FlightSafety international

GULFSTREAM G150 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 G150 Series and other Gulfstream aircraft are taught at the following FlightSafety learning center:

FlightSafety International
DFW Learning Center
3201 East Airfield Drive
DFW Airport, TX 75261-9428
(972) 534-3200

INSERT LATEST REVISED PAGES, DESTROY SUPERSEDED PAGES

LIST OF EFFECTIVE PAGES

Dates of issue for original and changed pages are:

Original ...... 0........... July 2006
Revision..... 1....... August 2006
Revision .... 2........ March 2008

NOTE:

For printing purposes, revision numbers in footers occur at the bottom of every page that has changed in any way (grammatical or typographical revisions, reflow of pages, and other changes that do not necessarily affect the meaning of the manual).

THIS PUBLICATION CONSISTS OF THE FOLLOWING:

Page No.	*Revision No.
Cover	0
Copyright	2
iii–iv	2
v–viii	0
1-i–1-iv	0
1-1	2
1-2	0
1-3–1-6	2
2-i	0
2-ii–2-iv	2
2-iv	0
2-1	2
2-2	0
2-3–2-20	2
3-i–3-iv	0
3-1–3-2	2
3-3	0
3-4	2
3-5–3-10	0
3-11	2
3-12	0
4-i–4-iv	0
4-1–4-5	2
4-6–4-7	0
4-8–4-9	2
4-10	0
4-11–4-16	2

Page No.	*Revision No.
5-i	0
5-ii–5-iv	2
5-1–5-3	2
5-4–5-5	0
5-6–5-16	2
5-17–5-18	0
6-i–6-iv	0
6-1	2
6-2	0
6-3–6-10	2
7-i–7-iv	0
7-1	2
7-2	0
7-3–7-4	2
7-5–7-6	0
7-7	2
7-8	0
7-9	2
7-10–7-11	0
7-12	2
7-13	0
7-14–7-15	2
7-16	0
8-i–8-ii	0
8-iii–8-iv	2
8-1–8-3	2
8-4	0

*Zero in this column indicates an original page.

Page No.	*Revision No.
8-5–8-6	2
8-7	0
8-8–8-12	2
8-13	0
8-14–8-15	2
8-16–8-17	0
8-18–8-19	2
8-20	0
9-i–9-iv	2
9-1–9-14	2
10-i–10-iv	0
10-1–10-2	2
10-3	0
10-4	2
10-5–10-9	0
10-10–10-12	2
10-13–10-15	0
10-16–10-17	2
10-18	0
11-i–11-iv	2
11-1	2
11-2	0
11-3–11-10	2
12-i–12-ii	2
12-iii–12-iv	0
12-1–12-2	2
12-3	0
12-4	2
12-5–12-7	0
12-8	2
12-9	0
12-10–12-12	2
13-i–13-ii	0
13-iii–13-iv	2
13-1–13-3	2
13-4	0
13-5	2
13-6	0
13-7–13-8	2
13-9	0
13-10	2
13-11	0
13-12	2
13-13–13-14	0
13-15	2
13-16	0

Page No.	*Revision No.
13-17	2
13-18	0
14-i–14-iv	2
14-1	2
14-2–14-4	0
14-5	2
14-6	0
14-7	2
14-8–14-10	0
14-11–14-30	2
15-i–15-iv	2
15-1	2
15-2–15-13	0
15-14–15-21	2
15-22	0
16-i	0
16-0–16-2	0
17-i–17-iv	0
17-1	2
17-2	0
17-3–17-4	2
17-5–17-6	0
17-7	2
17-8	0
17-9–17-14	2
MAP-i	2
MAP-ii–MAP-iv	0
MAP-1–MAP-2	0
MAP-3–MAP-5	1
MAP-6–MAP-8	0
MAP-9–MAP-10	2
MAP-11–MAP-12	1
MAP-13–MAP-16	0
MAP-17–MAP-19	1
MAP-20–MAP-26	2
APP-i–APP-ii	0
APP-1–APP-10	0
APP-11–APP-12	2

NOTICE

The material contained in this training manual is based on information obtained from the aircraft manufacturer's Airplane Flight Manual, Pilot Manual 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

ILLUSTRATION

TABLE

CHAPTER 1
AIRCRAFT GENERAL

INTRODUCTION

This chapter is an overview of G150 systems and characteristics.

GENERAL

The Gulfstream G150 is a swept-wing, twin-engine monoplane designed to accommodate a crew of two and a maximum of eight to nine passengers. The aircraft features a large, optimized cabin, highly integrated avionics, and exceptional performance capabilities. It is a pressurized transport-category aircraft utilizing an all- metal airframe. Composite materials are used in secondary structures where appropriate (Figure 1-1).

The aircraft incorporates a low-drag/high-lift wing that has been optimized for both high-speed/high-altitude flight and low approach speeds. The aircraft's double-swept, varied dimensional airfoil allows excellent high-speed performance. The leading-edge slats combine with trailing-edge slotted fowler flaps for excellent low-speed characteristics. Winglets provide drag reduction by controlling tip vortices, allowing higher takeoff weights under high and hot conditions and improving long-range cruise performance.

The custom design includes the cockpit, passenger cabin, and lavatory. Ample allowance for baggage stowage is provided in the 55 cubic foot baggage compartment with access through a side opening door.

Figure 1-1. G150 Aircraft

DIMENSIONS

Table 1-1 gives aircraft dimensions.

Table 1-1. DIMENSIONS

EXTERIOR DIMENSIONS	
Overall Length	56 ft 9 in.
Overall Height	18 ft 5 in.
Overall Span	55 ft 7 in.
CABIN DIMENSIONS	
Length	17 ft 8 in.
Height	5 ft 9 in.
Width	5 ft 9 in.
Volume	465.0 cubic ft
Baggage Volume	25 cubic ft
BAGGAGE COMPARTMENT	
Weight	1,100 lb
Floor Loading	105 lb/sq ft^2
Volume	55 cubic ft
KEY AREA SIZE	
Wing Area	337.2 sq ft
Flap Area	44.8 sq ft
Spoiler Area	12.7 sq ft
Aileron Area	12.9 sq ft
Winglet Area	5.56 sq ft
Vertical Stabilizer Area	99.03 sq ft
Rudder Area	12.6 sq ft
Horizontal Area	150.67 sq ft
Elevator Area	21.7 sq ft

AIR CONDITIONING

The air conditioning system provides conditioned air for ventilation, cabin pressurization, and cabin temperature control. The electronically controlled system is designed to be highly automatic. It uses LP and HP air from engines, APU, or RAM air. The system consists of a single air cycle machine (ACM), an ozone filter, and cabin ducting.

APU

An auxiliary power unit (APU) is standard aboard the aircraft with in-flight start and use capabilities. The APU provides electrical power, air conditioning, and heating during ground operations. These same capabilities are available in flight if required.

ENGINE INDICATING AND CREW ALERTING SYSTEM (EICAS)

The EICAS is the interface between the avionics system and the aircraft systems. The EICAS collects and concentrates engine and other subsystem information for display on the adaptive flight displays (AFDs).

The EICAS is made up of one dual-channel data concentrator unit (DCU), two radio interface units (RIU), and two cursor control panels (CCPs). The DCU accepts discrete, analog, and serial digital inputs from the engines, flight control surfaces, and other aircraft subsystems and concentrates these inputs onto high speed serial digital buses for delivery to the AFDs. The DCU generates the aircraft EICAS voice messages, aural alerts and controls aural alert prioritization (including TCAS and TAWS). The RIUs receive commands from the DCU to generate voice messages and aural alerts that are broadcast over the flight deck audio system. Processed EICAS data is supplied to the cross-side DCU channel, the AFDs, and the RIUs via the intergrated avionics processing system IAPS and system bus structure. Controls on the CCPs are used to operate the EICAS message list.

ELECTRICAL

Each engine is equipped with a 28V, 300 AMP starter-generator. The APU is equipped with an identical 28V, 300 AMP starter-generator. The electrical system can be powered by batteries (two nicad), external power, or with any one of three 28V, 300 AMP generators (two starter generators and one APU starter-generator). Two 24V, 27 AMP-HR batteries are in the baggage compartment.

FIRE PROTECTION

Fire protection is available to both engines as well as the APU through a fixed fire fighting system. Each engine is equipped with fire detection and suppression equipment. The right fire bottle also protects the APU. The system is constantly and automatically monitored.

The cabin is equipped with two portable fire extinguishers: one Halon and one water. Another Halon extinguisher is in the cockpit.

FLIGHT CONTROLS

Primary flight controls consist of ailerons, rudder, and elevators. Rudder and elevators are manually actuated. Ailerons are manually actuated with hydraulic assistance. The rudder and the elevator are protected against gust damage while on the ground. Trim is provided in all three axes.

Secondary flight controls consist of flaps, slats, and airbrakes. Flaps and slats are electrically actuated while airbrakes are hydraulically actuated. Dual-channel autopilot (elevator and aileron) and series yaw damper are standard. Autopilot must be disengaged below 200 AGL enroute, 400 AGL during non-precision approach, and 80 AGL on precision approach with full flaps. Yaw damper must be disengaged during takeoff.

FUEL

Fuel is stored in tanks contained in the fuselage and wing sections of the aircraft. The fuel system consists of fuel storage and tank venting, distribution and feed, control and indication, jettison, and pressure or gravity refueling. Maximum capacity is 10,300 lbs.

HYDRAULICS

The hydraulic system consists of a main and an auxiliary system. Both systems operate at 3,000 psi.

An engine-driven pump on each engine powers the main system. The main system components are as follows:

- Ailerons
- Normal brakes with anti-skid
- Landing gear
- Nosewheel steering
- Airbrakes

An electric pump powers the auxiliary system. The auxiliary system components are as follows:

- Ailerons
- Emergency/parking brakes
- Thrust reverser

A high-pressure nitrogen bottle is available to extend the landing gear should the main hydraulic system fail. Additionally, a nitrogen charged accumulator is available for either emergency/parking brake operation or thrust reverser deployment should a total hydraulic failure occur.

ICE AND RAIN PROTECTION

Ice and rain protection consists of pneumatic deicing and bleed air or electrical anti-ice. Deicing is provided for the inboard leading edge of the wing, for the slats , and for the horizontal stabilizer. Bleed-air anti-icing systems are provided to prevent ice formation on the critical areas of the nacelle air inlet. Electrical anti-icing systems are provided to prevent ice formation on the engine $P_{T2}T_{T2}$ probe, the pitot probes, the static ports, the angle of attack sensor, and the windshields.

LANDING GEAR AND BRAKES

All landing gear have dual wheels for safety and stability. The aircraft is easy to land with its low-wing, leading-edge devices and tricycle type with air/oil shock struts trailing link gear. Aircraft main landing gear are equipped with an anti-skid device on each wheel. The nose landing gear is a direct telescope type and is equipped with an electro-hydraulic steering system.

LIGHTING

The lighting system provides interior and exterior lighting required for ground and flight operations of the aircraft. Both systems use 28-VDC power and are controlled by switches on the cockpit overhead panel, instrument panels, and in the passenger compartment. An emergency lighting system is installed to provide the crew and passengers lighting in case of an emergency.

OXYGEN

Oxygen is provided for all occupants. Crew positions are equipped with the EROS quick-donning mask and oxygen is available at all times. Passengers are provided with drop-down masks, which can be deployed either manually or automatically.

PNEUMATICS

The pneumatic system provides pressurized bleed air for pressurization, deice, air conditioning, anti-ice, door seal, and hydraulic tank pressurization. Either or both engines or the APU can provide pressurized bleed air.

Emergency pressurization is available from the right engine LP bleed.

POWERPLANT

The aircraft is powered by two Honeywell TFE731-40AR-200G advanced-technology, turbofan engines that generate 4,420 pounds of static thrust up to 76°F. The aircraft engines including thrust reversers are pylon mounted on the upper aft fuselage.

PRESSURIZATION

The pressurization system regulates conditioned-air outflow from the cabin at the rates required to maintain cabin altitude pressure according to preset schedules. The pressurized space includes the flight compartment, and passenger cabin. The system primarily operates in the automatic mode but may operate manually as the situation requires.

LIMITATIONS

PERFORMANCE

The Gulfstream G150 has been certified as a transport category aircraft under FAA Part 25 as well as corresponding JAA, and CAAI regulations.

Range	2,950 NM
Total usable fuel weight	10,300 lb

Weight	
Max ramp	26,250 lb
Max takeoff	26,100 lb
Max landing	21,700 lb
Max zero fuel	17,500 lb
Min flight	13,200 lb

V_{MO}/M_{MO} (Normal Ops, autopilot engaged or Mach trim operative):

Sea level–8,000 ft	310 KIAS
8,000–12,000 ft	310–330 KIAS
12,000–29,260 ft	330 KIAS
Above 29,260 ft	0.85 M_i

V_{MO}/M_{MO} (autopilot disengaged and Mach trim becoming inoperative in flight):

Sea level—8,000 ft	310 KIAS
8,000—12,000 ft	310—330 KIAS
12,000—24,770 ft	330 KIAS
Above 24,770 ft	0.78 M_i

Maneuvering Speed (V_A):

Sea level—20,000 ft	272–287 KIAS
20,00 0—29,300 ft	287–330 KIAS
Above 29,300 ft	0.85 M_i

RVSM OPERATIONAL LIMITATIONS

Altitude (ft)	Speed Limits
29,000 TO 41,000	0.54 M_I–0.84 M_I

The difference between pilot and copilot altimeter readings shall not exceed 45 ft.

Maximum slats/flaps operating/extended (V_{SE}/V_{FE})
Slats 250 KIAS
Flaps 12° 250 KIAS
Flaps 20° 225 KIAS
Flaps 40° 180 KIAS

Do not operate slats and flaps above 20,000 ft.

Auto slats operation and manual retraction are not altitude limited.

Maximum landing gear operating/extended (V_{LO}/ V_{LE}) 180 KIAS

Do not operate landing gear above 20,000 ft.

Maximum tire ground speed 182 KTS

Minimum control speed/air (V_{MCA}):
Flaps 0°, 12° and 20° 91 KIAS

Minimum control speed/ground (V_{MCG}):
Flaps 0°, 12° and 20° 103 KIAS

Maneuvering flight load limits:
Flaps and landing gear up +2.77/–1.00 g
Gear or flaps extended +2.00/0.00 g
Slats extended +2.00/0.00 g

Maximum landing altitude 14,000 ft

Maximum operating altitude 45,000 ft

Maximum altitude for autopilot and yaw damper inoperative 31,000 ft

PUBLICATIONS

- *Airplane Flight Manual*
- *Operations Manual*
- Training Materials

AIRCRAFT WALKAROUND

Refer to Aircraft Walkaround Video/ Presentation.

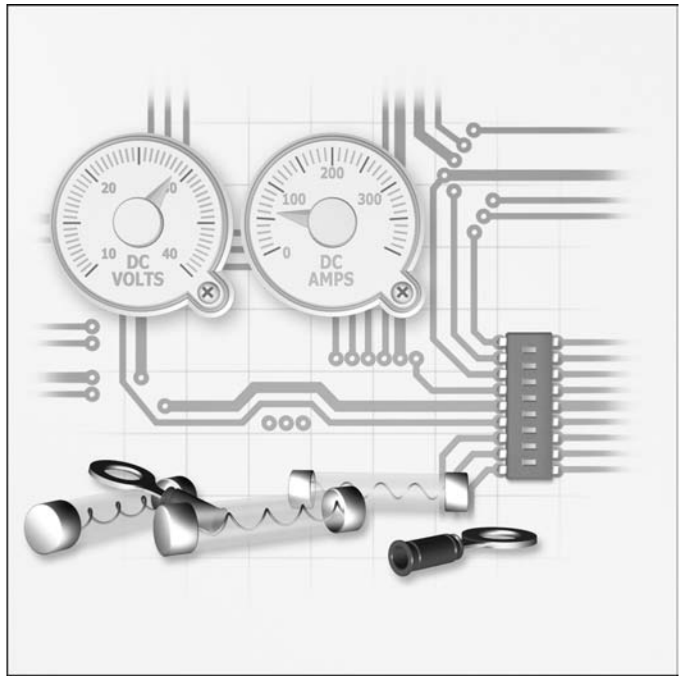
CHAPTER 2
ELECTRICAL POWER SYSTEMS

CONTENTS

Page

ILLUSTRATIONS

CHAPTER 2
ELECTRICAL POWER SYSTEMS



INTRODUCTION

The Gulfstream G150 aircraft's primary electrical power supply is a 28-VDC system. A 60 Hz 115 VAC system is provided for passenger convenience items only.

GENERAL

The main power supply system includes:

- Two 28-VDC starter-generators driven by the engines
- Two nickel-cadmium batteries
- DC external power receptacle
- Auxiliary power unit (APU) equipped with a starter generator

The APU starter generator can be connected to the aircraft DC buses and will operate in parallel with the engine driven starter generators. The APU is non-essential for the engine start and flight operations.

The two batteries are used for aircraft engine starting, APU starting, and for emergency flight operations.

The main electrical system is divided into two subsystems: No. 1 and No. 2. Each subsystem includes a battery, a generator power source, and respective buses that supply power throughout the aircraft.

The electrical system consists of the following major components (Figure 2-1):

- Overhead switch and circuit breaker panel
- A Forward relay panel
- Two 24 volt nickel-cadmium batteries
- An aft relay panel
- Two DC contactor boxes
- Three generator control units
- Two engine driven starter-generators
- APU starter-generator
- External power connector
- External power relay
- 60 Hz 115 VAC inverter
- EICAS display

DC POWER SYSTEM

GENERAL

The DC generators normally operate in parallel. Each generator is connected to a separate main bus (No. 1 or No. 2), through generator line contactors (GLCs) (Figure 2-2).

The starter-generators function primarily as generators, and also starter-motors. They never perform both functions simultaneously. The starter energizes the starter function from the battery bus and through the respective generator start contactors (GSCs).

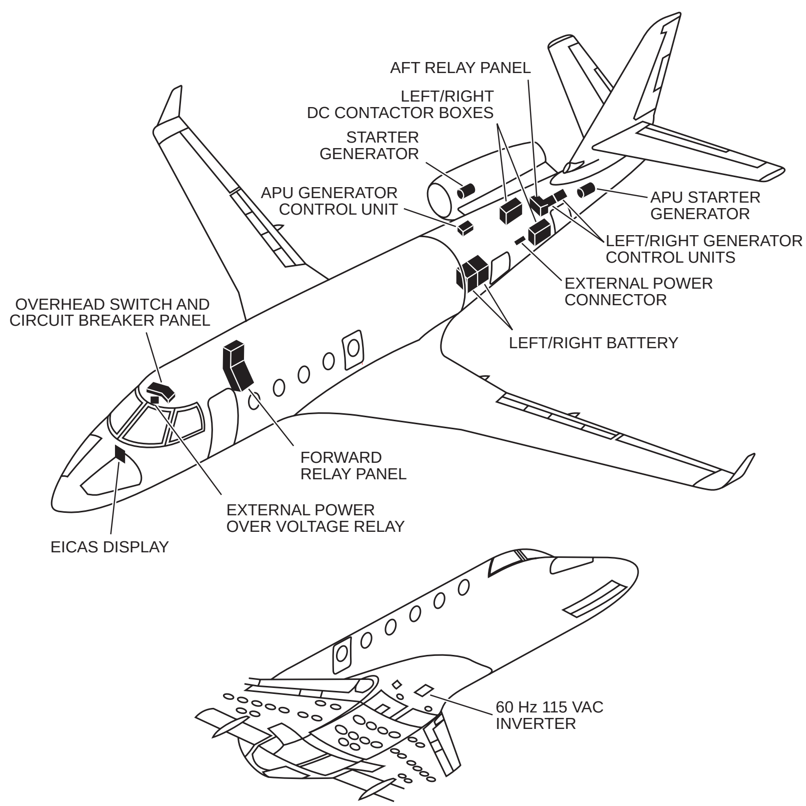


Figure 2-1. Electrical Component Locations

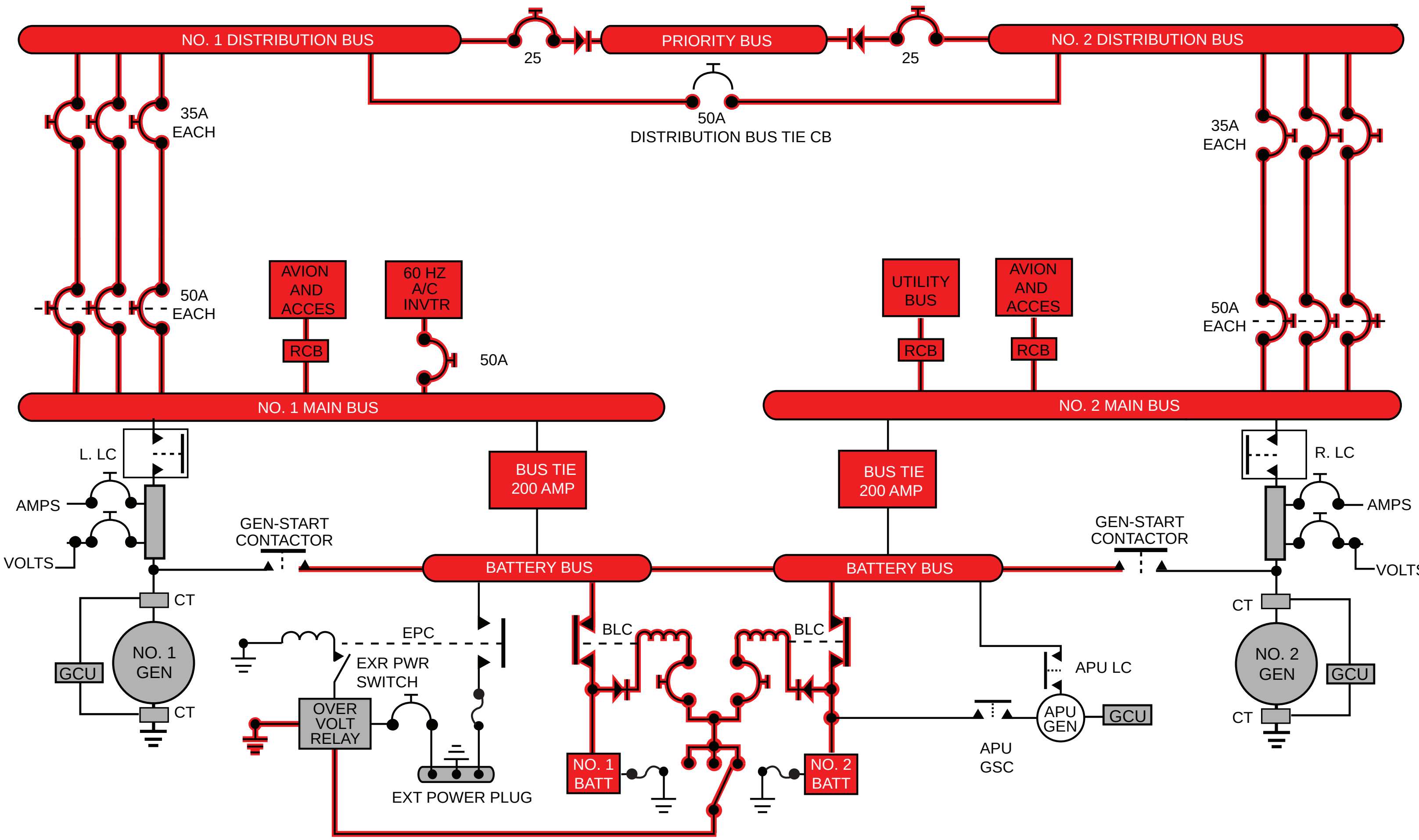


Figure 2-2. DC Electrical System

Two batteries connected in parallel are the secondary source of DC power and are available for engine and APU starting. They also provide assistance to the generators or external power during engine start.

An external power system with overvoltage protection is available to charge batteries, start engines, and energize the entire electrical system.

The auxiliary power unit (APU) has a 28-VDC starter-generator which is electrically connected to the DC buses, and operates in parallel with the primary DC power system. The APU is nonessential for engine start and flight operations.

A network of buses interconnected by circuit breakers and contactors distribute DC power. All components, controls and wiring are installed so that failure of one unit does not adversely affect the operation of other units essential for safe operation.

Associated switches for the DC system are located on the overhead panel(Figure 2-3).

Visual indications of the operating items are displayed on the engine indicating and crew alerting system (EICAS). These messages appear on the upper right corner of the multi function displays (MFDs).

BATTERIES

GENERAL

Two 24-VDC, 20 cell, 27 AH nickel-cadmium batteries are located in a sealed chamber in the forward LH side of the baggage compartment. The top of the chamber acts as the floor of the baggage compartment and has the same floor loading of 105 lbs/sq ft.

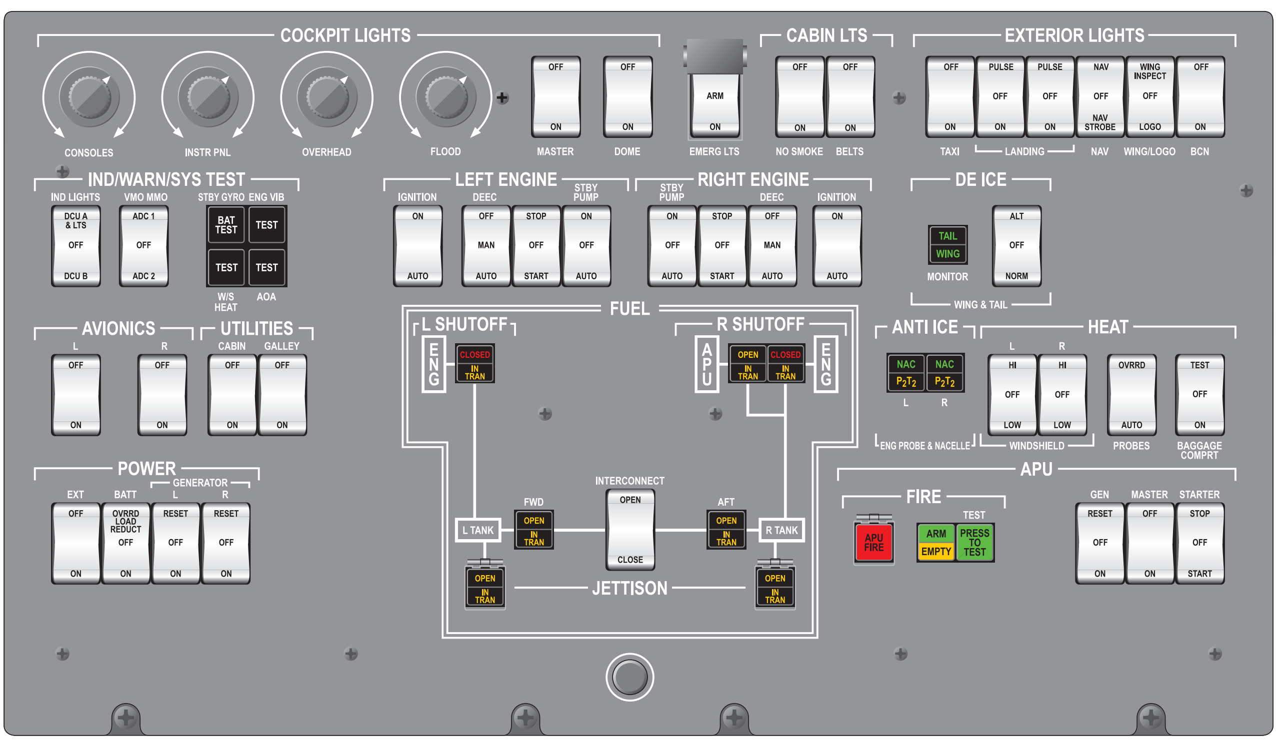


Figure 2-3. Overhead Switch Panel

The batteries vent overboard through hoses connected to the battery case vent ports. The vent system produces airflow through the batteries to aid in venting. Visual indications of battery temperatures and voltages appear on the secondary EICAS page (Figure 2-4).

OPERATIONS

When the BATT switch is selected to the ON position, the battery line contactors (BLCs) connect the batteries to the battery bus. Operating in parallel, the batteries supply power to start the first engine. With the first generator on line, both batteries begin charging.

As soon as the generators start supplying power, generator voltage output is reduced 1.5 volts below normal VDC for two minutes to prevent overheating of the batteries. After two minutes, generator voltage increases to the normal output level of 28 ±0.1 VDC.

The external power source may also be used to charge the batteries.

A caution of battery discharge while the aircraft is on the ground is provided by the beacon (BCN) light as follows:

- When one battery is connected to the main bus AND
- External power or one of the generators is not connected to the bus system.

This caution occurs regardless of the BCN switch position.

EMERGENCY POWER

The batteries provide the emergency power supply to essential flight instruments and emergency equipment.

Hot battery circuits for each battery are as follows:

No. 1 battery:

- Entrance lights
- Step lights
- Baggage lights
- Left battery voltage

No. 2 battery:

- Pressure refueling test
- Right battery voltage

BATTERY TEMPERATURE MONITORING

The battery temperature monitoring system displays dual temperature and a caution message on the MFD display. Testing capability is available through the MFD (Figure 2-4).

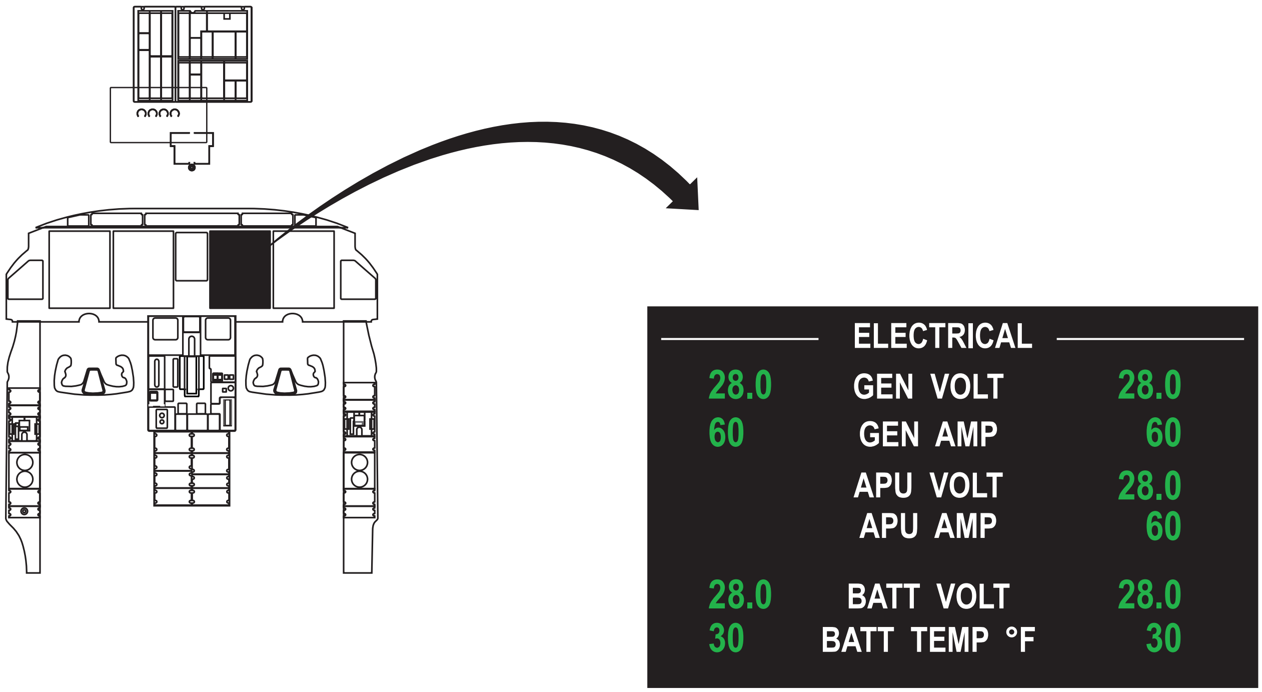


Figure 2-4. Electrical Visual Indications

The system consists of a thermistor (temperature sensor) in each battery and a circuit breaker on the overhead panel. The thermistor transmits the battery temperature to the MFD display. The dual battery temperature displays the temperature of each battery as follows:

Normal—50 to 140°F (10 to 60°C)

Caution—140 to 160°F (60 to 71°C). Battery should be manually disconnected through the appropriate (L or R) BATT DISC circuit breaker. EICAS Message and temperature readout will be amber.

Emergency—160 to 200°F (71 to 93°C). Land as soon as possible. Temperature readout will turn red.

EXTERNAL DC POWER

GENERAL

The external power system (Figure 2-5) consists of a 28-VDC external power receptacle beneath the left engine and an EXT POWER switch on the overhead panel.

External power is available to charge the batteries, assist during engine start, or supply power to the electrical systems. Overvoltage and reverse polarity protection is also provided by the external power source. The unit must furnish 28 VDC at a minimum rating of 1,000 amperes for engine starting.

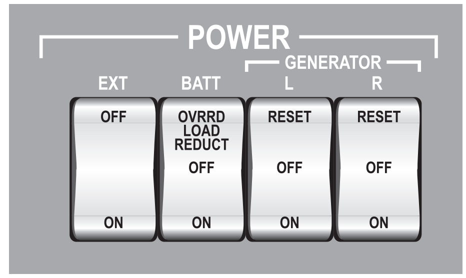


Figure 2-5. External Power Receptacle and Controls

OPERATION

When the aircraft is connected to an external power source and the EXT switch is selected to the ON position, external power flows to the battery bus through a 250-ampere fuse and the external power contactor in the left DC power box.

If the external voltage is 29.5 volts or above, the overvoltage relay disconnects the external power. Battery line contactors open automatically to prevent battery drain.

When external power is connected, external voltage from the battery bus charges the batteries through the battery line contactor. The external power relay disables the line contactor to prevent a conflict between starter-generator power and external power on the main buses.

CONTROL

The EXT POWER is a two position switch (Figure 2-5):

ON—Connects the external power source to the battery bus through the external power contactor and prevents the generators from coming on line.

OFF—Disconnects the external power circuit.

DC POWER DISTRIBUTION

GENERAL

During normal operation, the No. 1 and No. 2 generators supply DC power to the both main buses (see Figure 2-2). Each main bus then supplies respective power as follows:

- Battery bus
- Main bus
- Avionics and accessory bus
- Utility bus
- Priority bus
- Distribution bus

BUSES

Battery Bus

Each battery is connected through a battery line contactor (BLC) to the respective battery bus. The battery buses are also connected to each other by a hard wire to form one battery bus.

The battery buses are connected to the main buses through the main bus tie contactors. External power is connected to the battery bus through the external power contactor. The APU generator is connected to the battery bus through the APU generator line contactor. The battery provides power directly to the APU starter generator for APU engine start. The battery bus provides power to the desired engine for start through the generator start contactor.

Main Bus

The No. 1 and No. 2 main buses supply power to charge the batteries through bus-tie contactors (normally closed), and current sensors. Since both generator systems are interconnected through bus-tie contactors and the battery bus, they operate in parallel and form an integrated electrical system.

Each main bus also supplies power directly to its respective avionics and accessories bus and to some high load consumers. This prevents overloading of the distribution buses.

Avionics and Accessory Bus

The avionics and accessory buses are connected to their respective avionics equipment, and customer non-essential load. Each bus is connected to its main bus, respectively, via a remote controlled circuit breaker (RCCB). The RCCBs are controlled by the L and R avionics switches and control circuit breakers. The avionics switches operate in series with the control circuit breakers. Selecting the avionics switch to the OFF position or pulling the control circuit breaker opens the RCCB and removes power from the respective avionics and accessory bus.

The left and right avionics (AV) and accessory (ACC) switches are located on the overhead switch panel.

Utilities Bus

The 28-VDC utilities bus powers non-essential equipment including the cabin power and galley equipment. A RCCB protects the bus.

Priority Bus

A priority bus in the electrical system ensures a continuous supply of electrical power to certain critical equipment. Both distribution buses power the priority bus through circuit breakers and diodes.

Distribution Bus

Each main bus supplies power to its respective distribution bus through three feeder cables (Figure 2-6). Each feeder cable is protected by a circuit breaker at each end.

There are three (35 amp) feeder circuit breakers in the flight compartment overhead for each distribution bus. The breakers are labeled DISTR BUS L FDRS and DISTR BUS R FDRS (see Figure 2-2).

The feeder circuit breakers at the aft end of each feeder are in the DC contactor box, and are 50 amp circuit breakers. The circuit breakers on the DC contactor boxes are inaccessible in flight.

The 50 amp circuit breakers at the aft end of each feeder contain a built in switch. The switches are wired in parallel. If any of the aft feeder circuit breakers open it will illuminate the respective DISTR FEEDER OPEN message on the MFD.

Any unaffected feeder cable will continue to supply power to the distribution bus.

Failure of two feeder cables will likely be followed by a failure of the third feeder cable. When all feeder cables fail (indicated by failure of all consumers of the affected distribution bus), the respective distribution bus is disabled.

Closing the normally open DISTR BUS TIE circuit breaker on the overhead panel allows the operative bus to repower the disabled bus.

CAUTION

All three DISTR BUS FDRS circuit breakers of the affected bus must be opened before closing the DISTR BUS TIE circuit breaker to isolate the failure and avoid supplying power to the failed bus. Do not reset any tripped feeder circuit breaker when the DISTR FEEDER OPEN message is visible or while the DISTR BUS TIE circuit breaker is closed.

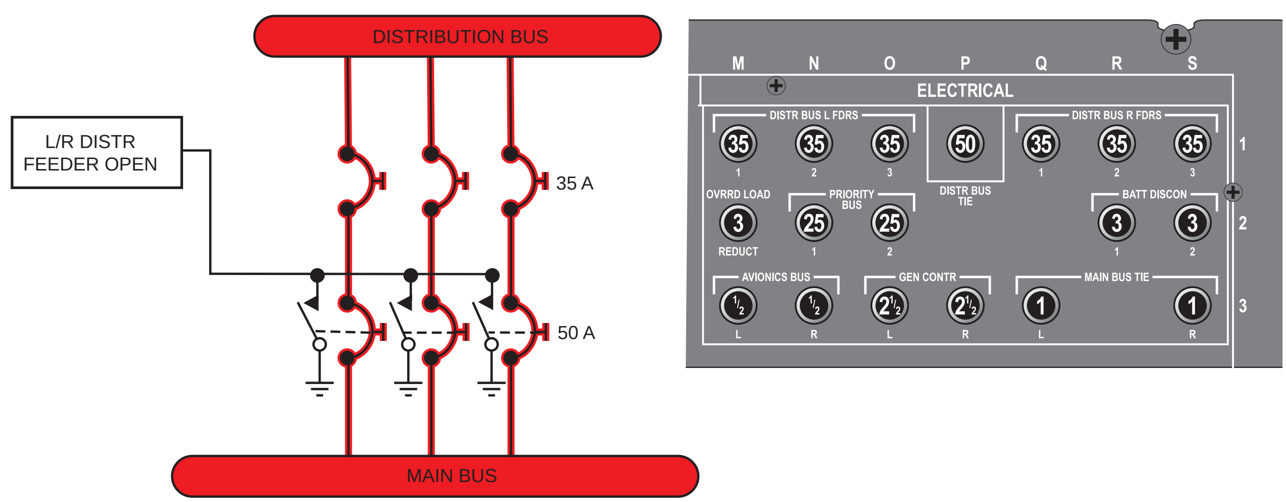


Figure 2-6. Distribution Feeder Cables

GENERATORS

GENERAL

Each engine has one starter-generator mounted on the accessory gearbox center drive pad (Figure 2-7). Ambient air cools the generators through a scoop located just below the engine air inlet. Each generator produces 28 volts DC and is rated at 300 amperes.

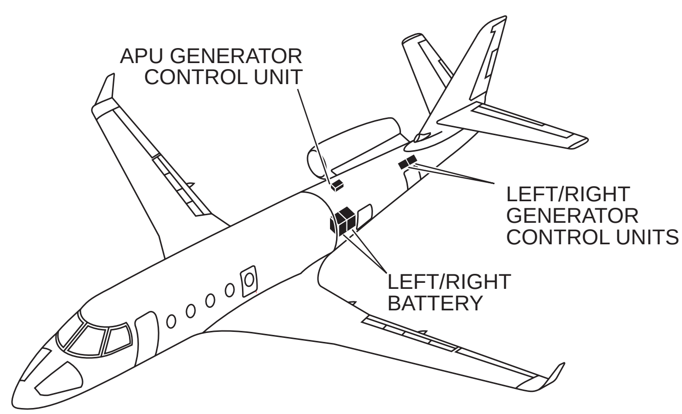
Generator output and load paralleling are automatically regulated. A generator control unit (GCU) protects each generator against over-voltage and feeder faults (over-excitation). These conditions trip the respective GCU, thereby deenergizing the generator magnetic field.

If there is a loss of power from one of the generators, the main bus-tie contactor (BTC) permits the opposite power source to supply power to the buses. The aircraft structure is used as the negative side of the aircraft electrical system.

GENERATOR CONTROL UNITS

There are three GCUs: one for each engine and the APU. The engine GCUs are located above the service compartment and APU access doors and provide the following functions:

- Voltage regulation—Regulates generator output to 28.5 ±0.1 volts during all load and environmental conditions for which the generator is designed.
- Load division—Load difference between the generators does not exceed 30 amperes(with both generators functioning).
- Over-voltage protection—An over-voltage sensing circuit continually monitors the generator output voltage. When the voltage exceeds a preset level, the respective generator is deenergized without deenergizing the other generator.
- Starter field current control (field weakening)—The starter field current control circuit controls the field flux of the starter. Allows maximum output torque at a low starter rpm while still permitting the starter to reach a relatively high rpm occurring at starter cutout.



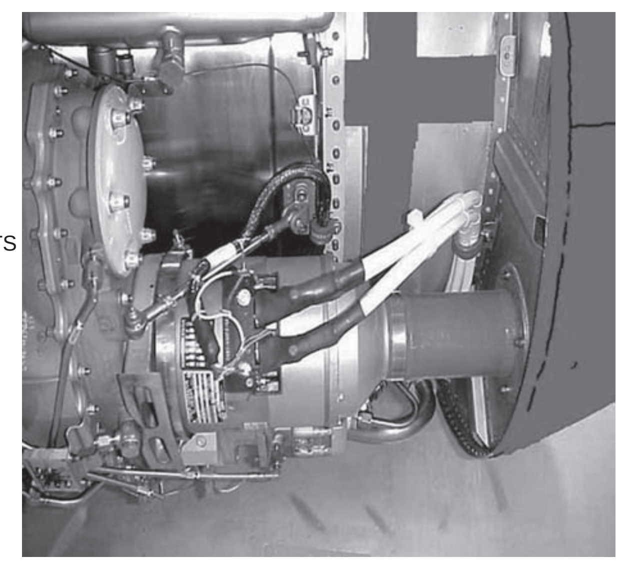


Figure 2-7. Generator Location

- Current limit (generator) control—Regulates its generator to a load no greater than 400 amps by controlling the voltage output of the generator electronically.
- Overspeed (runaway) protection—Prevents an unloaded unit from running away (if a shaft breaks) during starting.
- Generator (line) contactor control—Automatically connects the generator output to the generator bus when the control switch is in the ON position and the generator output voltage has risen to within 0.3 VDC of the main bus voltage.
- Reverse-current protection—Voltage differential greater than +0.22 volts automatically trips the generator line contactor, thereby removing the generator from the bus.
- Feeder (ground) fault protection—A current flow to ground removes that generator from the generator bus.

OPERATION

BATTERY CHARGING

External Power

With external power connected to the aircraft and the EXT POWER switch selected to the ON position, the external power contactor closes to energize the battery bus. When the BATT switch is in the ON position, the batteries close the contactors BLC-1 and BLC-2 to charge the batteries from the battery bus.

APU Generator

With the APU running and the generator switch selected to the ON position, the APU LC (line contactor) closes and energizes the battery bus with 28 VDC. When the battery switch is in the ON position, contactors BLC-1 and BLC-2 close to charge the batteries from the battery bus.

Starter-Generators

With an engine running, the starter-generator energizes the main bus through the GLC. When the BATT switch is in the ON position, the batteries close the contactors BLC-1 and BLC-2. This charges the batteries from the battery bus.

LOAD OPERATING CONDITIONS

With the BATT switch in the ON position, BLC-1 and BLC-2 close to supply 24 VDC from the batteries to the battery bus. Simultaneously the BTC-1 and BTC-2 close.

With external power applied and the EXT POWER switch ON, 28-VDC external power is applied to the battery bus. Simultaneously the BTC-1 and BTC-2 close.

Complete distribution of DC power is possible if both bus-tie contactors are closed. Normally, they are automatically energized when power is available on the battery bus. However, a contactor deenergizes if:

- Opposite engine is being started during battery only start
- Too much amperage goes through the contactor (more than 200 amperes)
- Its main bus-tie circuit breaker in the overhead panel is open.

Automatic Load-Shed

Under normal operating conditions, the generators operate in parallel to share the entire DC electrical load equally.

During single generator operation due to generator failure or crew selection, an automatic load-shedding circuit disconnects these nonessential loads:

- Galley equipment
- Windshield heat

- Baggage heat
- 60 Hz 115 VAC inverter

Opening of either generator line contactor (GLC) results in loss of power to emergency disconnect relay causing an automatic load-shedding of non-essential systems.

Placing the battery switch in the OVRRD LOAD REDUCT position bypasses the EDR relay circuit allowing the crew to reselect necessary systems. In this manner, the crew may override the load-shed system.

APU Starting

APU start power is supplied by the right battery only. A second power source must provide power to the ECU for APU engine control. The start sequence is as follows (Figure 2-8):

- Select the APU master switch to the ON position.
- APU PRESS TO TEST Pushbutton, press to test APU fire loop and squib.

NOTE

Completing the APU press to test after the master switch is on will lock in an APU fire fault in the ECU preventing an APU start.

- Select the APU start switch momentarily to START position
  - When the APU energizes, it removes power from the right battery line contactor (BLC-R) causing it to open and disconnects the right battery from the rest of the electrical system.
  - APU start relay receives its ground from the right battery relay, only if it is deenergized.
  - Right battery line contactor opens, disconnecting it from the battery bus. A secondary power source is required for APU ECU operation.
  - When the APU start relay energizes, it closes the APU start contactor and starts the APU.
  - The APU electronic control unit (ECU) provides starter cut-out at 77% rpm.

NOTE

APU start may also be terminated by momentarily selecting the APU START switch to the STOP position.

ENGINE START

Each engine is started with its starter-generator. The following can power the starter:

- Aircraft batteries
- External power
- APU

The battery bus powers the engine start control.

Engine Start Sequence—Batteries

Figure 2-9 depicts the after engine start sequence as follows:

- Select right start switch momentarily to START position—Switch returns to OFF position when released.
- Select right GEN switch—Closes the right generator start contactor (GSC) which powers the right starter-generator (S/G).
- Digital engine electronic controller (DEEC) starter cut-out is provided at 50% N_2 engine speed—Terminates the start.
- When the GCU receives the cut-out signal it depowers, which deenergizes the engine start relay.
- When the relay deenergizes, it opens the GSC and terminates the start.
- The opposite side main bus tie contactor opens, deenergizing the opposite side to reduce load during battery start.

