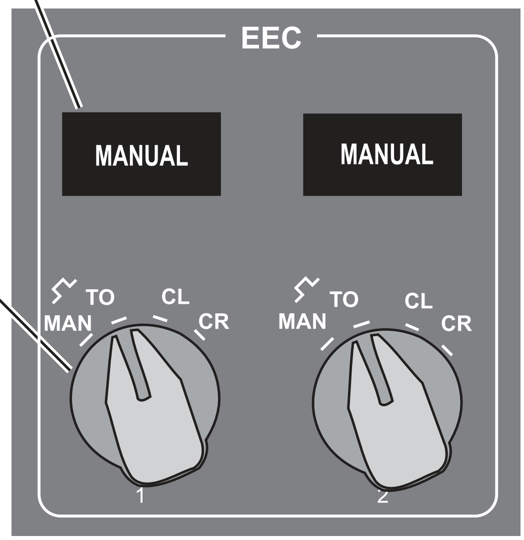
YELLOW ARC: 40 TO 55 PSID
AND 65 TO 70 PSID

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



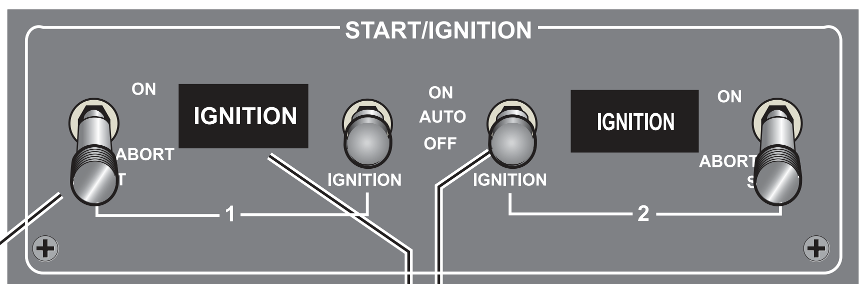
POWER RATING SELECTOR

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 LIGHT (WHITE)

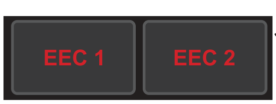
ILLUMINATED—WHEN THE IGNITION CIRCUIT IS ENERGIZED

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).



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)