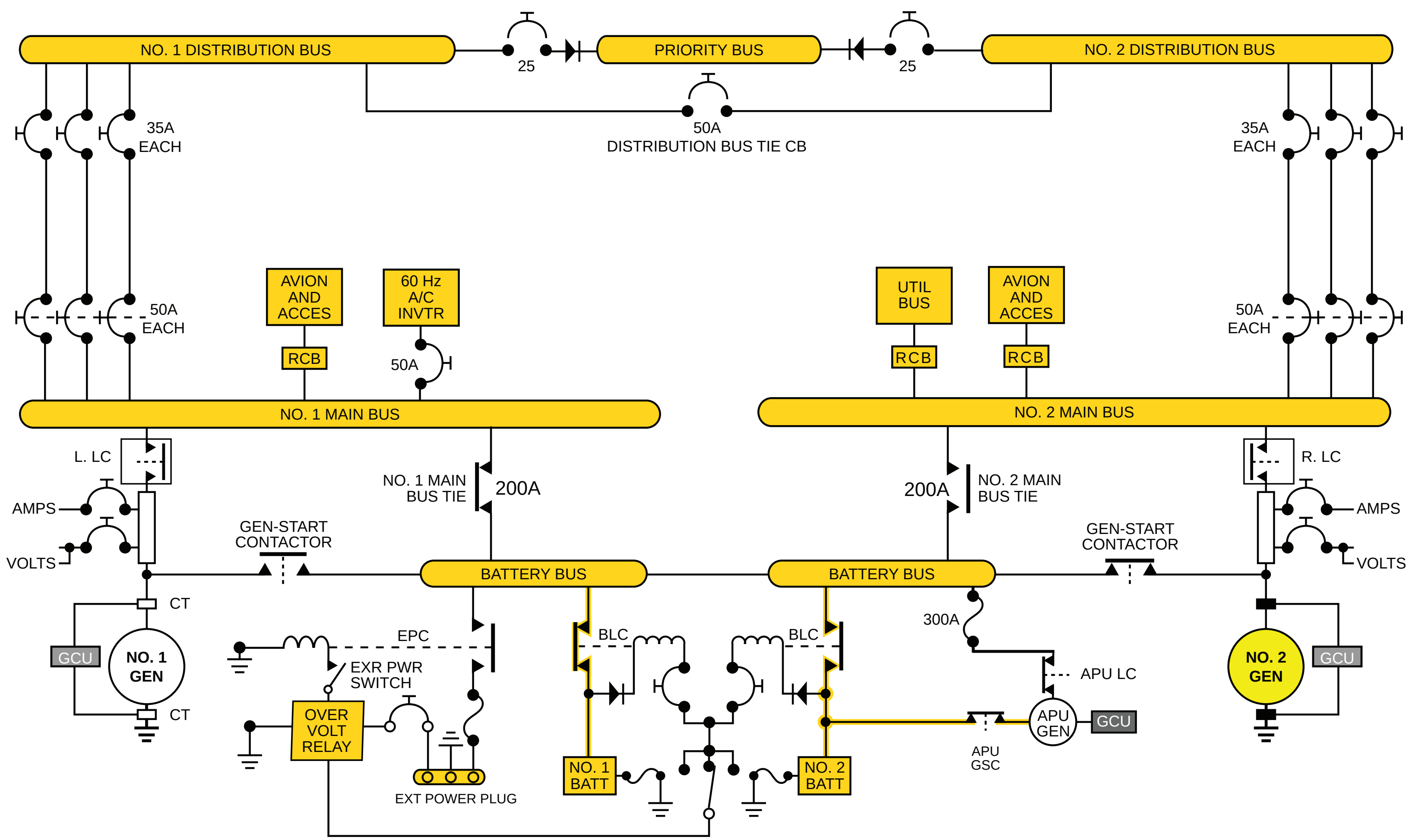


Figure 2-8. APU Running/Right Engine Start

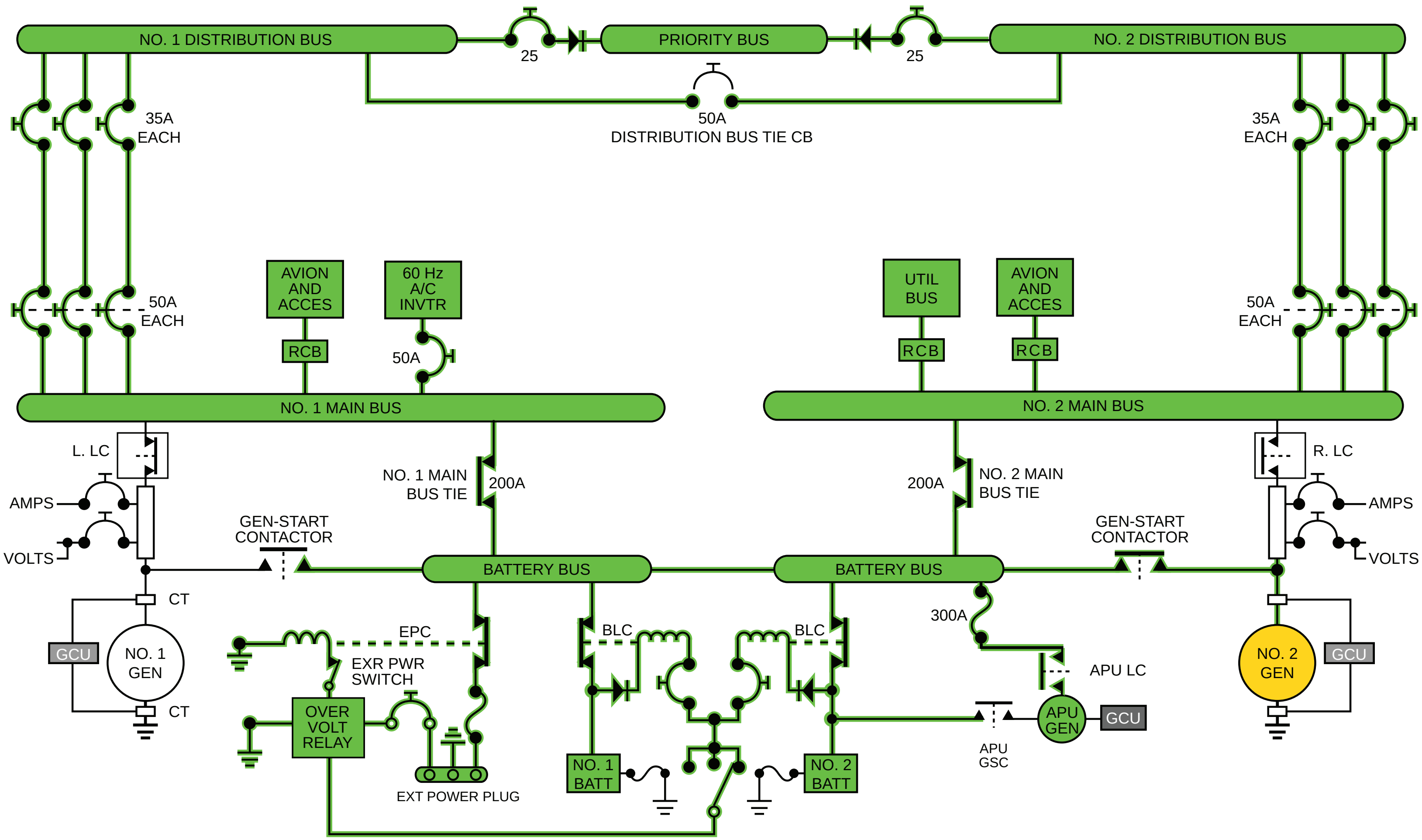


Figure 2-9. External Power/Right Engine Start

NOTE

The start sequence may also be terminated by momentarily selecting the START switch to STOP position.

The second engine start is normally a cross start. In this mode the operating engine and APU, if running, will aid the batteries in the start. Both generator start contactors (GSCs) will close on the second engine start. No buses will lose power at this time as the operating engine generator will continue to power its respective bus. The current limiting circuit of the GCU will limit the operating generators output to 400 amps.

AC SYSTEM

GENERAL

One KVA static inverter provides the 115-VAC, 60 Hz power for optional customer items.

SYSTEM OPERATION

The left main bus powers the static inverter to supply 115 VAC, 60 Hz.

AC power from the static inverter is supplied to the electrical outlets in the cabin and lavatory areas.

CONTROLS AND INDICATIONS

Control Switches

The overhead switch panel (see Figure 2-3) power section has the following switches:

EXT

OFF—Disconnects the external power circuit.

ON—Connects external power to the battery bus through an overvoltage relay and prevents generator line contactors from closing.

BATT

OFF—Disconnects both batteries from the battery bus.

ON—Connects both batteries in parallel to the Battery bus, if the BATT DISCONNECT circuit breakers are selected to the IN position

OVRRD LOAD REDUCT—Overrides automatic load reduction resulting from an engine generator off-line. Enables the windshield heat, baggage heat, 115 VAC inverter, cabin and the galley loads to operate with only one engine generator off-line.

GENERATOR (L/R)

OFF—Disconnects the generator output from the main bus. Illuminates the respective CAS message GEN OFF (does not deenergize the generator).

ON—Connects the generator output to the main bus. Extinguishes the respective CAS message GEN OFF if its respective generator voltage is sufficient.

RESET—Momentary position spring-loaded to OFF. Reflash the magnetic field in the generator.

AVIONICS

OFF—Opens the RCBs between respective main bus and avionics bus.

ON—Closes the RCB and powers the bus from its respective main bus.

UTILITIES—CABIN and GALLEY

OFF—Opens the RCBs between #2 main bus and the utility bus, thereby depowering the bus.

ON—Closes the RCB and powers the utility bus from the #2 main bus.

Engine Start

STOP—Opens the respective start circuit to abort a start cycle below 50% N_2.

OFF—No electrical power.

START—Initiates a start cycle.

MFD INDICATIONS

The following indications appear on the electrical synoptic page of MFD (Figure 2-10):

- Battery temperature and voltage
- Generator amps and voltage
- APU amps and voltage

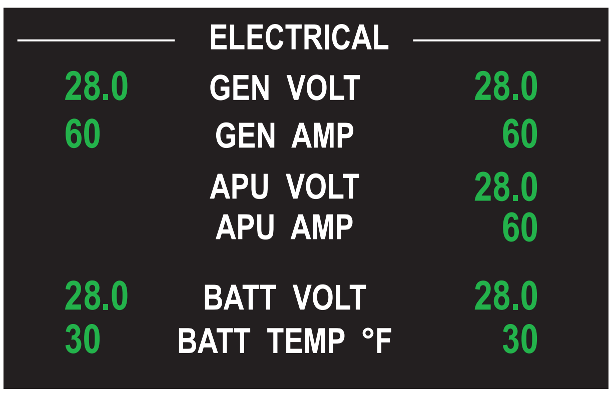


Figure 2-10. MFD Indications

CIRCUIT BREAKERS

DISTR BUS TIE—Normally open. When closed, connects both distribution buses. Ensure all three DISTR BUS FDRS CBs of the disabled bus are disconnected before closing the DISTR BUS TIE (Figure 2-11).

DISTR BUS L/R FDRS—Each set of three (Land R) DISTR BUS FDRS is used for disconnection of the distribution bus from the respective main bus. There are no associated EICAS messages for a tripped CB.

GEN CONTR and APU GEN—When tripped, the field circuit opens, disabling the generator. Trips when GCU anti-cycle circuit fails. Used to trip the generator manually.

MAIN BUS TIE—Trips when excessive current flows through the bus tie contactor (from main bus to battery bus). Also used to open bus tie contactors.

BATT DISCON—Disconnects the battery from the BATTERY BUS when required.

OVERRD LOAD REDUCT—Provides the control power to the EDR operation. When the EDR is deenergized in case of an engine generator failure or the CB is pulled manually, the non-essential loads automatically disconnect. CB connected to No.1 distribution bus.

ENGINE START—Provides the control for the starting circuit relay.

AVIONICS BUS—Intended for control of RCCBs from the overhead panel which connect the L and R AV BUSES to L/R MAIN BUSES accordingly.

PRIORITY BUS 1 and 2—Intended for protection of priority bus.

UTILITIES BUS—Controls the RCCB that connects the utilities bus to MAIN BUS 2.

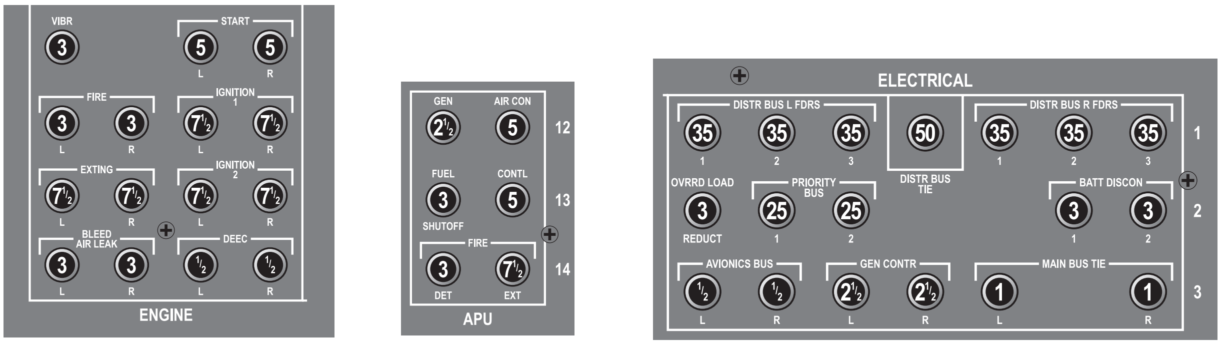


Figure 2-11. Electrical Circuit Breaker Locations

LIMITATIONS

The limitations contained in Section One of the *Airplane Flight Manual (AFM)* must be complied with regardless of the type of operation. Consult the *AFM* for references to specific pages or figures within this section.

DC STARTER-GENERATOR LIMITS

Ground and flight operations—300 A maximum continous.

BATTERY LIMITS

Do not take off if battery temperature remains above 140°F or BATT OVERHEAT (L/R) message appears.

- Minimum voltage above 0°C is 24 V
- Minimum voltage below 0°C is 23 V

EMERGENCY AND ABNORMAL OPERATIONS

EICAS MESSAGES

All messages appear on the MFD primary page. Caution messages are accompanied by a single-chime aural tone. All electrical system messages appear as follows (Figure 2-12):

CAUTION

- APU GEN OVERHEAT—APU overheat condition
- APU GEN OVERLOAD—Load is above 300 amps for more than 40 seconds or 400 amps for more than 10 seconds.
- BATT DISCHARGE—Both main battery voltages are below 25 VDC when at least one generator is active and one battery is connected.
- BATT OFF (L/R)—Indicates that left or right battery is disconnected from the battery bus.
- BATT OVERHEAT (L/R)—Temperature above 140°F (60°C).
- DISTR FEEDER OPEN (L/R)—One or more of the aft three distribution bus feeder circuit breakers are open. Normally accompanied by the respective DISTR BUS FDRS circuit breaker trip.

- GEN OFF—Generator not connected to the appropriate main bus or the generator switch is selected to the OFF position:
  - Green—400 amps or less
  - Amber—After 10 seconds if greater than 400 amps
- GEN OVERHEAT—Overheat condition.
- GEN OVERLOAD—Load is above 300 amps for more than 40 seconds or 400 amps for more than 10 seconds.

STATUS

- APU GEN OFF—APU operating but APU generator disconnected.

Figure 2-12. Electrical EICAS Indications

QUESTIONS

1. Automatic DC load shed will occur if:
   A. One battery is disconnected
   B. The battery switch is at OVRRD LOAD REDUCT
   C. A BATT OVERHT light illuminates
   D. One engine generator is off or failed

2. Which of the following will cause a generator to automatically trip?
   A. Overvoltage
   B. Feeder fault
   C. Over excitation
   D. All of the above

3. If the BATTERY MASTER is left ON or OVRRD LOAD REDUCT and neither external power nor any of the generators are connected, what indication of battery discharge will occur during ground operations?
   A. Warning CAS message displayed on the MFD
   B. Position lights illuminate
   C. Beacon light illuminates
   D. BATT DISCHG amber CAS message appears on the MFD

4. If the battery temperature readout is amber, the appropriate action is:
   A. Land as soon as possible
   B. Battery switch to OFF
   C. Battery manually disconnected through the L/R BATT DISC circuit breaker
   D. A and C

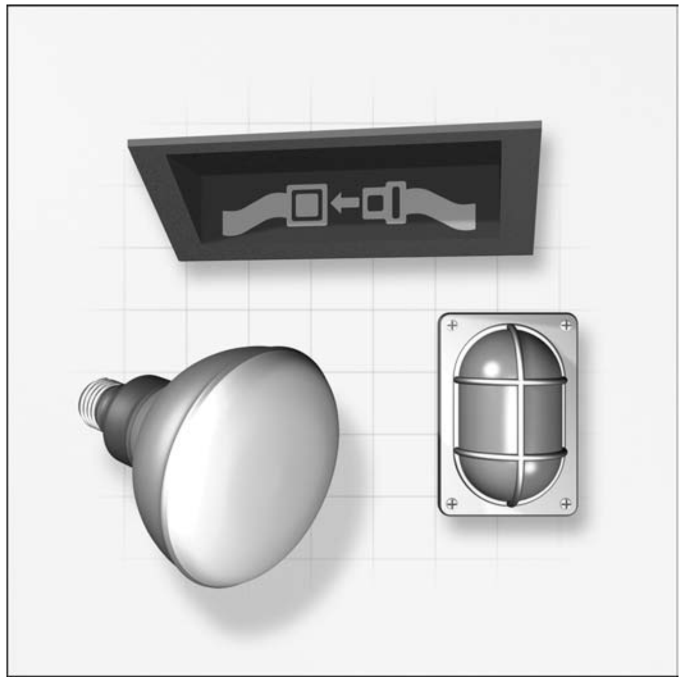
5. Which of the following electrical circuit breakers on the overhead panel is normally in the OPEN position?
   A. Galley
   B. Utility
   C. No. 2 main bus
   D. Distribution bus tie

CHAPTER 3
LIGHTING

CONTENTS

ILLUSTRATIONS

CHAPTER 3
LIGHTING



INTRODUCTION

The Gulfstream G150 lighting system incorporates flight compartment, passenger, baggage area, exterior and emergency lighting.

GENERAL

The G150 lighting system provides interior and exterior lighting required for ground and flight operations of the aircraft. Both systems use 28 VDC and are controlled by switches on the cockpit overhead panel, instrument panels and in the passenger compartment. An emergency lighting system is installed to provide the crew and passengers lighting in case of an emergency.

SYSTEM DESCRIPTION

The lighting system (Figure 3-1) is divided into five subsystems:

- Flight compartment (cockpit)
- Passenger compartment (cabin)
- Baggage compartment
- Exterior
- Emergency

FLIGHT COMPARTMENT LIGHTS (COCKPIT)

The flight compartment lights provide lighting for the flight crew. The DCU A provides the crew with the capability to test the flight compartment annunciators.

The overhead and pedestal lighting provides dimming control of the edge light panels and some instrument displays in the flight compartment.

Pilot and copilot lighting system includes controls for the cockpit floodlights, dome lights, map lights and step lights.

Potentiometers

The COCKPIT LIGHTS—CONSOLES, INSTR PNL, OVERHEAD and FLOOD potentiometers (Figure 3-1) adjust the brilliance of the equipment displays and overhead panels.

CONSOLES potentiometer illuminates the following display devices:

- Passenger oxygen control panel—right console
- Cockpit voice recorder control panel—left console
- Audio control panels—both consoles
- Selector switch—pedestal
- Nosewheel steering handle—left console
- Audio options panels—both consoles
- Control display units— pedestal
- Pedestal edge light panels
- Reversion switch panel—pedestal
- Cursor control panels—both pedestals

INSTR PNL potentiometer illuminates the following display devices:

- Digital clocks—instrument panel
- Standby instruments—center instrument panel
- Display control panel—glarescreen
- Display dimming panels—instrument panel
- Flight guidance panel—center

OVERHEAD potentiometer illuminates the cockpit overhead circuit breaker panel and the cockpit overhead panel. The MASTER switch on the cockpit overhead panel controls the application of electrical power. The panels receive 28 VDC from the No. 1 distribution bus through VARIABLE DIMMING circuit breaker.

FLOOD potentiometer on the overhead panel controls the flood lights which are located on the lower surface of the instrument panel glareshield.

MASTER Light

The MASTER light switch (Figure 3-1) illuminates the cockpit overhead panel. When in the ON position, the switch supplies electrical power to the OVERHEAD potentiometer which illuminates the instrument, edge and ice detection lights. The switch supplies 28 VDC to the DC/DC converters.

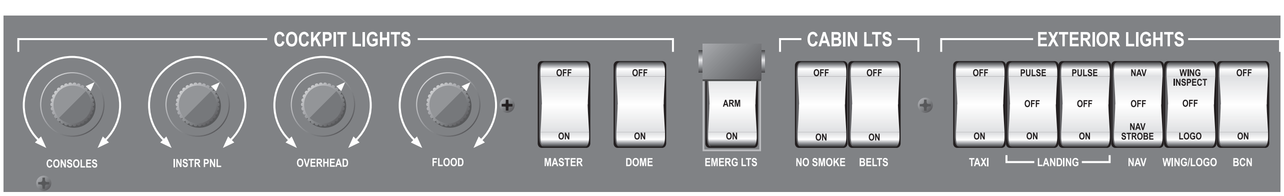


Figure 3-1. Lights

DC/DC Converters

Two DC/DC converters are located in the left cockpit console and receive 28 VDC from the No. 1 distribution bus. The converters receive a variable resistance value from the CONSOLES, INSTR PNL, OVERHEAD and FLOOD potentiometer to control the brilliance of instrument displays.

Dome Light

One dome light is located above the left ceiling panel above the pilot. This light is controlled by the DOME Light switch located in the cockpit overhead panel and receives 28 VDC power from the No. 1 distribution bus.

Pilot and Copilot Lights

The flight compartment pilot and copilot lighting system includes controls for the step lights and map lights which receive electrical power from the No. 2 distribution bus through MAP FLOOD–DOME/CMPSS circuit breaker.

Step Light

Two step lights are located on the face of the step leading into the cockpit and illuminate whenever electrical power is applied to the aircraft.

Map Light

Two swiveling map lights are above the pilot and copilot. Each map light is controlled by a switch on the respective control wheel (Figure 3-2).

The intensity of each map light is controlled by a dimming control located next to each light.

Emergency Lights

The EMERG LTS guarded switch (Figure 3-1) on overhead panel operates the emergency lights manually or arms them automatically in the event of electrical power loss.

Emergency Lighting Distribution Unit (ELDU)

There are two emergency light distribution units (ELDUs). The ELDUs are controlled by the EMERG LTS switch on the cockpit overhead panel.

- With EMERG LTS switch selected to the ON position the ELDUs supply emergency battery power to the interior and exterior emergency lights.
- With EMERG LTS switch selected to the ARM position the ELDUs supply emergency battery power to all the emergency lights during electrical power loss.
- With EMERG LTS switch selected to the OFF position the emergency lighting system is disabled.

The forward ELDU supplies power to the right overwing emergency light and the entrance door emergency light.

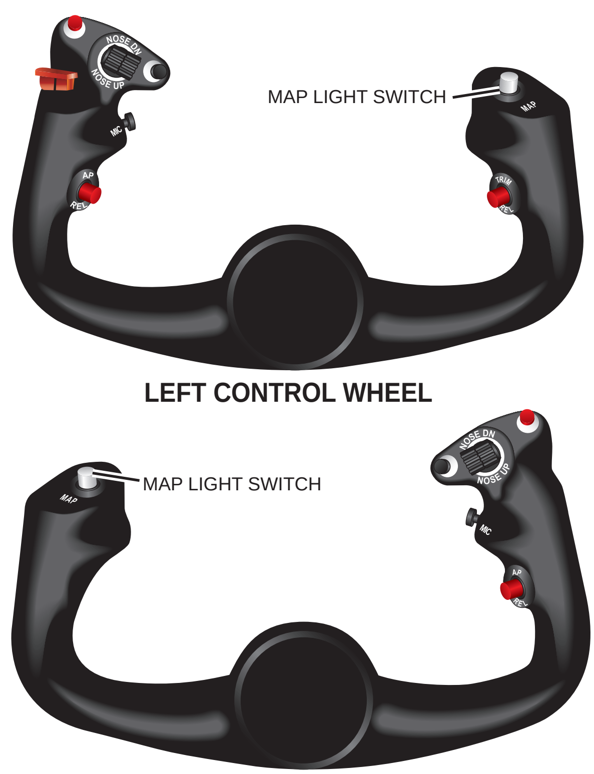


Figure 3-2. Map Light Switch

The aft ELDU supplies power to the left over-wing emergency light.

Both ELDUs supply power to the left and right ground emergency lights.

Emergency Lights Battery

There are two four-ampere-hour emergency light batteries designed to provide power for a duration of ten minutes, one for each of the ELDUs. The emergency light batteries are located adjacent to their associated ELDU. The batteries are charged by aircraft power through the ELDUs.

In the event the aircraft loses electrical power, the ELDUs connect the emergency lights batteries to the emergency lights. Each emergency battery has a circuit breaker, battery condition LEDs and a TEST SWITCH.

NOTE

Emergency batteries are designed to provide power for ten minutes only.

PASSENGER COMPARTMENT LIGHTING(CABIN)

The passenger compartment (cabin) lighting system consists of the following:

- Instruction lights
- Reading lights
- Indirect lighting
- Airstair lights

Instruction Lights

The instruction lights consist of seat belt and no smoking lights. These lights, mounted on the forward and aft cabin partition walls, are controlled by the pilot control switch.

NO SMOKE and BELTS switches provide selective control (see Figure 3-1).

Reading Lights and Indirect Lighting

Reading lights and indirect lighting are provided based on the individual operator’s specified configuration.

Indirect lighting is supplied by fluorescent tubes installed in the roof lining behind the window frames.

Airstair Lights

The aircraft entrance door airstair is provided with lighting on each step of the airstair for safety and visibility at night (Figure 3-3).

There are two airstair light switches:

- Main entrance door
- Right side of the airstair

When the door is opened and either the airstair switch or the entrance door switch is selected to the ON position, lighting is provided to the airstair.

When the door is closed, the switch disconnects the power from the airstair lights without placing either switch to the OFF position. Airstair lights are powered by the No. 1 battery.

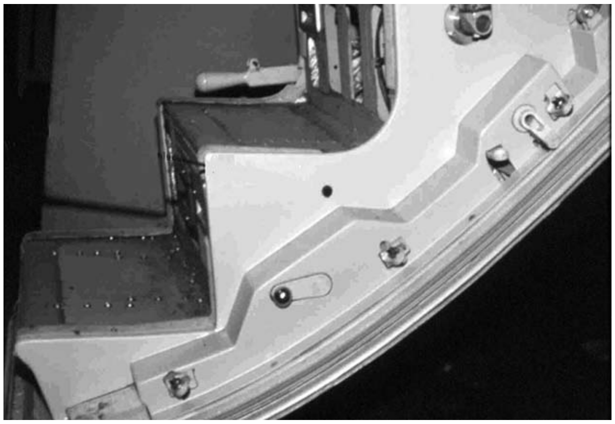


Figure 3-3. Airstair Lights

EXTERIOR LIGHTING

The exterior lighting (Figure 3-4) provides aircraft identification and is divided into four subcategories:

- Landing and taxi lights
- Navigation lights
- Exterior surface inspection lights
- Provisional lighting

Landing and Taxi Lights

The landing and taxi lighting systems are used for landing, takeoff and taxiing operations at night and under conditions requiring enhanced visibility.

Both landing and taxi lights are equipped with adjustable sealed-beams. The landing light is installed in each wing root leading edge and the taxi light is mounted on each side of the nose landing gear strut (Figure 3-4).

LANDING Switches

The EXTERIOR LIGHTS–LANDING switches are located on the cockpit overhead panel (see Figure 3-1). These switches provide control signals to the pulse light control unit.

The LANDING switches have three positions as follows:

- ON—Landing lights illuminate
- PULSE—Enable the pulse light control unit to flash the left and right landing lights alternately
- OFF—Landing lights off

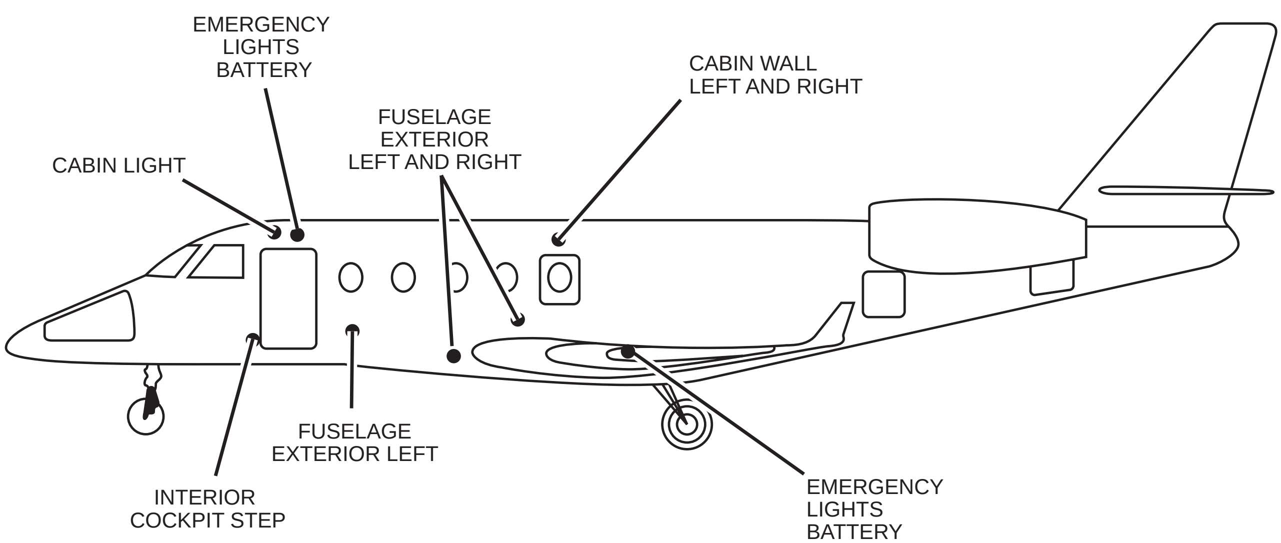


Figure 3-4. Exterior Lighting

Both L and R landing lights are supplied 28 VDC power. The left landing light is powered by the No.2 distribution bus through the L LANDING & TAXI circuit breaker and the right light is powered by the No. 1 distribution bus through the R LANDING & TAXI circuit breaker.

TAXI Switch

The EXTERIOR LIGHTS–TAXI switch (see Figure 3-1) is located on the cockpit overhead panel. When the TAXI switch is ON, the switch applies electrical ground to the nose landing gear down lock switch. When the nosewheel is down and locked, the switch closes enabling power to the PLCU, lighting the taxi lights.

Both taxi lights are supplied with 28 VDC. The right taxi light is powered by the No. 2 distribution bus through the R LANDING & TAXI circuit breaker. The left taxi light is powered by the No. 1 distribution bus, through the L LANDING and TAXI circuit breaker.

Pulse Light Control Unit

The pulse light control unit (PLCU) is a dual channel controller which receives control signals from the TAXI and LANDING switches. These switches enable both taxi and or landing lights to flash on alternately.

The pulse light control unit also receives an input from the traffic collision avoidance system transmitter/receiver which causes the landing lights to flash regardless of the landing light switch position.

Navigation System Lights

The navigation, strobe and beacon lights allow the aircraft to be more visible in low light conditions and provide the aircraft with required navigational lighting.

Navigation Lights

The navigation lights (see Figure 3-4) provide the aircraft directional recognition at night or in reduced visibility and are located in each wing tip. Each NAV light is made up of an array of five green or red LED elements. The tail cone houses two (white) navigation lights and a strobe light.

The NAV switch receives 28 VDC from the No. 1 distribution bus through the NAV circuit breaker. When the NAV switch is in the NAV–STROBE position, 28 VDC is applied to both wings and tail cone Navigation lights.

The power supply provides high voltage to the strobe light flash tubes, causing them to flash. Each strobe assembly receives high voltage by a separate power supply.

The EXTERIOR LIGHTS–NAV switch is a three-position switch as follows:

- NAV—enables the Navigation lights
- OFF
- NAV–STROBE—enables the navigation and strobe lights

Strobe Lights

Strobe lights (see Figure 3-4) are installed in each wing tip. A tail cone strobe light is also installed and is housed in common with the navigation light.

The NAV switch receives 28 VDC from the No. 1 distribution bus through the STROBE circuit breaker. When the NAV switch is in the NAV–STROBE position, 28 VDC is applied to the strobe power supply. They are synchronized by a SYNC signal connected between them.

The power supply provides high voltage to the strobe light flash tubes, causing them to flash. Each strobe assembly is supplied high voltage by a separate power supply.

Beacon Light (BCN)

The red flashing beacon light (see Figure 3-4) is mounted atop the vertical stabilizer. The EXTERIOR LIGHTS–BCN switch is located on the cockpit overhead panel (see Figure 3-1). The BCN switch receives 28 VDC from the No. 2 distribution bus through the BCN circuit breaker.

The BCN energizes automatically when the BATT master switch is in the ON position and the following conditions are met:

- Generators—OFF
- APU—OFF
- EPU—OFF

Wing Inspection and Logo Lights

The wing inspection lights located on each side of the fuselage illuminate the wing leading edges.

The logo lights are installed in the upper surface of each horizontal stabilizer and illuminate both sides of the vertical stabilizer and rudder .

The WING/LOGO switch on the cockpit overhead panel is a two position switch as follows:

- WING INSPECT—Power applied to the left and right wing inspection lights
- LOGO—Illuminates both sides of the vertical stabilizer

The switch receives power from the No. 2 distribution bus through the LIGHTS–WING INSPECT circuit breaker.

EMERGENCY LIGHTING

The emergency lighting system provides the crew, passengers and ground crew lighting in case of an emergency.

The interior emergency lighting system provides lighting in the entrance door exit sign and the over wing emergency exit signs. Escape path lighting is provided on the cabin floor.

The exterior emergency lighting system provides lighting of the wing surface, the left and right emergency exits and below the wing surface during emergency evacuation.

Main Entrance Door Exit Sign

The main entrance door exit sign (Figure 3-5) is located above the main cabin door and illuminates to indicate where the main cabin door is located.

Entrance Door Emergency Light

The entrance door emergency light (Figure 3-5) is located on the lower right corner of the main entrance door and illuminates the main entry door stairs.

Overwing Emergency Exit Sign

The overwing emergency exit signs (Figure 3-5) are located over each overwing emergency exit. They illuminate to indicate where the overwing emergency exits are located.

Overwing Emergency Light

The overwing emergency lights (Figure 3-5) are located on the lower right corner of the left and right emergency window exits. These lights illuminate the top portion of the wing.

Ground Emergency Lights

The ground emergency lights (Figure 3-5) are located under the aircraft on each side of the bottom fairing. These lights illuminate the ground just in front of the forward wing root.

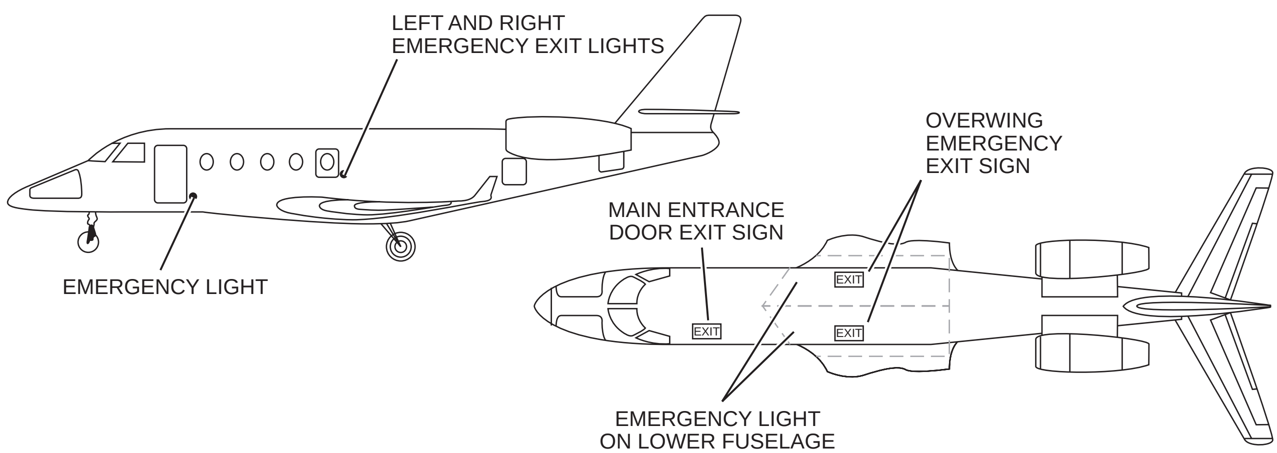


Figure 3-5. Emergency Lighting

BAGGAGE COMPARTMENT LIGHTING

Baggage compartment lighting provides general illumination for baggage loading.

Baggage Dome Light

The baggage compartment dome light is located in the ceiling of the baggage compartment. The light is controlled by a two position toggle switch on the left side of the baggage doorway.

A microswitch mounted in the door frame applies power to the baggage compartment light switch and sends a signal to the integrated avionics processor system (IAPS) that the baggage compartment door is open.

Baggage Inspection Light

The baggage compartment inspection light is located on the underside of the left engine pylon. This light is connected in parallel with the baggage compartment light. The inspection light illuminates whenever the baggage compartment light is selected to the ON position.

The baggage compartment lighting is controlled by two switches in series. The baggage compartment door switch provides an electrical ground to the baggage compartment light and the baggage inspection light. The baggage compartment switch receives 28 VDC from the battery bus through circuit breaker.

When the baggage compartment switch is in the ON position, power is applied to the baggage compartment and inspection lights. The baggage compartment door switch removes electrical power from the baggage compartment switch when the baggage compartment door is closed, preventing the baggage compartment light from draining the main battery.

ANNUNCIATOR LIGHTS

The annunciator lights test and dim systems are actually two subsystems:

- Indication test subsystem
- Day/night subsystem

Indication Test Subsystem

IND/WARN/SYS TEST–IND LIGHTS switch positions are as follows:

- DCU A AND LTS
    - All pushbutton switches to illuminate
    - Integrated avionics processor system (IAPS) initiates the left aural test
- DCU B
    - IAPS initiates the right aural test

Day/Night Subsystem

The day/night subsystem controls the common supply voltage to the annunciator and switch indicators. It also applies 28 VDC for full brilliance or a reduced voltage and controls the brilliance of the lighted pushbutton switches and annunciators.

PUSH DIM Switch

The PUSH DIM switch is part of the day/night subsystem and is located on the pilot side of the instrument panel. The PUSH DIM switch selects full brilliance or dim. When the PUSH DIM switch is depressed, it energizes control relays in the dimming box.

LIMITATIONS

The limitations contained in Section One of the *Airplane Flight Manual (AFM)* must be complied with regardless of the type of operation.

EMERGENCY AND ABNORMAL OPERATIONS

Refer to Figure 3-6 for the EICAS message displayed on the MFD for this system. The only EICAS message associated with the lighting system is as follows:

EMERG LIGHT SW—EMERG LTS switch not in ARM position

Figure 3-6. Lighting EICAS Indications

QUESTIONS

1. An illuminated CAS message EMERG LT SW indicates:

A. Emergency lights switch is in the ON position
B. Emergency lights switch is not in the ARM position
C. Emergency lights switch is in the ON position and no generator is on line
D. Emergency lights switch is in the OFF position

2. The Emergency Lights can be armed by:

A. Pushing the switch on the light panel over the main entry door
B. Pushing the EMERG LT SW to ON
C. Pushing the switch near the CEO seat
D. Pushing the EMERG LT SW to ARM

3. The EMERG LT SW CAS message will illuminate on the MFD when:

A. DC power is available and the EMERG LT switch is not ARM
B. EMERG LT switch is in ARM and one generator is off
C. Emergency batteries are discharged
D. DC power is not available

4. The MASTER SWITCH located on the overhead panel supplies power to COCKPIT LTS–OVERHEAD potentiometer which illuminates:

A. Instrument, console and panel lights
B. Instrument, console and ice detection lights
C. Instrument, ice detection and edge lights
D. Instrument, flood and console lights

5. In the event of a dual generator failure and loss of DC power, the EMERG LT system will activate. The recommended practice is to:

A. Position the EMERG LT switch from ARM to OFF until landing is assured
B. Keep the EMERG LT switch in the ARM position
C. Position the EMERG LT switch to ON
D. Position the EMERG LT switch to OFF

CHAPTER 4
MASTER WARNING SYSTEM

CONTENTS

ILLUSTRATION

TABLES

CHAPTER 4
MASTER WARNING SYSTEM

INTRODUCTION

The master warning system for the Gulfstream G150 employs a multi-function display (MFD) on the center instrument panel which illuminates all engine indicating and crew alerting system (EICAS messages).

GENERAL

The individual warning, caution, advisory and status messages on the MFD (Figure 4-1) illuminate to provide a means of alerting the crew to specific equipment failure, unsafe system operating conditions, or operation of a particular airplane system. A message can be extinguished only by correcting the condition that caused it to illuminate.

COMPONENTS

MASTER CAUTION lights (pilot and copilot) illuminate when any warning (red) or caution (amber) message appears (Figure 4-2).

When any message appears, confirm that the appropriate switches and circuit breakers are in the correct configuration. Messages having the horn symbol are accompanied by aural alert. EICAS messages are inhibited during takeoff and landing.

NOTE

On ground, with any warning message on, takeoff is prohibited.

Advisory (green) or status (white) messages do not cause the MASTER CAUTION light to illuminate.

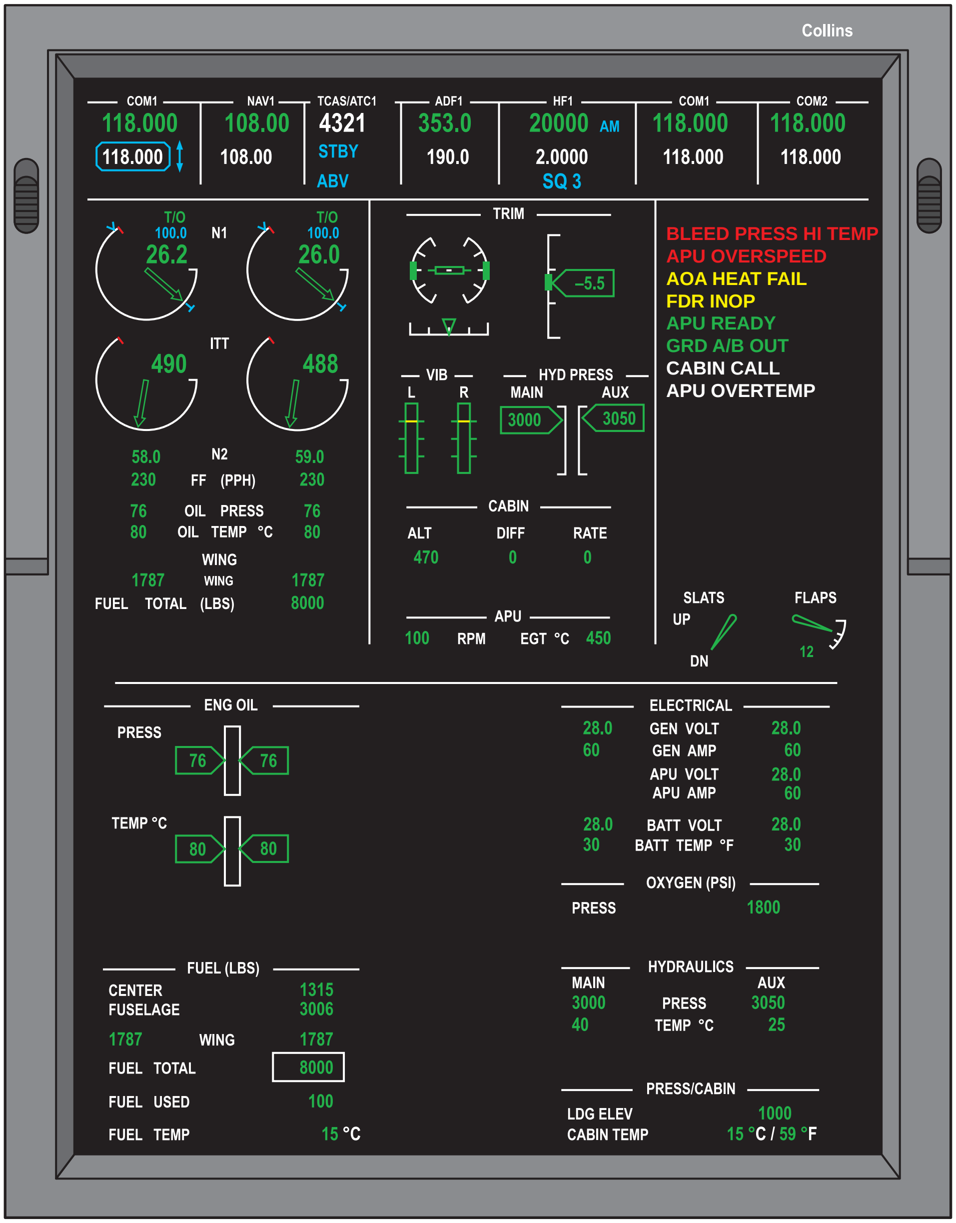


Figure 4-1. Multi-Function Display (MFD)

Figure 4-2. MASTER WARNING/CAUTION

Table 4-1 lists the warning messages. Table 4-2 lists the caution messages. Table 4-3 lists the advisory messages. Table 4-4 lists the status messages.

Table 4-1 WARNING MESSAGES

MESSAGE	MESSAGE MEANING	ACTION/REFERENCE
AILERON FAIL	Mechanical failure of one or both servoactuators or main hydraulic pressure loss	See AILERON FAILURES—in the *AFM*
APU FIRE	APU fire—APU enters automatic shutdown sequence	APU FIRE pushbutton—PRESS APU ARM/EMPTY pushbutton—PRESS
APU OVERSPEED	APU RPM too high—APU did not enter automatic shutdown sequence	1. APU STARTER switch—STOP When APU RPM drops below 9%: 2. APU GENERATOR switch—OFF 3. APU MASTER switch—OFF
APU OVERTEMP	APU excessive temperature—APU did not enter automatic shutdown sequence	1. APU STARTER switch—STOP When APU RPM drops below 9%: 2. APU GENERATOR switch—OFF 3. APU MASTER switch—OFF
BLEED PRESS/TEMP HI	Excessive pressure or temperature downstream of either bleed-air switching valve	See BLEED LINE OVERPRESSURE/OVER-TEMPERATURE—in the *AFM*
CAB ALT 15000	Cabin altitude above 15,000 ft—Cabin pressure controller malfunction	See CABIN UNDERPRESSURIZATION—in the *AFM*
CAB ALT HI	Cabin altitude above 10,000 ft—Cabin pressure controller auto mode malfunction	See CABIN UNDERPRESSURIZATION—in the *AFM*
CABIN DOOR UNLOCK	Cabin door is unlocked	Cycle door (if on ground)—See CABIN DOOR UNLOCK in the *AFM*
COLLECTOR LVL LOW (L/R)	Less than 35 lb fuel in either collector tank	FUEL INTERCONNECT switch—OPEN
CONFIG AIRBRAKE	Aircraft on ground, both engines thrust beyond 70% N_1 and airbrakes are unlocked	Configure aircraft as required

Table 4-1 WARNING MESSAGES (Cont)

MESSAGE	MESSAGE MEANING	ACTION/REFERENCE
CONFIG FLAPS	Aircraft on ground, both engines thrust beyond 70% N_1 and flaps position more than 22°	Configure aircraft as required
CONFIG PARKING	Aircraft on ground, both engines thrust beyond 70% N_1 and parking brake engaged	Configure aircraft as required
CONFIG SLATS	Aircraft on ground, both engines thrust beyond 70% N_1 and slats position less than 23° with SLAT BYPASS pushbutton off	Configure aircraft as required
CONFIG TRIM	Aircraft on ground, both thrust levers beyond 70% N_1 and horizontal stabilizer trim out of green band for the selected flap setting	Configure aircraft as required
DUCT TEMP HI	Excessive duct-air temperature downstream of mixing plenum	See DUCT TEMPERATURE HIGH— in the *AFM*
ENG EXCEEDANCE (L/R)	Engine limits exceeded: N_1, N_2, ITT	Affected engine—MONITOR Affected engine thrust—Reduce to bring within limits
ENG FIRE	Overheating or fire in zone 1—accessories section	See ENGINE FIRE— in the *AFM*
ENG OVERHEAT	Overheating or fire in zone 2—combustor section	See ENGINE FIRE— in the *AFM*
FUEL PRESS LOW (L/R)	If steady ON, boost jet pump has failed and automatic changeover to standby pump did not occur	See FUEL PRESSURE LOW— in the *AFM*
GEAR NOT DOWN	Landing gear is not down and locked with radio altimeter altitude less than 400 ft and one thrust lever at or below maximum cruise or flaps position more than 22°	Landing gear position—VERIFY Configure aircraft as required

Table 4-1 WARNING MESSAGES (Cont)

MESSAGE	MESSAGE MEANING	ACTION/REFERENCE
L ENG/APU BLEED LEAK	Leak or rupture in bleed-air ducting from left engine or APU	ECS selector—R ENG See BLEED AIR LEAK—in the *AFM*
OIL PRESS HI (L/R)	Engine oil pressure above limit	Thrust—Reduce as necessary to bring pressure within limits If still above limits—Idle engine for the remainder of flight Expect possible altitude driftdown
OIL PRESS LOW (L/R)	Engine oil pressure is low	See ENGINE OIL PRESSURE LOW—in the *AFM*
OIL TEMP HI (L/R)	Engine oil temperature is high	See ENGINE OIL TEMPERATURE HIGH—in the *AFM*
R ENG BLEED LEAK	Leak or rupture in bleed-air ducting from right engine	ECS selector—L ENG See Bleed Air Leak—in the *AFM*
STALL	Aircraft is approaching stall—Autopilot disconnects	AOA—DECREASE

Table 4-2. CAUTION MESSAGES

MESSAGE	MESSAGE MEANING	ACTION/REFERENCE
A/B T/O NOT ARMED	Ground airbrakes not armed for takeoff	GROUND A/B switch—T/O
A/P NOSE DOWN/UP	Excessive autopilot pitch trim	See AUTOPILOT SYSTEM—in the *AFM*
A/P NOSE LEFT/RIGHT	Excessive autopilot rudder trim	See AUTOPILOT SYSTEM—in the *AFM*
A/P PITCH TRIM FAIL	Autopilot pitch trim failure	See AUTOPILOT SYSTEM—in the *AFM*
A/P WING DOWN (L/R)	Excessive autopilot aileron trim	See AUTOPILOT SYSTEM—in the *AFM*
AOA HEAT FAIL	Discontinuity in power line	PROBS HEAT switch—OVRRD, as applicable
AOA PREHEAT FAIL	Discontinuity in power line to AOA transmitter/transmitter case	AOA PROBE HEAT CB (9Q)—Reset
APU FAULT	APU malfunction—APU automatically shuts down	APU MASTER switch—OFF
APU EXHAUST OVERTEMP	Excessive temperature around APU exhaust duct—APU automatically shuts down	APU MASTER switch—OFF
APU GEN OVERHEAT	APU generator temperature high	See APU GENERATOR FAILURE—in the *AFM*
APU GEN OVERLOAD	APU generator load above limits	See APU GENERATOR FAILURE—in the *AFM*
APU OIL PRESS LOW	APU oil pressure too low—APU automatically shuts down	APU MASTER switch—OFF

Table 4-2. CAUTION MESSAGES (Cont)

MESSAGE	MESSAGE MEANING	ACTION/REFERENCE
AUTO SLATS EXTENDED	Slats automatic extension has been activated	Slats extended automatically due to high AOA
AUTO SLATS FAIL	Failure in slats automatic extension system	Airspeed and AOA—Monitor
AUX HYD LEVEL LOW	Auxiliary hydraulic tank fluid level is low	1. AUX HYD PUMP switch—OFF 2. MAIN HYD PRESS—Monitor
AUX HYD PRESS HI	Hydraulic pressure above 3,500 psi	1. AUX HYD PRESSURE and TEMP—Monitor 2. AUX HYD PUMP switch—OFF, whenever possible
AUX HYD PRESS LOW	Hydraulic pressure below 1,200 psi	1. AUX HYD PRESSURE and TEMP—Monitor 2. AUX HYD PUMP switch—OFF, whenever possible
AUX HYD TEMP HI	Auxiliary hydraulic system temperature above 85°C	1. AUX HYD PRESSURE and TEMP—Monitor 2. AUX HYD PUMP switch—OFF, whenever possible
BAGG/FUEL DOOR	Either baggage compartment or refueling door is unlocked	See BAGGAGE COMPARTMENT DOOR UNLOCKED—in the *AFM*
BATT DISCHARGE	Engine is running and voltage of both batteries less than 25V	See BATTERIES DISCHARGE—in the *AFM*
BATT OFF (L/R)	Battery is disconnected from battery bus	See BATTERY DISCONNECTED—in the *AFM*
BATT OVERHEAT (L/R)	Battery temperature exceeds 140°F	See BATTERY OVERHEAT—in the *AFM*
CAB AUTO PRESS FAIL	Automatic cabin pressurization failure	1. MODE SEL pushbutton—MAN 2. CABIN ALT selector—Set desired cabin altitude

Table 4-2. CAUTION MESSAGES (Cont)

MESSAGE	MESSAGE MEANING	ACTION/REFERENCE
CAB AUTO TEMP FAIL	Cabin automatic temperature control malfunction	1. TEMP CONTR CABIN pushbutton—MAN 2. TEMP CONTR CABIN selector—SET
DCU FAULT	Data concentrator unit malfunction—EICAS operation not affected	If all EICAS data is displayed, only one DCU channel has failed
DEEC MAJOR (L/R)	Engine fuel controller malfunction	See DEEC MAJOR (L/R)—in the *AFM*
DEEC MAN MODE (L/R)	Engine fuel controller switched to manual mode either manually or automatically	See DEEC MAN MODE (L/R)—in the *AFM*
DEEC MAN XFER INOP (L/R)	Engine fuel controller cannot automatically switch to manual mode	See DEEC MAN XFER INOP (L/R)—in the *AFM*
DEICE LOW VACUUM	Insufficient deice boost selection	See SURFACE DEICE SYSTEM—in the *AFM*
DEICE HIGH PRESSURE	Overpressuring in deice boots system	See SURFACE DEICE SYSTEM—in the *AFM*
DISTR FEEDER OPEN (L/R)	One or more of three aft distribution bus feeder CBs are open	See DISTRIBUTION BUS FEEDER FAILURE—in the *AFM*
DOOR SEAL PRESS LOW	Cabin door seal is not inflated	See CABIN DOOR SEAL FAILURE—in the *AFM*
EFIS COMPARE INOP	EFIS comparator system malfunction	Messages and aural warning data—Monitor
EFIS MISCOMPARE	EFIS data difference (heading, attitude, LOC, G/S etc.)	Compare data with other side and determine which system is accurate
EGPW SYSTEM FAIL	EGPWS system failure. Ground proximity, windshear and terrain warnings unavailable	
EMERG LIGHT SW	EMER LTS switch is not in ARM position	EMERGENCY LTS switch—ARM
ENG FUEL TEMP HI (L/R)	Engine fuel temperature is high	See ABNORMAL ENGINE FUEL TEMPERATURE—in the *AFM*

Table 4-2. CAUTION MESSAGES (Cont)

MESSAGE	MESSAGE MEANING	ACTION/REFERENCE
ENG FUEL TEMP LOW (L/R)	Engine fuel temperature is low	See ABNORMAL ENGINE FUEL TEMPERATURE—in the *AFM*
ENG MISCOMPARE (L/R)	Engine data channels difference	Engine indications—Monitor
FDR INOP	Flight data recorder has failed	Optional system
FLAPS UNBALANCE	Failure of flap system or asymmetry between left and right flaps is more than 3°	See FLAPS OR SLATS FAILURE—in the *AFM*
FLT A/B INBD FAIL	Inboard flight airbrakes position differs from switch command	Outboard flight A/B are used
FLT A/B OUTBD FAIL	Outboard flight airbrakes position differs from switch command	Inboard flight A/B are used
FQMC FAIL (L/R)	Fuel quantity measurement computer failed	See FQMC FAILURE—in the *AFM*
FUEL FILT BYPASS (L/R)	Fuel filter is clogged	See FUEL FILTER CLOGGED—in the *AFM*
FUEL STBY PUMP ON (L/R)	Fuel standby pump is operating—Comes on when: • Fuel pressure drops, or • STBY PUMP switch is ON, or • FUEL JETTISON pushbutton is pressed	See FUEL PRESSURE LOW—in the *AFM*
FUEL UNBALANCE	Asymmetry between left and right wing fuel quantity greater than 400 lb	See WING FUEL UNBALANCE—in the *AFM*
GEN OFF (L/R)	Generator failure or generator off line	See FAILURE OF ONE GENERATOR—in the *AFM*
GEN OVERHEAT (L/R)	Generator temperature high	See FAILURE OF ONE GENERATOR—in the *AFM*

Table 4-2. CAUTION MESSAGES (Cont)

MESSAGE	MESSAGE MEANING	ACTION/REFERENCE
GEN OVERLOAD (L/R)	Generator load above limits	See GENERATOR OVER LOAD—in the *AFM*
GND BRK WOW MISCOMP	Ground brakes weight on wheels switches miscompare	Do not arm GROUND A/B before touchdown
GROUND PROX FAIL	EGPWS system ground proximity warning failure	
HYD PUMP PRESS LOW (L/R)	Hyd pump failure	See FAILURE OF ONE HYDRAULIC PUMP—in the *AFM* If both messages are on: See FAILURE OF COMPLETE MAIN HYDRAULIC SYSTEM—in the *AFM*
HYD TANK PRESS LOW	Hydraulic tank pressurization is low (both systems)	Descend to 35,000 ft or below. Landing gear extension above 8,000 ft is not recommended
JETT AUTO STOP INOP	Fuel jettison automatic stop unavailable	See FUEL JETTISON—in the *AFM*
MACH TRIM FAIL	Autopilot Mach trim failure	See MACH TRIM FAILURE—in the *AFM*
MAIN HYD LEVEL LOW	Main hydraulic tank fluid level is low	Main and auxiliary hydraulic pressure—Monitor (auxiliary hydraulic pump comes on automatically)
MAIN HYD PRESS HI	Hydraulic pressure above 3,500 psi	HYD PRESSURE and TEMP—Monitor
MAIN HYD TEMP HI	Main hydraulic system temperature above limits	See HYDRAULIC SYSTEM OVERHEAT—in the *AFM*
NAC ANTI ICE FAIL (L/R)	Engine bleed pressure insufficient for anti-icing or engine/nacelle anti-ice control has failed	See ENGINE/NACELLE ANTI-ICE SYSTEM FAILURE—in the *AFM*
NOSE TEMP HI	Nose compartment blowers malfunction	See NOSE COMPARTMENT TEMPERATURE HIGH—in the *AFM*

Table 4-2. CAUTION MESSAGES (Cont)

MESSAGE	MESSAGE MEANING	ACTION/REFERENCE
NWS INOP	Nosewheel is down and locked and nosewheel steering system is off or not aligned	NWS CONNECT/DISCONNECT switch—Check/Reset If system stays off, use differential braking
OIL PRESS HI (L/R)	Engine oil pressure above limit	1. Thrust—Reduce as necessary to bring pressure within limits as soon as practicable 2. If still above limits—Idle engine for the remainder of flight
OIL PRESS LOW (L/R)	Engine oil pressure is low	Change engine thrust and/or altitude
OIL TEMP HI (L/R)	Engine oil temperature is high	Change engine thrust and/or altitude
OIL TEMP LOW (L/R)	Engine oil temperature is low	Heat engine prior to ground start
OXY MASKS PRESS LOW	Oxygen pressure to crew and passenger oxygen controller below 55 psi	1. OXYGEN SHUTOFF switch—Check ON 2. Oxygen pressure—Check 3. Descend to 12,500 ft or below
OXY QTY LOW	Oxygen cylinder pressure is less than 800 psi	Check time remaining at current oxygen consumption. See Figure 7-35-6 or Figure 7-35-7 in the *AFM*
PITOT HEAT FAIL (L/R)	Power supply failure	PROBES HEAT switch—OVRRD as applicable—If message still on, use cross-side ADC
SAT/TAS HEAT FAIL	Power supply failure	PROBES HEAT switch—OVRRD as applicable
SIDE WDO HEAT FAIL (L/R)	Failure in one side window heat system	See SIDE WINDOW MESSAGE ON—in the *AFM*
SLATS UNBALANCE	Failure of slat system or asymmetry between left and right slats is more than 3°	See FLAPS OR SLATS FAILURE—in the *AFM*
STALL WARNING FAIL	Stall warning system has failed	Information only

Table 4-2. CAUTION MESSAGES (Cont)

MESSAGE	MESSAGE MEANING	ACTION/REFERENCE
T/R FAIL (L/R)	1. Thrust reverser system failure 2. In flight—If ***both*** L and R messages simultaneously ON—weight on wheel switch failure	1. See THRUST REVERSER FAILURE—in the *AFM* 2. If ***both*** L and R messages simultaneously ON, do not arm GROUND A/B before touchdown. See THRUST REVERSER FAILURE—in the *AFM*
TERRAIN FAIL	EGPWS system terrain warning failure	
WINDSHEAR FAIL	EGPWS system windshear warning failure	
WING FUEL LVL LOW (L/R)	Low fuel quantity—approximately 450 lb in either wing tank	See FUEL LEVEL LOW—in the *AFM*
WNDSHLD HEAT FAIL (L/R)	Failure in one windshield heat system	If CB tripped—Attempt reset

Table 4-3. ADVISORY MESSAGES

MESSAGE	MESSAGE MEANING	ACTION/REFERENCE
APU READY	APU is on and running and ready to load up	
CAB PRESS TEST OK	Successful cabin pressure control system test	
CONFIG SLATS BYPASS	Takeoff configuration setting of flaps—20° and SLATS BYPASS selected	
FLT A/B INBD OUT	Flight (inboard) airbrakes extended	
FLT A/B OUTBD OUT	Flight (outboard) airbrakes extended	
GND A/B OUT	Ground airbrakes extended	
GND A/B OUTBD OUT	Ground (outboard) airbrakes extended	
IGNITION ON (L/R)	Engine ignition is on	Monitor appropriate engine
PARKING BRAKE ON	Parking brake lever is in the PARK position and adequate hydraulic pressure is sensed at the brakes	
SELCAL DATALINK	SELCAL datalink extablished. Aural alert—SELCAL	
SELCAL HF 1/2	Incoming SELCAL call on HF 1 or 2 Aural alert—SELCAL	
SELCAL VHF 1/2/3	Incoming SELCAL call on VHF 1, 2, or 3 Aural alert—SELCAL	

Table 4-4. STATUS MESSAGES

MESSAGE	MESSAGE MEANING	ACTION/REFERENCE
APU GEN OFF	APU is operating and APU generator is disconnected	See APU GENERATOR FAILURE, in the *AFM*
APU OIL LEVEL LOW	APU oil quantity low	
APU OVERSPEED	APU overspeed—APU automatically shuts down	APU MASTER switch—OFF
APU OVERTEMP	APU overtemperature—APU automatically shuts down	APU MASTER switch—OFF
AURAL DISABLE (A/B)	Aural alert channel is disabeled	
AUX HEADSET CONN	Maintenance headset connected	
CAB PRESS IN TEST	Cabin pressure control system test is in progress	Verify TEST OK message on at end of test
CAB PRESS MONITOR	Malfunction in cabin pressure control monitoring	Cabin altitude—Monitor
CABIN CALL	Cabin call pushbutton pressed	
CABIN DOOR UNLOCK	Cabin door verified locked and cabin reset pushbutton pressed after CABIN DOOR UNLOCK caution message	See CABIN DOOR UNLOCKED, in the *AFM*
CAS MISCOMPARE	Difference in crew alert system channels data	
DCU FAN FAIL	DCU fan failure	

Table 4-4. STATUS MESSAGES (Cont)

MESSAGE	MESSAGE MEANING	ACTION/REFERENCE
DEEC COMM FAIL (L/R)	DEEC Arinc 429 communication failure	
ECS BLEED OFF	ECS selector in RAM position or APU position and APU not running	
ECTM DOWNLOAD (L/R)	Engine condition trend monitoring download—to be performed on ground	See ECTM DOWNLOAD (L/R), in the *AFM*
ENG CHIP DETECT (L/R)	Metal particals found in engine oil	See ENGINE CHIP DETECTOR, in the *AFM*
ENG COMPARE INOP (L/R)	Engine data comparator inoperative between DCU and DEEC	
FUEL FILTER BYPASS (L/R)	Respective fuel filter is clogged	See OIL FILTER CLOGGED, in the *AFM*
FUEL QTY COMPNS FAIL	Fuel quanity indication degraded	
GEN LOAD UNBALANCE	Unbalanced electrical load on generators	Check generators current
LAV CALL	Lavatory call pushbutton pressed	
MAINTENANCE	New maintenance information available in MDC	
MDC DATA ERROR	MDC data error	
OIL FILTER BYPASS (L/R)	Engine oil filter is clogged	See OIL FILTER CLOGGED, in the *AFM*
PARKING BRAKE ON	Parking brake engaged	
PHONE CALL	Cabin interphone call	

CHAPTER 5
FUEL SYSTEM

CONTENTS

ILLUSTRATIONS

CHAPTER 5
FUEL SYSTEM

INTRODUCTION

The G150 fuel system supplies fuel to the engines and the auxiliary power unit (APU). The fuel system consists of storage, distribution, refuel and defuel capability as well as indicating, warning and control.

GENERAL

Fuel is stored in tanks located in the fuselage and wing sections of the aircraft. The fuel tanks in the fuselage include the center tank, two collector boxes, and fuselage tank. The wing tanks are integral to each wing (Figure 5-1).

Fuel is supplied to each engine by two independent systems. Two collector tanks (boxes) in the center most section of the center tank house boost jet pumps that directly feed the engines.

There is one standby electrical pump in each collector box and is used for engine start, fuel jettison, and in the event of boost jet pump failure.

Motive flow is also supplied by either the boost jet pump or standby pump to two transfer jet pumps located in the center section of the center tank. Additional motive flow is provided by two transfer jet pumps at the inboard section of each wing.

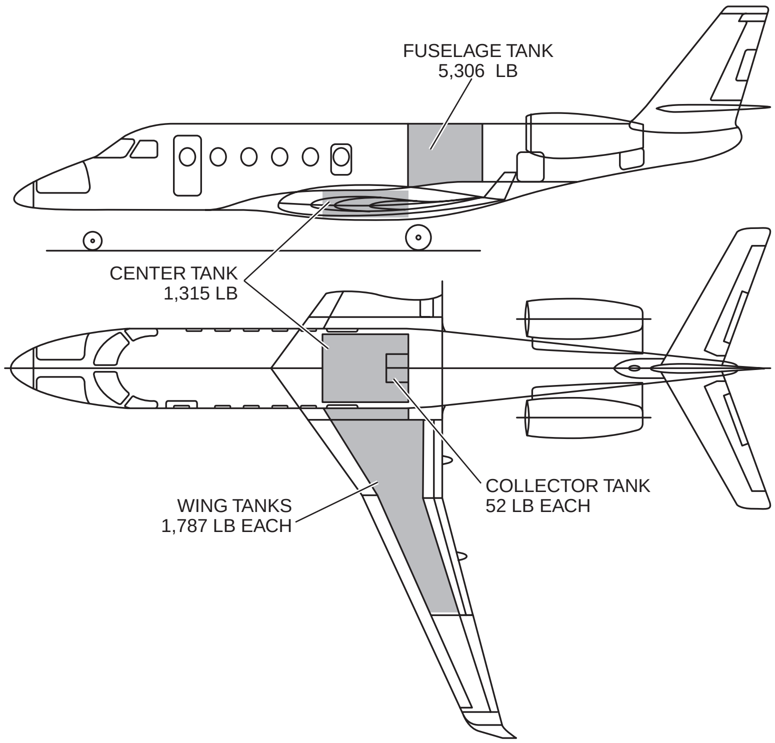


Figure 5-1. Fuel Tank Arrangement

Fuel shutoff valves are installed in the engine feed lines just aft of the center fuel tank in the wing carry-through structure. The auxiliary power unit (APU) feed line branches from the right engine feed line with a separate fuel shutoff valve.

The two wing tanks can be interconnected by an electrically controlled interconnect valve in the center tank, which enables lateral fuel balancing. The two collector tanks are linked through an interconnect valve in the left collector tank that balances its fuel and allows all usable fuel to flow to the engines.

The fuel distribution system transfers the fuel automatically in a specific sequence by either gravity or transfer jet pumps to maintaining the center of gravity (CG) within the desired envelope. Forward and aft electrically controlled interconnect valves enable lateral fuel balancing.

The fuel vent system vents all tanks into a common expansion space at the uppermost part of the fuselage tank. The expansion space drains overboard through two independent outlets near the wingtips.

The aircraft is refueled by pressure or gravity fueling. Total fuel load is reduced by 655 lb when using gravity refueling.

The fuel jettison system reduces aircraft weight by jettisoning fuel overboard during flight.

Fuel quantity indications in the flight compartment indicate and monitor the fuel status.

Drain valves located at various low points in the system facilitate drainage of fuel or contaminants. Flush mounted access panels on the lower wing skin and fuselage surface provide access to each tank.

FUEL STORAGE SYSTEM

FUSELAGE TANK

The bladder type, vapor tight fuselage tank is located within the fuselage aft of the aft pressure bulkhead. A stand pipe divides the fuselage tank into an upper and lower section. The fuel from the upper portion of the fuselage tank constantly fills the center tank via gravity while fuel from the lower portion constantly fills the wing tanks (Figure 5-2).

The fuselage tank contains the following components:

- Overflow standpipe
- Two fuel probes
- Vent lines
- Refueling shutoff valve
- Refueling pilot valve
- Gravity refueling valve

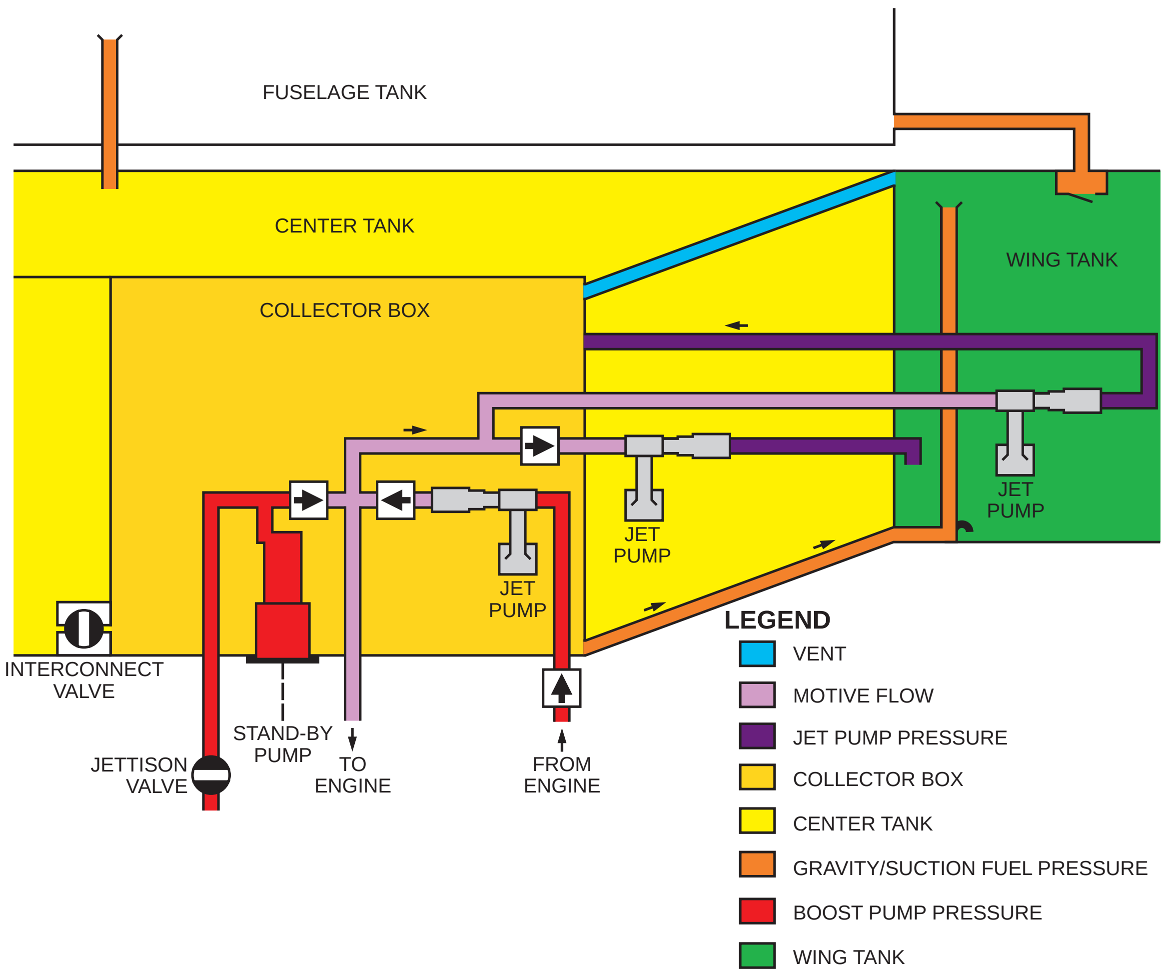


Figure 5-2. Engine and APU Feed System

CENTER TANK

The center tank is a metal tank located in the carry-through structure at the bottom of the lowest point in the fuselage. The center tank contains two sealed collector boxes, two transfer jet pumps, the wing interconnect valve, and two drain valves. The center tank is filled through the overflow standpipe from the fuselage tank. Fuel is constantly being transferred to the wing tanks by the two transfer jet pumps just forward of the collector boxes.

Collector Boxes

The left and right collector boxes are in the aft center section of the center tank at its lowest point. They are divided by a metal partition that houses an interconnect valve. The collector box interconnect valve is located in the left collector box. Each collector box supplies fuel directly to its respective engine. There is no way to pump fuel from one collector box to the opposite engine.

Fuel is supplied to the collector box by gravity and a transfer jet pump in the respective wing. Each collector box contains a booster jet pump, an electrically-driven standby pump, drain valve, float switch, check valves, flapper valves, connecting pipes, and electrical wiring.

WING TANKS

Each wing tank is divided into three compartments with flapper check valves that allow the fuel to only flow inboard. The all metal wing tanks are integrated with the torsion box. They extend outboard and are contained between the front and rear wing spars. The wing tanks are initially filled by fuel from the center tank, through the transfer jet pump. When the center tank is empty, the wing tanks are filled from the lower section of the fuselage via the overflow standpipe. A gravity-balance line equipped with a motor operated interconnect shutoff valve connects the inboard compartments of the two wings and allows balancing of the fuel between the wings.

DRAIN VALVES

Fuel drains are used to drain all moisture and foreign matter from the fuel system. The drains are located at the lowest points of the wing, center, collector box tanks as follows:

- One for vent drain plenum
- Each collector box
- Center tank left and right sides
- Each wing tank at the inboard compartments

FUEL TANK VENT SYSTEM

Venting of the fuel system is accomplished through internal pipes and vent lines (Figure 5-3).

All the tanks are vented into the common expansion space at the uppermost part of the fuselage tank. The expansion space is vented overboard through two independent vent lines through non-icing NACA scoop outlets located on the underside of each wing. Fuel expansion space is assured by the height difference between the refueling shutoff pilot valve location and the inlets to the overboard vent lines. The outboard compartments of the wing tanks are also vented through float valves to the same NACA scoops.

The vent system prevents overpressure in case of failure of the refueling shutoff valve or during climb. It also prevents excessive negative pressure during emergency descent.

A small line is routed from the main wing vents to a fuel vent plenum sump located at the bottom of the fuselage forward of the left main landing gear wheel well. This plenum collects any moisture or fuel that accumulates in the vent lines and should be drained during preflight.

FUEL DISTRIBUTION SYSTEM

SYSTEM DESCRIPTION

The fuel distribution system includes five subsystems.

- Engine feed
- APU feed
- Fuel transfer
- Refueling and defueling
- Fuel jettison

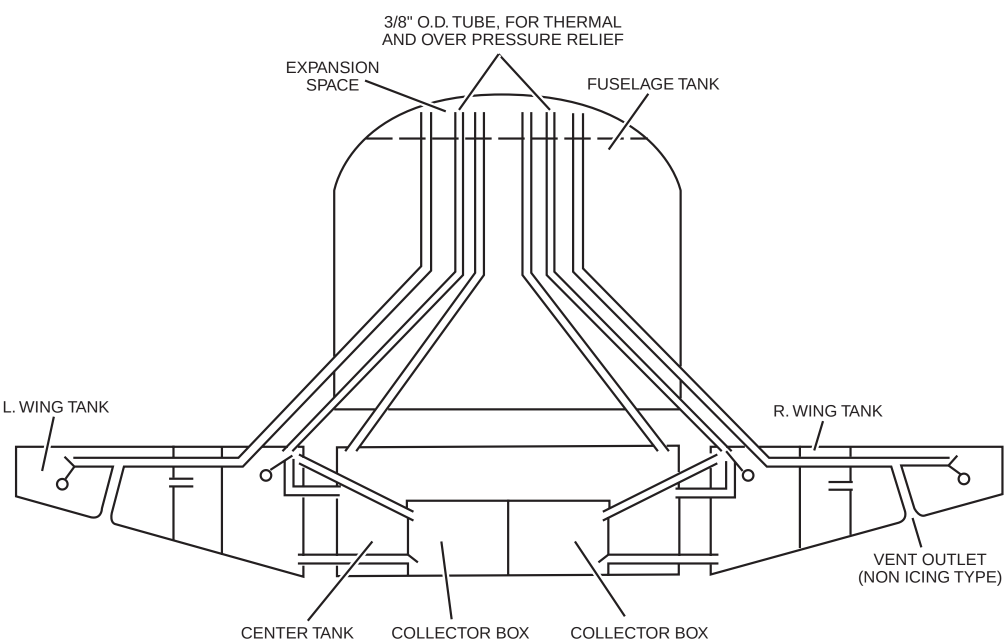


Figure 5-3. Fuel Vent System

ENGINE FEED

The separate collector boxes located in the center tank, supply all the fuel to their respective engines and are completely independent (Figure 5-4). Fuel enters the collector boxes by gravity or from the transfer jet pumps in the respective wing. Each supply consists of one boost jet pump and one electric DC motor-driven standby pump in each collector box. The boost jet pump and electric standby pump are separated by check valves that ensure independent operation of the two pumps.

The transfer system ensures availability of all fuel to the collector boxes, providing a continuous fuel supply throughout the flight. An interconnect valve connects the collector boxes in case of one engine malfunction or fuel unbalance.

The boost jet pump is actuated by motive flow fed back from the engine-driven HP fuel pump. The motive flow, filtered by the engine filter, allows very reliable and trouble-free operation of the boost jet pumps under all operating conditions. The engine motive flow lines to the collector boxes pass through check valves to prevent any loss of fuel from the collector box in the event of a motive flow line rupture.

Depending on the engine motive flow, the output of the boost jet pump is approximately 12 to 20 psig. This output pressure is sufficient for all engine requirements and for actuating the transfer jet pumps, except for jettison. During jettison operations, the standby pump supplies sufficient pressure.

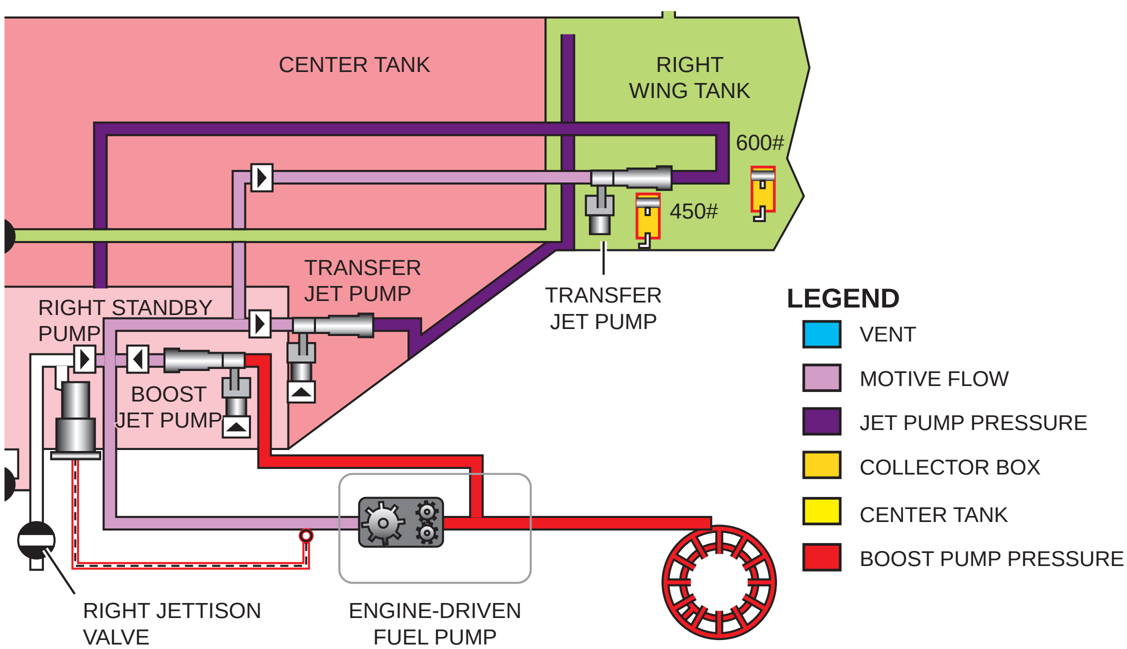


Figure 5-4. Engine and APU Feed System (Aft Looking Forward)

APU FEED

The APU feed line branches off from the right engine main feed line through a motor-operated APU shutoff valve (SOV). The shutoff valve is installed as close as possible to the branch. Installing the APU SOV in this location decreases the number of potential main engine feed line leakage points. The APU SOV opens only during APU operation. The APU feed line is routed directly to the APU firewall. A thermal relief valve, which is an integral part of the APU shutoff valve, allows for the release of thermal expansion pressure.

FUEL USAGE

Fuel usage sequence is intended to keep the CG within the envelope. The sequence is automatically controlled by the center tank transfer jet pumps, the fuselage tank overflow stand pipe, the fuselage transfer lines, and by the wing transfer jet pumps (Figure 5-5).

The fuel usage sequence is as follows:

- Fuel is first used from the upper part of the fuselage tank through the stand pipe to the center tank.
- Then from the center tank to the wings through the center tank transfer jet pumps.
- When the center tank is empty, the lower part of the fuselage tank is depleted by constant transfer to the wings.
- Fuel is then depleted from the wing tanks.
- Collector box fuel burns last to the engines.

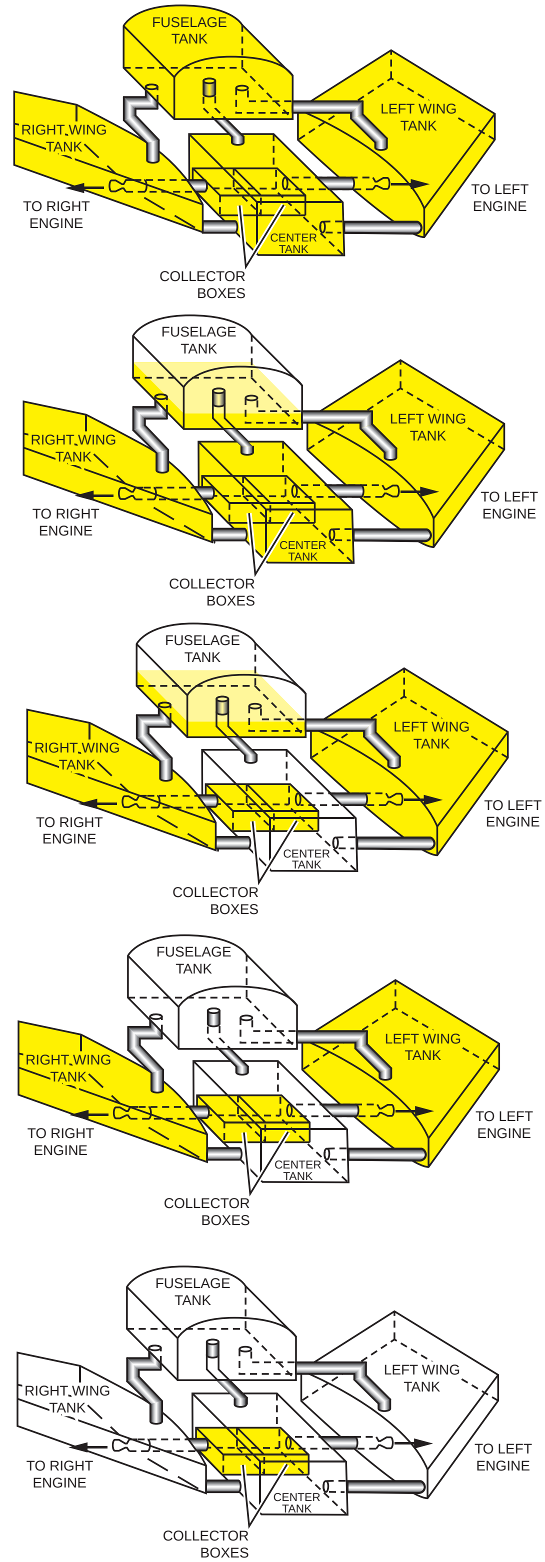


Figure 5-5. Fuel Usage Sequence

Standby Pump

One standby pump (Figure 5-6) is mounted in each collector box. It is a 28-VDC submerged, centrifugal pump. Power for the left pump is from the No. 1 main bus with automatic control from the No. 1 distribution bus through a 3-amp circuit breaker. The right pump is powered from the No. 2 main bus with automatic control through the No. 2 distribution bus and 3-amp circuit breaker.

The pumps may be selected to the AUTO–OFF–ON position by their respective switch on the overhead panel as follows:

- AUTO mode—Pump activates automatically if the fuel pressure at the engine inlet drops below 6.0 psig.
- OFF mode—Pressure switch does not activate the pump. The pump operates regardless of switch position when jettison system is activated.
- ON mode—Pump activates regardless of other conditions.

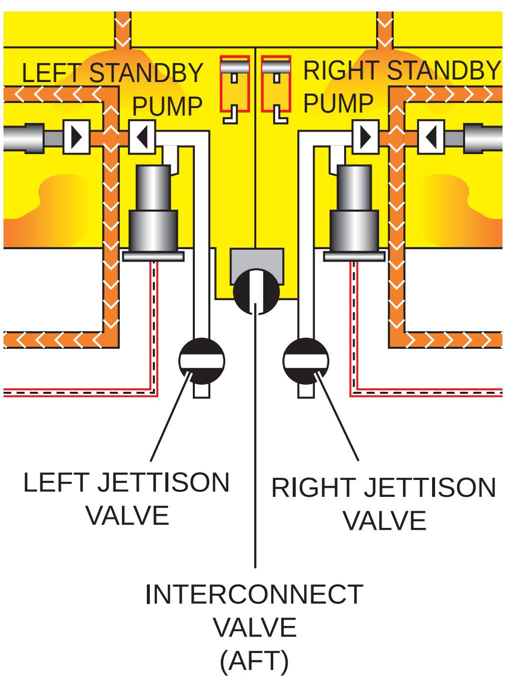


Figure 5-6. Standby Pump (Typical)

NOTE

If the standby pump automatically activates due to a drop in pressure to the engine inlet, it is necessary to cycle the switch from OFF to AUTO to reset the pump.

Transfer Jet Pumps

The transfer jet pumps are operated by motive flow from the output of either the boost jet pump or the standby pump respectively (Figure 5-7).

Check valves prevent reverse motive flow into the respective tanks. The check valves also prevent suction of air through the jet pumps when the engine operates in the suction mode.

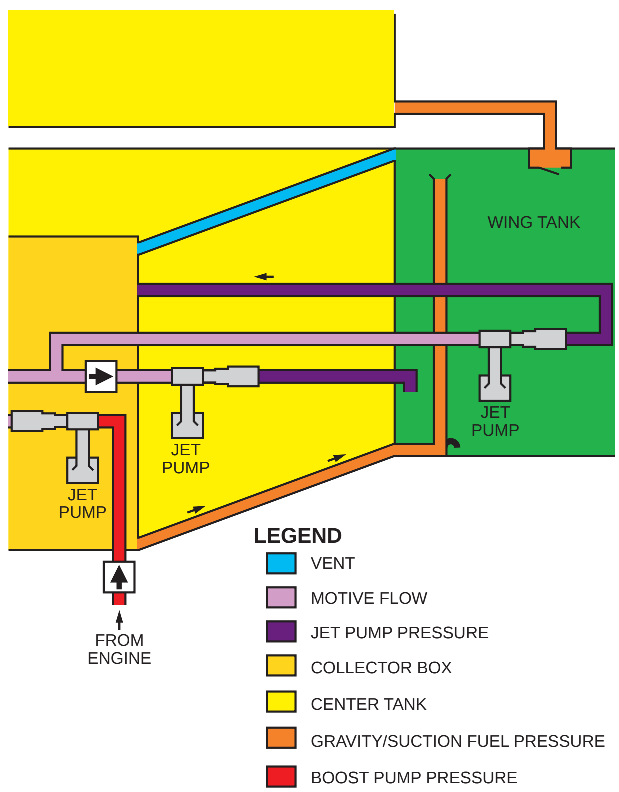


Figure 5-7. Transfer Jet Pumps

Interconnect Valves

The fuel system includes two motor-operated interconnect valves (Figure 5-8):

- The forward valve interconnects the left and right wing tanks in a line that passes through the center tank forward of the collector boxes
- The aft valve in the left collector box interconnects the left and right collector boxes

These valves provide lateral fuel balancing. Both interconnect valves are controlled by a two-position switch on the overhead panel labeled INTERCONNECT OPEN–CLOSE.

Operation

Valve Position

Both the FWD and AFT valves are monitored by amber lights as follows:

- OPEN light remains illuminated when associated valve is open
- IN TRAN light indicates that the associated valve is in transit or is not in the same position as the switch. The light extinguishes when the associated valve completes its travel to the selected position.

Fuel Unbalance

When the unbalance between the wing tanks reaches or exceeds 400 lbs., the FUEL UNBALANCE EICAS message appears. To correct the unbalance, select the INTERCONNECT switch to the OPEN position which opens both interconnect valves allowing fuel to flow to the low side by gravity.

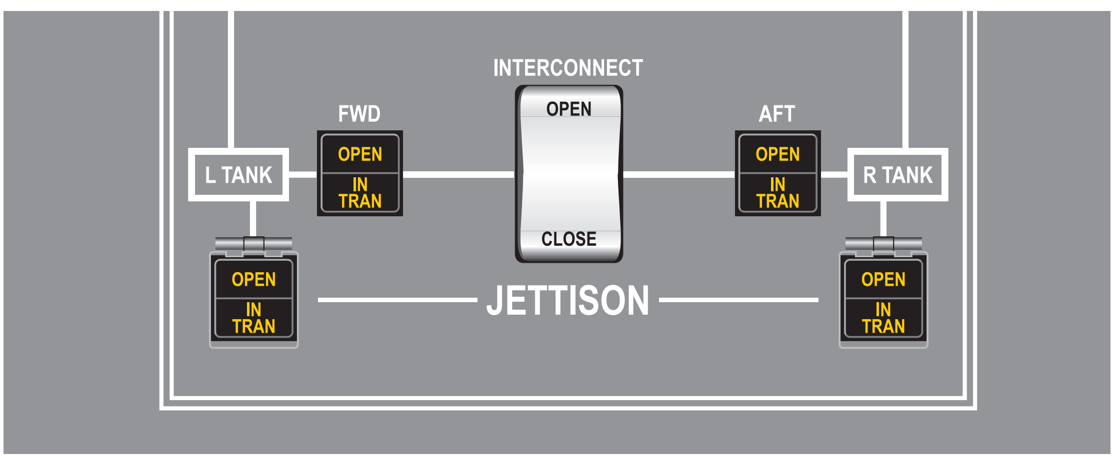


Figure 5-8. Interconnect Switch Valves

FUEL QUANTITY SYSTEM

SYSTEM DESCRIPTION

The fuel quantity system comprises systems for both measuring and indicating fuel quantity (Figure 5-9).

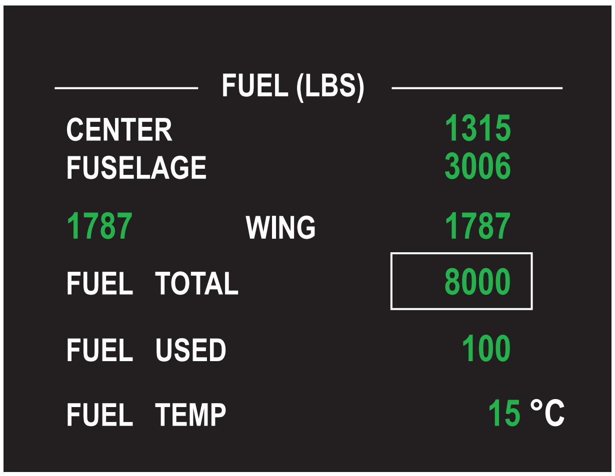


Figure 5-9. Fuel Indication EICAS Display

MEASUREMENT SYSTEM

The fuel quantity measurement computer (FQMC) is in the aft left side of the cabin.

INDICATION SYSTEM

On the MFD, the fuel quantity indication system provides the flight crew with fuel quantity.The center tank, fuselage tank, wing tanks, and total fuel quantities are displayed on the secondary page of the MFD. The tank quantities are displayed from the top down in the order that the tanks are emptied.

The quantities are displayed as kilograms or pounds depending on the system configuration.

Fuel temperature in degrees Celsius, and fuel used are also displayed on the secondary page of the MFD.

REFUELING

The aircraft is refueled either by gravity or through a single-point pressure refueling (SPPR) adaptor.

The single-point pressure refueling (SPPR) adaptor is inside an access door on the aft right side of the fuselage aft of the right wing (Figure 5-10).

Figure 5-10. Aircraft Fuel Panel

A microswitch in the door causes the BAGG/FUEL DOOR caution message on the EICAS to illuminate when the refueling door is secured. The refuel door must be securely latched.

The pressure refueling system is designed to refuel the tanks at a maximum pressure of 55 psi with automatic fuel level control. Pressure refueling allows partial refueling or complete refueling.

A decal inside the refueling panel door provides the procedures for refueling the aircraft.

The gravity refueling port is located on the upper right side of the fuselage forward of the right engine inlet and provides another means to fill the tanks. The tanks fill in the same sequence as with the pressure refueling system, thereby ensuring the aircraft remains within its CG.

Maximum fuel capacity is 655 lbs less when using gravity refueling over pressure refueling.

The refueling sequence allows the aircraft to be ready for immediate takeoff with any fuel quantity. The tank filling sequence is as follows:

- Wings and collector boxes
- Fuselage tank lower part
- Center tank
- Fuselage tank upper part

During refueling, fuel enters the fuselage tank through the refueling shutoff valve at the bottom of the fuselage tank and flows by gravity to the remaining tanks. The refueling shutoff valve is operated using the incoming fuel pressure and is controlled by a float valve at the top of the fuselage tank. The float valve controls the maximum fuel level and establishes the expansion space at the top of the fuselage tank (Figure 5-11).

Operation

The float valve operation is as follows:

- The tank has a drain port with an attached electrical solenoid and a tube to the shutoff valve.
- With full tank, the float rises and closes the port to the shutoff valve to cause incoming truck pressure to force the shutoff valve closed.
- The solenoid allows partial fueling. When the solenoid is energized, it closes the drain port. This causes the float to close the port to the shutoff valve, which, in turn, causes the shutoff valve to close.
- The solenoid receives its electrical power from the right hot battery bus and is controlled by the refuel TEST/NORMAL switch located at the refueling panel.

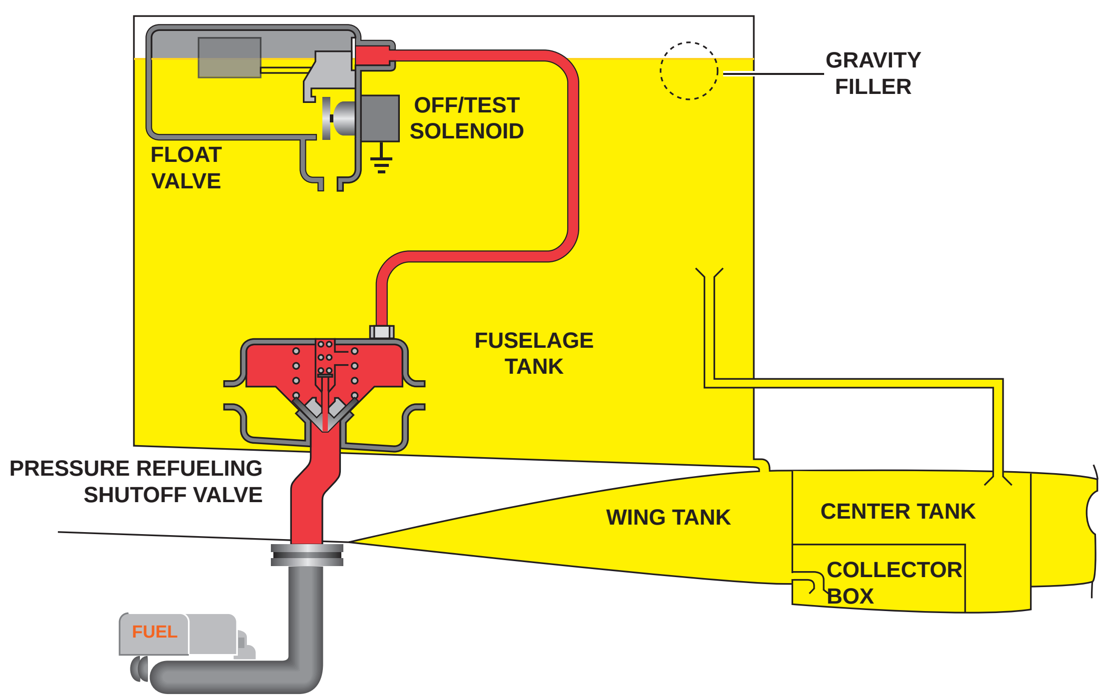


Figure 5-11. Pressure Refueling System

The solenoid may also be energized by the REFUEL OFF switch on the center console (Figure 5-12).

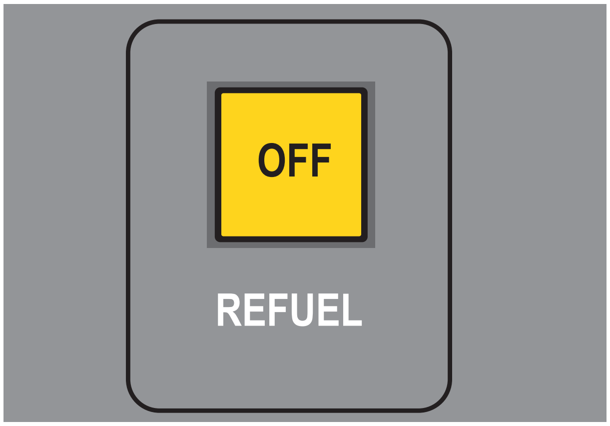


Figure 5-12. Center Console Refuel Switch

DEFUELING

The jettison system provides defueling of the aircraft (see Figure 5-10).

The guarded DEFUEL switch ensures the refueling door cannot be closed with the switch selected to the DEFUEL position.

NOTE

For maintenance purposes only, the 600 lb switches in the wings may be bypassed by placing the NORMAL/DEFUEL switch on the pressure refueling panel in the DEFUEL position.

Fuel Jettison System

Fuel jettison during flight is possible from either one or both wings simultaneously. Each jettison valve circuit has a float switch that automatically closes the valve to stop the jettison when fuel level in the respective wing reaches 600 lbs of fuel remaining. This allows the other wing to continue until it reaches the 600 lb fuel remaining level.

The motor-operated jettison valves are in the jettison line. The jettison outlets are between the outboard flap and aileron on each wing.

Control

Each jettison valve is controlled by a split-lens, alternate-action type switchlight (Figure 5-13) labeled JETTISON, OPEN-IN TRAN.