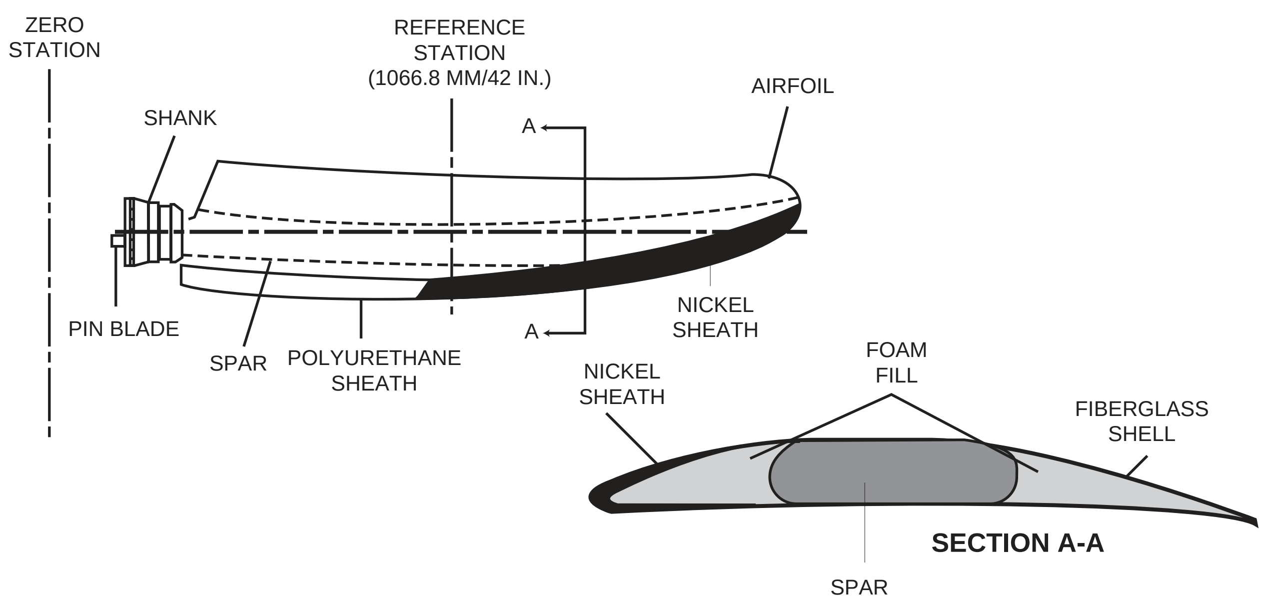


Figure 7-21. Blade Construction

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.

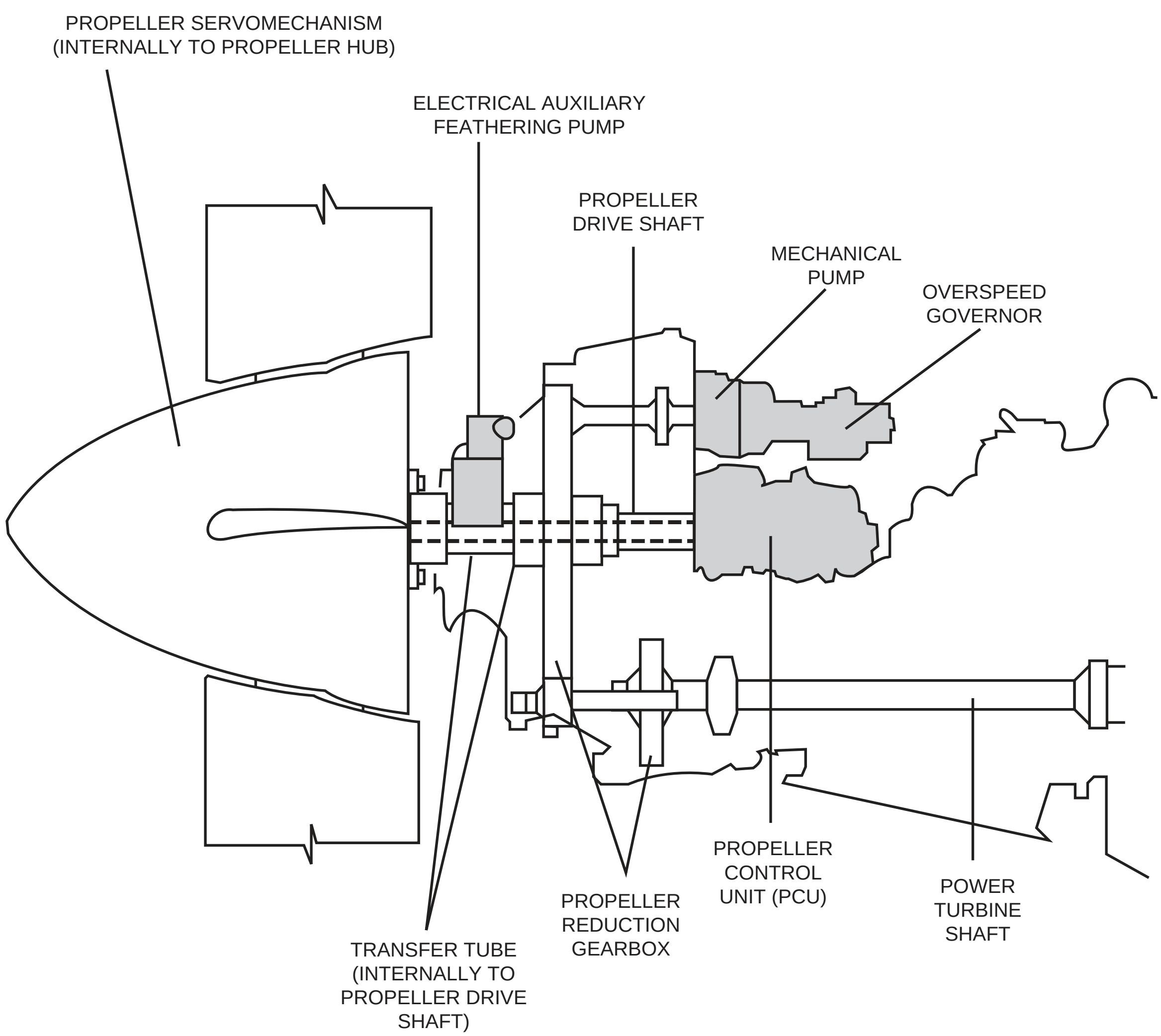


Figure 7-22. Propeller Control Components

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).

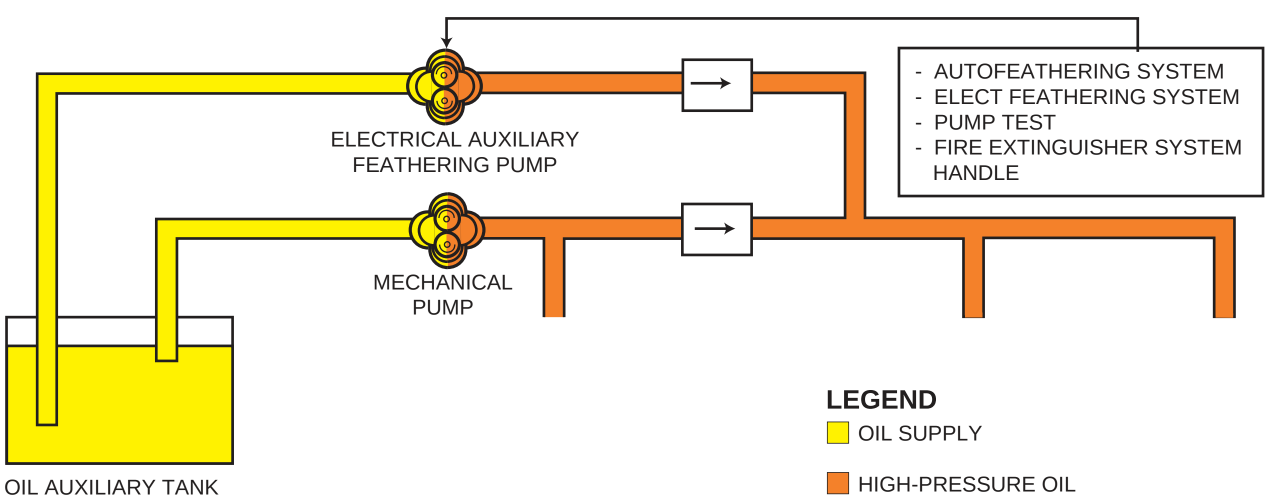


Figure 7-23. Propeller Oil Supply

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.