- OPEN—Illuminated when the associated jettison valve is open
- IN TRAN—Illuminated when valve travels to the selected position

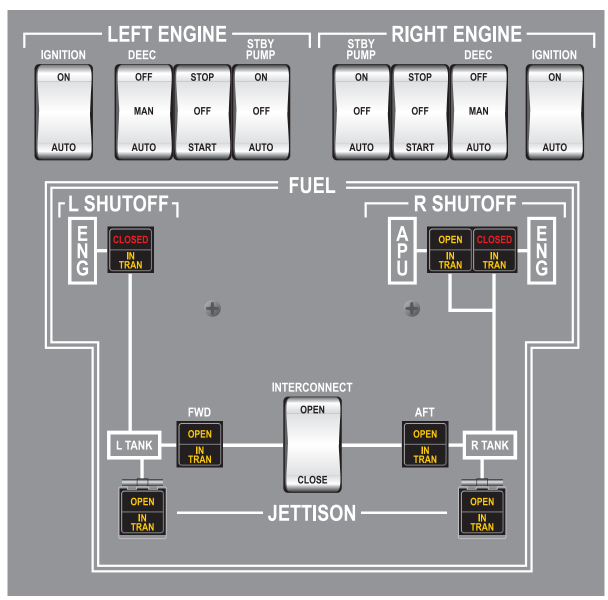


Figure 5-13. Overhead Fuel Panel

Operation

- Selecting fuel jettison switchlight on overhead fuel panel:
  - Opens the respective jettison valve and the amber IN TRAN light illuminates
  - IN TRAN light extinguishes and the OPEN light illuminates when jettison valve reaches fully open position.
  - Selecting jettison button automatically activates the standby fuel pump regardless of its switch position.
  - The jettison circuit may be deactivated, either by pushing the jettison pushbutton or automatically when the fuel level in that wing reaches the 600 lb fuel remaining setting, the standby pump deactivates

CONTROLS AND INDICATIONS

OVERHEAD PANEL

Figure 5-13 shows the fuel switches on the flight compartment overhead panel and all related switches.

STBY PUMP Switches

The STBY PUMP switches have the following three positions:

- ON—The standby fuel pump runs continuously.
- OFF—The standby pump is off unless the jettison system has been activated.
- AUTO—The standby pump is off except for the following conditions:
  - Jettison valve selected
  - Engine inlet pressure 6 psi or less
  - During engine start

INTERCONNECT Switch

The INTERCONNECT switch has the following two positions:

- OPEN—Both the wing and collector box interconnect valves are open.
- CLOSE—Both the wing and collector box interconnect valves are closed.

FWD and AFT Lights

- OPEN—Lights illuminate when the respective interconnect valve is open.
- IN TRAN—Lights illuminate when the respective interconnect valve is in transit.

JETTISON Switchlights

There are two jettison switchlights, one for each side:

- OPEN—Illuminates when the respective jettison valve is powered open.
- IN TRAN—Illuminates when the respective jettison valve is in transit.

APU MASTER Switch

The APU MASTER switch (Figure 5-14) has two positions:

- ON—Opens the APU fuel shutoff valve
- OFF—Closes the APU fuel shutoff valve

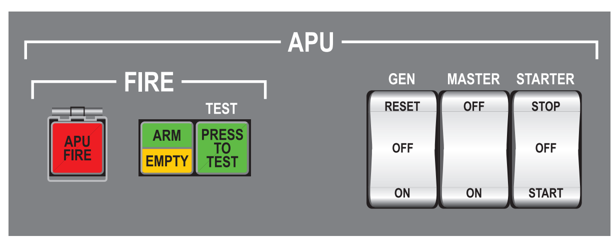


Figure 5-14. APU Control Panel

APU FIRE Switch

The APU FIRE switchlight closes the APU fuel shutoff valve. Pushing the APU FIRE switchlight a second time reopens the APU fuel shutoff valve.

CENTER PEDESTAL

REFUEL OFF

The REFUEL OFF switchlight is located on the pedestal. It illuminates OFF when pushed to stop pressure refueling from the flight compartment. This light receives its electrical power from the right hot battery bus.

INSTRUMENT PANEL

FIRE-OVERHT Switchlights

Figure 5-15 shows the engine and APU fuel shutoff valve switches.

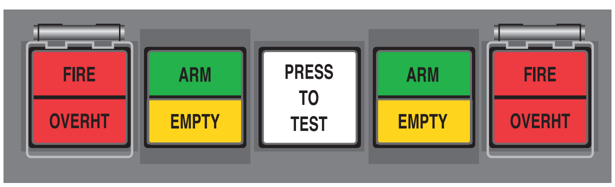


Figure 5-15. Engine and APU Fuel Shutoff Switches

The FIRE–OVERHT switches located on the center instrument panel control the position of the engine fuel shutoff valves which default to the OPEN position.

Selecting the pushbutton closes the valve, which stops fuel flow to the engine. Pressing the pushbutton a second time reopens the valve, allowing fuel flow to the engine.

Each valve has a CLOSE/IN TRAN light located on the flight compartment overhead fuel panel (see Figure 5-13).

The IN TRAN light illuminates only when the valve travel is between the open and closed position. The CLOSE light illuminates when the valve is powered to the closed position.

Fire Shutoff Valves

The fire shutoff valves are installed on the engine feed lines at the engine firewall. The fire shutoff valves are actuated by the flight crew through the FIRE–OVERHT switchlights to stop the fuel flow to the engine in case of fire.

LIMITATIONS

APPROVED FUELS

Approved fuel types are:

- Commercial Kerosene per specification (Jet A and Jet A-1)
- Commercial wide turbine fuel (Jet-B)
- JP-4
- JP-5
- JP-8

FUEL TANK CAPACITIES

Wing tanks (each)	1,787 lbs
Collector box (each)	52.5 lbs
Center tank	1,315 lbs
Fuselage tank	5,306 lbs
Total fuel capacity	10,300 lbs at 6.7 lb/U.S. gallon
	1,537 US gallons

FUEL UNBALANCE

Maximum lateral unbalance:

Takeoff	400 lbs
Cruise and landing	600 lbs

EMERGENCY AND ABNORMAL OPERATIONS

FUEL EICAS MESSAGES

Figure 5-16 depicts the fuel messages as they appear on the multifunction display (MFD).

Warning Messages

COLLECTOR LVL LOW (L/R)—Less than 35 pounds of fuel is in the respective collector box.

FUEL PRESS LOW (L/R)—Indicates a drop in fuel pressure to the engine fuel pump. If the message remains illuminated, it indicates that the boost jet pump has failed and the automatic changeover to the standby pump did not occur.

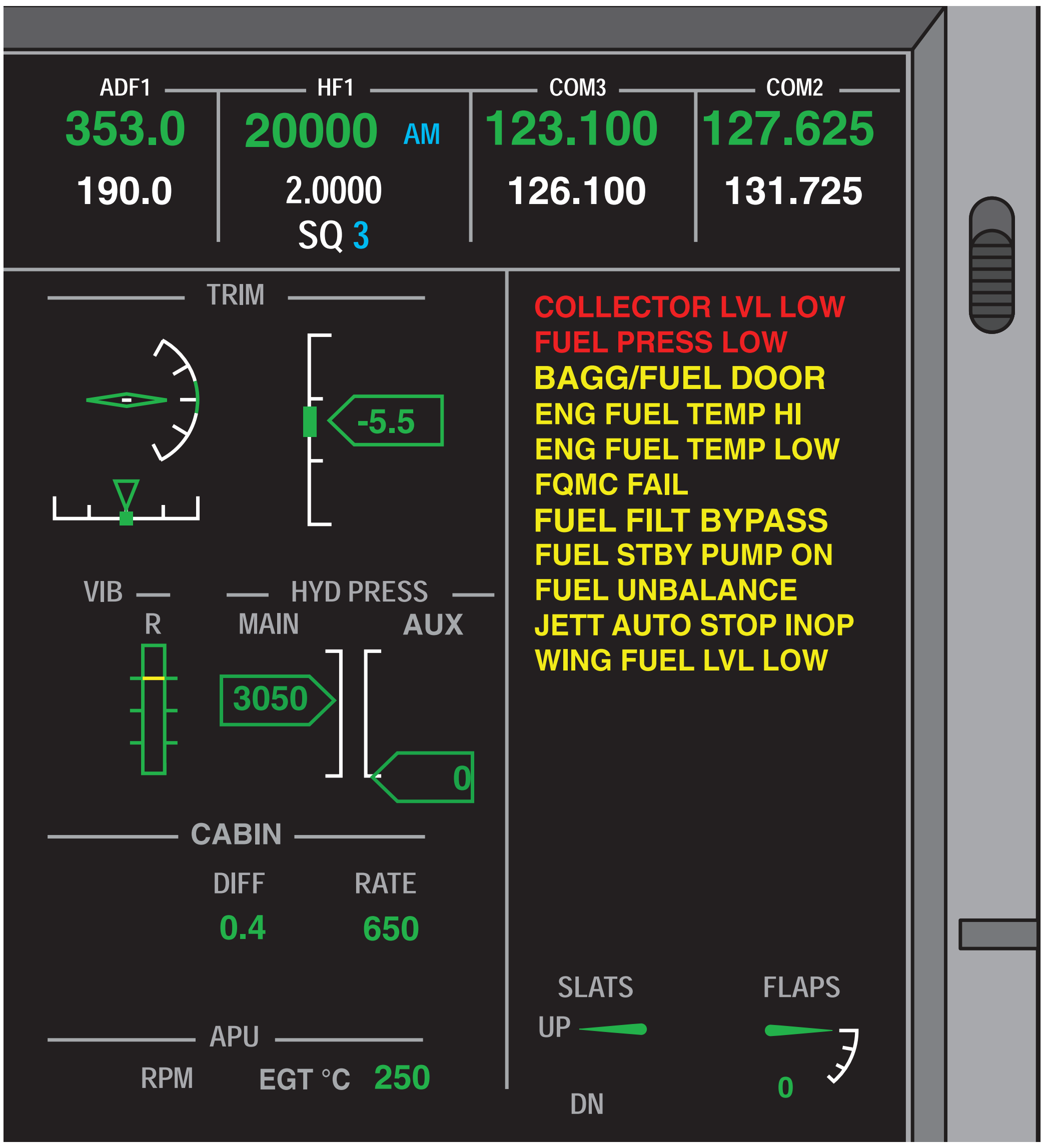


Figure 5-16. Fuel EICAS Indications

Caution Messages

BAGG/FUEL DOOR—Either the baggage compartment or refueling door is not secured.

ENG FUEL TEMP HI (L/R)—Engine fuel inlet temperature is above 275°F.

ENG FUEL TEMP LOW (L/R)—Engine fuel inlet temperature is lower than 50°F.

FQMC FAIL (L/R)—The respective fuel quantity measurement computer (FQMC) failed.

FUEL FILT BYPASS (L/R)—Both fuel filters are clogged.

FUEL STBY PUMP ON (L/R)—Illuminates when the respective standby fuel pump is operating.

FUEL UNBALANCE—Asymmetry between left and right wing fuel greater than 400 pounds.

JETT AUTO STOP INOP—Fuel jettison automatic stop is unavailable.

WING FUEL LVL LOW (L/R)—Respective wing tank is approximately 400 lbs.

Status Messages

FUEL FILTER BYPASS (L/R)—Respective fuel filter is clogged.

FUEL QTY COMPNS FAIL—Fuel quantity monitoring degraded.

QUESTIONS

1. Engine fuel is supplied from the:
A. Associated collector box
B. Fuselage tank
C. Associated wing tank
D. Center tank

2. When the standby pump switch is at AUTO, the pump will run when:
A. The interconnect switch is CLOSED
B. Line pressure drops to 12 psi
C. Wing tank fuel level depletes to 450 pounds
D. Fuel pressure to the engine inlet is low

3. Maximum allowable lateral fuel unbalance is:
A. 600 pounds for all conditions
B. 370 pounds for landing
C. Not limited for cruise
D. 400 pounds for takeoff and 600 pound for cruise and landing

4. WING FUEL LVL LOW (L/R) CAS message indicates:
A. Collector box has 35 pounds of fuel remaining
B. L/R wing tanks contain 600 pounds of fuel
C. Asymmetry between the L/R wing is more than 450 pounds
D. Respective wing tank is approximately 450 pounds

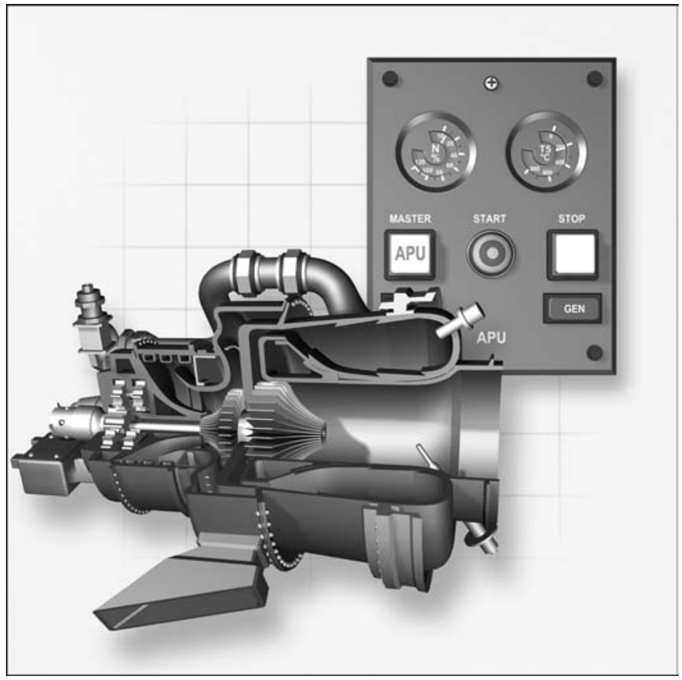
5. Fuel usage sequence is:
A. Upper fuselage tank through the standpipe, center tank, lower fuselage tank, wing tank, collector box
B. Fuselage tank, wing tank, center tank, collector box
C. Center tank, fuselage tank, wing tank, collector box
D. Lower fuselage tank, center tank upper fuselage tank, collector box

CHAPTER 6
AUXILIARY POWER SYSTEM

CONTENTS

ILLUSTRATIONS

CHAPTER 6
AUXILIARY POWER SYSTEM



INTRODUCTION

This chapter covers the general information about the G150 auxiliary power unit.

GENERAL

The Gulfstream G150 is equipped with the Honeywell RE100 auxiliary power unit (Figure 6-1). The purpose of the airborne auxiliary power system, essentially a self-contained power source, is to provide an alternate source of pneumatic power for the environmental control system and shaft power to drive the auxiliary direct current (DC) starter/generator. It is a fully automatic, constant speed gas turbine engine designed to provide power while on the ground or in the air.

SYSTEM DESCRIPTION

The APU is a gas turbine engine that starts with aircraft or ground DC power, and then operates on aircraft-supplied fuel. It is certified as a nonessential system and provides electrical power to the battery bus and bleed air to the ECS ducting when necessary. The APU is mounted in the aircraft tail section with access provided through the equipment bay door in the left aft fuselage.

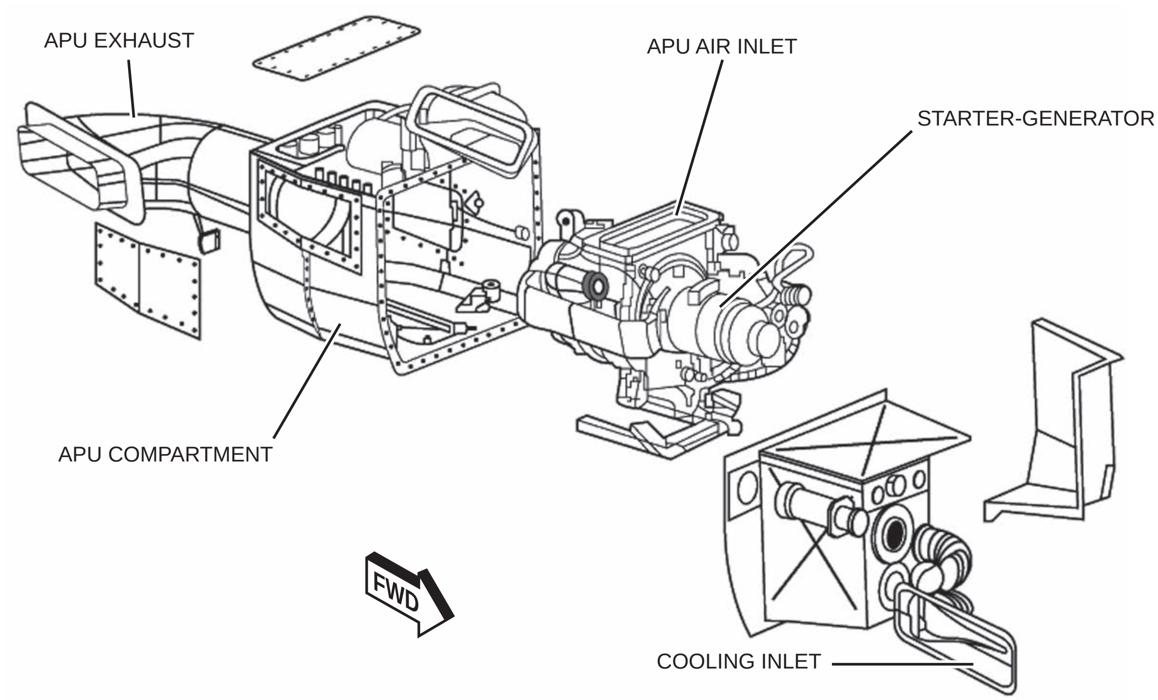


Figure 6-1. APU Components

The engine is computer controlled by sensors that measure the APU operating parameters to control the engine. This ensures that the APU and all of its subsystems operate correctly in response to all environmental and load conditions. It is approved for unattended operation.

COMPONENTS

APU Enclosure

The APU is enclosed within a fire-protecting shield made of titanium sheet metal that also serves as a shield for APU debris containment should rotor disc failure occur.

Intake Duct

The air inlet system consists of an intake duct mounted between the top of the APU and the top right aft side of the fuselage. The intake duct is made of 0.6-inch thick titanium. The air intake with a screened inlet is above the right pylon abeam the leading edge of the vertical stabilizer. A fireproof gasket separates the intake duct from the aircraft skin.

Cooling Duct

A NACA scoop style cooling air inlet duct is below the right engine pylon abeam the right HP blowout disc. The duct is divided into two paths as follows:

- Upper—Cools starter-generator
- Lower—Cools APU enclosure

Exhaust Duct

The exhaust duct is on the right rear fuselage directly below the horizontal stabilizer. It is made of metal and vents the APU exhaust overboard. There is a chrome heat shield mounted with a small air space between it and the fuselage to protect the skin from overheating.

Starter-Generator

The APU is equipped with a 28-VDC starter-generator mounted on the accessory gearbox. The starter-generator is cooled by ambient air drawn in from an inlet on the right side of the empennage, forward of the APU tail cone. The starter-generator is identical to the power-plant starter-generator.

Air System

APU bleed air is supplied via the APU air valve that operates two solenoids. Selecting the APU position on the ECS selector switch (Figure 6-2) energizes solenoid #1. This opens the valve and supplies low-pressure bleed air to the air conditioner.

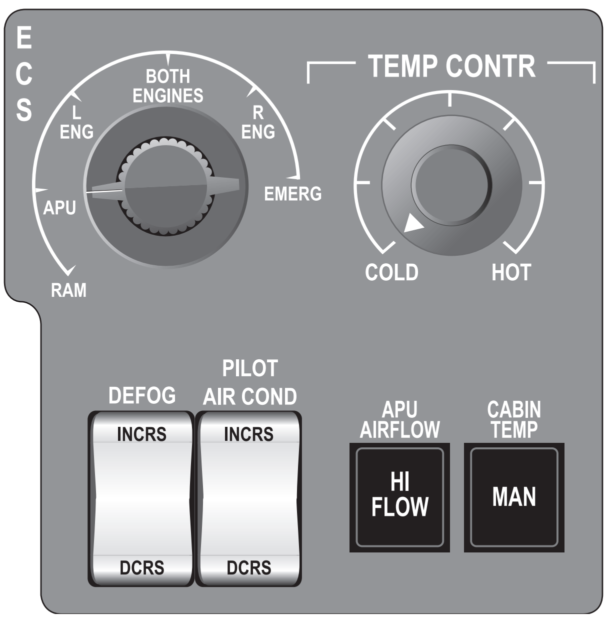


Figure 6-2. APU ECS Panel

Selecting the APU AIRFLOW switchlight energizes solenoid #2. With both solenoids energized, the APU air valve operates in the high-pressure mode. Under these conditions the APU AIRFLOW switch illuminates the words HI FLOW within the switch.

Selecting the APU AIRFLOW switch a second time:

- Deenergizes solenoid # 2
- Extinguishes the words HI FLOW
- Returns valve to the low-pressure mode

Electronic Control Unit

APU operation is controlled through the electronic control unit (ECU). The ECU operates on 28-VDC power. It is an electrically-driven, digital computer based controller that is programmed to control all APU electrical functions. The APU is fully dependent on the ECU for its electrical inputs.

The ECU performs four primary functions:

- Sends signals for engine start and timed acceleration to 100% speed
- Controls the fuel control unit (FCU) torque motor current, and thus fuel supply, to maintain the APU at governed speed for all electrical and pneumatic loads
- Continually monitors engine parameters to ensure the APU operates within specified limits
- Safely shuts down the APU when a normal or protective shutdown signal is received
- Performs shutdown for overspeed, overtemp, or low oil pressure

The ECU receives electrical inputs from the APU, flight compartment switches, and some aircraft subsystems. The inputs are crew commands, data from the APU sensors, conditions of main engine operation, and APU electrical and pneumatic load demands. The ECU analyzes the signals and, as necessary to agree with the system logic, sends a control signal output or ignores the input. The ECU automatically adjusts control parameters for APU nonessential mode (aircraft on ground) and APU essential mode (aircraft airborne) to agree with the status of the essential mode hardwire discrete that is linked to weight-on-wheels (WOW).

OPERATION

CONTROLS

APU controls consist of switches, relays, and electronic equipment to control starting, stopping, and operation of the APU. The APU controls are on the overhead panel (Figure 6-3) and center pedestal (see Figure 6-2) in the flight compartment .

APU indication and monitoring are provided on the MFD display.

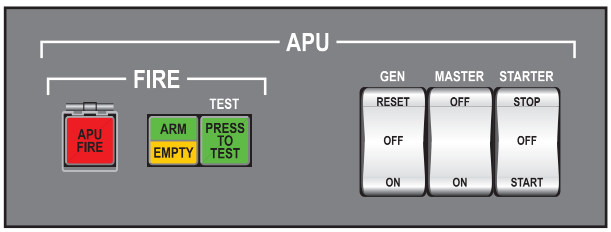


Figure 6-3. APU Panel

Generator Switch

The APU GEN switch is a three-position switch spring-loaded to the OFF position. It operates as follows:

- RESET—Restores magnetic field in the generator
- OFF—Disconnects generator output from aircraft DC buses without energizing the generator
- ON—Connects generator's output to aircraft DC buses

MASTER Switch

The APU MASTER switch is a two-position rocker switch that functions as follows:

- OFF—Turns off power to ECU, closes APU fuel valve and disables APU displays on EICAS
- ON—Provides electrical power to the APU ECU and energizes the APU circuits. It also initiates the APU displays on the MFD, and opens the fuel feed valve.

Start Switch

The APU STARTER switch is a three-position switch spring-loaded to the OFF position. It operates as follows:

- STOP—Inserts a false overspeed signal into the speed sensing circuit to cause the APU to shut down; the inlet door remains in the open position
- OFF—No signal is supplied to the APU ECU
- START—Initiates the start cycle

ECS Selector

The environmental control system (ECS) selector is on the center pedestal. The control knob has an APU position that is used to select APU bleed air for air conditioning or pressurization (see Figure 6-2).

APU AIRFLOW Pushbutton

The APU AIRFLOW pushbutton is on the center pedestal and selects either high or low flow from the APU for air conditioning or pressurization. When the button light is off, the APU is providing low airflow. When the button is pushed and the light comes on, the APU provides high airflow.

APU FIRE PROTECTION

For more information about the APU fire protection system, see Chapter 8, "Fire Protection."

IGNITION/STARTING

When the APU MASTER switch is placed in the ON position, the ECU performs a circuitry check prior to start. A built-in-test (BIT) checks the state of the electronics while a built-in-test-equipment (BITE) test checks the line replaceable units (LRUs). If the ECU finds problems with the LRUs, the ECU issues a start inhibit command. If no problems are found, the ECU transmits its ready status to the cockpit so the APU can be started. The ready status is when the numbers associated with RPM and EGT are displayed.

Upon receiving a signal to actuate the APU, the ECU provides power to the starter-generator to commence rotating the APU turbine engine. When the engine speed reaches approximately 5%, the ECU completes a circuit to the fuel shutoff valve and ignition unit. The fuel shutoff valve opens, allowing fuel flow to the fuel nozzles. The ignition unit commands the igniter plug to fire and ignite the air-fuel mixture in the combustion chamber.

NORMAL OPERATION

The engine has four basic operating modes:

- Ready-to-load (full rpm with no shaft or bleed load)
- Environmental control (bleed load)
- Electrical power generation (shaft load)
- Combination operation (simultaneous shaft and bleed loads)

At engine-governed speed, the fuel control assembly controls fuel flow and automatically controls turbine discharge temperature and rpm within safe, set limits.

During engine operation, the ECU monitors engine speed, oil temperature, exhaust gas temperature (EGT), and oil pressure. At engine speeds above 95%, the ECU records engine operating time and number of engine starts.

NORMAL SHUTDOWN

Under normal conditions, the APU is shut down by positioning the STARTER switch on the overhead panel to the STOP position. This inserts a false overspeed signal into the speed-sensing circuit causing the APU to shut down.

Terminate APU operation at the existing operating condition. If the APU is operating in the loaded condition, shutdown is accomplished from the loaded condition. If the APU is operating in the unloaded condition, shutdown is accomplished from the unloaded condition.

If the APU does not shut down as noted above, an alternate shutdown may be caused by placing the MASTER switch to OFF. When the MASTER switch is positioned to OFF, the APU shuts down.

The APU may be shut down without damage while starting or operating by either of the two methods described above. Electrical power then is removed from the APU and aircraft fuel solenoid valves, and the APU ECU is disarmed. No idle speed running is required prior to APU shut down.

LIMITATIONS

The limitations contained in Section One of the *Airplane Flight Manual* (*AFM*) must be complied with regardless of the type of operation.

- Maximum altitude for APU start 20,000 ft
- Maximum altitude for APU operation 35,000 ft
- Do not operate APU if right fire extinguisher has been activated
- Allow five-minute cool-down periods between APU starts or between shutdown and subsequent APU start. Allow 30 minutes for cool down period after three start attempts
- Do not operate APU if right standby fuel pump is inoperative
- Approved fuels:
  - EMS 53111 (Jet A)
  - EMS 53112 (Jet A-1 and JP-8)
  - EMS 53113 (Jet B and JP-4)
  - EMS 53116 (JP-5)
- Approved oils:
  - Aeroshell/Royce 560
  - Esso/Exxon 2380 Turbo Oil
  - Castrol 5000
  - Mobil Jet Oil II
  - Mobil Jet 254
- APU generator load limits:
  - On ground—300A
  - In flight below 25,000 ft—300A
  - In flight above 25,000 ft—250A
- Exhaust gas temprature:
  - Maximum at governed speed—690°C
  - Maximum transient during start-up to 50% rpm—870°C
- Rotor speed:
  - Normal operations—100% ±1%
  - Overspeed—108%

EMERGENCY AND ABNORMAL OPERATIONS

INDICATING

The indicating system supplies the aural and visual APU operational data necessary for the crew.

The APU system status is available on the flight deck on the engine instruments and crew alerting system (EICAS) and multi-function display (MFD).

EICAS MESSAGES

There are several warning/caution (red/amber) and advisory/status (green/white) messages associated with the APU. Messages having symbol are accompanied by an optional aural alert (Figure 6-4).

Warning Messages (Red)

APU FIRE— APU fire; APU enters automatic shut-down sequence.

APU OVERSPEED—APU rpm too high; APU did not enter automatic shutdown sequence

APU OVERTEMP—APU excessive temperature; APU did not enter automatic shutdown sequence.

L ENG/APU BLEED LEAK—Leak or rupture in bleed air ducting from left engine or APU.

Caution Messages (Amber)

APU FAULT—APU malfunction; APU automatically shuts down.

APU EXHAUST OVERTEMP—Excessive temperature around APU exhaust duct; APU automatically shuts down.

APU GEN OVERHEAT—APU generator temperature high.

APU GEN OVERLOAD—APU generator load above limits.

APU OIL PRESS LOW—APU oil pressure too low; APU automatically shuts down.

Advisory Messages (Green)

APU READY—APU on and running and ready to load up.

Status Messages (White)

APU GEN OFF—APU operating and APU generator disconnected.

APU OIL LEVEL LOW—APU oil quantity low.

APU OVERSPEED—APU overspeed; APU automatically shuts down.

APU OVERTEMP—APU overtemperature; APU automatically shuts down.

Figure 6-4. APU EICAS Indications

QUESTIONS

1. The airborne APU provides:

A. An alternate source of pneumatic power for the ECS (environmental control system)

B. Shaft power to drive the DC starter-generator

C. Power on the ground or in the air

D. All of the above

2. The APU should not be operated when:

A. Right STBY fuel pump in inoperative

B. Aircraft de-icing is in progress

C. Right fire bottle has been activated

D. All of the above

3. The APU load limit on the ground and in-flight below 25,000 feet is:

A. 200 Amps

B. 275 Amps

C. 250 Amps

D. 300 Amps

4. The APU generator and aircraft generators will run in parallel:

A. True

B. False

5. Maximum altitude for operation of the APU generator is:

A. 20,000 feet

B. 35,000 feet

C. 25,000 feet

D. For ground only

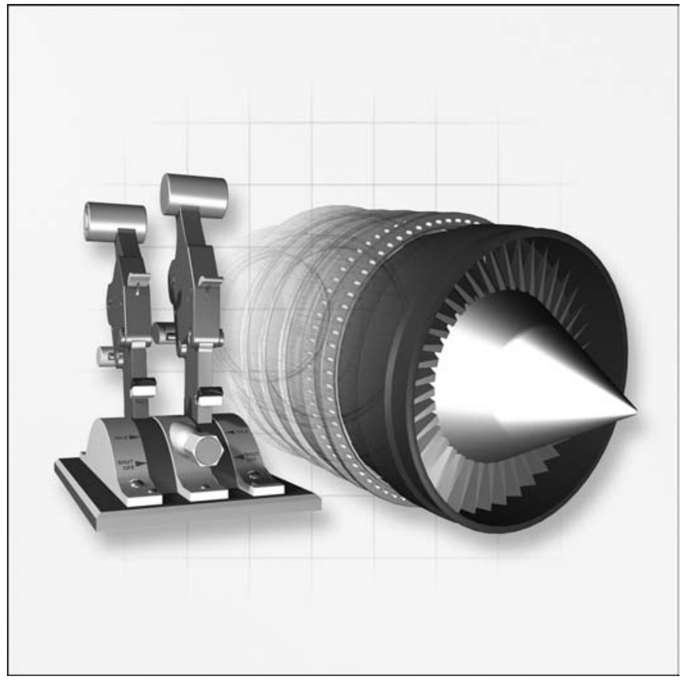
CHAPTER 7
POWERPLANT

CONTENTS

Page

ILLUSTRATIONS

CHAPTER 7 POWERPLANT



INTRODUCTION

The Gulfstream G150 is powered by two aft fuselage-mounted Honeywell TFE731 medium-bypass turbofan engines. Early research and design was done by Garrett AiResearch when they initially developed the engine as an APU for the DC10 aircraft. Shorty after introduction, the engine was adapted for use on the Learjet 35/36, the Falcon 10s and 20s, and is used extensively on many corporate aircraft today. Corporate mergers have resulted in the TFE731 being branded by Honeywell Corporation.

GENERAL

The Honeywell TFE731-40AR-200G is a two-spool, geared front fan, medium bypass ratio turbofan engine (Figure7-1). The low-pressure spool (N_1) consists of a four-stage axial flow compressor coupled through the center shaft to a three-stage axial turbine. It drives the fan through a planetary gearbox.

The high-pressure spool (N_2) consists of a single-stage centrifugal compressor driven by a single stage axial turbine through the outer concentric shaft. It drives the accessory gearbox through a tower shaft.

A reverse-flow annular combustor provides heated high-energy air to the turbines. Each engine is enclosed in a nacelle assembly and mounted on an engine pylon by means of a forward and aft mount.

The digital electronic engine control (DEEC) and a hydromechanical fuel control unit (FCU) provide operational control of this engine. Standard equipment includes automatic power reserve (APR), manual power reserve (MPR), engine synchronization, and thrust reversers.

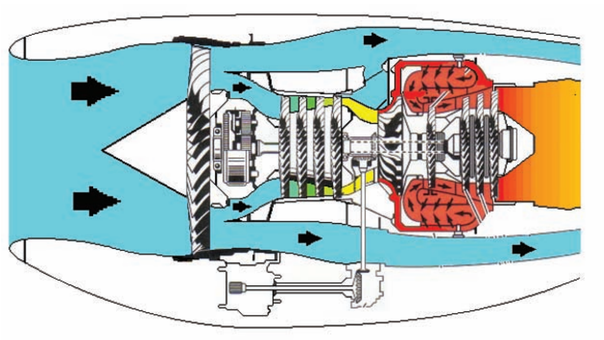


Figure 7-1. Engine Airflow

POWERPLANT CONTROLS

Engine thrust is controlled from the throttle quadrant with two main thrust levers (Figure 7-2). Motion is transmitted from the throttle quadrant by a control cable to the engine fuel control unit.

The throttle quadrant in the pedestal comprises main thrust levers, thrust reverser levers, and main thrust lever latches. The main thrust levers control forward and reverse engine thrust from IDLE to T/O thrust.

- Forward thrust—Thrust reverser levers retained in the stowed position; movement of thrust levers forward or aft controls engine power

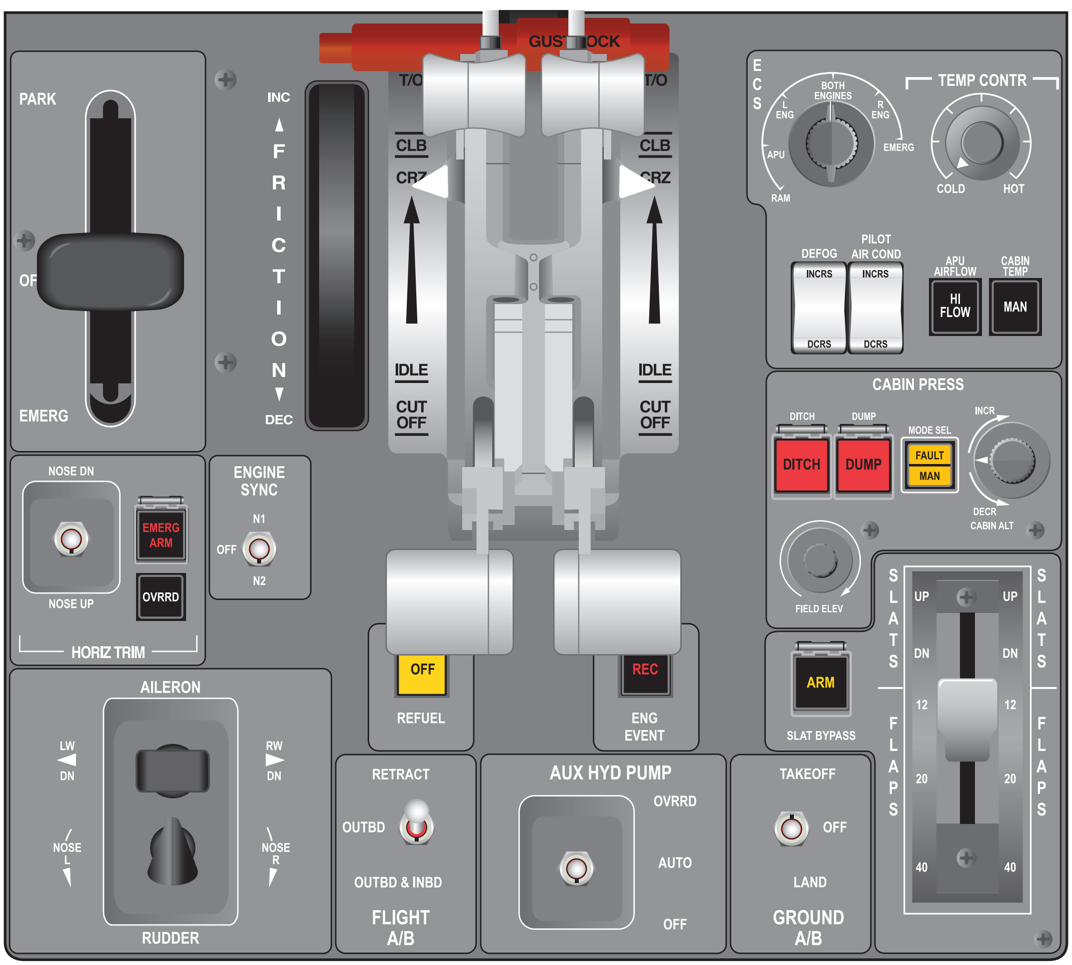


Figure 7-2. Center Console Throttle Quadrant

- Reverse thrust—
  - o Main power levers in IDLE position
  - o Thrust reverser levers in DEPLOY detent position deploys the thrust reverser buckets

When the thrust reverse buckets are fully deployed, a lock solenoid in the throttle quadrant is energized, enabling the thrust reverser levers to function in reverse thrust mode from IDLE to MAX reverse.

Each main thrust lever has three positive stops:

- CUT OFF
- IDLE
- T/O

Additionally, two intermediate ranges determined by thrust lever position are labeled CRZ (cruise) and CLB (climb).

Moving the main thrust lever forward from the IDLE and T/O thrust position and back through to the IDLE position is a non-obstructed movement.

To move the thrust power lever aft from the IDLE position to the CUT OFF position and back through to IDLE position, the thrust lever latch must be raised to prevent inadvertent engine shutdown. A thrust lever retarder automatically retards the main thrust lever to IDLE position during an inadvertent deployment of thrust reverse buckets.

Switches controlling the ignition system, DEEC, starter, and standby pump are in the center of the overhead panel.

Switches controlling fire protection, APU, and reverse thrust are on the center instrument panel.

DIGITAL ELECTRONIC ENGINE CONTROL (DEEC)

The TFE731-40AR-200G is controlled by an engine mounted DEEC. Each receives power from its respective main bus. The DEEC reduces pilot workload during engine operation. It provides spool speed and temperature limiting, surge-free acceleration and deceleration, engine synchronization, automatic performance reserve, and engine condition trend monitoring.

Additionally, the DEEC provides for fully automatic engine starts with automatic termination for parameter exceedances. Required control inputs are N_1, N_2, ITT, P_2T_2, and power lever angle (PLA).

A three-position switch on the overhead panel controls electrical power to the DEEC as follows (Figure 7-3):

- AUTO—DEEC has full authority over engine operation through PLA inputs from the cockpit
- MAN—DEEC provides only supervisory control for the engine ultimate overspeed protection
  - o Parameter monitoring remains available
  - o Overspeed protection fully functional

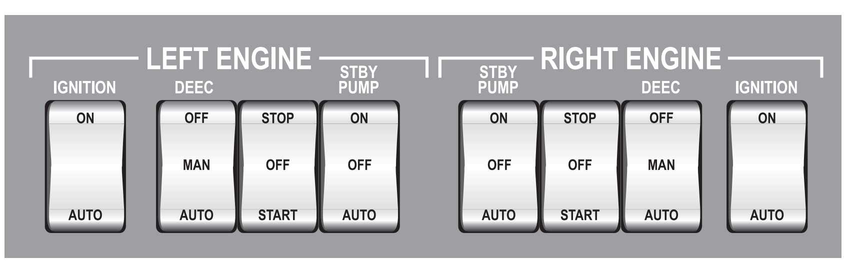


Figure 7-3. Engine Start Panel

- OFF—Engine operation controlled through mechanical linkage to the fuel metering system of the FCU
  - o Only ultimate overspeed protection through the FCU remains available

IGNITION SYSTEM

The ignition system for each engine includes the ignition exciter, two igniter plugs, and two high voltage shielded output cables. The ignition switch on the overhead panel is labeled AUTO and ON (Figure 7-3).

In the AUTO position, the DEEC commands ignition on for three modes of operation as follows:

- Normal engine start—N_2 between 6.8% and 45%
- Uncommanded deceleration—N_1 below the PLA set point and N_2 not accelerating
- Excessive deceleration—physical deceleration of the engine exceeds the commanded deceleration by the PLA. This mode protects the engine in the event of flameout

In the ON position, the ignition system operates continuously.

START SYSTEM

A three-position switch on the overhead panel controls each starter-generator as follows (see Figure 7-3):

- START—Momentary position that latches the starter relay and energizes the starter
- STOP—Momentary position that unlatches the starter relay and secures the starter
- Neutral position—Normal, default position

Under normal conditions, the starter is actuated by the pilot, and thereafter, controlled by the DEEC. The starter will be secured either as commanded by the DEEC or by selecting the STOP position of the starter switch.

ENGINE FUEL SYSTEM

The engine fuel system provides clean fuel to the engine in a form suitable for combustion (Figure 7-4).

It controls the flow in the required quantity for easy starting, acceleration, and stable running in all operating conditions.

A high-pressure fuel pump delivers fuel to the fuel spray nozzles, injecting the fuel into the combustion chamber in the form of an atomized spray. The flow rate must vary in order to maintain a constant selected engine speed. This is accomplished by the controlling devices that are fully automatic with the exception of power selection, which is achieved by the thrust lever.

Fuel flows from the fuel pump to the hydromechanical metering unit (HMU) that controls the amount of fuel supplied to the engine. In normal operation, the metering section of the fuel control is fully open, and fuel metering is controlled by the DEEC through the torque motor.

In manual operation, the governor limits rpm to 105% and acts as an onspeed governor. The shutoff (rotary) valve in the fuel control is operated by throttle position (between IDLE and CUTOFF) to facilitate engine start and shut down.

The fuel then passes through the fuel heater/oil cooler and flow divider before going to the fuel nozzles. Fuel from the HMU in excess of the engine requirements flows to the return to the fuel filter inlet.

Metered fuel is delivered to 12 fuel nozzles divided between the primary and secondary fuel circuits. The primary fuel circuit initiates engine starting. Before reaching ground idle,

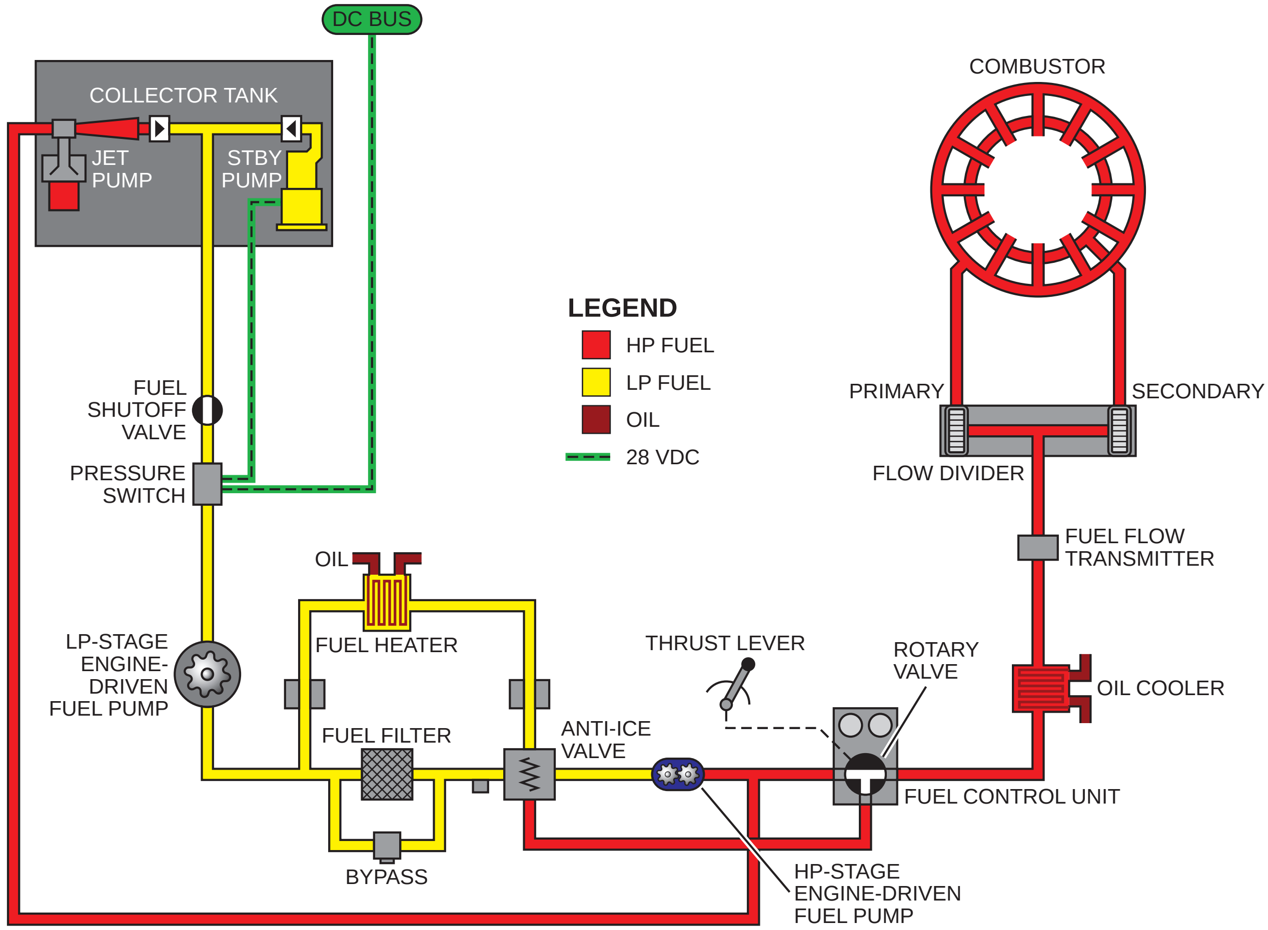


Figure 7-4. Engine Fuel System

fuel flow divider allows fuel to the secondary circuit and returns the system to normal operation. To prevent fuel accumulation in the combustion section after engine shutdown, a fuel dump valve purges remaining fuel from both the primary and secondary fuel manifolds. The purged fuel flows to a fuel waste ejector tank on the outer bypass duct. Occasionally, a small amount of fuel may be pooled in the bypass duct.

A mechanical fuel shutoff valve mounts on the dump valve. This shutoff valve provides automatic engine shutdown if the low-pressure (LP) compressor fan separates from the LP turbines.

Engine operation is controlled by the FCU, which contains fuel shutoff and metering sections. This unit provides the throttle linkage connection point and N_2 overspeed protection (flyweight manual mode governor) when the DEEC is in use.

During normal operation, the DEEC performs the functions of thrust setting, speed governing, and acceleration and deceleration limiting through electrical inputs to a servo motor on the fuel control unit. In the event of DC electrical or DEEC failure, the fuel control metering valve assumes control of the engine.

ENGINE OIL SYSTEM

The TFE731-40AR-200G lubrication system is a dry sump, high pressure regulated system designed to provide cooled, clean oil at a nearly constant pressure to the engine bearings and the gears and bearings in the planetary gearbox, transfer gearbox, and accessory drive gearbox (Figure 7-5).

The lubrication system on each engine is self-contained and supplies oil that lubricates and cools the engine bearings and other rotating parts. The system consists of storage, distribution, and indicating systems.

The engine oil is contained in a five quart oil tank on the engine right side but the overall engine capacity is 7.3 quarts. The tank has a sight gauge to check oil quantity.

The pressure element on the lube and scavenge pump supplies pressurized oil for lubrication to the accessory drive and transfer gearbox assemblies, the planetary gear assembly and high and low pressure shaft bearings. A four-element scavenge pump scavenges the oil from the bearing cases, fan planetary gear case, accessory drive gear case, and transfer gear case. The scavenged oil is then returned through a common scavenge line to the oil tank.

A filter removes impurities from oil before it is routed for lubrication. The filter is protected against clogging by a bypass valve adjacent to the oil filter. The bypass valve opens when oil pressure drop across the filter is excessive, and to bypass oil around the filter to avoid possible bearing starvation. The impending bypass indication (but not actual bypass function) is inhibited by a thermal lockout mechanism when the oil temperature is less than approximately 100°F (38°C). This feature displays EICAS indications on the MFD during engine start due to high oil viscosity at cold temperatures. A thermal lockout device on the indicator prevents actuation in cold oil conditions although the bypass valve bypasses oil under these conditions.

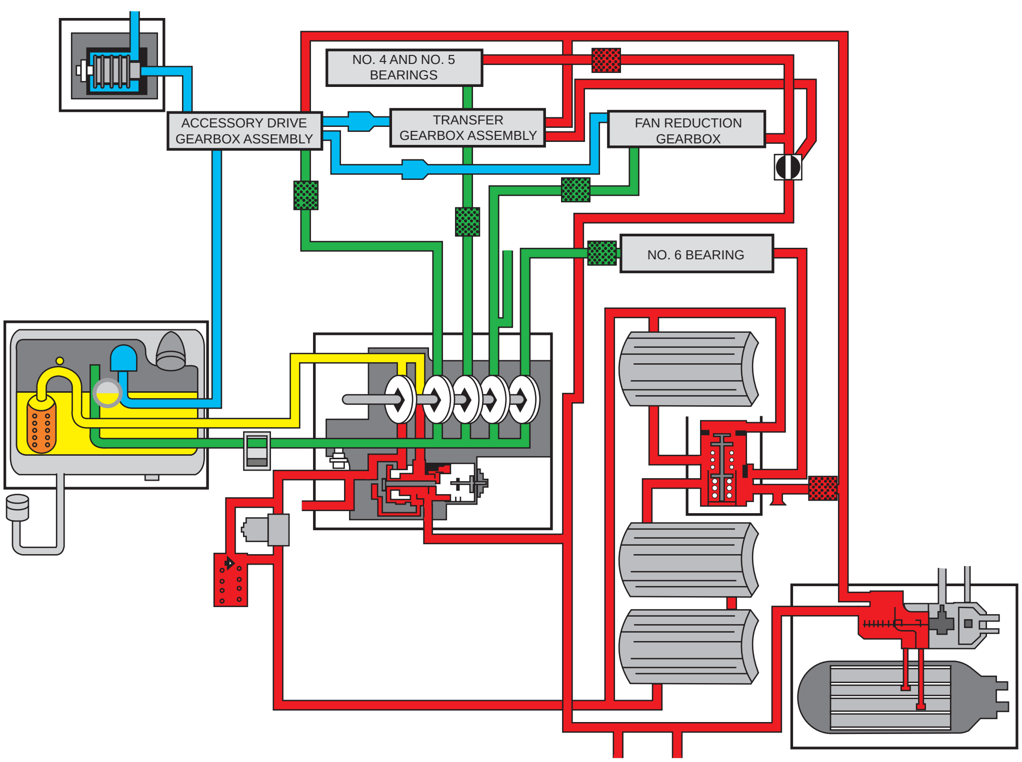


Figure 7-5. Engine Oil System

An oil temperature regulator (fuel heater/oil cooler) and three finned surface oil (air/oil) heat exchangers provide temperature control of fuel and oil. A fuel heater provides oil-to-fuel heat exchanging to prevent ice formation in the fuel system from clogging the fuel filter and other components. Thermostatic bypass valves in the air/oil and fuel heater/oil coolers maintain the oil at the desired minimum temperature during operation.

The breather pressurizing valve vents the oil system at low altitudes, and increases the internal engine vent and tank pressure to ensure proper oil pump operation.

SYNCHRONIZING

The primary purpose of the synchronization system is to reduce the cabin noise level for passenger comfort by synchronizing the speed of the engines. The sync system consists of the DEEC, data crosslink communication lines, and a three-position cockpit switch labeled SYNC switch (Figure 7-6).

The factory sets the master/slave identification with the left engine generally identified as the master engine. The synchronizer will function from flight idle to the maximum power rating, with an authority of ±5% N_1 during mid-range operations.

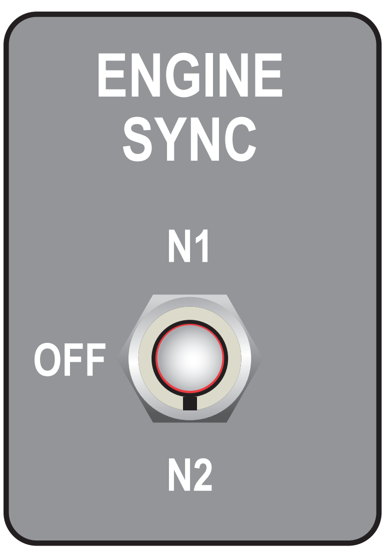


Figure 7-6. ENGINE SYNC Switch

ITT SYSTEM

The interstage turbine temperature (ITT) is sensed by thermocouples arranged between the high and low pressure turbines. These thermocouples supply signals to the EICAS system data concentrator unit (DCU) and on to the MFD or PFD display. The ITT for each engine is displayed in degrees centigrade (°C). This indication is in digital and analog format (Figure 7-7).

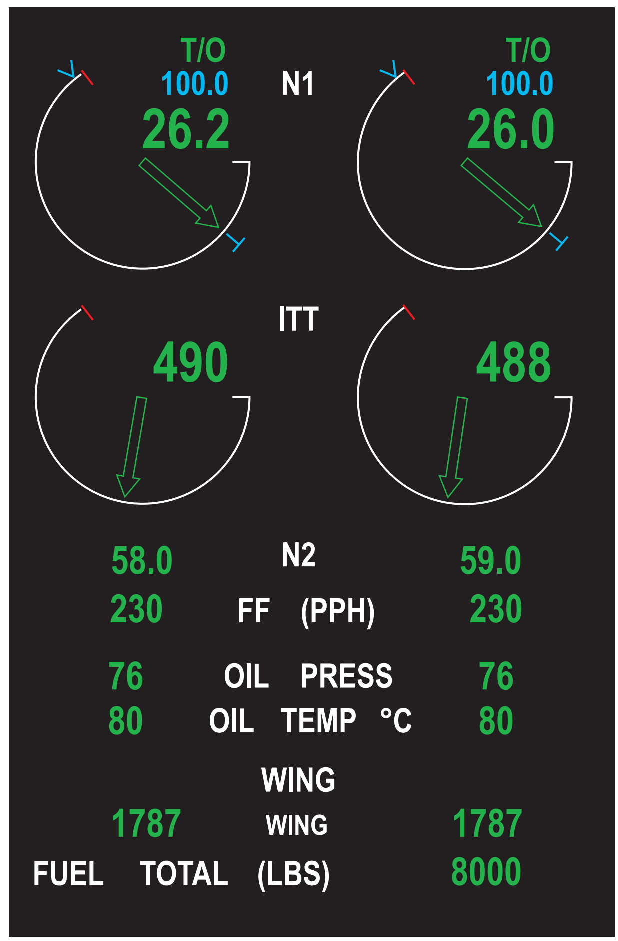


Figure 7-7. Engine Indications

AUTOMATIC POWER RESERVE (APR)

APR provides additional thrust in the event of an engine failure. Pressing the APR button in the cockpit, arms the system (Figure 7-8). Once armed, the system compares N_2 speeds of both engines. If a reduction of 15% N_2 is detected on one engine during takeoff, the system will allow both DEECs to deliver additional thrust.

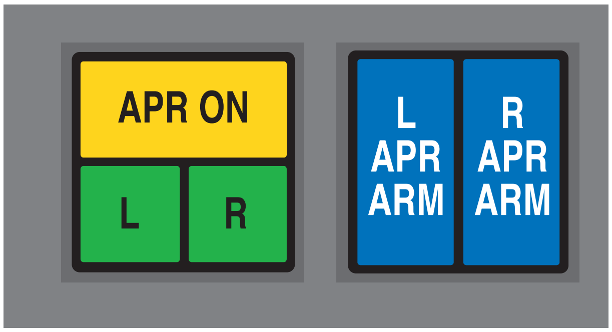


Figure 7-8. APR Controls

The increased thrust is achieved by allowing the ITT limit to increase from 1,004°C to 1,022°C. As such APR will only yield additional thrust if the engines are operating in a temperature limited regime. At sea level this occurs at 76°F.

ENGINE INDICATING

The engine indicating system gathers information from different monitoring systems and displays them on the engine indicating and crew alerting system (EICAS) and adaptive flight displays (AFDs) in the pilot and copilot front instrument panel as primary flight displays (PFDs) and multifunction displays (MFDs) (see Figure 7-6).

Primary engine indications are N_1 and N_2 rpm, ITT temperature, and engine vibration. Other engine indications are engine oil pressure and temperature, and fuel flow.

The low pressure rotor speed (N_1) for each engine is displayed in analog and digital display and calibrated in percentage rpm. A dual monopole transducer at the rear end of the engine low pressure rotor shaft supplies N_1 input signal to the EICAS system DCU and on to the MFD or PFD. Since the N_1 is directly proportional to the thrust, the N_1 indication is the primary thrust indication.

The high pressure rotor speed (N_2) for each engine is displayed in digital format and calibrated in percentage rpm. A dual monopole transducer on the transfer gearbox supplies N_2 input signals to the EICAS system DCU and on to the MFD or PFD display.

VIBRATION MONITORING

The vibration indication (VIB) display consists of two analog scales for right and left engines (Figure 7-9). A special test pushbutton is provided on the overhead panel. The vibration accelerometer is on the engine front frame. The accelerometer supplies signals to the EICAS system DCU and on to the MFD and PFD display.

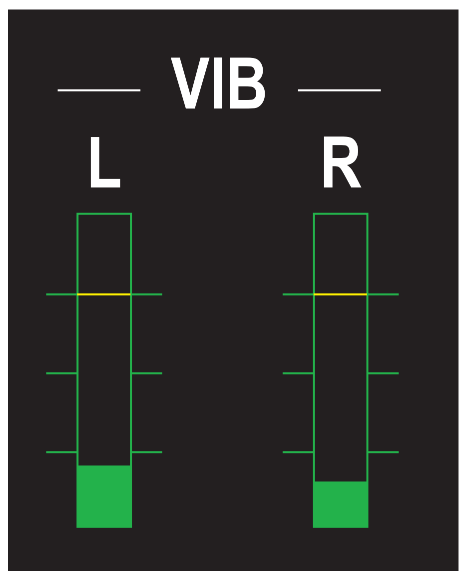


Figure 7-9. Engine Vibration Indicator

THRUST REVERSERS

The aircraft is equipped with the model TR 5040AS thrust reverser system to aid in slowing the aircraft upon landing. Engine thrust is redirected in a forward direction using target-type clamshell doors designed to stow in two seconds and deploy in one second.

The system is hydraulically actuated from the auxiliary system and electrically controlled using 28 VDC from the priority bus. Each thrust reverser includes dual primary hydraulic actuators to ensure symmetrical door actuation. Each system includes latch monitoring circuitry that initiates an automatic stow (AUTOSTOW) sequence if door latches inadvertently move to an unlocked position. If a thrust reverser is inadvertently deployed, throttle retard actuators automatically reduce the respective throttle to IDLE thrust if the TR is not in the fully stowed or fully deployed position.

COMPONENTS

The main structural components of the thrust reverser are the following:

- Jetpipe afterbody
- Two thrust reverser door assemblies
- Tailpipe assembly
- Inner fan duct assembly
- Fixed afterbody

The primary hydraulic and electrical components of the TR system are the latchbox assemblies, thrust reverser control box (TRCB), and throttle retard actuator.

Latchbox Assemblies

Each thrust reverser assembly is equipped with two latchbox assemblies mounted on the inboard and outboard sides of the tailpipe. The function of the latchbox is to latch and unlatch the doors.

The latchbox consists of a support bracket, a fitting assembly, a hydraulic actuator, a pair of latch hooks, leaf springs, and two switch assemblies.

Thrust Reverser Control Boxes

There are two thrust reverser control boxes (TRCB) per aircraft with each thrust reverser electrically isolated. The box receives input from the throttle quadrant position switches, landing gear squat switches, and thrust reverser assembly in order to determine the position of the reverser.

Based on this information, the thrust reverser control box signals positioning commands to the hydraulic isolation valve and the hydraulic control valve to produce thrust reverser overstow, deploy, and stow actions. Thrust reverser position signals are also simultaneously sent to the engine DEECs and cockpit annunciators.

Throttle Retard Actuator

The throttle retard system is designed to retard or keep the engines in idle whenever the thrust reverser is not in the fully stowed or fully deployed position. This safety feature prevents damage to the aircraft and thrust reverser.

Although it is possible to overcome throttle retard forcibly, damage to related components is probable. The throttle retard assembly is installed on the underside of the afterbody, inside an access panel. The unit consists of a hydraulic actuator and a crankshaft/lever assembly mounted inside a housing.

The FCU has been modified as part of the throttle retard installation. An arm added to the teleflex control box is designed to contact the throttle retard assembly lever, preventing engine acceleration.

In order to use the thrust reverser system, it must first be armed. Depressing the thrust reverse (L/R) buttons on the pedestal arms the thrust reversers if aircraft electrical power is available and STOW and DEPLOY circuit breaker are closed. If these conditions are met, the EICAS displays a white thrust reverser (T/R) icon inside the left and/or right N_1 scales and each thrust reverser switchlight illuminates.

Deploying either or both thrust reversers requires the respective thrust reverser system to be armed and the aircraft to be on the ground (both main landing gear weight on wheels signals present).

CONTROLS AND INDICATIONS

System Controls

The thrust reverser system is armed using the left and right thrust reverse ARM switchlights.

They are located on the pedestal between the FMSs and are marked THRUST REVERSE. When pushed, the button illuminates indicating electrical power is available to the TRCB (Figure 7-10).

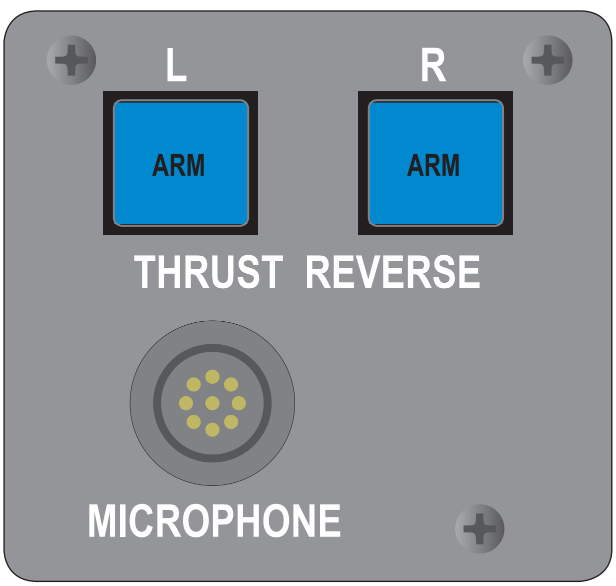


Figure 7-10. Thrust Reverser Indicator

Piggyback levers on the throttle quadrant control the system. Raising each respective piggyback lever to the idle detent initiates the deploy sequence. An interlock will prevent the lever from being raised unless the respective power lever is at IDLE. An additional interlock prevents raising the piggyback lever past the IDLE detent unless the thrust reversers are fully deployed.

System Indications

Cockpit indications consist of an amber CAS message (L or R T/R FAIL) and a T/R icon displayed within each respective N_1 display whenever the system is armed. The L or R T/R FAIL CAS message indicates a failure of the left or right thrust reverser system. The T/R icon can be displayed in green, white, or amber.

- Green indicates thrust reverser is deployed on the ground
- White indicates thrust reverser system is armed
- Amber displays whenever the L or R T/R FAIL CAS message is displayed.

OPERATION

Deploy Cycle

Once the system is armed, both main mounts are on the ground and the reverse thrust levers are moved to the idle deploy position, the deploy sequence will be initiated. The deploy sequence begins with pressure being routed to the stow side of the actuator. This pressure places the thrust reversers in the overstow position, retracts the latches, and activates the latch switches. This causes the TRCB to deenergize the stow valve solenoid and energize the deploy valve solenoid routing pressure to the deploy side of the actuator. The deploy actuators will remain pressurized for as long as the thrust reversers are commanded to the deploy position. As the thrust reverser reaches full deployment, the TRCB detects the signal from the full deploy switch on the outboard primary actuator, signaling that the thrust reverser doors are deployed.

This satisfies the TRCB requirements that activate the thrust reverser deployed output which causes the T/R icon to switch from white to green. Simultaneously, the TRCB reverse thrust schedule output signals the DEEC that the thrust reverser is deployed. This activates the reverse thrust via the power lever angle (PLA) schedule.

Stow Cycle

Movement of the piggybacks from reverse thrust position to the stowed position initiates the stow sequence. Hydraulic pressure is routed to the stow side of the actuator that causes the thrust reversers to move toward the overstow position. This will terminate the deployed signal to the TRCB causing the T/R icon to switch from green to white. Simultaneously, stow pressure will fully retract the latch hooks allowing the thrust reversers to reach the overstow position. Once in overstow, pressure will be removed from the actuator. This will release the latch hooks and allow the latches and thrust reversers to return to the stowed position.

Overstow

Pressurizing the stow side moves the thrust reverser doors into the overstow position. This door position removes the interference of the latch hook barbs with the door receptacles and allows the latch hooks to fully retract with the pressurization of the latch actuators. The throttle retard actuator and the primary actuators receive the stow pressure simultaneously. The throttle retard actuator always retards the engine to idle (regardless of throttle lever position) during the overstow sequence.

Deployment of the thrust reverser outside the overstow envelope (power setting, altitude, and airspeed constraints) cannot be accomplished. The thrust reverser doors must overstow prior to unlatching the thrust reverser. Overstow will require that the thrust reverser doors overcome the forces that result from the differential of internal and external pressures.

Autostow

During forward thrust operation (power on aircraft, thrust lever in forward quadrant), electrical power through the stow circuit breaker is always present in the stow array of the thrust reverser control. This feature assures that an autostow sequence is initiated in the event that any two unlock switches are detected to be unlocked (or one switch unlocked and hydraulic pressure is detected). An electrical ground to the airframe, through the common terminal of pole A of each unlock switch, ensures that the unlocked condition described above causes activation of the thrust reverser unlocked output of the TRCB resulting in the UNSAFE indication.

Movement of two unlock switches from their unlocked position (or one switch plus hydraulic pressure) advises the TRCB to energize the isolation and stow valve solenoids. This opens the valves and directs hydraulic pressure to the stow sides of the primary actuators and the throttle retard actuators. This reduces the thrust of the affected engine to idle thrust.

When the engine is throttled to idle, the force of the primary actuators will exceed the thrust that results from the differential pressures exerted on the doors. This will overstow the thrust reverser doors and allow the latch hooks to return to their locked positions. The isolation and stow valve solenoids are subsequently deenergized depressurizing the primary actuators and allowing the thrust reverser to return to the normally stowed position.

LIMITATIONS

The limitations contained in Section 1 of the *Airplane Flight Manual (AFM)* must be complied with regardless of the type of operation.

GENERAL

Oil Temperature

Starting, minmum continous operation:

- Up to 30,000 ft—30°C to 127°C
- Above 30,000 ft—30°C to 140°C
- Transient—149°C (two minutes at any operational altitude

NOTE

At temperatures below –40°C for extended periods, preheat engine before attempting start. During cold oil temperature starts, oil pressure may exceed maximum allowable transients.

Ambient Temperature and Altitude Envelope

For normal operation and ground starts, refer to the *AFM*, Figure 5-19.

For air starts, refer to the *AFM*, Figure 3-1.

ENGINE OPERATING LIMITATIONS

Setting and control of engine thrust is based upon N_1 (fan) speed. N_2 is high pressure rotor speed.

Rotor Speeds

Takeoff and maximum continous set per Figures 5-1 through 5-6 in the *AFM*.

Do not exceed:

- Maximum N_1—100%
- Maximum N_2 APR off—100.4%; APR on—101.0%
- Transient N_1—100% to 100.8% for 10 seconds
- Transient N_2 APR off—100.4% to 102.5% for 10 seconds; APR on—101.0% to 102.5% for 10 seconds
- Overspeed N_1—Red indications (log overspeed duration in aircraft logbook)
- Overspeed N_2—Red indications (log overspeed duration in aircraft logbook)
- Shutdown—Operate two minutes at 38% N_1 or below before shutdown (including taxi time)

interstage Turbine Temperature

- Starting—990°C
- Takeoff—1,004°C (1,022°C with APR on) for five minutes maximum
- Maximum continuous—990°C
- Maximum climb—974°C (recommended)
- Maximum cruise—949°C (recommended)

Start Times

- Light-off from initial fuel flow—10 seconds maximum
- From light-off to idle—60 seconds maximum
- Air starts (from initial fuel flow to 60% N_2)—45 seconds maximum
- Clearing engines (motoring or restart)—15 seconds on; two minutes off for three cycles, then 20 minutes off

Oil Pressure

At normal operating temperature:

- Engine start—Indication within 10 seconds after light-off
- Idle—50 to 150 psi when oil temperature below 30°C
- Idle—62 to 83 psi when oil temperature above 30°C
- Takeoff, climb, and cruise—62 to 83 psi
- Transient—100 psi maximum (three minutes maximum)

NOTE

At temperatures below –40°C for extended periods, preheat engine before attempting start. During cold oil temperature starts, oil pressure may exceed maximum allowable transients.

POWERPLANT LIMITATIONS

Manu-Facturer	No. of Engines	Model And Type	Takeoff and Maximum Continuous Thrust Rating, Static, ISA
Honeywell	2	TFE-731-40AR	With and without APR 4,420 lb

Approved Oils

Lubricating oils conforming to Honeywell specification, EMS 53110, Type II

- Aeroshell/Royce 560
- Esso/Exxon 2380 Turbo Oil
- Castrol 5000
- Mobil Jet Oil II
- Mobil Jet 254

NOTE

The listed brands of oil may be mixed. Other types of oil are not approved.

Approved Fuels

Fuels conforming to Honeywell specifications:

- EMS 53111 (Jet A)
- EMS 53112 (Jet A-1 and JP-8)
- EMS 53113 (Jet B and JP-4)
- EMS 53116 (JP-5)

EMERGENCY AND ABNORMAL OPERATIONS

POWERPLANT SYSTEM EICAS MESSAGES

Following is a ready-reference guide which briefly outlines the message meaning and action to be taken when a message appears on the MFD (Figure 7-11).

WARNING Messages

ENG EXCEEDANCE (L/R)—Engine limits exceeded: N_1, N_2, ITT

ENG FIRE— Overheating or fire in Zone 1 (accessories section)

ENG OVERHEAT— Overheating or fire in Zone 2 (combustor section)

OIL PRESS HI (L/R)—Engine oil pressure above limit

OIL PRESS LOW (L/R)—Engine oil pressure is low

OIL TEMP HI (L/R)—Engine oil temperature is high

CAUTION Messages

DEEC MAJOR (L/R)—Engine fuel controller malfunction.

DEEC MAN MODE (L/R)—Engine fuel controller switched to manual mode either manually or automatically

DEEC MAN XFER INOP (L/R)—Engine fuel controller cannot automatically switch to manual mode

ENG FUEL TEMP HI (L/R)—Engine fuel temperature is high

ENG FUEL TEMP LOW (L/R)—Engine fuel temperature is low

ENG MISCOMPARE (L/R)—Engine data channels difference

OIL PRESS HI (L/R)—Engine oil pressure above limit

OIL PRESS LOW (L/R)—Engine oil pressure is low

OIL TEMP HI (L/R)—Engine oil temperature is high

OIL TEMP LOW (L/R—Engine oil temperature is low

ADVISORY Messages

IGNITION ON (L/R)—Engine ignition is on

STATUS Messages

DEEC COMM FAIL (L/R)—DEEC Arinc 429 communication failure

ECTM DOWNLOAD (L/R)—Engine condition trend monitoring download. To be performed on ground

ENG CHIP DETECT (L/R)—Metal particles found in engine oil

ENG COMPARE INOP (L/R)—Engine data comparator inoperative between DCU and DEEC

FUEL FILTER BYPASS (L/R)—Respective fuel filter is clogged

OIL FILTER BYPASS (L/R)—Engine oil filter is clogged

Figure 7-11. Powerplant EICAS Indications

QUESTIONS

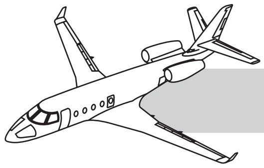
1. The rated thrust of the TFE 731-40AR engine is:
   A. 4,420 pounds, takeoff and max contionous thrust, static, ISA
   B. 3,500 pounds, sea level to 20,000 feet
   C. 4,250 pounds, static, sea level to 75°F
   D. 755 pounds at .8 M_1 and FL 400

2. Thrust setting for takeoff, climb, and cruise are based on:
   A. DEEC operation
   B. N_1 rpm
   C. ITT and fuel flow
   D. N_2 rpm and ITT

3. The planetary gear determines the rpm between the:
   A. LP and HP rotors
   B. HP compressor and HP turbine
   C. LP compressor and LP turbine
   D. LP rotor and the fan

4. The maximum allowable N_1 under all operating conditions is:
   A. 101% to 103% maximum continuous
   B. 105% for one minute
   C. 100%
   D. 103% for five minutes

5. When operating above FL 300, the maximum allowable oil temperature is:
   A. 127°C continuously
   B. 140°C continuously
   C. 120°at 55 psi
   D. 149°C for five minutes



CHAPTER 8
FIRE PROTECTION

CONTENTS

ILLUSTRATIONS

CHAPTER 8
FIRE PROTECTION

INTRODUCTION

The Gulfstream G150 aircraft provides for fire detection and protection for each engine, and the APU compartment.

GENERAL

The fire protection system provides the pilots with a simple, quick and reliable means of detecting and responding to hazardous overheat conditions and/or fires in either engine nacelle or in the area of the APU. The system incorporates two overheat/fire detectors routed inside fire Zone 1 (accessory and compressor section) and Zone 2 (core section), two containers and tubing system located in the aft fuselage, APU compartment fire extinguisher (and detectors) and warning and indicating system with continuous monitoring for failures. Three separate portable fire-extinguishers are provided for the occupied areas of the fuselage.

SYSTEM DESCRIPTION

The fire protection system incorporates detection, warning, fire extinguishing and testing devices for each engine and APU.

Each engine nacelle has two designated detection zones with each zone having a dedicated sensor tube and responder switch. Zone 1 (forward part of the nacelle) includes the accessories and compressor section of the engine. Zone 2 (aft part of the nacelle) includes the combustor, turbine and tail pipe section of the engine.

A continuous fire detector element is routed through Zone 1 and Zone 2 at each nacelle to provide fire and overheat signals to a cockpit warning light. Two warning lights, one per engine, are provided and a test button enables testing of the detector integrity. Even though the detector element is continuous, Zones 1 and 2 are separated by a fire seal.

The APU fire detector is similar to the engine fire overheat detectors with the exception of the length and factory set alarm temperature.

DETECTION

Engine

Two independent detection systems, fire and overheat, are installed on each engine. Each system consists of two units: sensing element and pressure switch (responder). The sensing element is a sealed gas-filled tube which activates the alarm. The systems are operated through power from the No. 1 and No. 2 distribution buses, through the L and R DET circuit breakers located on the overhead panel (Figure 8-1).

Zone 1 includes the accessory and compressor sections of the left and right engines. Zone 1 is protected by the fire extinguishing system. If a fire or an overheat condition is sensed in the zone, a FIRE pushbutton light illuminates, an EICAS message appears and an audio alarm sounds.

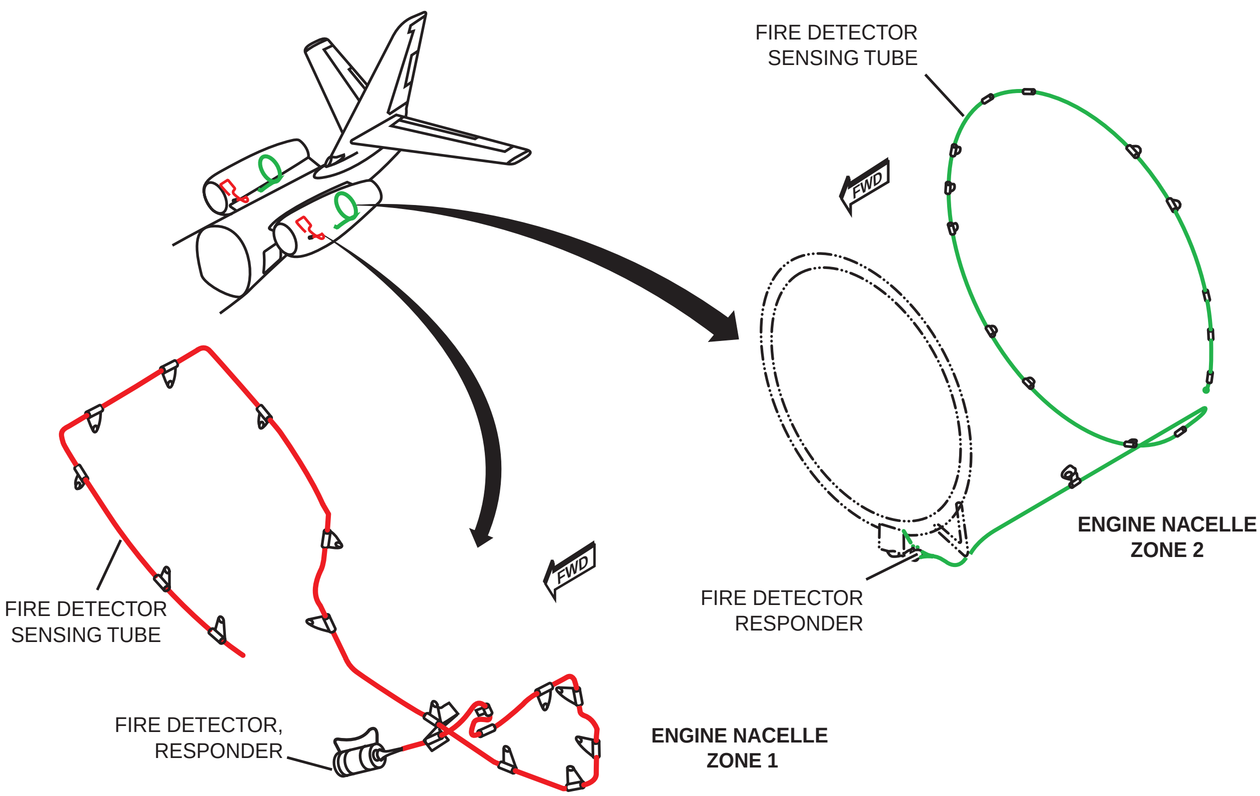


Figure 8-1. Engine Fire/Overheat Detection

Zone 2 includes the combustor section and is not protected by the fire extinguishing system, since fire in this section is self-controlled due to the limited amount of flammable fluids in this section. If a fire or an overheat condition is sensed in this section, the OVERHT pushbutton light illuminates, an EICAS message appears and an audio alarm sounds.

The detection assembly consists of two major subassemblies: a sensor loop assembly and a responder assembly. The detector loop and responder are joined to form a single unit. The hermetically sealed responder assembly contains two pressure switches, the alarm responder, and the integrity monitor responder.

The detector sensor element tube contains a fixed volume of helium gas. When the sensor detects an overall ambient temperature of 300°F in Zone 1 or 450°F in Zone 2, its internal pressure is proportionally raised. When a pressure responding to the preset temperature is reached, the pressure-actuated alarm switch closes and sends a signal to the fire warning light (Zone 1) or overheat warning light (Zone 2).

To detect and warn of a localized high-intensity temperature condition, such as a fire or ruptured combustion chamber, a capability for discrete sensing is built into the detector. The sensor element contains an absorbent core of halogen gas along its entire length. The halogen-charged core has the unique property of releasing an extremely large volume of gas when any finite section of the sensor is heated above a temperature setting of 950°F. The gas avalanche raises the internal pressure of the helium "averaging gas," activating the alarm switch. Thus, the sensor is capable of detecting two temperature hazards. It can provide alarm signals for both general overheating of the entire area and/or a high temperature condition resulting from a localized fire or combustion gas leak.

The PRESS TO TEST pushbutton works two ways. When the button is pushed it checks the integrity of the respective discharge cartridges.

The temperature-sensitive characteristic of both gases provides for automatic resetting of their detection qualities when temperatures decrease into the normal range.

Should the detector lose integrity (lose internal gas pressure) the integrity responder contacts will open, resulting in one of the PRESS TO TEST pushbutton lights to come on. The PRESS TO TEST pushbutton lights will also come on, along with the associated FIRE/OVHT lights, if a fault occurs in the detection system electrical circuit.

APU

The APU compartment is equipped with a fire detector, which is similar to the engine detector, except for length and factory set alarm temperature.

EXTINGUISHING

The fire extinguishing system provides a two shot fire extinguishing capability for each of the main engines and a one shot capability for the APU compartment.

The left (forward) bottle has two outlets and provides extinguishing agent to the left or right engine through one of two dedicated outlets. The right (aft) bottle has three outlets and provides extinguishing agent to the left or right engine or APU compartment through the corresponding dedicated outlet. Fire extinguishing agent expels through distribution piping into the appropriate compartment.

FIRE EXTINGUISHING BOTTLES

Two spherical steel bottles (Figure 8-2) made of corrosion resistant steel are installed on the upper left side of the aft fuselage section. Access is through the baggage compartment and through the equipment bay. Each bottle is filled with 4 pounds of Halon (CBrF3–Bromotrifluoromethane) and charged with dry nitrogen at 600 psi at 70°F.

Each bottle has a direct-reading pressure gauge which indicates the propellant charge relative to ambient temperature, a combined filling and safety (thermal) relief valve and two outlet terminals to left (forward) bottle and three outlets to right (aft) bottle. Each outlet contains an electrically initiated pyrotechnic cartridge.

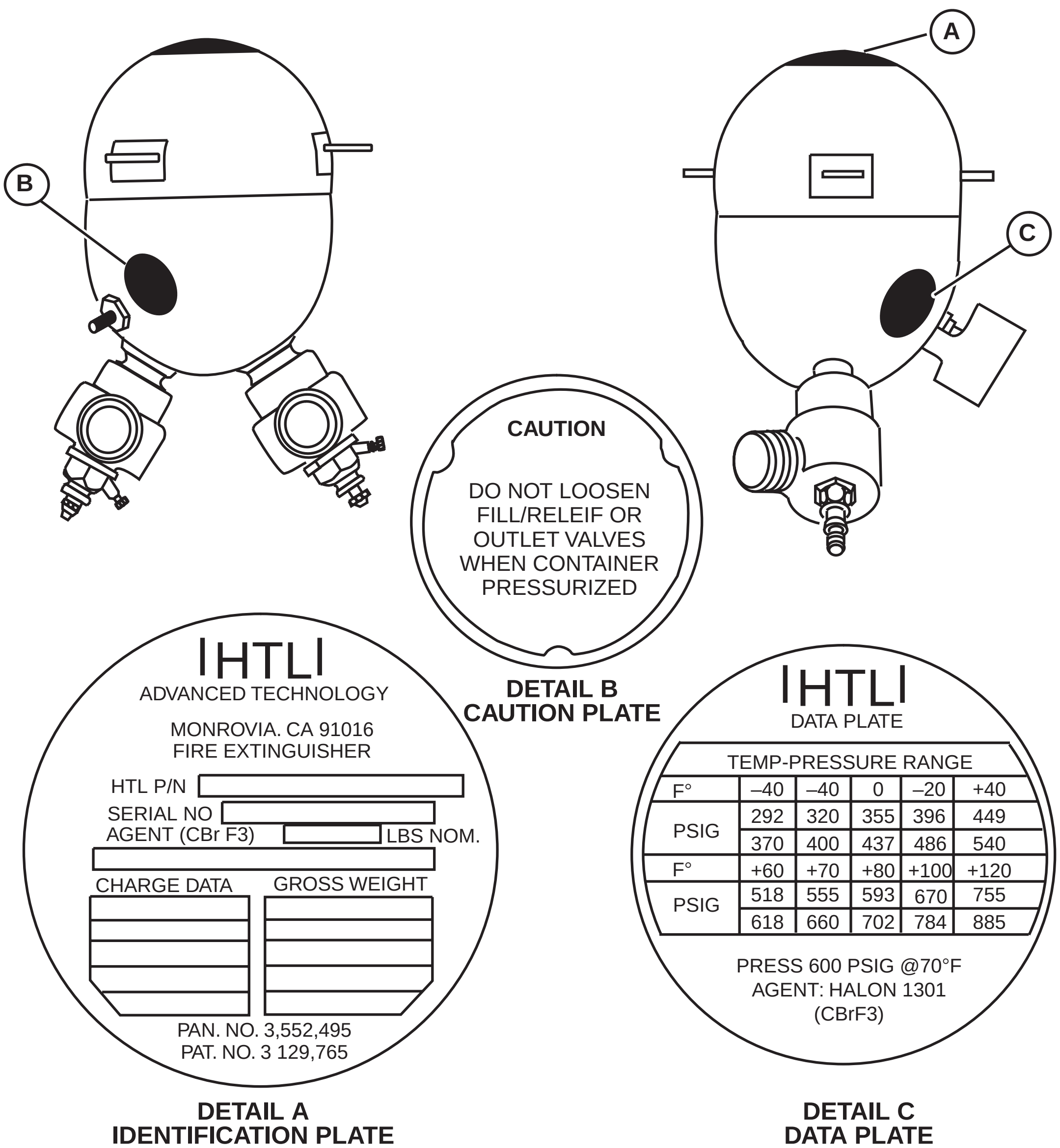


Figure 8-2. Fire Extinguisher Container Plates

TWO-WAY CHECK VALVE

There are two two-way check valves in the fire extinguishing system, one in each of the common discharge lines between the fire extinguisher bottles. Each check valve has three ports, two inlets and one outlet. One check valve joins the left outlet lines of both bottles to a discharge line and the second check valve joins the right outlet lines to both bottles to the second discharge line. This allows either bottle to discharge its contents into either engine and prevents extinguishing agent from back flowing from a full bottle to an empty one.

THERMAL RELIEF VALVE

A fusible check valve is incorporated in the safety (thermal) relief valve. If the ambient temperature rises abnormally, the check valve melts and the bottle discharges its contents through its thermal relief valve into the common line between the bottles and overboard by bursting the thermal discharge indicator disk.

THERMAL RELIEF DISC

A thermal relief valve on each bottle is connected in a common line terminating at a red frangible disc (Figure 8-3) on the left side of the aft fuselage skin. The thermal relief valve initiates a discharge of the bottle contents overboard if internal thermal conditions cause excessive pressure.

CAUTION

If the frangible disc is not in place during preflight, it indicates that one or both bottles have thermally discharged, but does not identify which one. The bottle pressure gauges must be checked and the disc and/or bottle replaced before flight.

Figure 8-3. Thermal Relief Disc

PORTABLE FIRE EXTINGUISHERS

There are three portable fire extinguisher bottles (Figure 8-4) in the aircraft; one in the flight compartment and two in the cabin on the aft partition.

The extinguishing agent, Halon 1211 (BCF) (Chemical formula of CBrC1F2) is used for the following types of fire:

- Class A carbonaceous fires
- Class B flammable liquid fires
- Class C electrical fires

The third bottle is filled with H_20. Since the location of these extinguishers may vary with each customized interior aircraft configuration, pilots should determine the specific locations for each aircraft.

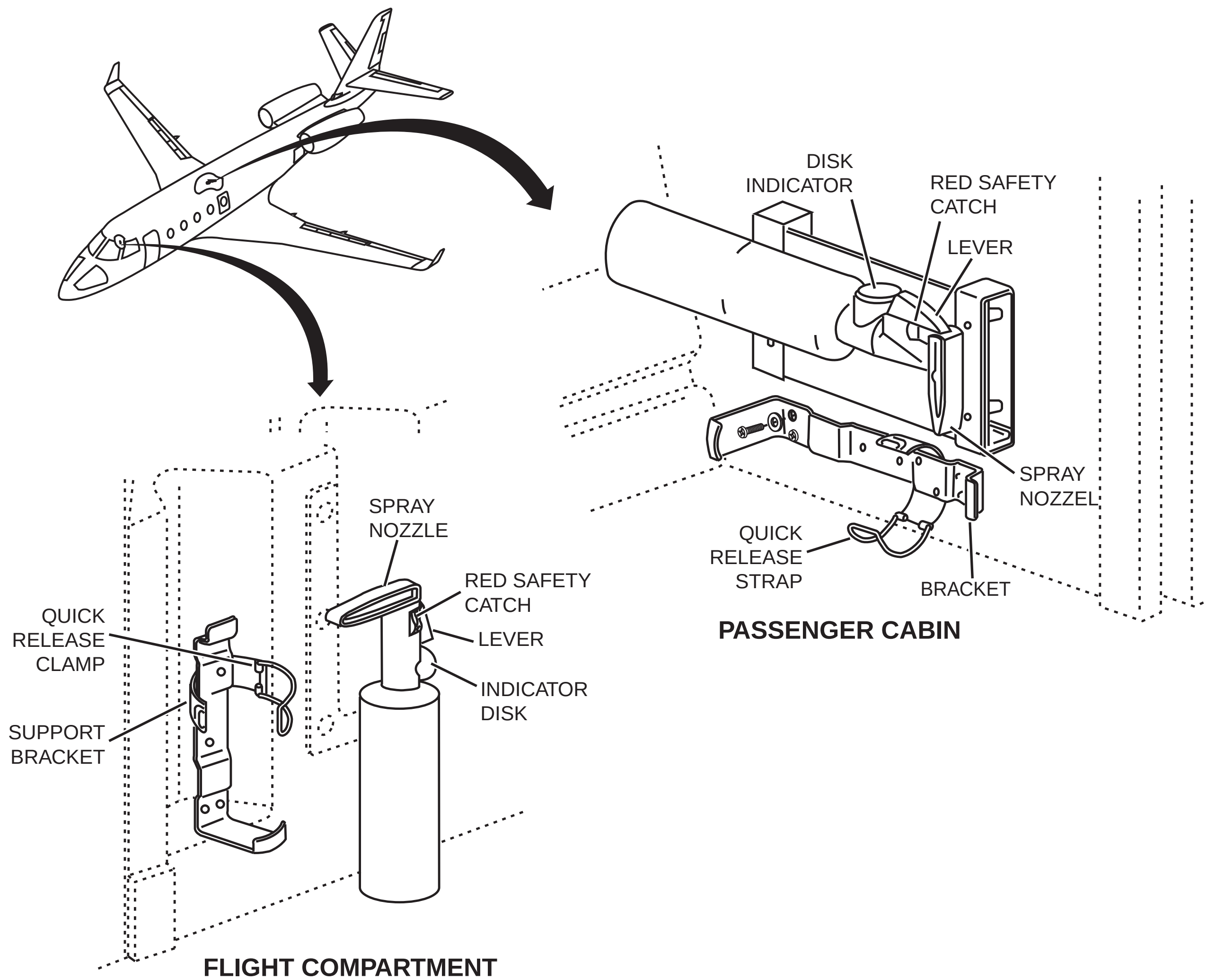


Figure 8-4. Portable Fire Extinguishers

OPERATION

ENGINE CONTROLS

The engine fire detection and extinguishing controls consist of lighted pushbuttons and circuit breakers (Figure 8-5).

Five lighted pushbuttons are mounted on the center instrument panel: two FIRE/OVERHT pushbuttons (one for each engine), two ARM/EMPTY discharge pushbuttons (one for each extinguishing bottle) and a PRESS TO TEST pushbutton.

The circuit breakers are mounted on the overhead circuit panel with controls for FIRE DETECT (L/R) and EXTING (L/R).

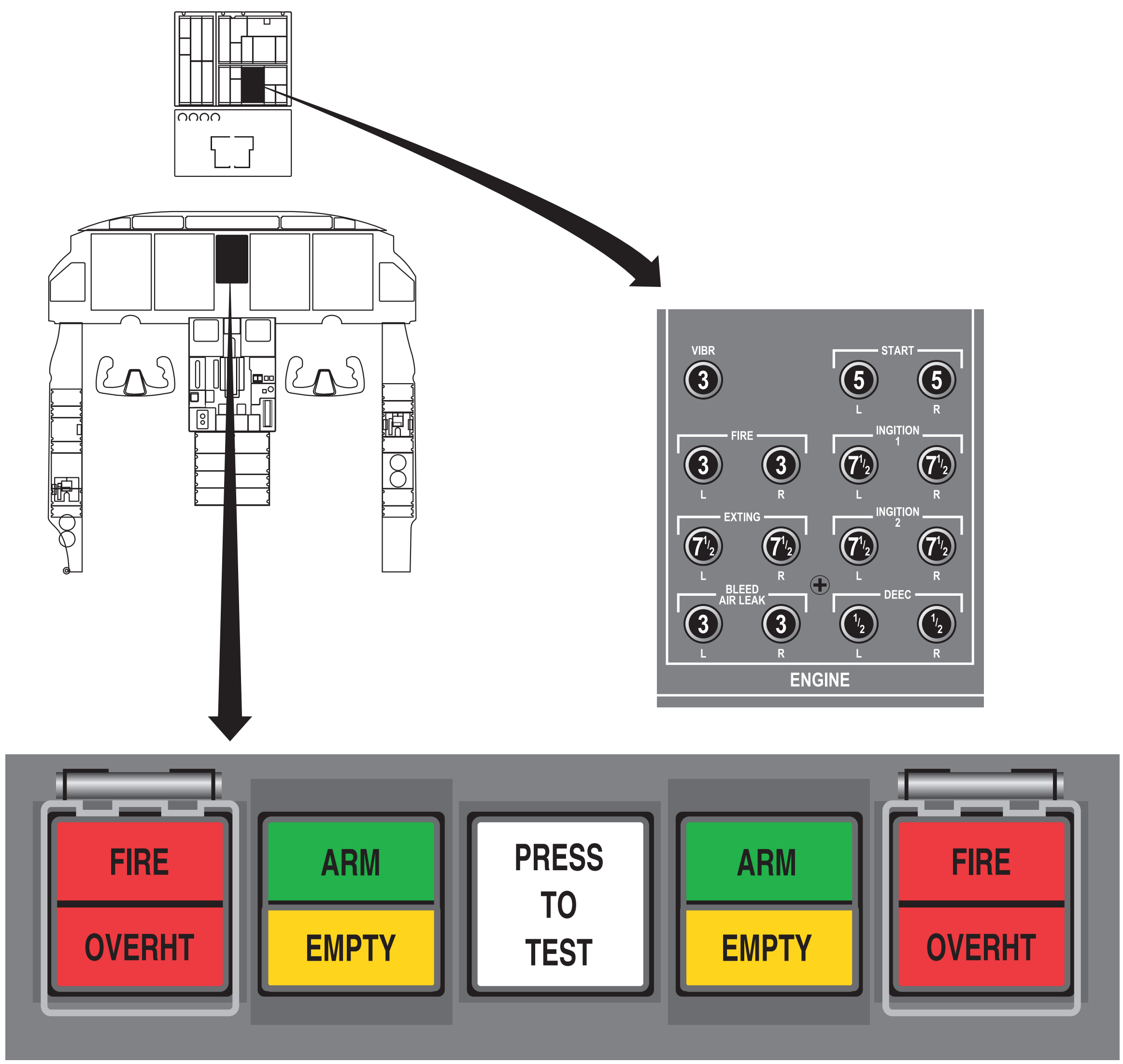


Figure 8-5. Engine Fire Extinguishing Controls

FIRE/OVERHT Pushbutton

There are two FIRE / OVERHT pushbuttons, one for each respective engine. The pushbuttons are divided into two parts and guarded to prevent inadvertent activation.

FIRE—Located on the upper part of the pushbutton, it illuminates (red) to indicate a fire or overheat condition in Zone 1 of the engine.

OVERHT—Located on the lower part of the pushbutton, it illuminates (red) to indicate a fire or overheat condition in Zone 2 of the engine.

ARM/EMPTY Pushbutton

There are two ARM/EMPTY pushbuttons, one for each respective fire extinguishing bottle. The pushbuttons are divided into two parts.

ARM—Located on the upper part of the pushbutton, it illuminates (green) to indicate that the discharge cartridges for the selected fire extinguishing bottle have been armed.

EMPTY—Located on the lower part of the pushbutton, it illuminates (yellow) to indicate that an electrical charge has been sent to the squibs. The light is solely a function of switch activation after system arming.

PRESS TO TEST Pushbutton

Primary testing of the fire protection system is achieved by the white PRESS TO TEST pushbutton. Pressing this button checks the integrity of the electrical circuits, the sensor (detector) tubes, and the fire bottle discharge cartridges. When the pushbutton is pressed and held, the fire protection system is tested as follows:

- Both FIRE lights illuminate to indicate the integrity of the detectors in Zone 1 for each respective engine.
- Both OVERHT lights illuminate to indicate the integrity of the detectors in Zone 2 for each respective engine.
- Both the green ARM and yellow EMPTY lights (Figure 8-6) illuminate to indicate continuity of the electrical circuits.
- APU EMPTY light illuminates
- The white lights in the four corners of the PRESS TO TEST pushbutton (Figure 8-7) should illuminate when pressed, indicating electrical continuity of the squibs through the fire control panel to the respective discharge cartridges on the fire bottles. If any of the lights in the pushbutton fail to illuminate, the cause

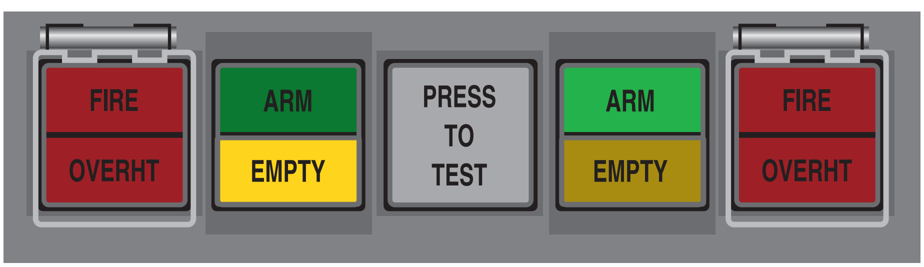


Figure 8-6. ARM/EMPTY Pushbutton

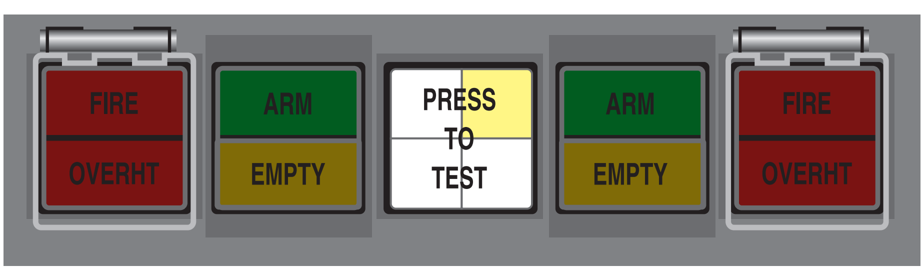


Figure 8-7. PRESS TO TEST Pushbutton

may be a discharged squib or simply a failed bulb. Releasing the PRESS TO TEST switchlight, place the IND TEST switch on the overhead panel to the DCU A & LTS position. If the lamps illuminate, a problem exists with the discharge cartridge and must be corrected prior to flight.