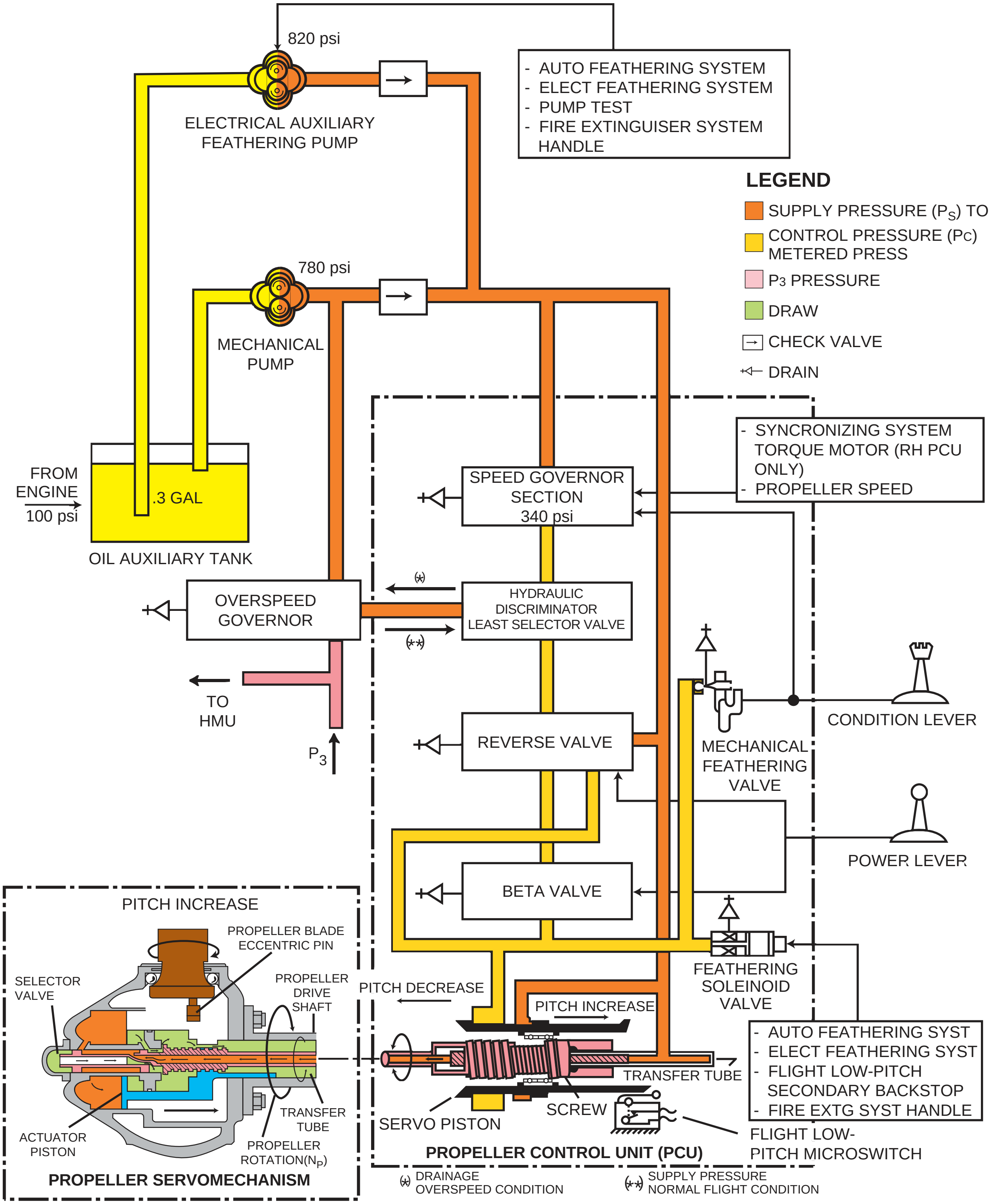


Figure 7-24. Propeller Control System

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

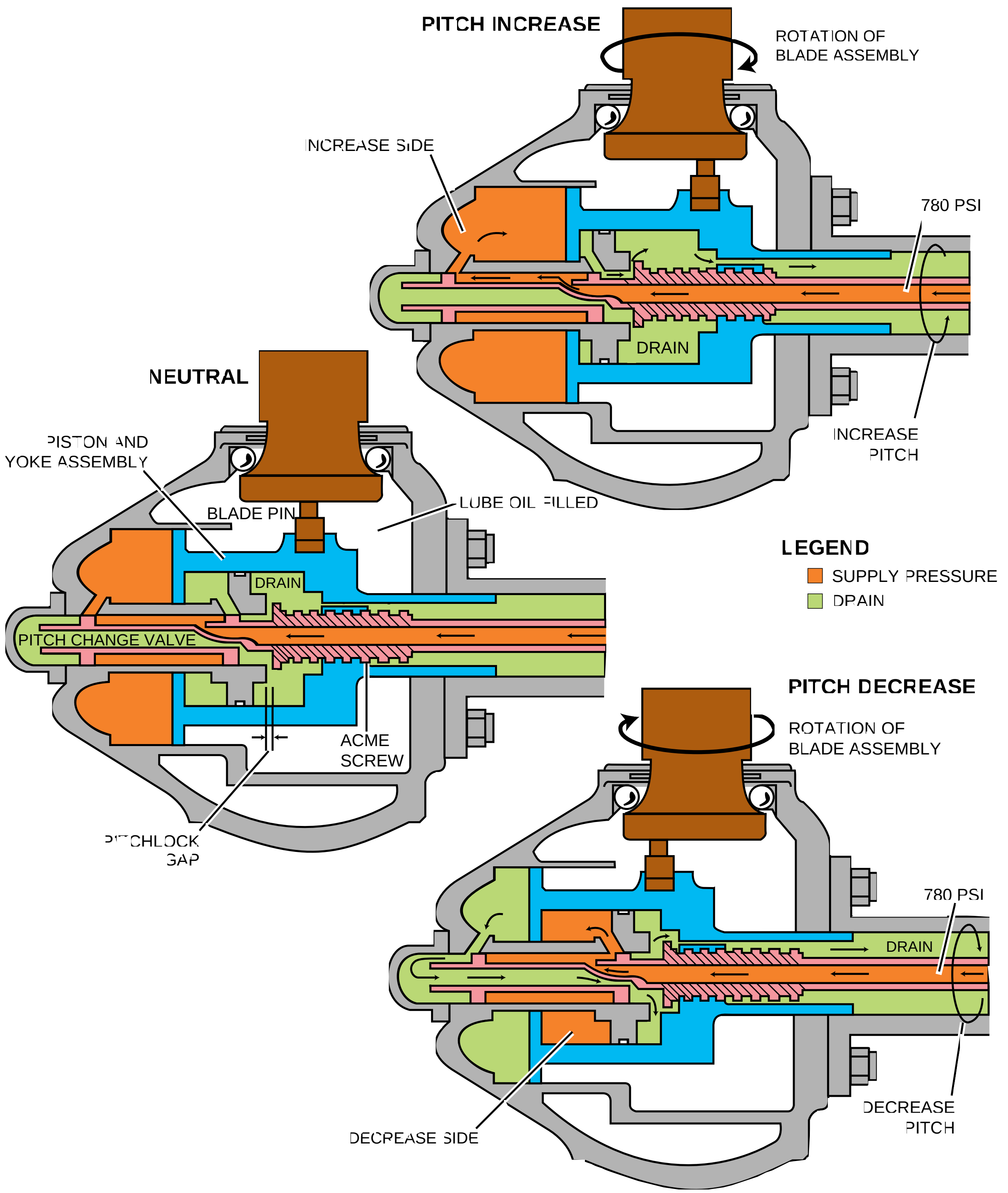


Figure 7-25. Propeller Servomechanism during Pitch Change Operations

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 ball-screw 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.

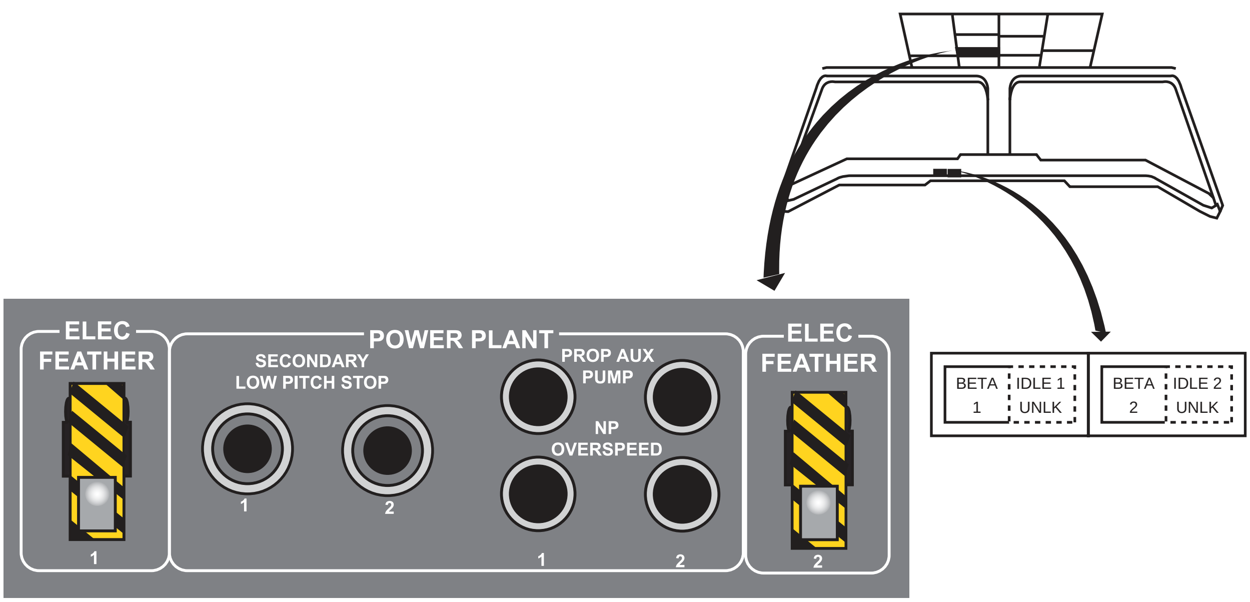


Figure 7-26. Powerplant Control Panel

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.

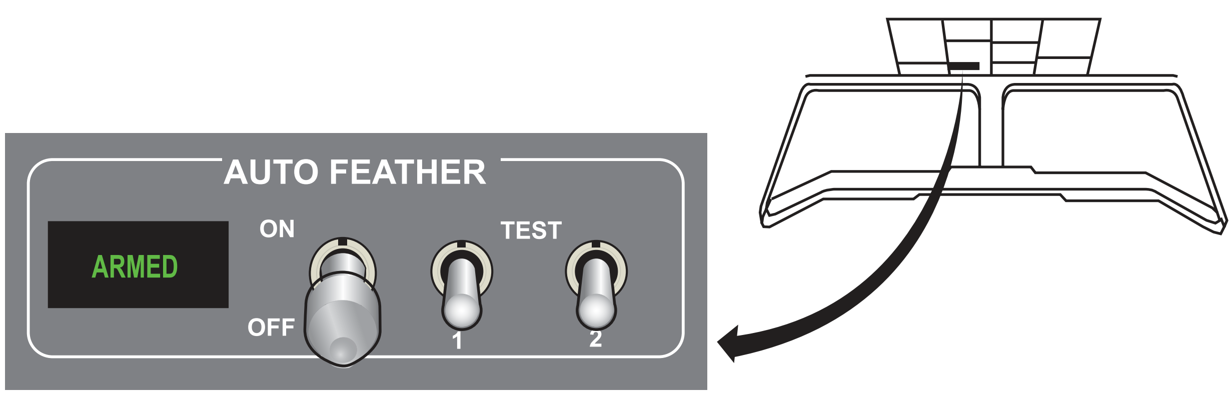


Figure 7-27. AUTO FEATHER Control Panel

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.

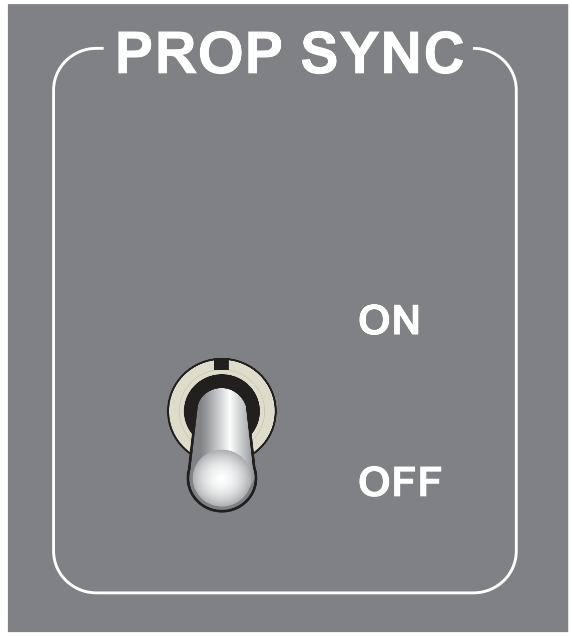


Figure 7-28. Synchronization Control Panel

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.