- The audible ENGINE FIRE will activate.

CAUTION

Pressing the fire PRESS TO TEST switchlight and the DCU A & LTS switch simultaneously will result in squib activation.

NOTE

When the PRESS TO TEST button is pressed, the L ENG/APU BLEED LEAK and R ENG BLEED LEAK warning messages will appear on the EICAS display verifying the integrity of the low pressure bleed leak detector loops.

FIRE PANEL OPERATION

Overheat

If an OVERHT light illuminates during engine start, it indicates excessive temperature or fire in zone 2 of the associated nacelle. The affected engine should be shut-down (thrust lever to cutoff), the FIRE/OVERHT switchlight pressed to close the fuel and hydraulic shutoff valves, and the engine should be motored using the START switch until the OVERHT light is out.

Fire

If a FIRE light illuminates, normal engine shutdown (by moving the power lever to cutoff) should be accomplished. Pushing the affected switchlight illuminates both ARM lights. Pushing the associated discharge switch actuates a discharge head, and the contents are discharged to zone 1 of the selected engine. At the same time, the ARM light for the associated switch extinguishes and the EMPTY light illuminates. If the FIRE light remains illuminated or if visual evidence of fire exists, discharge the second bottle.

NOTE

Illumination of the EMPTY light is not an indication that the selected bottle has discharged. This is simply a discharge switch function.

APU CONTROLS

The APU fire detection and extinguishing controls consist of lighted pushbuttons and circuit breakers (Figure 8-8).

Three lighted pushbuttons are mounted on the overhead panel for the APU FIRE warning indication: an APU FIRE pushbutton, an ARM/EMPTY discharge pushbutton and a PRESS TO TEST pushbutton.

The circuit breakers are mounted on the overhead circuit panel with controls for FIRE DET and EXT.

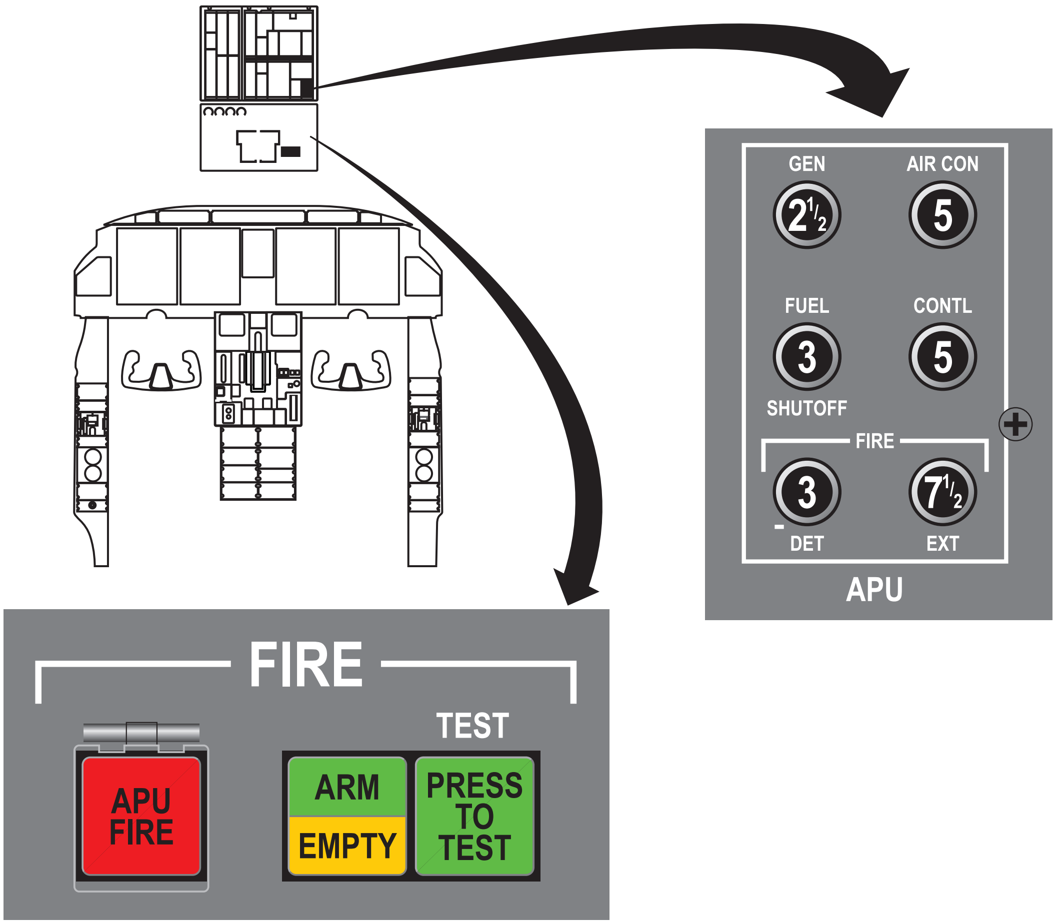


Figure 8-8. APU Fire Extinguishing Controls

APU FIRE Pushbutton

The APU FIRE pushbutton illuminates (red) if a fire or overheat condition occurs in the APU compartment. When pressed, the APU FIRE pushbutton (Figure 8-9) arms the right fire bottle's APU discharge cartridge. The pushbutton is guarded to prevent inadvertent activation.

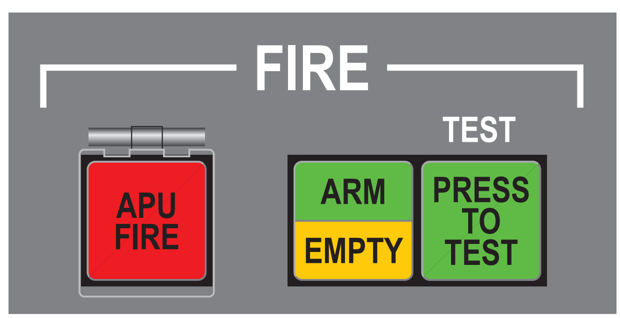


Figure 8-9. APU Fire Pushbutton

ARM/EMPTY Pushbutton

The ARM/EMPTY pushbutton (see Figure 8-6) is divided into two parts.

ARM—Located on the upper part of the pushbutton, it illuminates (green) to indicate that the APU discharge cartridge for the right fire extinguishing bottle has been armed.

EMPTY—Located on the lower part of the pushbutton, it illuminates (yellow) to indicate that the right switch has been depressed.

PRESS TO TEST Pushbutton

Testing of the APU fire protection system is achieved by the green PRESS TO TEST pushbutton (Figure 8-9). Pressing this button checks the integrity of the electrical circuits, the sensor (detector) tubes, and the fire bottle discharge cartridges. When the pushbutton is pressed and held, the fire protection system is tested as follows:

- The red FIRE lights illuminate to indicate the integrity of the APU compartment fire detector.
- The green ARM and yellow EMPTY lights illuminate to indicate continuity of the electrical circuits.
- The PRESS TO TEST pushbutton illuminates to indicate the integrity of the right fire bottle's discharge cartridge.
- On the main panel, the right EMPTY light illuminates
- APU FIRE audible activates
- Horn in nose wheel well activates

NOTE

When the PRESS TO TEST button is pressed, the APU FIRE warning message and APU EXHAUST OVERTEMP caution messagewill appear on the EICAS display.

ADDITIONAL CONTROLS

IND TEST Rocker Switch

An IND TEST switch is mounted on the overhead panel. When the IND TEST rocker switch is placed in the DCU A & LTS position, the integrity of the four white lamps in the engine PRESS TO TEST pushbutton and the APU PRESS TO TEST pushbutton light are tested. This tests whether an unlit lamp is due to a lamp failure or to the loss of continuity to the discharge cartridge (Figure 8-10).

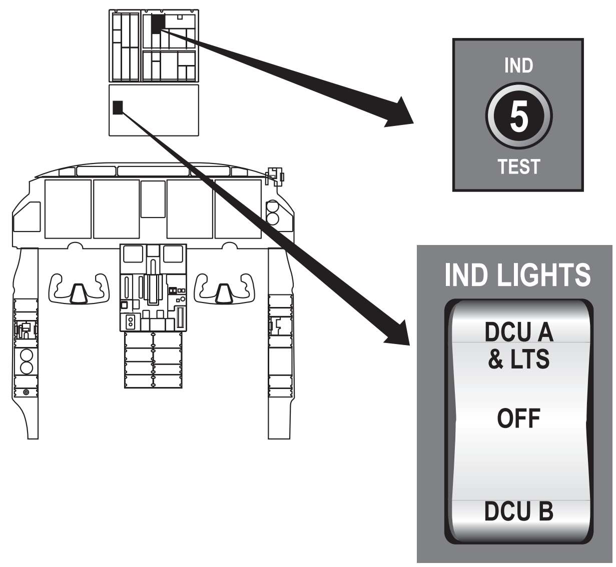


Figure 8-10. Additional Controls

WARNING

Do not press the IND TEST and fire PRESS TO TEST switches simultaneously. Discharge of the four fire bottle cartridges may occur.

DETECTION

The detection system consists of gas-filled sensor tube (Figure 8-11), pressure-operated diaphragm alarm responder switch and detector integrity responder switch. The sensor tube encircles the engine at all critical points to provide large area overheating detection and concentrated, small area, spot detection of fire.

Each sensor tube consists of stainless steel, hermetically sealed housing with permanently attached sensor element forming a single unit. Electrical connector and two pressure (responder) switches are enclosed in the housing. The sensor tube also contains a core (discrete element) which releases halogen gas when heated above a preset operating point. The sensor tube is also precharged with helium gas, which surrounds the core and provides arithmetic average gas response feature, enabling detection of general overheating condition within the nacelle when heated to a preset operating point. Therefore, increased pressure caused by general overheating or fire causes the FIRE or OVERHT light come on, as well as an EICAS message and aural warning, by closing the normally open contacts of alarm (responder) switch. Both average and discrete functions are reversible, therefore, when the sensor cools, averaging gas pressure lowers, halogen is reabsorbed into the discrete core and with the resulting pressure drop, it causes the alarm switch contacts to open and extinguish the warning lights.

The integrity responder switch is connected to the PRESS TO TEST switch and is installed in the sensor tube to check integrity of the sensor tube. It is similar to the alarm responder switch but operates at lower pressure and contacts are normally closed. If the sensor is ruptured and gas pressure is lost, contacts open, thus when the PRESS TO TEST pushbutton is pressed, FIRE / OVERHT warning lights, for affected side, will not come on, indicating sensor failure. Warning for overheat or fire condition in Zone 1 or Zone 2 in the engine nacelle, is indicated by the FIRE or OVERHT warning lights coming on in the fire control panel located at top center instrument panel.

The APU FIRE warning light on the overhead panel comes on indicating a fire or overheat condition and stays on as long as the temperature is above its rated setting. Simultaneously, the APU FIRE warning message is displayed on the EICAS and the aural warning is activated.

The fire detection system is continuously self-monitored for failures. The monitoring is indicated by continuous dim glowing of the FIRE/OVERHT lights. If a system failure is detected, one of the four lights in the switch-light illuminates.

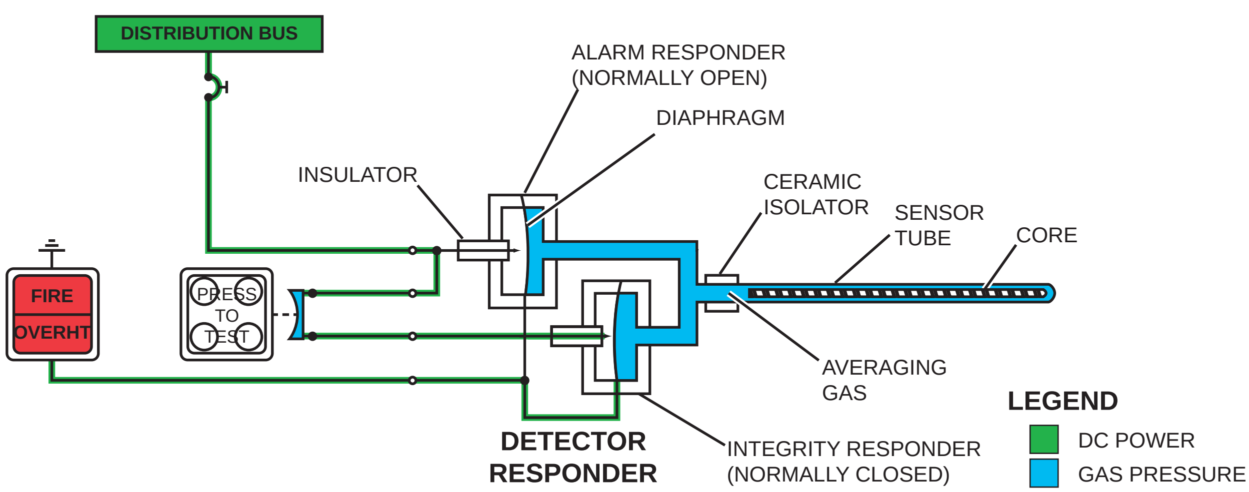


Figure 8-11. Detector Sensor Element

NOTE

Under some failures, FIRE or OVERHT may illuminate simultaneously with one of the PRESS TO TEST lights.

EXTINGUISHING

Engine

The engine fire extinguishing system (Figure 8-12) is powered from the priority bus through the L and R ENG FIRE EXT circuit breakers.

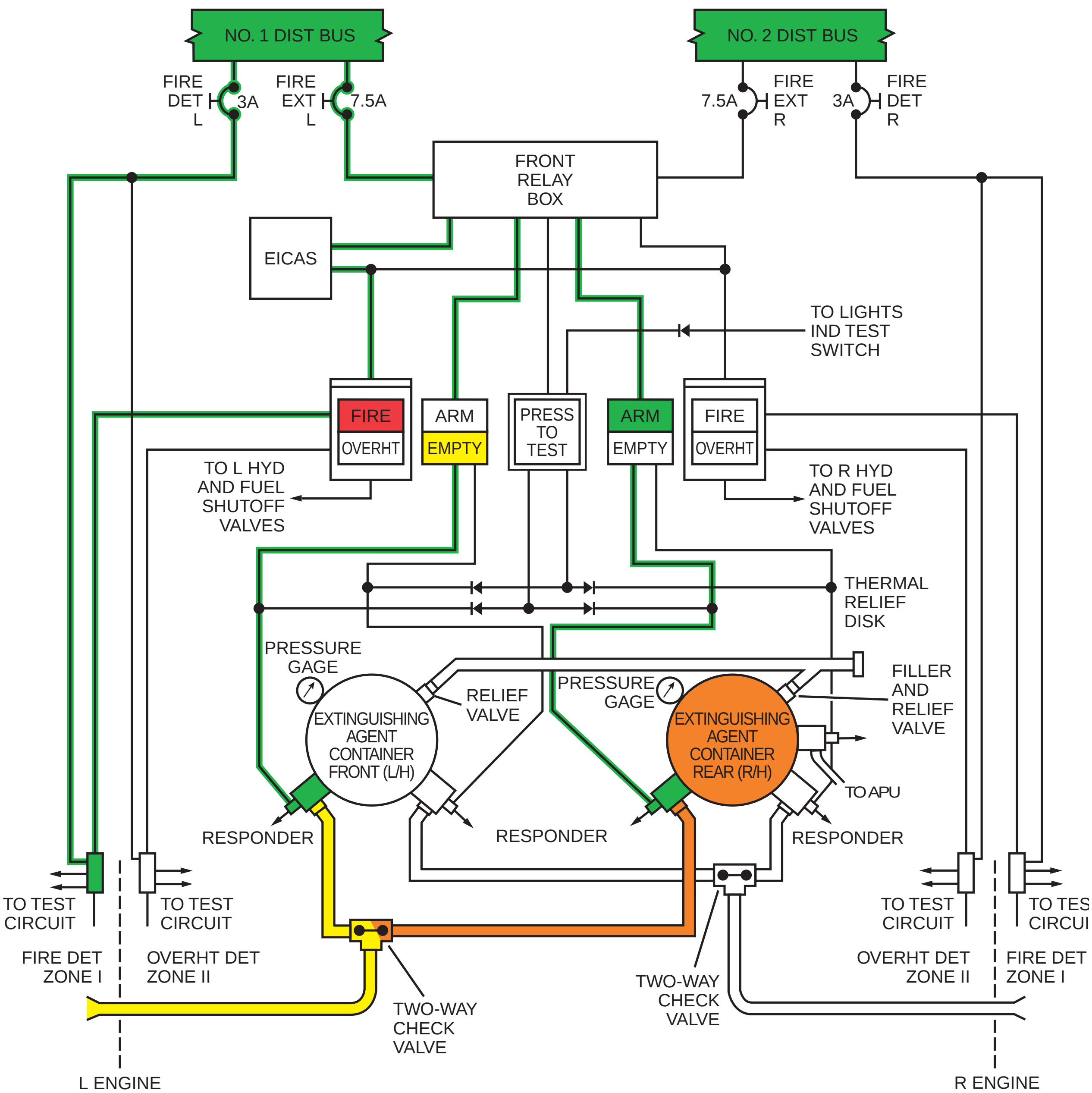


Figure 8-12. Engine System Operation

If a FIRE light comes on, the guard over the FIRE/OVERHT pushbutton will be lifted for depressing the pushbutton. Pressing the button has two functions:

- It arms both extinguisher discharge cartridges for the affected engine while lighting both ARM (green) lights
- It operates the relevant relays to close the shut-off valves in the affected engine fuel and hydraulic supply lines, which shut down the engine.

An ARM/EMPTY (discharge) pushbutton is depressed, firing the pyrotechnic cartridge on the side selected by the FIRE switch, discharging the contents of the bottle into the affected engine. The ARM (green) light goes off and the EMPTY (yellow) light illuminates. If the fire is not extinguished (FIRE light stays on), the action is repeated on the second ARM/EMPTY pushbutton. After the fire is extinguished, the FIRE light goes off. The FIRE/OVERHT pushbutton must be depressed again to release it.

The EMPTY light remains on as long as electrical power is provided to the system. If the right engine bottle button has been pressed, the EMPTY and APU EMPTY lights illuminate simultaneously.

If the FIRE pushbutton is pressed again, the ARM lights illuminate even if a bottle is empty. Therefore, a system check is performed through the PRESS TO TEST pushbutton which has four lights, one for each cartridge, as follows:

- IND TEST switch is pressed. All four PRESS TO TEST lights should illuminate, indicating the test system is serviceable
- PRESS TO TEST pushbutton is pressed.

Whichever light does not illuminate indicates that its respective cartridge has been fired, and therefore, the bottle is empty

APU

The APU fire extinguishing system is powered from the priority bus through the APU EXT circuit breaker.

If a FIRE light comes on, the guard over the APU FIRE pushbutton will be lifted for depressing the pushbutton. Pressing the button has two functions:

- Arms the system for the APU while lighting the ARM (green) light
- Operates the relevant relays to close the shut-off valve to the APU fuel supply line, which shut down the APU.

An ARM/EMPTY (discharge) pushbutton is depressed, firing the pyrotechnic cartridge on the right side fire bottle, discharging the contents of the bottle into the APU compartment. The ARM (green) light goes off and the EMPTY (yellow) light illuminates. After the fire is extinguished, the FIRE light goes out. The APU FIRE pushbutton must be depressed again to release it.

The EMPTY light for the APU as well as the right EMPTY light for the engine remain on as long as electrical power is provided to the system.

If the FIRE pushbutton is pressed again, the ARM lights illuminate even if a bottle is empty. Therefore, a system check is performed through the PRESS TO TEST pushbutton, as follows:

- IND TEST switch is pressed. The PRESS TO TEST light not should illuminate, indicating the test system is not serviceable

If during unattended ground operation the APU enters an over temperature situation, the APU fire system automatically shuts the APU down and an external horn sounds.

An APU OVERTEMP EICAS message illuminates to indicate excessive temperature and the APU did not enter the automatic shutdown sequence.

Portable Fire Extinguishing

To use portable extinguishers, remove from quick-release bracket, hold upright by gripping hand grip with spray nozzle pointing away from operator. Slide the red safety-catch down with the thumb and point nozzle to direct the spray at base of fire. Squeeze the lever in the hand grip with palm of hand to discharge the spray. Releasing the lever closes a secondary seal inside the operating head. A ruptured red indicator disk in the fire extinguisher indicates a partial or total discharge of extinguisher and should be replaced.

INDICATION

ENGINE

If there is a fire or overheat in Zone 1 of an engine, pushbutton indicating and warning lights illuminate and action is taken, as follows:

- FIRE/OVERHT: FIRE light (red) illuminates and the audio alarm sounds. If feasible, shut down engine. Lift transparent guard over pushbutton and press pushbutton. This closes the fuel and hydraulic shut-off valves to the respective engine and arms the extinguishing system.
- ARM/EMPTY: ARM light (green) illuminates. Press the pushbutton, firing the cartridge of the respective bottle to discharge the extinguishing agent into the affected engine. Thereafter, EMPTY light (yellow) illuminates. If fire light stays on, discharge the second bottle by the second ARM/EMPTY pushbutton.
- FIRE/OVERHT: FIRE light (red) goes out and audio alarm stops, after fire or overheat in Zone 1 is remedied. Pushbutton is pressed again to release it and put out the EMPTY lights.

If there is a fire or overheat in Zone 2 of an engine, pushbutton indicating and warning lights illuminate and action is taken as follows:

- FIRE/OVERHT: OVERHT light (red) illuminates and the audio alarm sounds. If feasible, shut down engine. Transparent guard over pushbutton is lifted and the pushbutton is pressed. This closes the fuel and hydraulic shut-off valves to the respective engine (both ARM lights illuminate).

- ARM/EMPTY: ARM light (green) illuminates. The pushbutton is pressed, firing the cartridge of the respective bottle to discharge the extinguishing agent into the affected engine. Thereafter, EMPTY light (yellow) illuminates. If fire light stays on, the second bottle is discharged by the second ARM/EMPTY pushbutton.
- OVERHT light (red) goes out and audio alarm stops, after fire or overheat in Zone 2 is remedied. Pushbutton is pressed again to release it and put out the ARM lights.

When the PRESS TO TEST pushbutton is pressed, four white lights in the pushbutton should illuminate and audio alarm sounds to indicate the integrity of the four cartridges in the bottles. If a light does not illuminate, it indicates that the respective cartridge has been fired.

If a detector loses integrity (loses internal gas pressure) or if a fault occurs in the detection system electrical circuit, the respective light of the PRESS TO TEST pushbutton will illuminate (Figure 8-13).

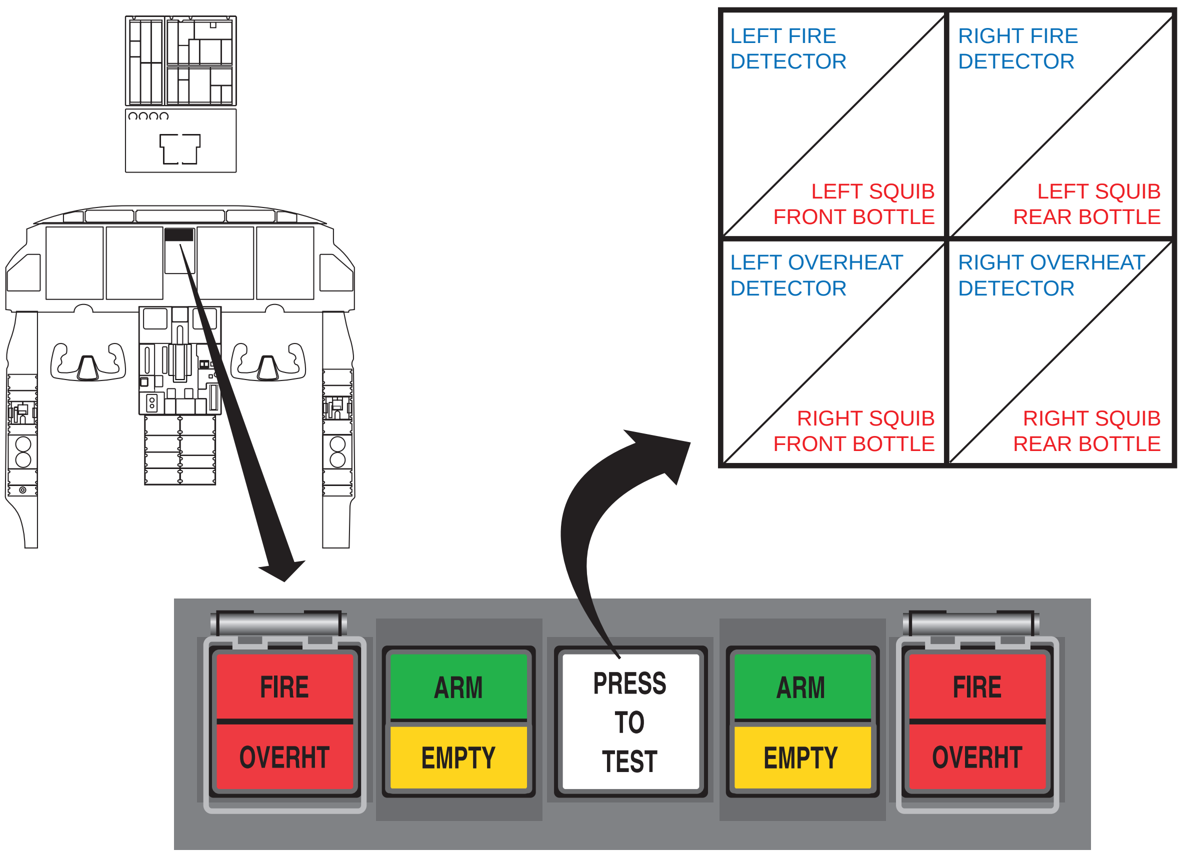


Figure 8-13. Engine Indication

APU

The APU FIRE pushbutton condition indicator light (red) illuminates when a fire or overheating condition occurs in the APU compartment. When pressed, the right container discharge cartridges are armed and the ARM/EMPTY pushbutton illuminates.

The upper half of the ARM/EMPTY pushbutton ARM light (green) illuminates to indicate the APU discharge cartridge in the right bottle is armed. The ARM light goes off and the EMPTY light (yellow) illuminates when the ARM pushbutton is pressed and the right bottle has been discharged.

The lower half of the ARM/EMPTY pushbutton EMPTY light illuminates to indicate the right bottle has been discharged. The right EMPTY and APU EMPTY indicator lights come on simultaneously (Figure 8-14).

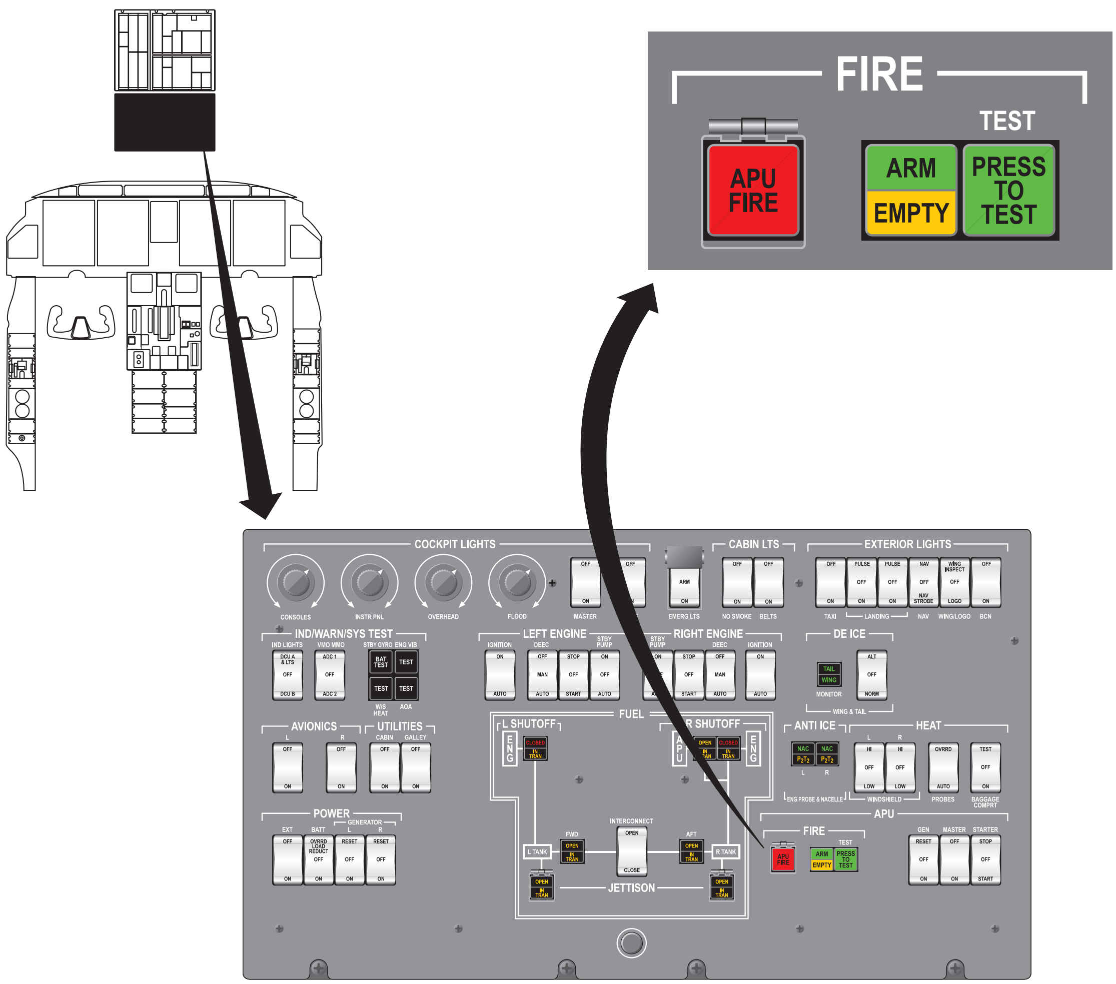


Figure 8-14. APU Indication

SMOKE DETECTOR SYSTEM

The cabin entertainment smoke detector system is installed in production aircraft beginning with S/N 227.

A fan and a smoke detector is below the aft end of the copilot side panel. The fan draws air from the outboard side of the entertainment system forward through the smoke detector, then blows it out under the cockpit floor to the outflow valve. The system also has an overheat warning system set at 125°F. The OVERHEAT switchlight illuminates at 125°F and extinguishes at 115°F.

WARNING SYSTEM

ENT SMOKE and ENT O'HEAT switchlights along with a pushbutton test switch are below the Davtron clock on each side of the cockpit (Figure 8-15).

The amber O'HEAT switchlight illuminates at 125°F. This light may illuminate on a hot day until the fan starts.

The red SMOKE switchlight flashes when smoke is detected. When the SMOKE switchlight is pushed, it stops flashing and illuminates steady as long as smoke is detected. If the smoke stops, the light extinguishes after it is pushed.

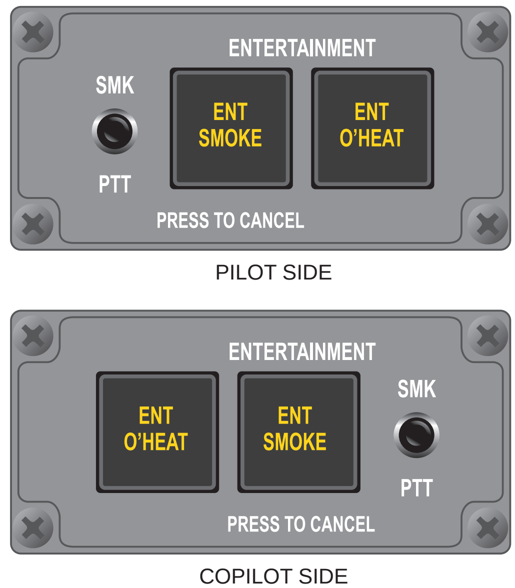


Figure 8-15. Cabin Entertainment Smoke Detector System

EMERGENCY AND ABNORMAL OPERATIONS

Following is a ready-reference guide which briefly outlines the message meaning and action to be taken when a message appears on the MFD (Figure 8-16).

Messages having symbol are accompanied by an aural alert.

WARNING MESSAGES

APU FIRE —APU fire. APU enters automatic shut-down sequence

ENG FIRE —Overheating or fire in Zone 1 (accessories section)

ENG OVERHEAT —Overheating or fire in Zone 2 (combustor section)

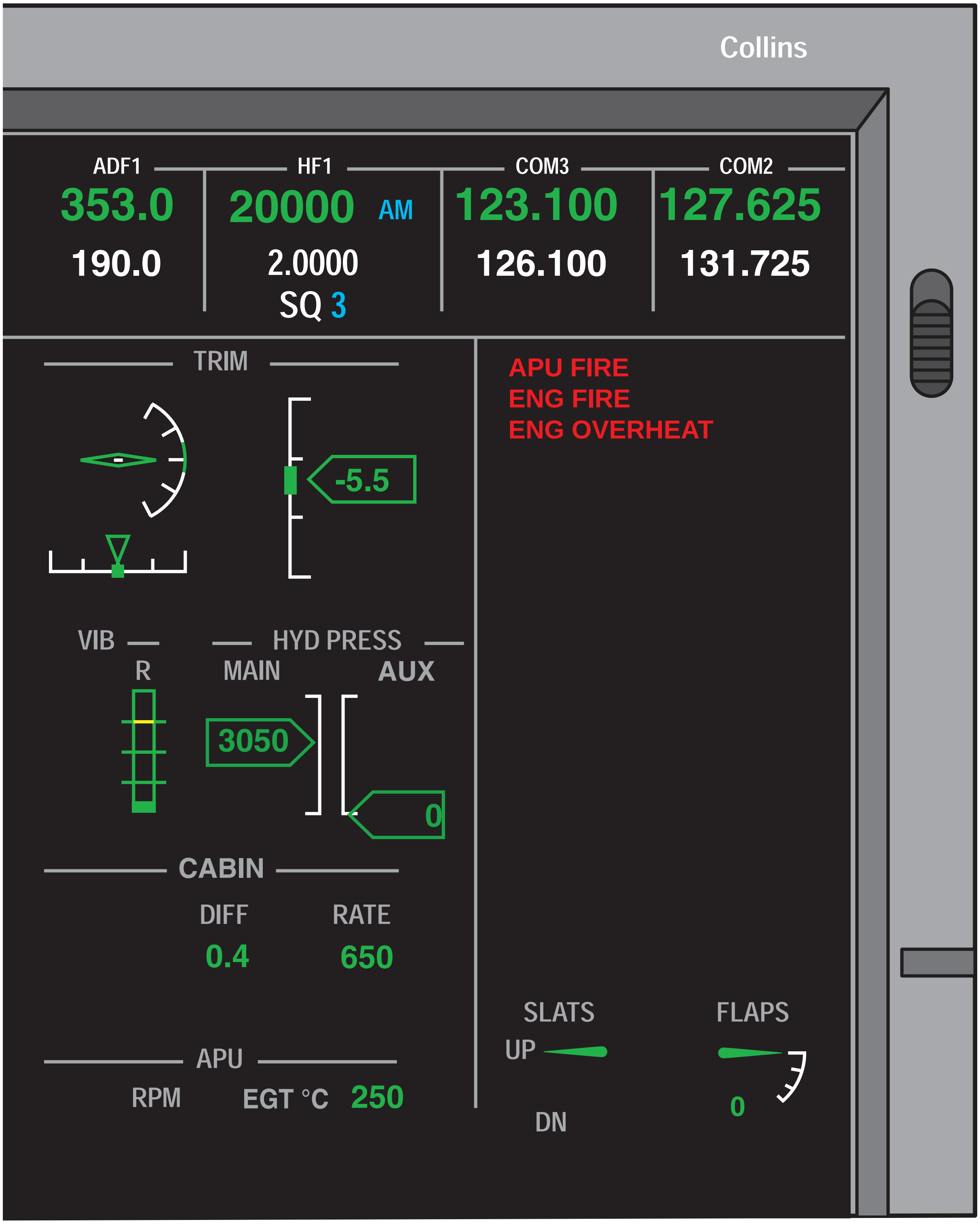


Figure 8-16. APU FIRE EICAS Indications

QUESTIONS

1. OVERHT annunciation is provided for:
   A. OVERHEAT in zone 1
   B. OVERHEAT in zone 2
   C. FIRE in zone 1
   D. Zones 1 and 2 of each nacelle

2. When the FIRE/OVERHT switchlight is pushed, the ARM lights will come on to indicate:
   A. Closing of the fuel and hydraulic shut-off valves
   B. Fully charged fire extinguishers
   C. Integrity of the fire/overheat elements
   D. Fire-extinguishing arming

3. Thermal overpressure discharge of a fire extinguisher is indicated by:
   A. EMPTY light illuminated in the cockpit
   B. FIRE light not illuminated
   C. There is no cockpit indication
   D. One bulb in the PRESS-TO-TEST button would not be illuminated

4. If a fire condition is sensed in Zone 1, the crew will have the following indication:
   A. ENGINE FIRE CAS message
   B. FIRE pushbutton, CAS message and audio alarm
   C. Audio alarm and illumination of the FIRE pushbutton
   D. FIRE/OVERHT pushbutton illuminates

5. The fire extinguishing system provides a two shot fire extinguishing capability for each of the engines and a one shot capability for the APU:
   A. True
   B. False

CHAPTER 9
PNEUMATICS

CONTENTS

ILLUSTRATIONS

CHAPTER 9
PNEUMATICS

INTRODUCTION

The pneumatic system for the G150 includes the air sources, conditioning, control, and distribution. In addition, related systems such as nacelle anti-icing, surface deicing, hydraulic reservoir pressurization, and cabin pressurization air sources are included. A backup air supply is incorporated in the system for aircraft pressurization in the event of primary system or controller failure.

GENERAL

The pneumatic system provides the sources, control, and warning systems associated with the use of bleed and ram air.

Both HP and LP air is supplied from each engine or the APU to a single air cycle machine (ACM). The ACM cools the hot bleed air for use in the cabin and cockpit.

The ECS selector panel has controls for selecting the source of ECS air, controlling the cabin temperature and secondary (backup) controls in the event of system degradation.

PNEUMATIC SYSTEM

The following subsystems comprise the pneumatic system:

- Engine bleed-air supply
- APU bleed-air supply
- Nacelle anti-icing air supply
- Surface deicing air supply
- Emergency air supply
- Ozone converter
- Hydraulic reservoir and cabin door air supply
- Indicating and warning system
- Ram-air supply

BLEED-AIR SOURCES DESCRIPTION

Engine Bleed-Air Supply

Bleed air is extracted from both the high pressure HP and low pressure LP ports on each engine. LP bleed air is supplied from the fourth stage of the low pressure compressor, and the HP bleed air is supplied from the single stage high pressure compressor of the engine. Each of these ports have a venturi at the point where the air leaves the engine. The purpose of these venturies is to prevent excessive air extraction from the engine if one of the ducts should rupture (Figure 9-1).

Both the HP and LP air ducts of each engine are routed to a bleed switching valve (BSV) which controls the amount of bleed air entering the environmental system.

HP bleed air from each engine is used for nacelle inlet anti-icing. HP bleed is tapped off the bleed-air duct before the bleed switching valve.

LP bleed air from each engine is used for surface deicing. It is tapped off the bleed-air duct before the bleed switching valve.

LP bleed air, from the right engine only, is tapped off the bleed-air duct before the bleed switching valve to provide an emergency source of bleed air for the pressurization system.

APU BLEED AIR SUPPLY

APU bleed air is available for use on the ground or airborne. Two modes, low flow and high flow, supply air to the pneumatic system when the air source selector switch on the ECS control panel is in APU, located on the top right side of the center pedestal. (Figure 9-2) Selection of any other bleed source will close the APU bleed-air valve.

Since bleed air for the deice system is tapped off the LP bleed air prior to the bleed-air switching valves, APU bleed air is not available to operate any of the deice or anti-ice systems on the G150.

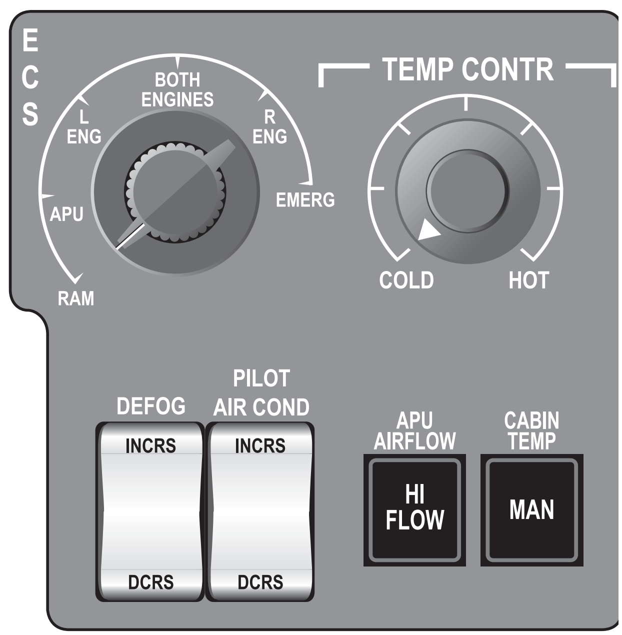


Figure 9-2. ECS Selector Panel

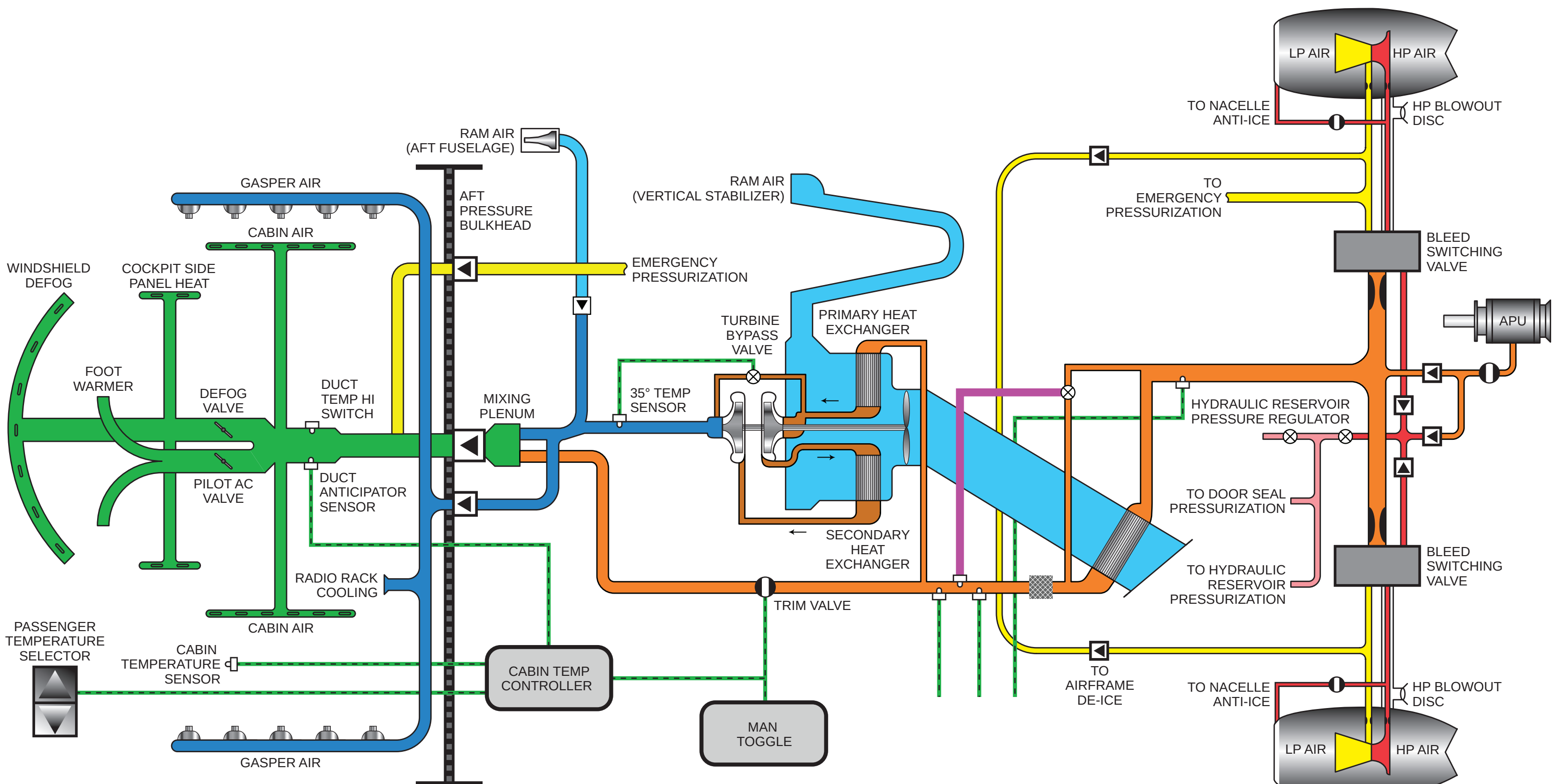


Figure 9-1. Pneumatic System Diagram

AIR SELECTION

Air source selection is accomplished by the air source selector switch on the ECS control.

The switch has the following positions:

- RAM
- APU
- L ENG
- BOTH ENGINES
- R ENG
- EMERG

When the switch is placed in one position that valve is open and all other valves are closed. One exception to this is the BOTH ENGINES position which opens both engine bleed switching valves (BSV) and closes all other valves. This is the normal position of the switch in flight.

The other exception is the RAM position. There is no ram-air shutoff valve. The ram- air inlet has a check valve. When RAM is selected it closes all other valves. When the pressure in the duct falls below the ram-air pressure, the check valve opens and allows ram air to enter the duct and ventilate the aircraft.

The switch has a detent for the L ENG, BOTH ENGINES, and R ENG positions. This is done to prevent accidental selection of one of the other positions. To select one of the other positions it is necessary to pull up on the switch and then make the selection.

COMPONENTS

BLEED SWITCHING VALVES

Two bleed switching valves, one for each engine, are in the upper aft fuselage. The bleed switching valves control the normal source of air to the common manifold of the temperature control system (see Figure 9-1).

Air from the high pressure inlet to the bleed switching valve (BSV) is used to pressurize the hydraulic reservoirs and the cabin door seal. The hydraulic system is covered in Chapter 13.

This air is also used for the service pressure regulator valve, which in turn provides the working pressure for the trim air valve (hot air valve).

The bleed switching valves are electrically controlled and pneumatically actuated. In the event of a loss of electrical control power, they fail-safe to the open position.

The valves incorporate LP and HP air-inlet ports, a mixing chamber, HP air pressure regulator, LP air-inlet check valve, solenoid valve, and a pneumatically actuated modulating and shutoff valve.

Control of the bleed switching valves is by the air selector switch located on the top right side of the center pedestal (see Figure 9-2).

OPERATING MODE

The bleed switching valve is sensitive to LP and HP air inlet pressure. The best air source is selected from either the LP or HP inlet pressures. Whenever LP inlet pressure is less than approximately 21.5 psi, the HP pressure regulator opens the HP poppet valve and the LP check valve closes. HP bleed air becomes the sole source in this condition (Figure 9-3).

The HP regulator modulates to maintain the HP bleed air pressure through the BSV to approximately 21.5 psi, regardless of the HP pressure available from the engine.

As engine power increases the LP bleed-air pressure also increases, along with a relative increase in HP air pressure. When the LP bleed air from the engine reaches approximately 21.5 psi the venturi action across the HP pressure regulator valve induces the LP check valve to open. This allows bleed air from both the LP and HP to pass through the BSV at approximately 21.5 psi.

As the engine power continues to increase eventually the LP bleed-air pressure from the engine exceeds the 21.5 psi setting and the HP pressure regulator closes and stops the use of the HP bleed air.

Eventually there could be too much LP bleed-air pressure. At this point the pneumatically operated modulating and shutoff valve regulates the BSV output pressure to a maximum of 27.5 ±2 psig.