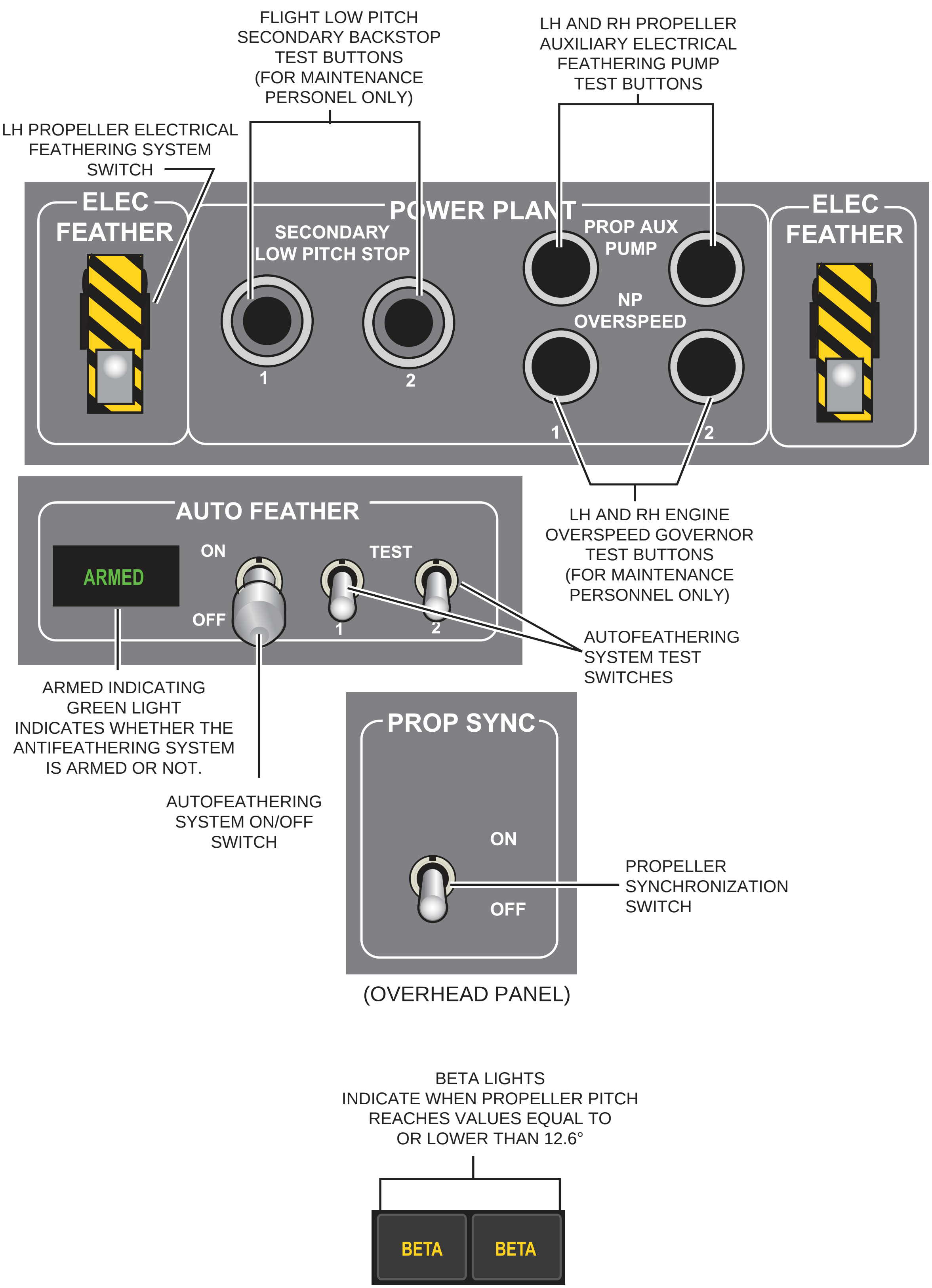


Figure 7-29. Propeller System Controls and Indicators

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

ILLUSTRATIONS

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.