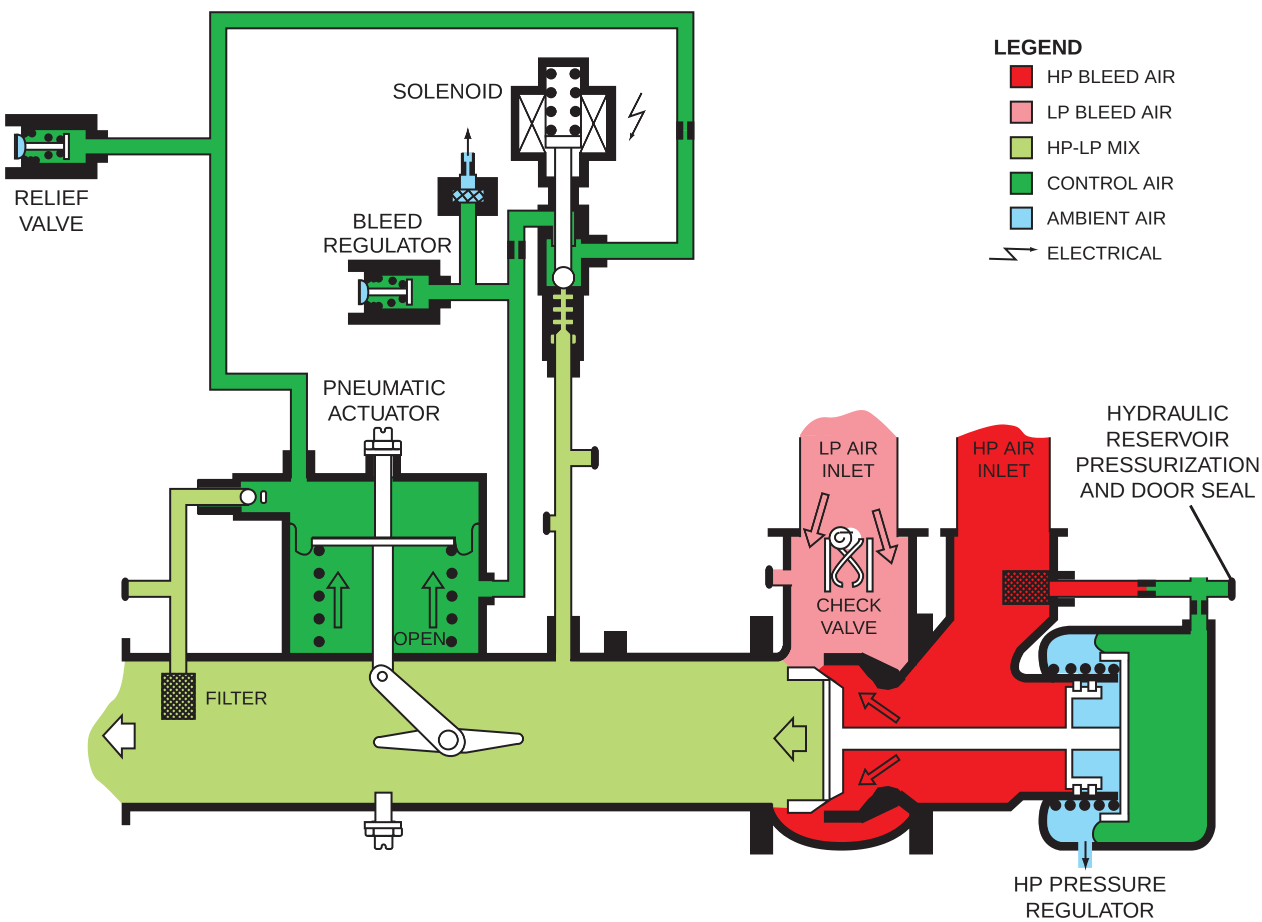


Figure 9-3. Bleed Switching Valve Open

If at any time during the flight the LP bleed air pressure drops to less than 21.5 psi the HP regulator will open and limit the minimum output pressure of the BSV to 21.5 psi. At this point the check valve at the LP input will close and prevent reverse flow to the engine.

CLOSED MODE

Both electricity to the solenoid and bleed-air pressure are required to close the BSV (Figure 9-4). A loss of either bleed air or electricity causes the shutoff section of the valve to open.

In the event of one engine shutdown, reverse airflow to the shutdown engine is prevented by the check valve at LP bleed port and the fact that the HP regulator closes if the pressure at the inlet to the BSV exceeds 21.5 psi.

APU AIR VALVE

The APU air valve is a pneumatically operated electrically controlled valve.

When the valve is open it allows air from the APU to enter the common air duct to the temperature control system.

The APU air valve has two solenoids, solenoid No. 1 and solenoid No. 2.The APU air valve is controlled by two switches on the ECS control panel (see Figure 9-2).

When the air selector switch is in the APU position it energizes solenoid No. 1, which opens the valve and operates in the low-pressure mode. If the APU AIRFLOW switchlight is also selected, it energizes solenoid No. 2.

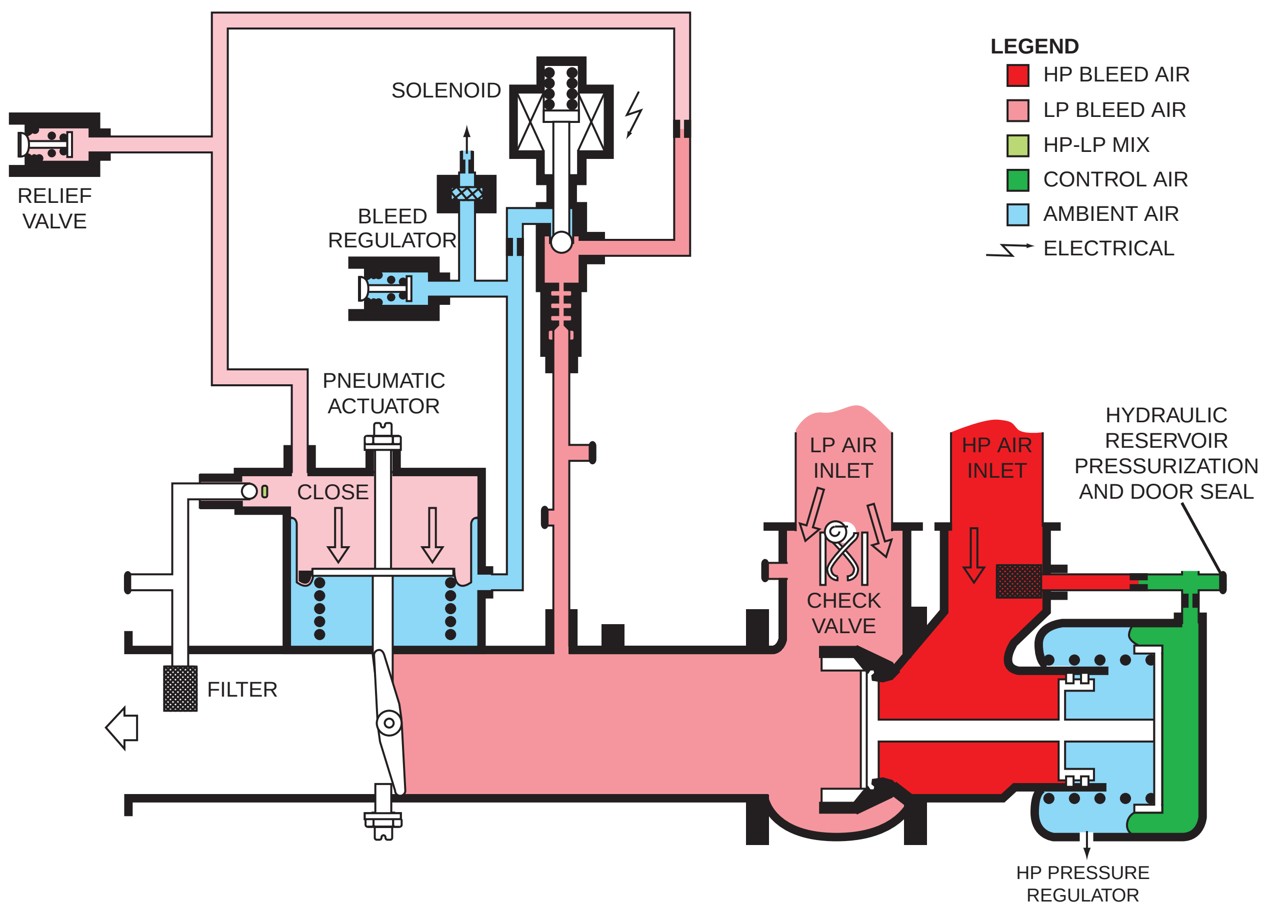


Figure 9-4. Bleed Switching Valve Closed

With both solenoids energized, the APU air valve operates in the high-pressure mode. Under these conditions, the APU AIRFLOW switch illuminates the words HI FLOW.

When the APU AIRFLOW switch is pressed a second time, it deenergizes solenoid No. 2 and extinguishes the words HI FLOW. This causes the valve to operate in the low-pressure mode.

PNEUMATIC SYSTEM OPERATION

When the air selector switch is placed in the L ENG, BOTH ENGINES, or R ENG position the respective BSV opens and all other valves close. (Figure 9-5)

Air passes through the mixed bleed-air venturi to the common bleed-air manifold for the temperature control system.

When the air selector switch is placed in the APU position it closes the bleed switching valves and opens the APU air valve. Bleed air can come from either the APU or engines but never both.

Air from the APU air valve passes through the APU check valve and provides air to the common bleed-air manifold. The purpose of the check valve is to prevent reverse airflow when the BSV is open and the APU is not operating.

The APU airflow is sufficient to provide full

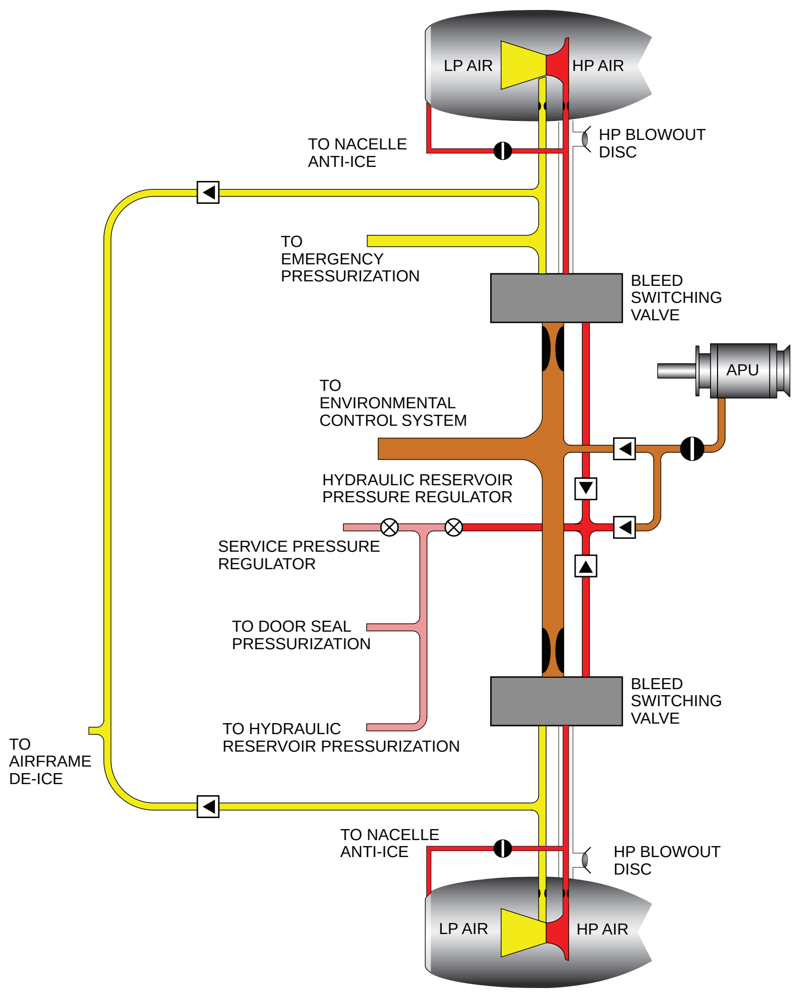


Figure 9-5. Pneumatic System Operation

heating, cooling, and pressurization, either on the ground or in flight.

Bleed air for the hydraulic reservoir and cabin door seal pressurization system is tapped off the APU bleed-air duct between the air valve and the check valve.

To provide more control of the airflow into the cabin, the APU air valve can be selected to either high or low flow. This is done with the APU AIRFLOW push button on the ECS control panel.

When it is selected to HI FLOW, the push-button illuminates the words HI FLOW. With the APU selected and the button not pushed, the air flow is lowered and the light is extinguished.

The HI FLOW is especially beneficial in providing more rapid heating or cooling of the cabin while the aircraft is on the ground. Once the cabin has reached the desired temperature, selecting low flow provides a quieter cabin.

Emergency Air Supply

In the event of a failure or malfunction of the normal cabin pressurization air supply, an emergency air source is available. The emergency bleed air comes from the right engine low-pressure bleed-air outlet upstream of the right bleed switching valve (BSV).

The emergency bleed-air supply consists of a pneumatically operated electrically controlled shutoff valve, air orifice, and a separate duct that goes directly to the aft pressure bulkhead. This duct bypasses all the normal ducts in the unpressurized area of the fuselage.

The emergency air shutoff valve is a solenoid controlled valve.

Placing the air selector switch to the EMERG position energizes the solenoid. This allows the LP bleed air from the right engine to overcome the spring and open the valve.

Selecting any other position of the air selector switch deenergizes the solenoid and a spring closes the valve.

Since the emergency bleed-air duct bypasses the temperature control ducts, the only way to control the emergency bleed air temperature is by adjusting the right engine power setting.

Use of windshield defog should be limited to lower engine power settings when emergency bleed air is selected. This is to avoid the possibility of damaging the windshield due to the potential of excessive bleed air temperature while in emergency.

WARNING AND PROTECTION

OVERPRESSURE SWITCH

An overpressure switch is installed in the mixed-air duct to warn the flight crew of an overpressure condition.

OVERTEMPERATURE SWITCH

Two temperature switches are installed in the mixed-air duct just prior to entering the air-cycle machine. Either of these switches warn the flight crew of an over temperature condition.

Both temperature switches and the pressure switch are wired in parallel. If any of the switches close, they supply a ground to the DCU, which illuminates the red BLEED PRESS/TEMP HI warning message on the EICAS. When the condition is corrected, the switch(es) open and the message is extinguished.

HIGH PRESSURE DUCTING

The high pressure (HP) ducts from the engines to the bleed switching valves (BSV) are double-wall ducts. Essentially they are ducts within ducts.

The HP bleed air flows through the inner duct. The outer duct is vented to atmosphere through a hole in a blow-out disc.

The blow-out disc is on the bottom of the engine pylon and must be inspected prior to flight.

If the inner duct ruptures, the hot bleed air is contained within the outer duct. If the pressure in the outer duct reaches 40 psi or more, it ruptures the disc and safely vents the hot air away from the fuselage.

Since the hot bleed air is safely directed away from the fuselage, there is no flight compartment indication that this duct has ruptured.

Bleed-Air Leak Sensing Elements

The low pressure (LP) ducts from the engines and the APU bleed-air supply duct have overheat sensing elements that provide visual warning messages on the EICAS when the temperature surrounding the bleed duct exceeds a predetermined temperature.

Each sensing element consists of a center and outer conductor separated by a semiconductor that does not normally conduct electricity unless it reaches a preset temperature.

Mounting clips and insulated grommets mount the sensing element to the bleed-air ducts, maintaining a minimum clearance between the sensing element and the surrounding ducts to avoid false alarms.

Bleed-Air Leak Controller

The bleed-air leak detectors are connected to controllers, which energize an internal relay when an overtemperature condition exists. The controller, in turn, sends a signal to the DCU to illuminate the CAS message.

The right bleed-air leak detection system is routed along the right engine bleed-air ducts up to the air cycle machine. If an overtemperature occurs along this duct, it causes the controller on that side to illuminate the R ENG BLEED LEAK message

The left bleed air-leak detection system is routed along the left engine and APU bleed-air ducts. If an overtemperature occurs along either of these ducts, it causes the left controller to illuminate the L ENG/APU BLEED LEAK message.

Illumination of the L ENG/APU BLEED LEAK message automatically shuts down the APU. It is important to remember that performing a FIRE PRESS TO TEST with the APU running also shuts down the APU.

The left APU bleed-air leak detection system is powered by 28 VDC from the No. 1 distribution bus. The right bleed-air leak detector system is powered from the No. 2 distribution bus.

The circuit breakers that protect the systems are labeled BLEED AIR LEAK L, R respectively, and are under the ENGINE section of the cockpit overhead circuit-breaker panel.

The test circuit for the system is powered from the No. 1 distribution bus and is activated by pushing the fire PRESS TO TEST pushbutton.

Bleed-Air Leak System Operation

The bleed-air leak detection system provides bleed-air leak detection for the following zones:

Right engine bleed-air ducts

Left engine and APU bleed-air ducts

Each zone consists of one or more sensing elements and a controller. When an overheat condition impinges on any portion of a sensing element, raising its temperature above sensor temperature setting, the semiconductor conducts the electricity from the center conductor to the outer conductor which goes to ground.

This completes the circuit, which causes the warning relay in the controller to energize and trigger the alarm signal. When the sensor cools below the preset temperature setting, the semiconductor stops conducting, causing the relay to deenergize and stop the alarm signal. This causes the warning message to extinguish.

The sensing elements connect to the control unit in a continuous 1oop arrangement with both ends of the center conductor going to the relay in the controller and both ends of the outer conductor going to ground. This installation ensures that a single break in the 1oop does not cause the system to become inoperative.

If more than one break occurs in the loop, only the portion of the sensing element between the breaks is lost. The remaining sections connected to the control unit retain the ability to detect and signal an overheat condition. A false warning does not occur as a result of one or more breaks in the sensing element circuit.

A test circuit is provided to ensure continuity of the center conductor. When the fire PRESS TO TEST pushbutton is pressed it energizes the bleed-air leak test relay.

This external relay energizes another relay within the controller. The relay in the controller disconnects one end of the center conductor from the warning relay, also in the controller, and connects it to ground. This way the center conductor must have good continuity to energize the warning relay and illuminate the respective CAS message.

RAM-AIR SYSTEM

Ram comes from three separate inlets. One inlet is at the base of the vertical stabilizer and provides ram air for the refrigeration unit (air cycle machine).

The second ram-air inlet is a flush-mounted scoop near the top right side of the fuselage. This inlet provides ram-air circulation for the cabin and flight compartment. This ram-air enters environmental ducts through a check valve forward of the water separator.

There is no ram-air shutoff valve. When the air selector switch on the pedestal is selected to RAM, it simply closes all other air valves.

As long as the ducts are pressurized by a bleed-air source, the check valve is closed. If there is a loss of bleed air, the ram air opens the check valve and provides air circulation for the cabin and flight compartment. This system also prevents negative pressure differential in the cabin in the event of a pressurization system malfunction. There is no temperature control of ram air.

The third ram-air inlet is a flush-mounted scoop on top of the fuselage just aft of the baggage compartment. The ram-air duct outlet faces the aft wall of the baggage compartment. The air deflects from the aft wall of the baggage compartment to ventilate the aft fuselage area except for the APU compartment.

TEMPERATURE CONTROL SYSTEM

The bleed air from the engines or APU enters the common duct. From there it goes through the precooler or bypasses the precooler depending on the bleed air temperature going to the refrigeration unit.

From the precooler, all the bleed air goes through the ozone converter to remove ozone from the air entering the cabin.

After the bleed air leaves the ozone converter, it branches off to go either through the refrigeration unit (air-cycle machine) or to the trim air valve.

The trim-air valve serves as a hot air control valve. When the trim-air valve is full open, it provides maximum heating of the cabin. If the valve is closed, all the air is forced through the refrigeration unit and changed to cold air. This condition results in maximum cooling of the cabin.

With the trim-air valve partially open, some hot and cold air enters the mixing chamber and provides conditioned air for temperature control of the cabin and flight compartment.

Ozone Converter

The ozone converter is essentially a catalytic ozone converter that removes ozone from the air entering the cabin and flight compartment.

The ozone converter is to the right and slightly forward of the refrigeration unit.

This unit contains a substance that is capable of breaking down the ozone in the air supply from the engines or APU.

Bleed-Air Precooler

The precooler lowers the temperature of the bleed air from either the engines or APU. The precooler is an air-to-air heat exchanger mounted just aft of the refrigeration unit.

The ram air exiting the refrigeration unit is used to cool the bleed air before it enters the refrigeration unit.

There is a temperature sensor and a bypass valve associated with the precooler.

The precooler bypass valve is a pneumatically operated valve controlled by a temperature sensor.

TEMPERATURE CONTROL SYSTEM

OPERATION

Cabin and flight compartment temperature is normally preset for automatic control. The temperature is set by positioning the TEMP CONTR knob on the ECS panel to the desired temperature. The temperature control system may be operated in either the automatic or manual mode (Figure 9-6). Cabin temperature is displayed on the secondary page of the EICAS.

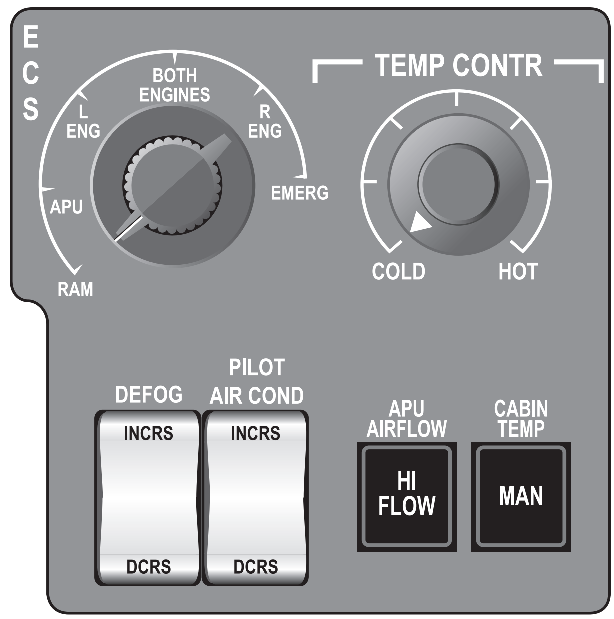


Figure 9-6. ECS Control Panel

AUTOMATIC MODE

In the automatic (normal) mode the temperature is set by positioning the TEMP CONTR knob to the desired temperature. There are no numbers on this knob. You can see the selected temperature by going to the maintenance diagnostic system and selecting system parameters.

Limited temperature control is provided by a temperature selector located in the cabin. The cabin temperature control only operates in the automatic mode, and can only vary the temperature a few degrees on either side of the temperature called for by the TEMP CONTR knob in the flight compartment.

The temperature controller receives inputs from the TEMP CONTR knob, the cabin temperature sensor, and the duct temperature sensor. It uses these signals to develop the drive signal to position trim air valve.

The trim air valve mixes hot and cold air to provide the desired temperature. Power for the automatic temperature control system is from the No. 2 distribution bus.

MANUAL MODE

The manual temperature mode is selected by pressing the CABIN TEMP switchlight. When selected to manual operation, this switchlight illuminates the word MAN.

In the manual temperature mode, only the crew members can control the cabin and flight compartment temperature. The passenger temperature selector, cabin temperature sensor, and duct temperature sensors are inoperative.

In manual mode positioning, the TEMP CONTR knob utilizes the second potentiometer on the common shaft and separate circuitry in the temperature controller, to directly position the trim air valve.

Power for the manual temperature system is from the No. 1 distribution bus.

EMERGENCY AND ABNORMAL OPERATION

ENVIRONMENTAL SYSTEM EICAS MESSAGES

The environmental system messages WARNING, CAUTION, ADVISORY, and STATUS are displayed on the EICAS section (Figure 9-7) of the upper right portion of the primary page.

WARNING MESSAGES

BLEED PRESS/TEMP HI—Excessive pressure or temperature downstream of either bleed switching valve.

L ENG/APU BLEED LEAK—Leak or rupture in bleed air ducting from left engine or APU. APU automatically shuts down.

R ENG BLEED LEAK—Leak or rupture in bleed air ducting from right engine.

STATUS MESSAGES

ECS BLEED OFF—The ECS selector switch is either in the RAM position or the APU position and the APU is not running.

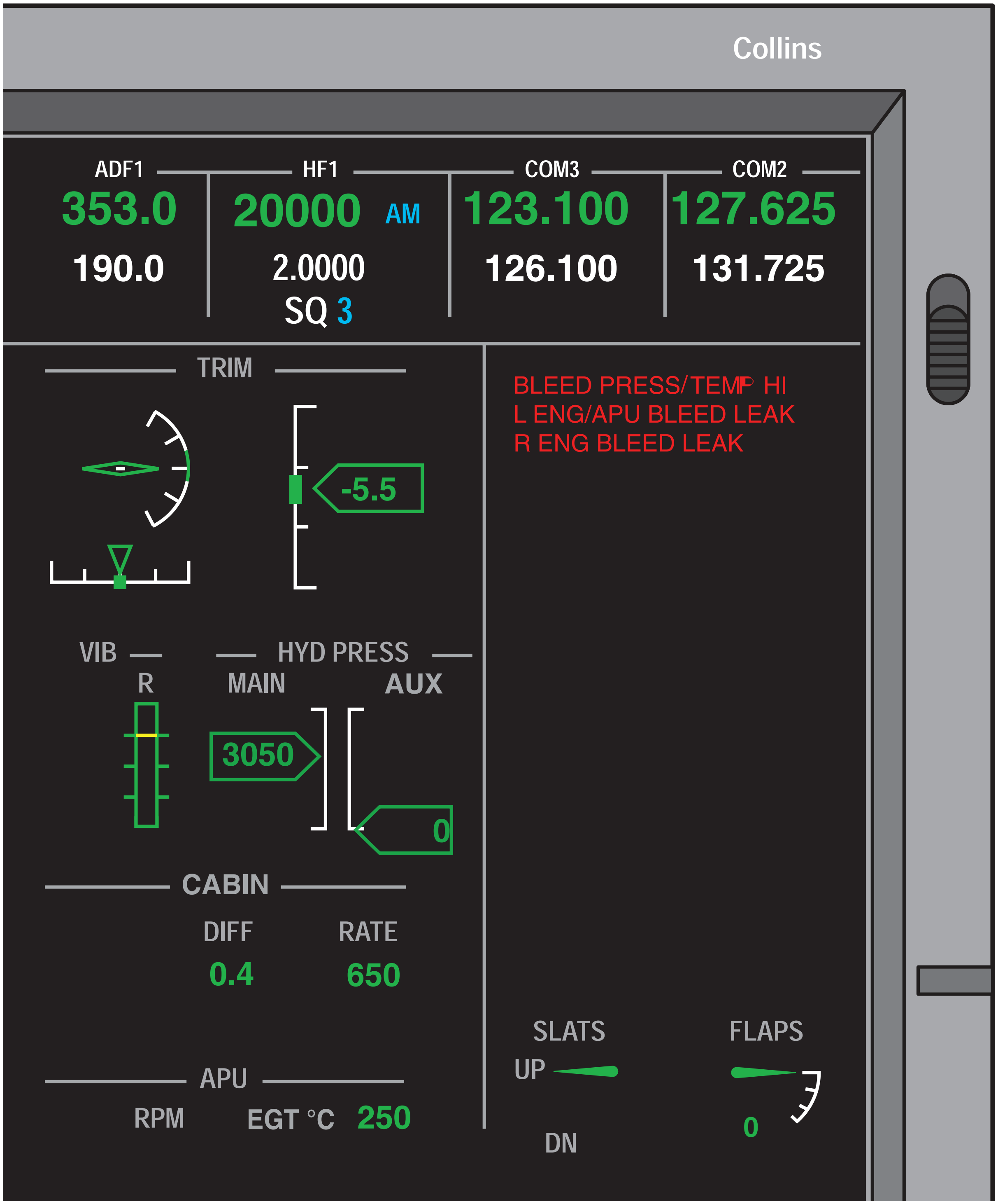


Figure 9-7. Pneumatic EICAS Indications

QUESTIONS

1. The pneumatic system extracts bleed air from:

A. First stage of the LP compressor
B. Single stage HP turbine
C. Both LP and HP ports on each engine
D. All of the above

2. The amount of HP/LP bleed air entering the environmental system is controlled by the:

A. Mixing chamber
B. Bleed switching valve (BSV)
C. HP pressure regulator
D. LP air inlet check valve

3. HP bleed air becomes the sole source of pressure when:

A. LP inlet pressure is less than 21.5 psi
B. HP pressure regulator opens the HP poppet valve
C. LP check valve closes
D. All of the above

4. APU airflow is selected and APU airflow switchlight illuminates HI FLOW on the ECS selector panel, this indicates:

A. APU air valves operate in the high pressure mode
B. APU air valves operate in the low and high pressure mode
C. APU air valves operate in the low pressure mode
D. Rapid cooling for the cabin only

5. BLEED PRESS/TEMP HI CAS message indicates:

A. Temperature around the L/R bleed air or APU air duct is above pre-set limits
B. Temperature in the duct upstream of the bleed switching valve (BSV) is too high
C. Pressure or temperature in the duct downstream of the bleed switching valve (BSV) is too high
D. All of the above

CHAPTER 10
ICE AND RAIN PROTECTION

CONTENTS

ILLUSTRATIONS

CHAPTER 10
ICE AND RAIN PROTECTION

INTRODUCTION

The G150 is equipped for anti-ice of pitot, SAT/TAS and AOA probes, static ports, windshields and engine nacelles. Additionally, wings and horizontal stabilizers are deiced with pneumatically powered boots. Rain removal from windshields is accomplished without mechanical wipers by a rain repellant coating.

GENERAL

Ice and rain protection consists of deice and anti-ice. Deicing is provided as follows:

- Inboard wings
- Outboard slats
- Horizontal Stabilizer.

Anti-icing systems are provided to prevent ice formation as follows:

- Nacelle air inlet
- P_2T_2 probes
- Pitot probes
- Static ports
- AOA sensor
- Windshield

SYSTEM DESCRIPTION

The ice and rain protection system (Figure 10-1) is divided into four major areas:

- Airframe Deice
- Nacelle Anti-Ice
- Pitot and Static Anti-Ice
- Windshield and Window Anti-Ice

Airframe de-icing is achieved by mechanical means. Electronically controlled pneumatically operated boot deicers, installed on the wing leading edge, slats leading edge and horizontal stabilizers cause any accumulation to break.

Nacelle anti-ice utilizes hot engine bleed air to prevent the formation of ice on the engine inlet. The P_2T_2 probe on the engine is heated electrically.

The pitot and static ports, angle of attack sensor and total air temperature (TAT) probes are heated electrically.

Windshields are heated electrically, with provision for rain removal by means of a rain repellent coating on the windshields.

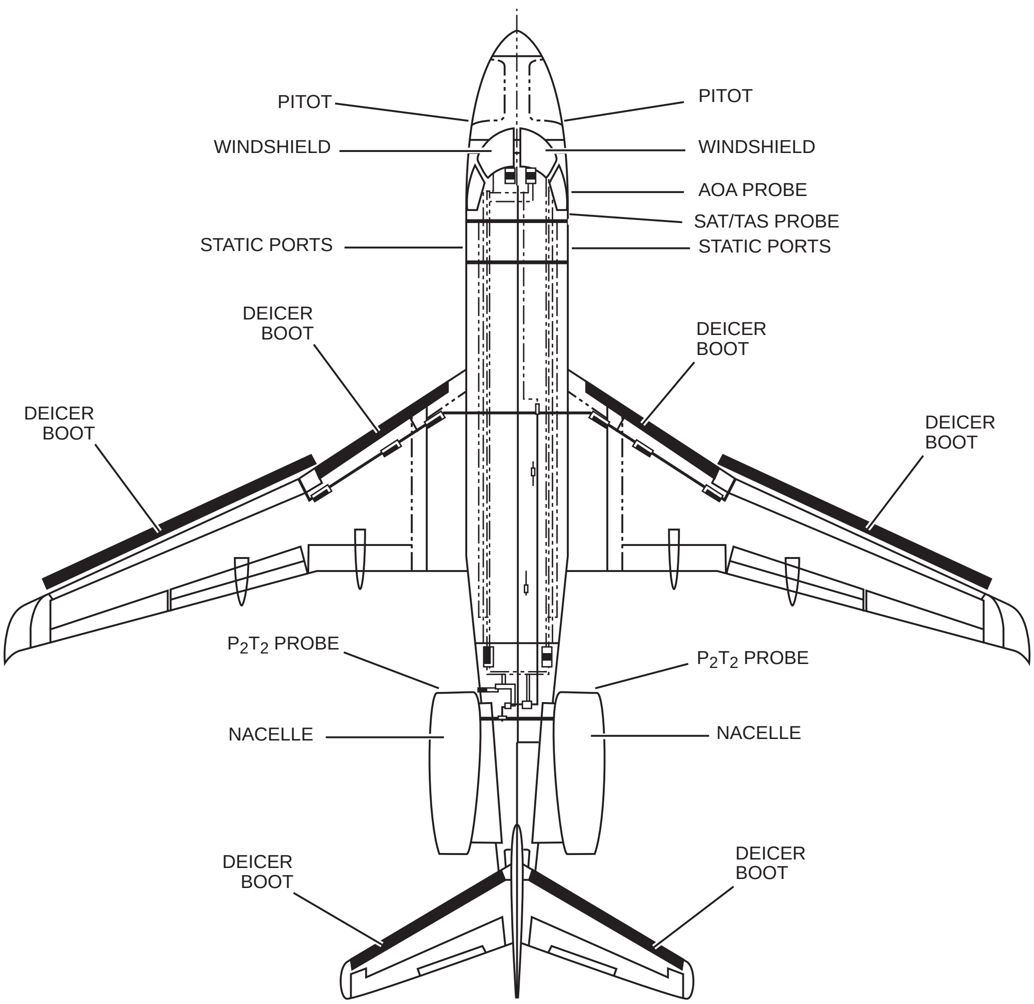


Figure 10-1. Ice and Rain Protection System

AIRFOIL DEICING SYSTEM

Airfoil de-icing (Figure 10-2), utilizing engine bleed air, provides air pressure and vacuum to the de-icing boots. The boots are installed on the leading edges of the wings (fixed and slatted) and horizontal stabilizer.

High pressure, low vacuum and inflation pressure indication switches are provided in the system to warn of excessive pressure, insufficient vacuum and boot inflation status.

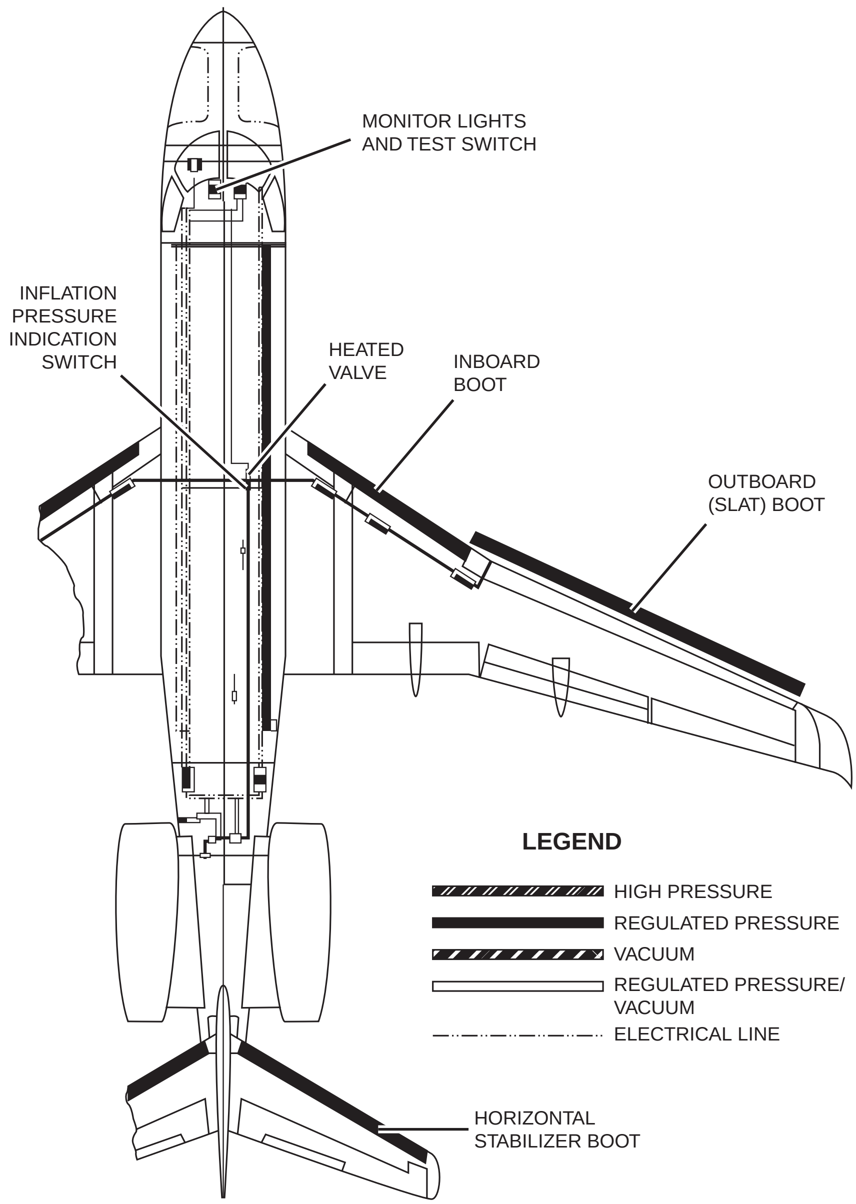


Figure 10-2. Airfoil Deicing Protection Control

COMPONENTS

Deicer Boots

Deicing of the wing and horizontal stabilizer leading edges are by pneumatically inflated boots. The boots are installed on the leading edges of the wings, slats, and horizontal stabilizer.

Distributor Valve

The distributor valve alternately supplies pressure and vacuum to the air lines of the wing and horizontal stabilizer deicer boots. When the deicing system is not in use, the spring-loaded solenoids position the valves to apply constant vacuum to the boots.

Pressure Regulator/Relief Valve

The pressure regulator maintains the air pressure in the deicing system within safe limits.

Air Ejector

The air ejector creates the vacuum used to deflate the deicer boots so they conform to the leading edges when the system is not in operation or between periods of inflation when the system is operating.

Drain Valve

The drain valve provides drainage from the air line to the wing deicer boots. An electrical heating blanket, incorporating a thermostat, is installed on the valve. Power for the blanket in flight is from the No. 1 distribution bus and, on the ground, through the right main gear oleo switch. The blanket receives power on the ground or in flight with the system activated and a temperature of 40 ±5°F or less. Selecting the DE-ICE MONITOR TAIL-WING pushbutton switchlight bypasses the thermostat so that operation of the heating blanket can be checked on the ground. The TAIL portion of the pushbutton switch light illuminates when system check is normal.

High Pressure Switch

The high-pressure switch senses the regulated air pressure at the air ejector inlet. An amber DE-ICE HI PRESSURE message appears on the MFD to indicate a failure in the system if the switch closes.

Low Vacuum Switch

The low vacuum switch senses the vacuum in the air ejector vacuum outlet. The low vacuum switch closes if vacuum in the boots becomes too low and sends a signal to display an amber DE-ICE LOW VACUUM message on the MFD.

Inflation Pressure Indication Switch

The inflation pressure switch in the wing and horizontal stabilizer supply lines sends a signal to the WING / TAIL lights in sequence on the overhead panel confirming inflation of deicing boots.

OPERATION

Controls

Electrical control of the airfoil deice system (Figure 10-3) is located in the ICE PROTECTION area of the overhead circuit breaker panel. The NORM circuit breaker is powered from the No. 1 distribution bus while the ALT circuit breaker is powered by the No. 2 distribution bus.

System Operation

Redundancy is provided in the deicing system (Figure 10-4) by installation of two identical timers, each of which can control boot inflation and deflation when selected. Actuation of either timer is by selecting the NORM or ALT position on a three-position switch on the overhead panel.

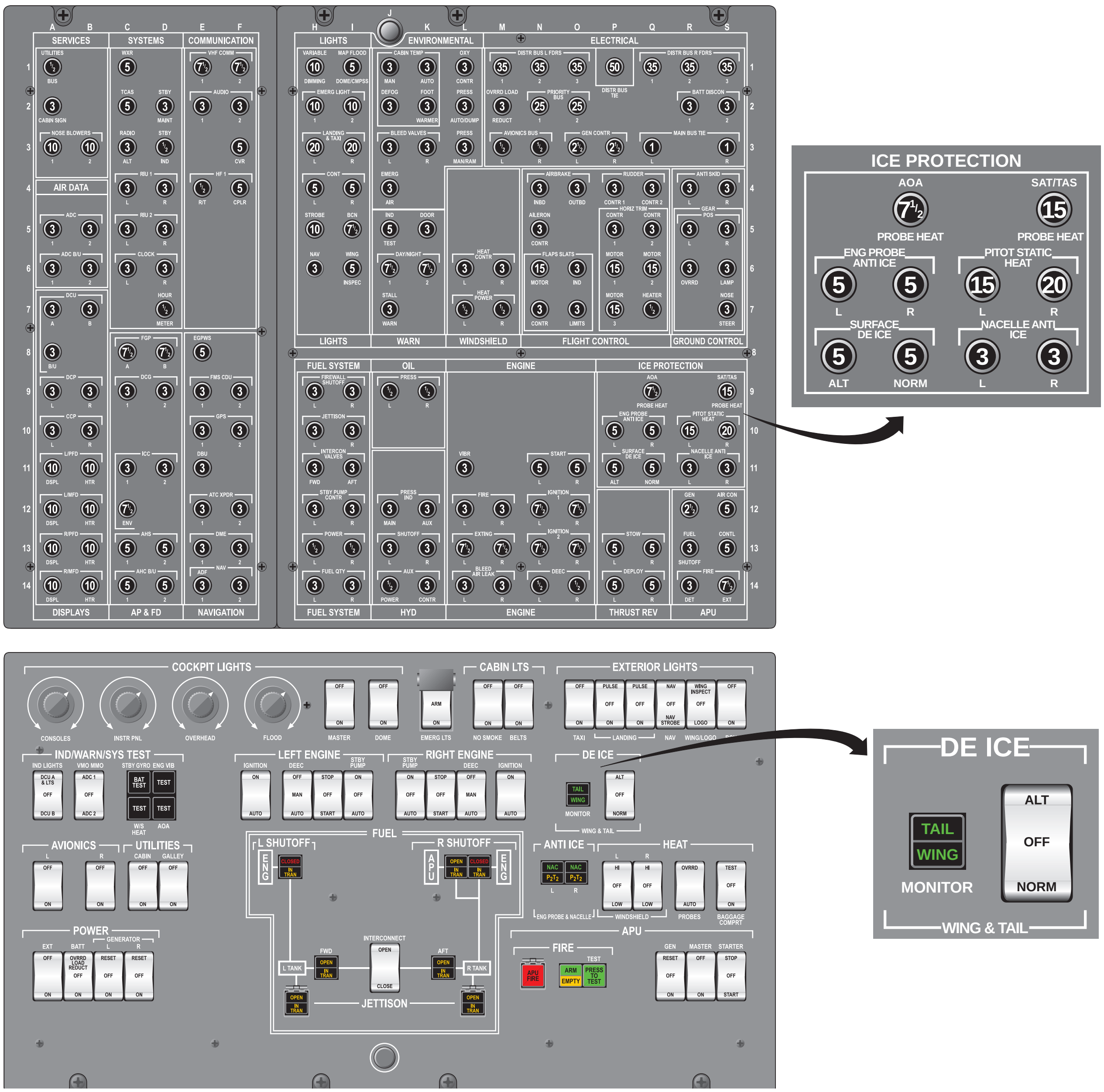


Figure 10-3. Airfoil Deicing Controls

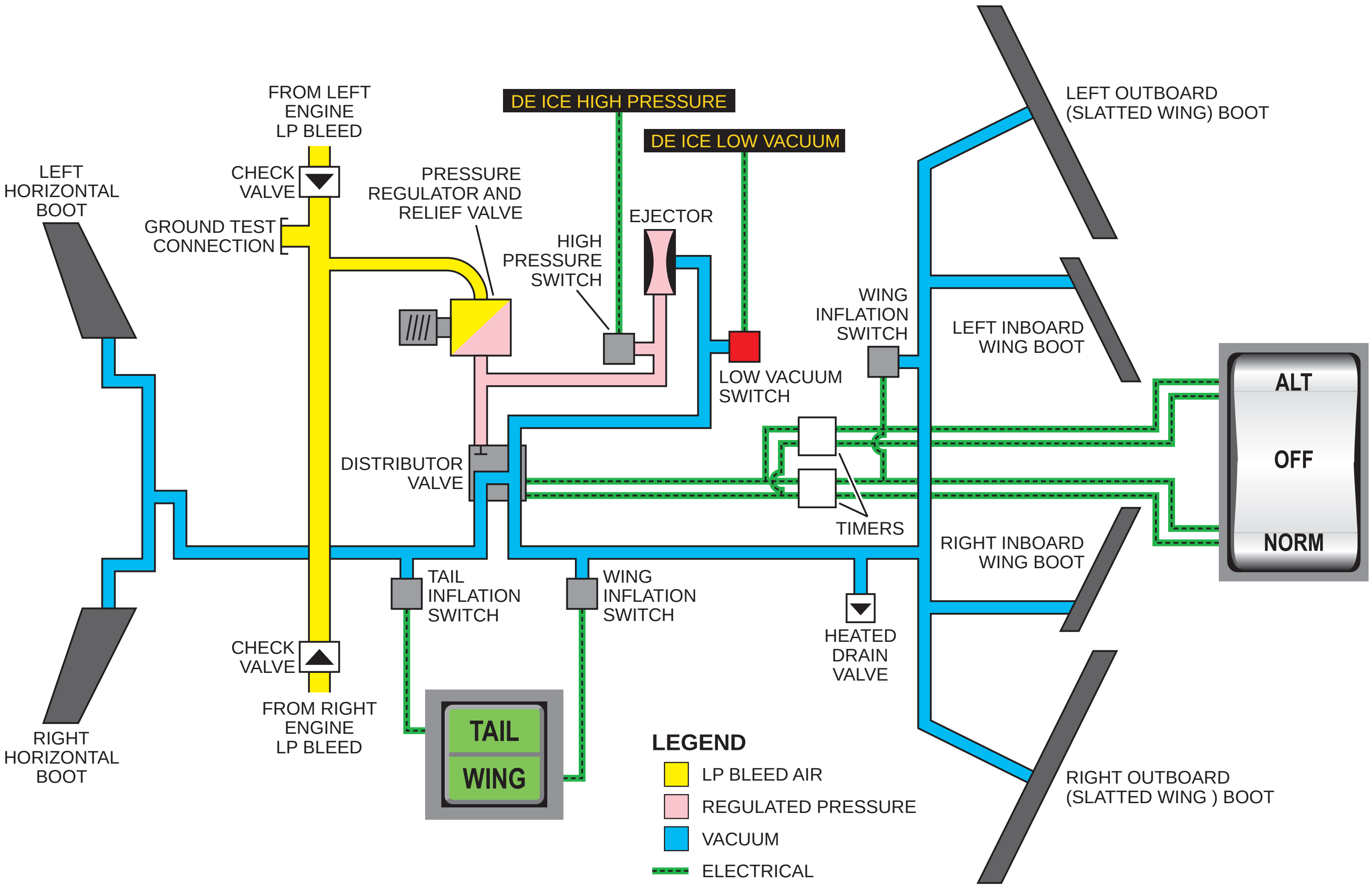


Figure 10-4. Airfoil Deicing Schematic

Bleed air from both aircraft engines' low pressure ports is used for system operation. High pressure and low vacuum switches warn of excessive pressure or insufficient vaccum. A check valve is installed on each engine bleed line which allows for single engine operation.

The bleed air pressure is controlled by means of a pressure regulator equipped with a relief valve so that a failure of regulator does not overpressure the boots. From the pressure regulator, the air is fed into a distributor valve and an ejector. The distributor valve is connected to the boot deicer as well as to the ejector and high pressure supply in order to provide them with pressure or vacuum. The ejector evacuates air from the boot deicers and operates continuously during engine operation independently of the surface ice protection system operation.

In order to prevent freezing due to any condensation within the system, a drain valve with heating blanket is installed at the lowest point of the system. The drain valve is closed while vacuum is available to the wing boots deicer and is open when pressure is available to the boots. With the system in the OFF position, vacuum is continuously supplied to the wing and empennage boots through the distributor valve by the ejector.

INDICATIONS

Indication of adequate boot inflation pressure during system operation is provided by a split-lens switchlight TAIL-WING MONITOR (see Figure 10-3). The TAIL light is operated by a pressure switch installed in the supply lines to the horizontal stabilizer boots while the WING light is operated by a switch in the wing boot supply lines. When selecting the switch to NORM or ALT, the lights illuminate momentarily in sequence to indicate adequate inflation pressure for that system. All lights extinguish when the timer cycles the distributor valve to the next system and the pressure in the supply lines is reduced.

The switchlights include a push-to-test function. When depressed, the TAIL lens illuminates, indicating that the drain heating element is operating.

Do not prolong the tail-wing test function as power is applied to the heating element during the test.

High-pressure and low vacuum indication switches are provided in the system to warn of excessive pressure and insufficient vacuum (Figure 10-4). The high pressure switch senses the regulated air pressure at the air ejector inlet. An amber DE-ICE HI PRESSURE message appears on the MFD to indicate a failure in the system if the switch closes. The DE-ICE LOW VACUUM message also indicates insufficient vacuum. Momentary illumination of the WING/TAIL switch indicates inflation.

The EICAS message is controlled by either of two pressure switches, one located on the motive-flow line from the regulator to the air ejector and one located on the ejector suction line to the distrbutor valve.

NACELLE ANTI-ICE SYSTEM

Each engine nacelle air intake (nose cowl) has an anti-ice system using hot engine bleed air to prevent formation of ice in and around the air intake. Unmanifolded hot engine bleed air from the HP compressor is supplied to an electropneumatic valve for each nacelle. Unused anti-icing air is exhausted overboard via a duct and an exit in the nose cowl outer skin. The nacelle anti-icing valve and P_2T_2 inlet air probe are electrically controlled.

Nacelle anti-ice system indication is provided by pressure switches which relay signals to the cockpit.

COMPONENTS

Nacelle Anti-ice Valve

Nacelle anti-ice valve is a spring-loaded, normally open, fail-safe valve which opens when electrical power fails or is removed, thus enabling continuous heat supply to the nacelle.

Thermal Anti-icing Pressure Switch

Pressure switches illuminate the NAC P_2T_2 light on the overhead panel to show proper pressure in the nacelle anti-ice system during system operation.

Inlet Probe

The inlet P_2T_2 probe heaters (Figure 10-5) receive power from distribution buses through the ANTI ICE ENG PROBE & NACELLE L and R pushbutton switches.