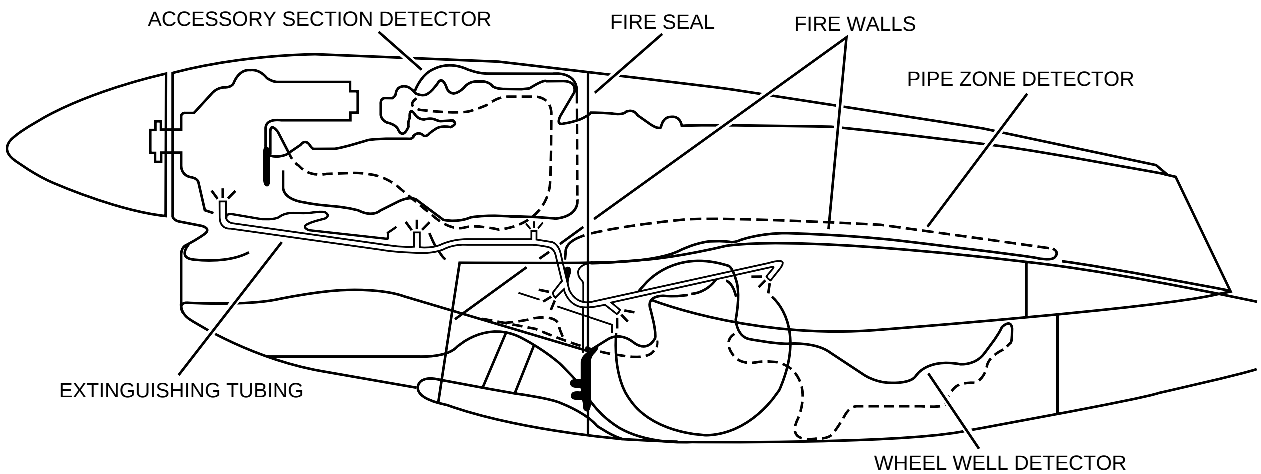


Figure 8-1. Engine/Wheel Well Fire Detector Location

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.

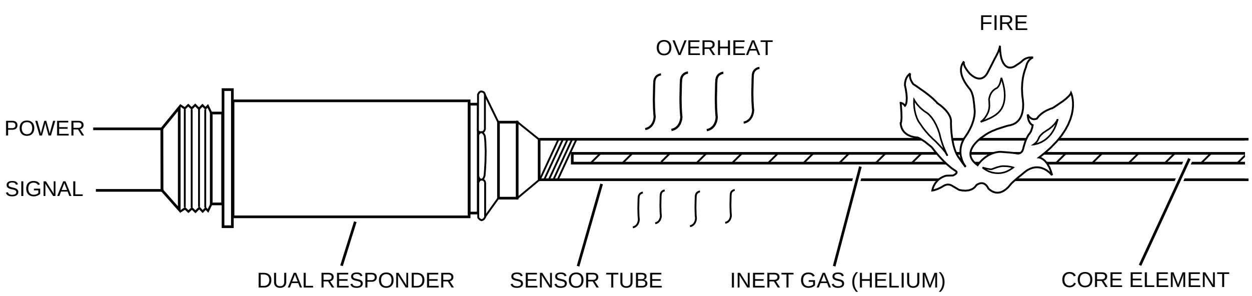


Figure 8-2. Fire Detector