The engine inlet P_2T_2 probe supplies air pressure and temperature information to the engine fuel computer. The P_2 inlet of the probe is susceptible to ice formation and is therefore protected by an electrical anti-icing heating element. The power for the element is routed through the associated engine oil low-pressure switch. This switch is open when the engine is not operating, or if it is operating and oil pressure drops below the engine idle value.

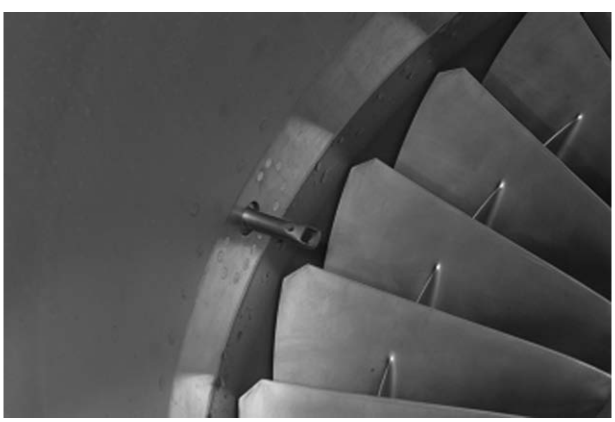


Figure 10-5. Nacelle P_2T_2 Probe

CONTROLS

The nacelle anti-icing system is controlled by a push-on/push-off splitlens switchlight labeled L and R ENG PROBE & NACELLE. (The term probe applies to the P_2T_2 probe in the engine inlet). The switchlight lenses are labeled NAC and P_2T_2. The power supply for nacelle anti-ice is supplied from the associated distribution bus.

Depressing an ENG PROBE & NACELLE switchlight (Figure 10-6) routes power to the associated engine oil pressure switch. If the engine is operating, the switch opens and removes power from a relay, completing a circuit from the associated distribution bus to the P_2 heating element of the probe.

System Operation

Ice protection is provided by ducting high pressure, hot engine bleed air through a solenoid-controlled shutoff valve. Air is distributed around the O chamber by means of a spray ring. Hot engine bleed air is directed to a piccolo tube inside the engine nose cowl. The air is distributed inside the nose cowl through holes in the piccolo tube and then discharged overboard through louvers. The bleed air flow is controlled by the nacelle anti-ice valve, which is spring-loaded open ensuring continuous anti-ice protection in case of electrical failure.

Control of the P_2T_2 probe anti-icing is common with the associated nacelle anti-icing.

NOTE

No indication is provided for P_2T_2 probe heat failure.

The top switchlight segment labeled NAC (Figure 10-6) receives 28-VDC power from the ENG ANTI-ICE circuit breaker. Pressing the NAC-P_2T_2 switchlight lenses opens the engine anti-ice valve, allowing HP air to flow to the nacelle anti-ice duct. The anti-ice pressure

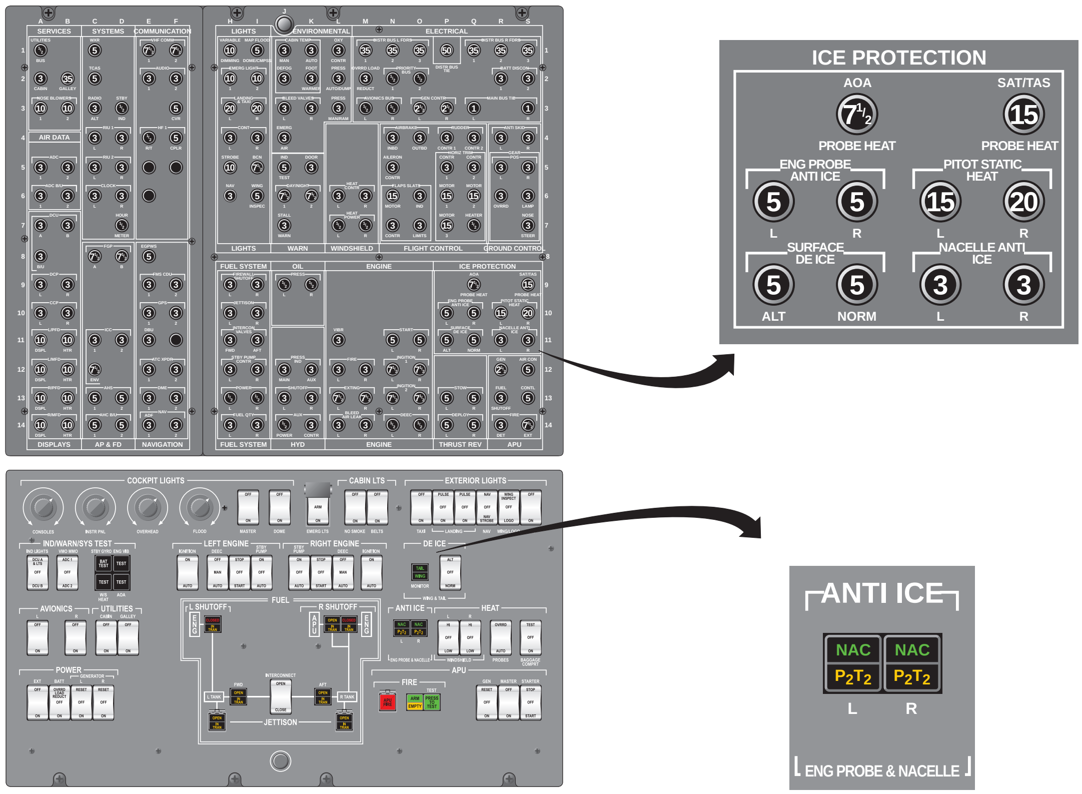


Figure 10-6. Nacelle Anti-Icing Controls

switch monitors anti-ice air pressure. When bleed-air pressure greater than 5 psi is present downstream of the engine anti-ice valve, the NAC light illuminates. A amber CAS message: NAC ANTI-ICE, illuminates whenever nacelle anti-ice has been selected but anti-ice air pressure is less than 5 psi.

During a groundcheck of the engine anti-ice system, engine N_1 speed will decrease approximately 2.4% due to bleed air extraction. During flight, the engine digital electronic engine controller (DEEC) increases N_1 speed approximately 2.0%, never exceeding thrust or rpm limitations.

INDICATIONS

Nacelle and probe anti-ice is controlled by the ANTI ICE ENG PROBE & NACELLE pushbutton switches. The switches illuminate when the system is activated and sufficient pressure is sensed in the system for flight or ground operation, respectively. The amber L NAC ANTI ICE FAIL or R NAC ANTI ICE FAIL message appears on the MFD if the system activates and one or both NAC lights on the overhead panel fails to illuminate.

PITOT AND STATIC SYSTEM

Ice formation is prevented on the air data sensing probes which include the pilot and copilot pitot probes, the angle-of-attack (AOA) probe, the static air temperature/total air temperature probe (SAT/TAT), and the pilot and copilot static ports. Typical heated sensors are shown in Figure 10-7.

Left and right pitot probes, three left and three right static ports and the TAT probe are electrically heated to prevent ice formation. A current sensor in each heating element circuit detects any interruption of current to the heating element and operates the appropriate message on EICAS.

There is a fourth static port on the right side of the fuselage which is not heated and is available for cabin pressure.

COMPONENTS

Static Ports and Probes Heaters

Static ports heater, pitot probes heater, TAT probe heater and AOA heaters are used to prevent ice formation. All heaters are electrically heated and are controlled by a single PROBES HEAT switch located on the overhead panel.

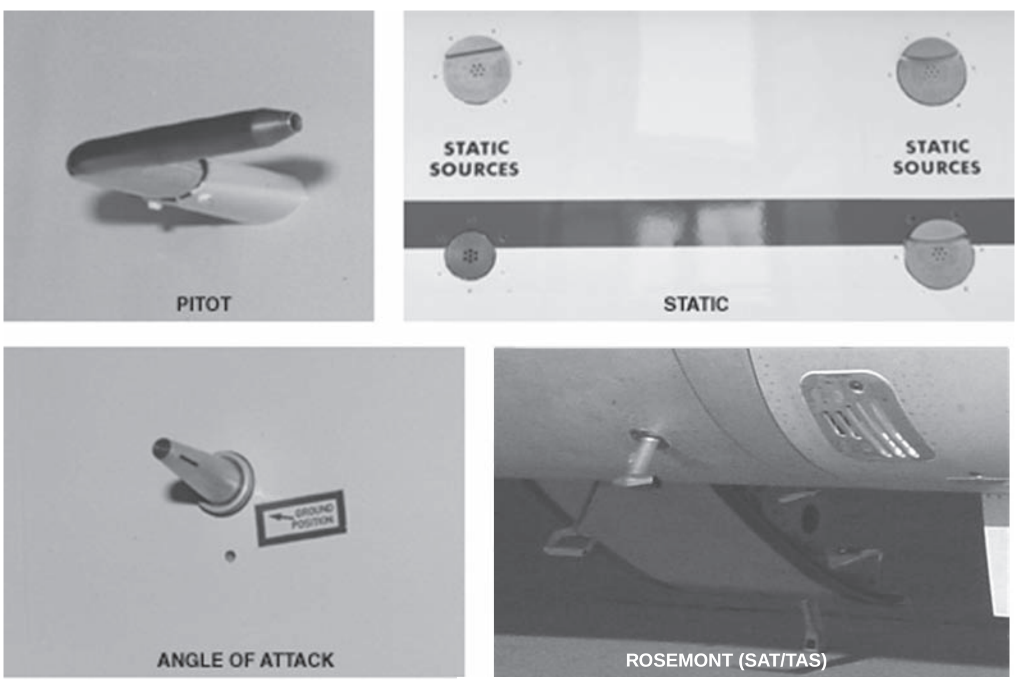


Figure 10-7. Pitot and Static System Heated Sensors

OPERATION

Controls

Probe and static port anti-icing is controlled by a two-position (OVRRD and AUTO) switch labeled PROBES (Figure 10-8). When the switch is in the AUTO position and the aircraft is on the ground, the pitot probes, static ports, and SAT/TAT systems are deactivated; however, power is provided to the AOA probe. The OVRRD position of the switch applies power directly to all heating elements.

The OVRRD position of the PROBES HEAT switch is limited to less than 30 seconds on the ground.

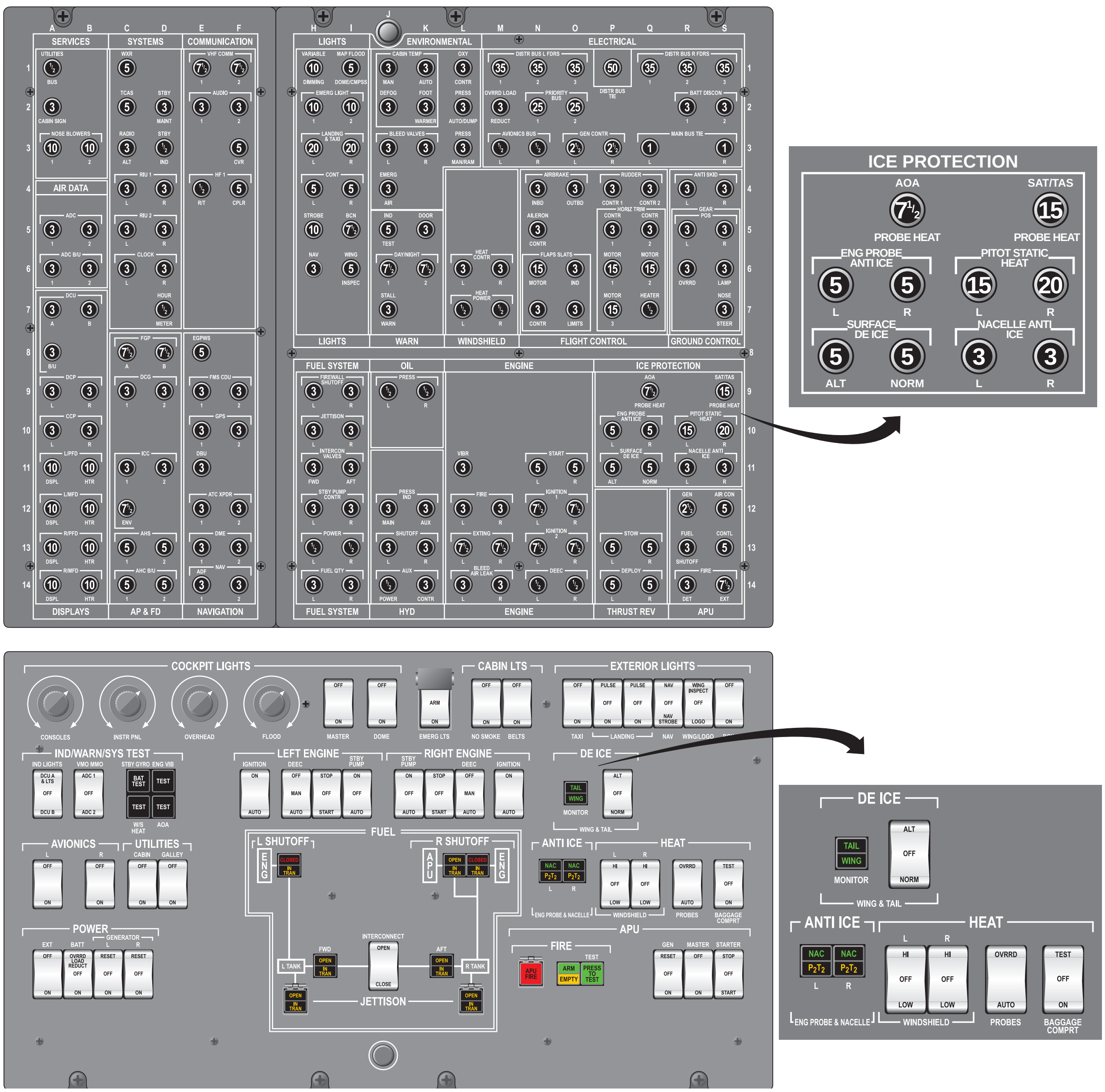


Figure 10-8. Pitot and Static System Controls

System Operation

Power is provided for the left pitot probe, one left and one right static port heaters from the No. 1 distribution bus through the L PITOT STATIC HEAT circuit breaker. The right pitot probe, two right and two left static port heaters receive power from the No. 2 distribution bus through the R PITOT STATIC HEAT circuit breaker. The nose landing gear WOW switch interrupts power to the heaters when the switch is in AUTO position and the aircraft is on the ground.

The angle of attack (AOA) probe, on the aircraft right side, is electrically heated. The system is controlled by the PROBES HEAT switch, located on the overhead panel. Whenever No. 2 distribution bus is powered, the aircraft is on the ground and PROBES HEAT switch is in the AUTO position, the AOA probe is heated with low current (preheat). There are two heating elements in the AOA system, one for the probe and one for the case. The case heater is powered directly from the No. 2 distribution bus. A thermal switch will cycle this system in the 95° range.

The SAT/TAS probe is electrically heated in flight from the No. 2 distribution bus. The system is controlled by the PROBES HEAT switch on the overhead panel.

Following takeoff, an air-ground relay is energized by the airborne position of the nose gear ground contact switch, automatically applying power to all probes and static heaters.

INDICATIONS

The heating power lines to all probes and static heaters incorporate current sensors that control a failure relay. If sufficient power is being transmitted to the heaters, the current sensors will hold the failure relays open. Loss of power or reduced flow will illuminate the affected (L/R) PITOT HEAT FAIL, SAT/TAS HEAT FAIL or AOA HEAT FAIL EICAS messages.

The (L/R) PITOT HEAT FAIL EICAS message illuminates if the respective probe heat has failed in flight or unheated on the ground.

An additional current sensor is installed in the AOA system to provide a ground indication of AOA case heater failure by illuminating the AOA PREHEAT FAIL message.

During ground operation with the PROBES HEAT switch in AUTO, the (L/R) PITOT HEAT FAIL, the SAT/TAS HEAT FAIL, and the AOA HEAT FAIL EICAS messages will not illuminate. The AOA PREHEAT FAIL EICAS message extinguishes if power is available to the No. 2 distribution bus.

Interruption of power to the case heater will illuminate the AOA PREHEAT FAIL message. This is separate and distinct from the AOA HEAT FAIL message which indicates a failure of the high power AOA probe heat. The AOA HEAT FAIL message is only active airborne and inhibited on the ground.

Each heating element has a current sensor. If any heating element fails to draw current when the system is operating, the associated current sensing relay closes and an amber L PITOT HEAT FAIL or R PITOT HEAT FAIL message illuminates on the MFD.

If the case heater heating element fails AOA PREHEAT FAIL message illuminates. If AOA probe heat fails in flight, AOA HEAT FAIL message appears. On ground, the message is inhibited.

If the SAT/TAS heat fails in flight, the SAT/TAS HEAT FAIL message appears.

WINDSHIELD AND WINDOWS

The windshield heat control prevents accumulation of ice on both windshields and the side windows (Figure 10-9). It comprises two independent and identical systems, pilot and copilot. Each system includes a windshield temperature sensor and control unit, relays and WINDSHIELD HEAT switches (L and R) located on the overhead panel.

The pilot and copilot windshields and side windows incorporate multiple layer transparencies. Electrical heating elements are embedded between layers of each windshield and side window.

Each windshield is equipped with a temperature sensor (and two spares) which monitor windshield temperature. The temperature sensors supply their output to the temperature control units.

Power for the windshield heating elements and the temperature control units is supplied by the left and right main buses through remote control circuit breakers and cockpit overhead panel circuit breakers from the distribution buses respectively. Remote control circuit breakers are located in the DC control boxes in the aft equipment bay control power to the heating elements.

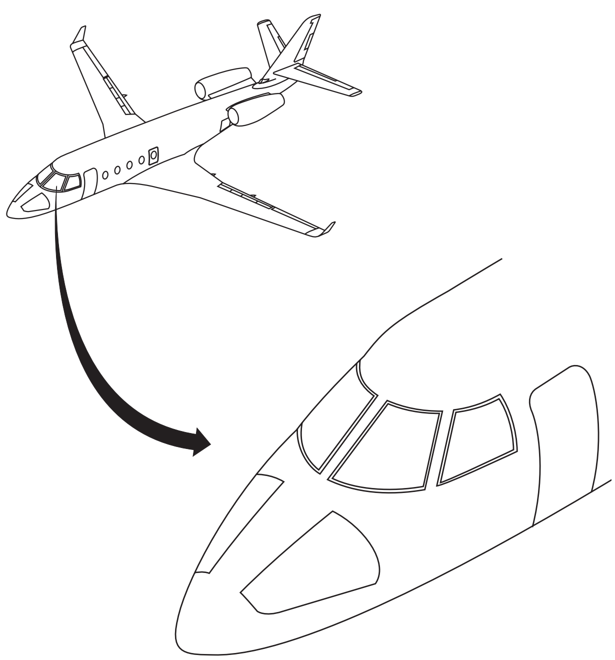


Figure 10-9. Windshields and Windows

The aircraft is not equipped with a windshield wiper or rain removal system; however, windshields are coated with rain repellent to aid in rain removal.

COMPONENTS

Windshield Temperature Control Unit

The windshield temperature control unit monitors the windshield temperature through sensors embedded in the windshield and automatically regulates and maintains constant temperature.

The windshield heat control units receive data on windshield temperature from the temperature sensor and, if the temperature is below the 90°F limit, activates a relay to supply power to the heating element. If the temperature is above the 110°F limit, the temperature control unit activates a relay which removes power to the heating element.

Defog Valve

The defog valve is manually controlled and routes air from the mixing plenum to the glareshield defog and the side window defog.

CONTROLS

The windshield heat system (Figure 10-10) is controlled by a three-position HI–OFF–LOW switch located on the overhead panel for each windshield (L and R). The WINDSHIELD HEAT switch controls three zones for the left and right windshields. In the LOW position, all three zones are powered by a fixed current. In the HI position the central zone will receive higher current.

The W/S HEAT TEST, a lighted push switch in the IND/WARN/SYS TEST section of the overhead panel, is used to test and indicate system failures by displaying the WNDSHLD HEAT FAIL message on the MFD.

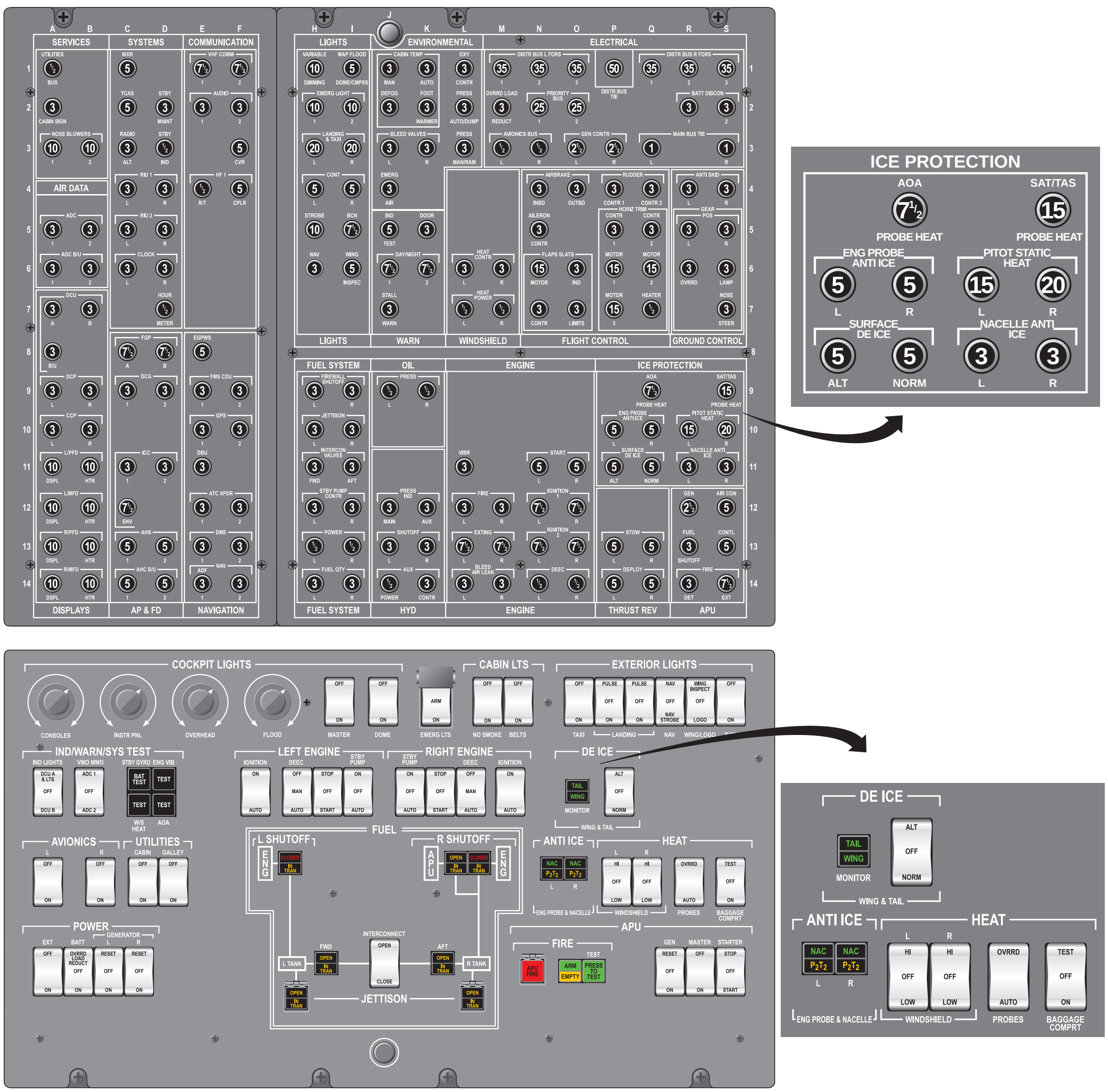


Figure 10-10. Windshield Heating System Controls

System Operation

Positioning the WINDSHIELD HEAT left and/or right switch to HI or LOW powers the temperature control unit. The temperature control unit continuously monitors the windshield temperature and cycles the power on and off as necessary to maintain the required temperature range.

The pilot side windshield is powered from the No. 1 main bus and its temperature controller by the No. 1 distribution bus. The copilot side is powered from the No. 2 main bus and its temperature controller by the No. 2 distribution bus. Each side has a temperature sensor embedded in the windshield, connected to a temperature controller behind the instrument panel, and L and R WINDSHIELD HEAT switch on the overhead panel.

On the ground, when the BATT MASTER switch selected to the ON position and WINDSHIELD HEAT switches are in the LOW or HI position, both side windows are electrically heated.

INDICATIONS

The temperature control unit contains a test circuit that monitors power input and temperature control functions whenever that windshield control switch is selected to the ON position. With power applied to the windshield heat system, pressing the W/S HEAT TEST switch performs a complete system test.

Automatic load shedding through the DC power emergency disconnect relay (EDR) will prevent windshield heat test or operation if both generators are not on line. To test the windshield heat system, select battery switch position OVRRD LOAD REDUCT whenever both generators are not operating and connected to the main buses. Monitor the operating generator for excessive generator load.

If the power supply to the system or the temperature control unit fails, the amber EICAS message (L/R) WNDSHLD HEAT FAIL and (L/R) SIDE WDO HEAT FAIL illuminates on the MFD. The message also illuminates to indicate correct system operation when the W/S HEAT control switch is positioned to HI or LOW and the W/S HEAT TEST pushbutton is depressed.

LIMITATIONS

ANTI-ICE SYSTEM LIMITATIONS

Anti-ice systems must be ON during all ground and flight operations when icing conditions exist or are anticipated.

ON GROUND:
Use engine anti-icing when OAT is 10°C or less and visible moisture in any form is present (such as: clouds, fog with visibility of one mile or less, rain, snow, sleet or ice crystals) or when operating on ramps, taxiways or runways where surface is with snow, ice, standing water or slush that may be ingested by the engines or freeze on engines, nacelles or sensor probes.

IN FLIGHT:
With TAT below 10°C (or SAT below 5°C if TAT is inoperative) and visible moisture in any form is present (such as: clouds, fog with visibility of one mile or less, rain, snow, sleet or ice crystals) that may be ingested by the engines or freeze on engines, nacelles or sensor probes, ice may form on windshield, wing and empennage leading edges and engine nacelles.

Operation of anti-ice is required during all ground and flight operation when icing conditions, as listed above, are imminent or immediately upon detection of ice formation on wings, winglets, near windshield center post and by ice detector light.

Engine/Nacelle Anti-Ice

ENG PROBE & NACELLE ANTI-ICE pushbuttons must be actuated for ice protection. NAC light should be on whenever icing flight conditions exist.

De-Ice System Limitations

Do not operate surface de-ice system when SAT is below –40°C.

EMERGENCY AND ABNORMAL OPERATIONS

The ice and rain messages as they appear on the multifunction display(MFD) are as follows (Figure 10-11):

Caution Messages

AOA HEAT FAIL—Discontinuity in power line

AOA PREHEAT FAIL—Discontinuity in power line to AOA transmitter/transmitter case

DE-ICE HI PRESSURE—Overpressure in boot pressure line

DE-ICE LOW VACUUM—Low vacuum pressure in boot deflation line

NAC ANTI ICE FAIL (L/R)—Engine bleed pressure insufficient for anti-icing or engine/nacelle anti-ice control has failed

PITOT HEAT FAIL (L/R)—Power supply failure

SAT/TAS HEAT FAIL—Power supply failure

SIDE WDO HEAT FAIL (L/R)—Failure in one side window heat system

WNDSHLD HEAT FAIL (L/R)—Failure in one windshield heat system

Figure 10-11. Ice and Rain EICAS Indications

QUESTIONS

1. The air for surface deicing is supplied from the:
A. A manifold interconnecting the LP compressors
B. The mixing chamber in each bleed switching valve
C. The air-conditioning bleed manifold
D. A manifold interconnecting the HP compressors

2. The nacelle anti-icing air is supplied from the:
A. Manifold interconnecting the bleed switching valve mixing chambers
B. LP bleed port of the associated engine
C. Manifold interconnecting the LP compressors
D. HP bleed line of the associated engine

3. Airfoil de-ice system operates on an electric timer to provide:
A. One minute de-ice boot inflate to the leading edge wing and horizontal stabilizer
B. Four second inflation to the leading edge wing, six second inflation to the horizontal stabilizer
C. Six seconds inflation to the leading edge wing, four second inflation to the horizontal stabilizer, 50 seconds off
D. Ten second inflation to the LE wing, 50 second inflation to the horizontal stabilizer

4. Which of the following statements is correct:
A. L/R pitot probes, seven static ports and TAT are electrically heated
B. L/R pitot probes and TAT are electrically heated
C. Seven static ports and L/R pitot probes are electric
D. Three left and three right static ports, L/R pitot probe, SAT/TAS and AOA are electrically heated.

5. Do not operate surface de-ice equipment at ambient temperatures below:
A. –37°C
B. 40°C
C. –40°C
D. –56°C

CHAPTER 11
AIR CONDITIONING

CONTENTS

ILLUSTRATIONS

CHAPTER 11
AIR CONDITIONING

INTRODUCTION

The air conditioning system on the G150 encompasses an air cycle machine, trim air valve, ducting, control air valves, and cockpit controls for the cabin and cockpit air. Additionally, this chapter covers nose compartment cooling.

GENERAL

The air conditioning section covers how the bleed air is conditioned and controlled to provide heating or cooling for the cabin and flight compartment.

Normal and abnormal operations are covered along with EICAS messages associated with the system.

AIR DISTRIBUTION SYSTEM DESCRIPTION

CONDITIONED AIR

Conditioned air is distributed to the cabin and flight compartment via a single supply duct (Figure 11-1)

The duct is routed from the mixing plenum, above the main landing gear wheel wells, directly into the aft pressure bulkhead. The conditioned air enters the cabin at floor level, circulates through the cabin to the return ducts. The return ducts run along the top of the cabin, above the panels on each side.

The flight compartment conditioned air branches to two separate paths. Each branch is equipped with an electrically controlled valve. One branch is for windshield defog. The other is for the foot warmers and side panels.

The amount of air going to the windshield defog and the foot warmers is controlled by the DEFOG and PILOT AIR COND switches on the ECS control panel (Figure 11-2).

The switches are momentary-type switches that are spring loaded to the center off position. The switches are labeled INCRS, and DCRS.

The defog and foot warmer valves are motor driven valves that remain where they are if power is lost.

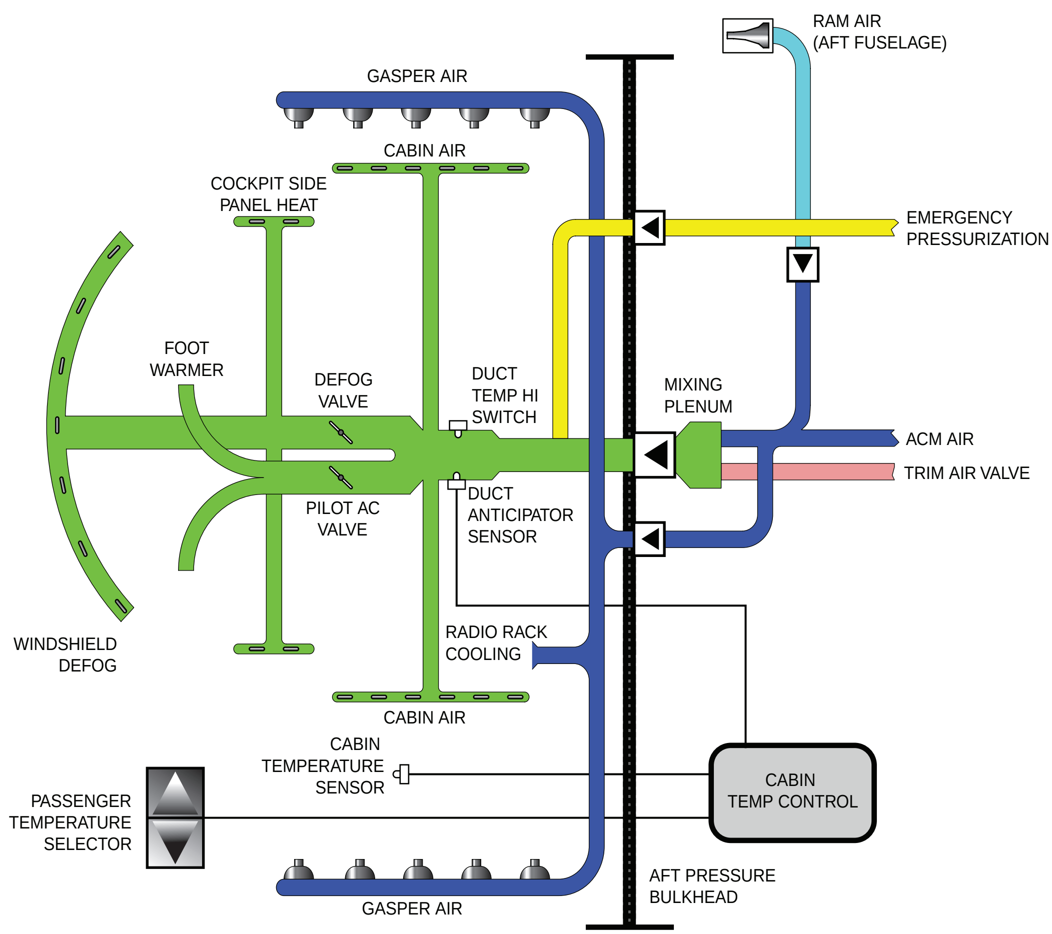


Figure 11-1. Air Distribution System—Conditioned Air

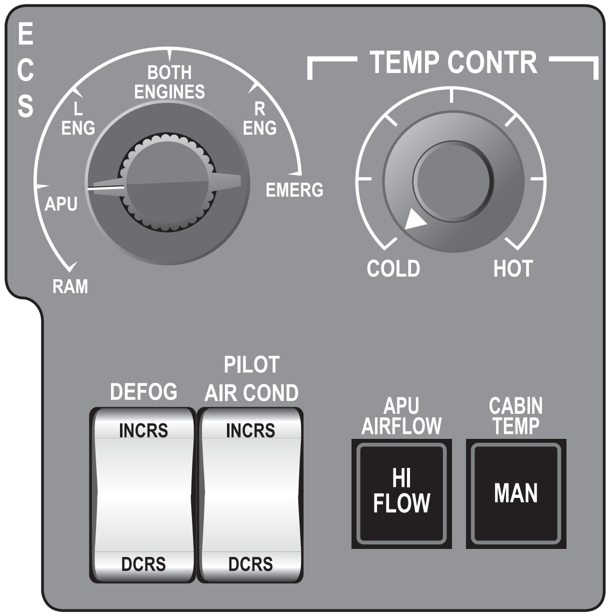


Figure 11-2. ECS Control Panel

The defog valve receives power from the No. 1 distribution bus.

The pilots conditioned air valve receives power from the No. 2 distribution bus.

GASPER AIR

Cold air from the air-cycle machine branches off before the mixing plenum, and enters the aft pressure bulkhead through a separate duct. Gasper air is cold air only.

The gasper duct in the cabin runs up the inside of the aft pressure bulkhead then branches off to each side.

Gasper outlets are provided in both the cabin and flight compartment.

EMERGENCY AIR

The emergency air has a separate duct that branches from the low pressure duct of the right engine only. The emergency air duct bypasses all the normal ducts and enters the aft pressure bulkhead through a separate inlet.

While in the EMERG position, the only way to control cabin temperature is by varying the right engine rpm.

CHECK VALVES

Each of the air ducts entering the aft pressure bulkhead pass through a check valve. These check valves prevent a rapid decompression of the cabin if any of the air ducts in the unpressurized area of the fuselage rupture or become disconnected.

BLEED AIR ROUTING

The bleed air from the engines or APU enters the common duct. From there it goes through the precooler or bypasses the precooler depending on the bleed air temperature going to the refrigeration unit.

All of the bleed air from the precooler and the turbine bypass is routed through the ozone converter.

Bleed air then branches off to either the refrigeration unit (air cycle machine) or the trim air valve.

The trim-air valve serves as a hot air control valve. When fully open, it provides maximum heating of the cabin. If the valve is closed, all the air is forced through the refrigeration unit and converted to cold air. This condition results in maximum cooling of the cabin.

With the trim-air valve partially open hot and cold air enter the mixing chamber and provide conditioned air for temperature control of the cabin and flight compartment.

OZONE CONVERTER

The ozone converter is essentially a catalytic ozone converter that removes ozone from the air entering the cabin and flight compartment.

It is located to the right and slightly forward of the refrigeration unit.

This unit breaks down the ozone in the air supply from the engines or APU.

BLEED-AIR PRECOOLER

The precooler lowers the temperature of the bleed air from either the engines or APU. The precooler is an air-to-air heat exchanger mounted aft of the refrigeration unit.

The ram exiting the refrigeration unit is used to cool the bleed air before it enters the refrigeration unit.

The precooler bypass valve is a pneumatically operated valve controlled by a temperature sensor (Figure 11-3).

When the temperature switch senses that the bleed air entering the refrigeration unit is 300°F (148°C) or higher, it commands the bypass valve to close which forces the air to pass through the precooler.

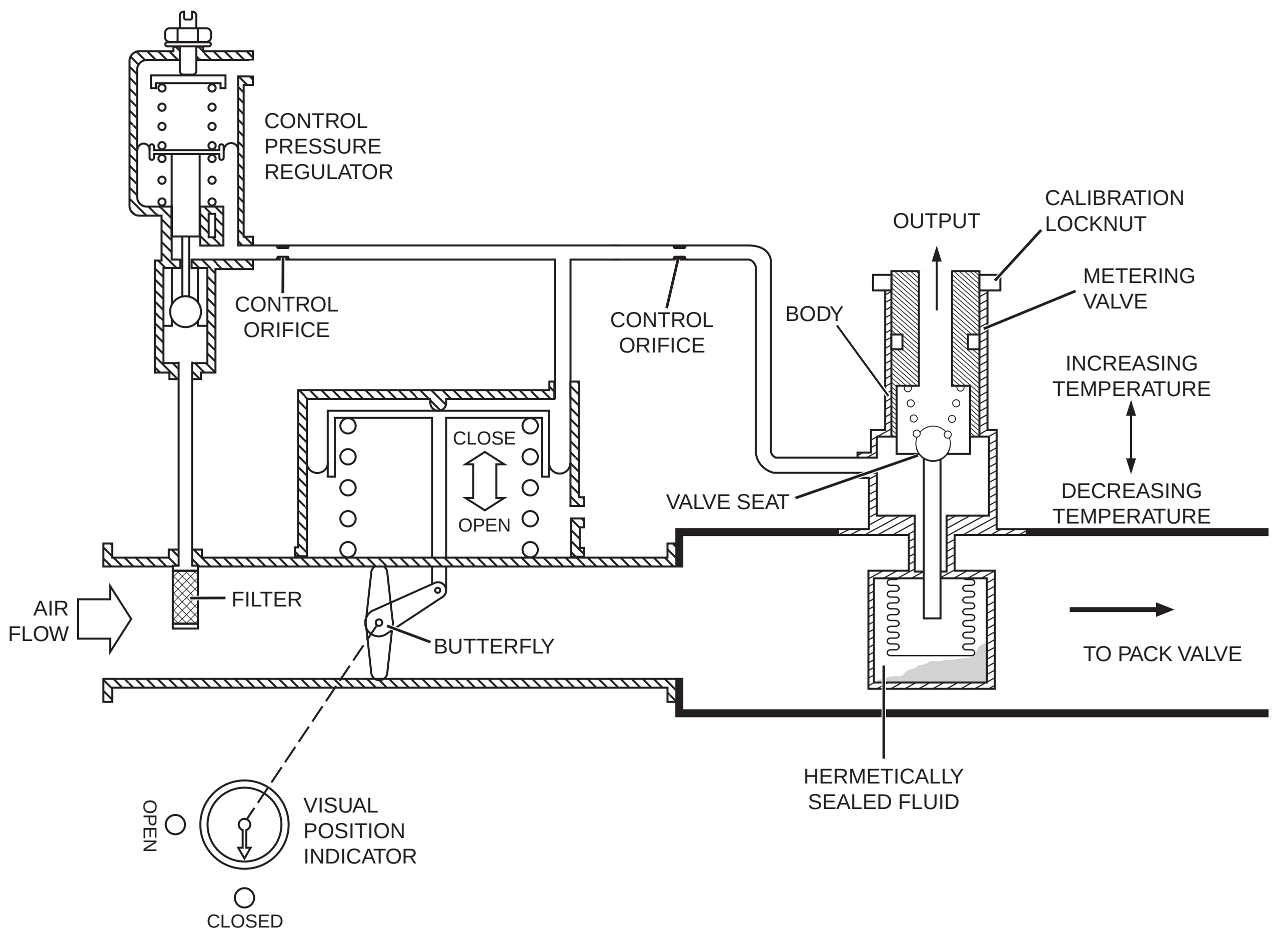


Figure 11-3. Precooler Bypass Valve

Temperatures cooler than 300°F, cause the bypass valve to open, allowing the air to bypass the cooler.

The precooler bypass valve does not maintain a particular temperature; it simply opens or closes as the bleed-air temperature fluctuates above or below 300°F.

REFRIGERATION UNIT

AIR CYCLE MACHINE

The three-wheel, air-bearing air-cycle machine consists of a compressor wheel, turbine wheel, and cooling fan mounted on a common shaft. The shaft is supported by foil-type air bearings that operate without scheduled maintenance. There are two journal bearings, one between the compressor and the turbine and the other between the compressor and the cooling fan. One double-acting thrust bearing is in the turbine-end cavity to accommodate axial loads. Both journal and axial bearings are self-acting. A small amount of air is supplied to both bearing cavities from the turbine inlet scroll through a reverse pilot pickup that minimizes dirt buildup.

Cooling of the bleed air is achieved by passing the air through heat exchangers, the air cycle machine compressor, and finally through the turbine section. The combination of compression and expansion of the bleed air by the air cycle machine compressor and turbine, along with the cooling effect of the heat exchangers, results in production of a cold air supply to the cabin environmental system.

The air cycle machine is considered a separate component of the refrigeration unit and may be replaced as such.

Figure 11-4 depicts various components of the refrigeration unit.

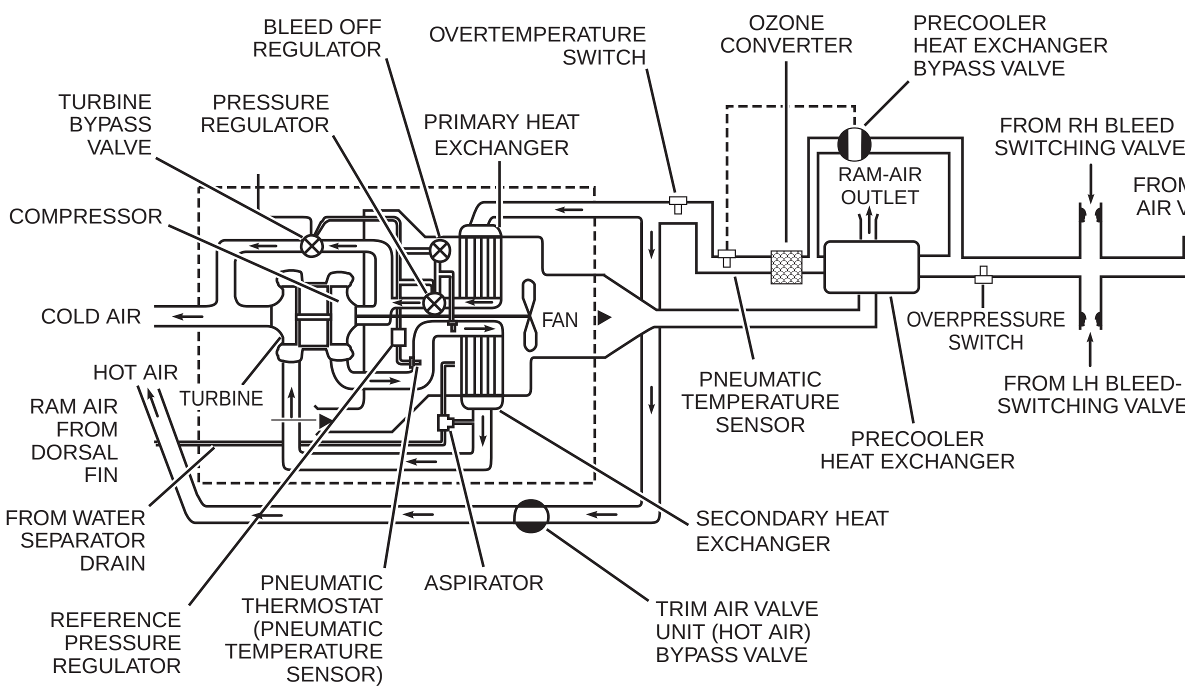


Figure 11-4. Refrigeration Unit

ANTI-ICING

Anti-icing of the air cycle machine turbine and water separator is accomplished with the normally closed turbine bypass valve (Figure 11-5) and a 35°F (1.7°C) temperature sensor.

The 35°F sensor is immediately downstream of the water separator. When the temperature of the air leaving the water separator drops to 35°F, the sensor sends a signal to the cabin temperature control unit, which modulates the turbine bypass valve open.

The turbine bypass valve is a pneumatically operated electrically controlled valve, located in a branch duct from the compressor discharge duct to the turbine outlet.

When the temperature of the air leaving the water separator rises above 35°F, the temperature controller will position the torque motor to close off the vent line of the turbine bypass valve.

This will increase the pressure on top of the diaphragm and close the turbine bypass valve.

The bypass valve is a fully modulated valve and will open only enough to mix the warm air from the compressor with the turbine output to keep the air from getting colder than 35°F.

WATER SEPARATOR

The water separator consists of a condenser unit, a separator shell, and a duct assembly. The condenser unit consists of a bypass valve and a conical fiberglass cloth condenser.

A drain fitting and strainer are in the duct assembly. The drain tube is electrically heated to prevent water from freezing in the tube. Power for the drain tube heating element is 28 VDC from the No. 1 distribution bus.

An aspirator, using a small amount of air from the compressor section, is used to draw the water from the water separator and spray it on the secondary heat exchanger.

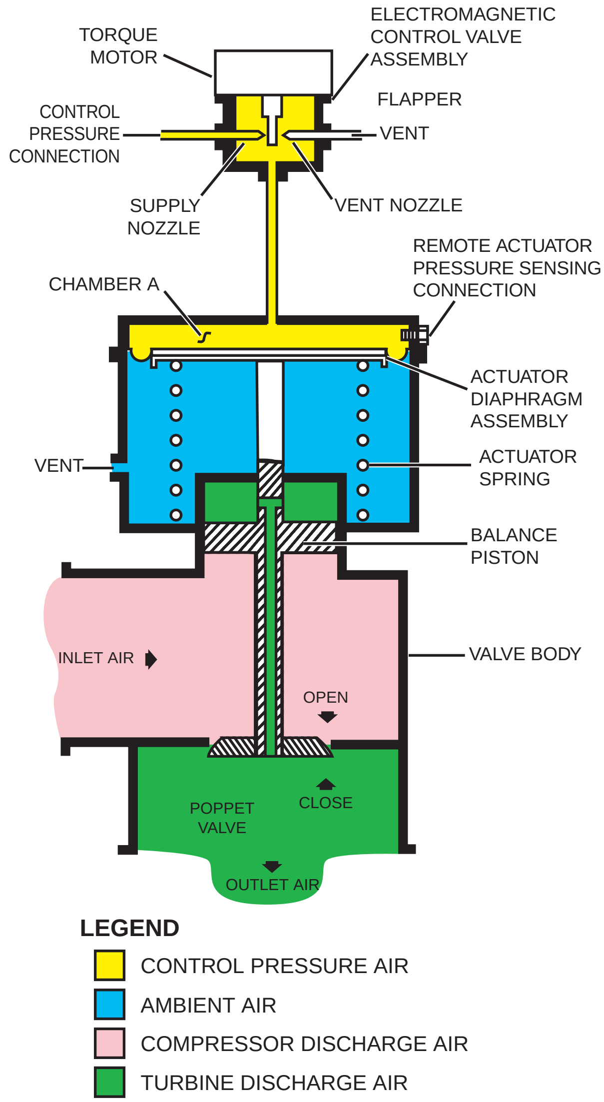


Figure 11-5. Turbine Bypass Valve

This water is used as an additional heat sink to help cool the bleed air going through the secondary heat exchanger.

TRIM-AIR VALVE

The trim-air valve allows bleed air to bypass the air cycle machine and enter the cabin ducts as hot air.

The trim-air valve is in a branch duct of the air entering the air cycle machine. This is a fully modulating valve that either diverts some of the bleed air from air cycle machine into the cabin or forces the air through the air cycle machine.

The hot bleed air from the trim air valve enters the mixing chamber where it is mixed with the cold air from the air cycle machine to provide the conditioned air for the cabin and flight compartment.

The position of the trim-air valve is either automatically controlled by the temperature controller or directly controlled by a crewmember when operating in the manual temperature control mode.

Operation

The trim-air valve is a pneumatically operated valve positioned by a built-in torque motor (Figure 11-6).

In the automatic temperature mode, the temperature controller operates the torque motor to manipulate the trim air valve. In the manual temperature mode, a crewmember directly positions the trim-air valve through a separate circuit of the temperature controller.

To prevent a malfunctioning trim-air valve from overheating the air ducts, a pneumatic duct temperature limiter is in the conditioned air duct. The duct temperature limiter is in the duct after the mixing plenum.

If the temperature of the duct reaches or exceeds approximately 200°F (93°C), it opens, dumping some of the working pressure from the trim-air valve.

This loss of working pressure causes the trim-air valve to modulate closed, preventing the duct from getting too hot.

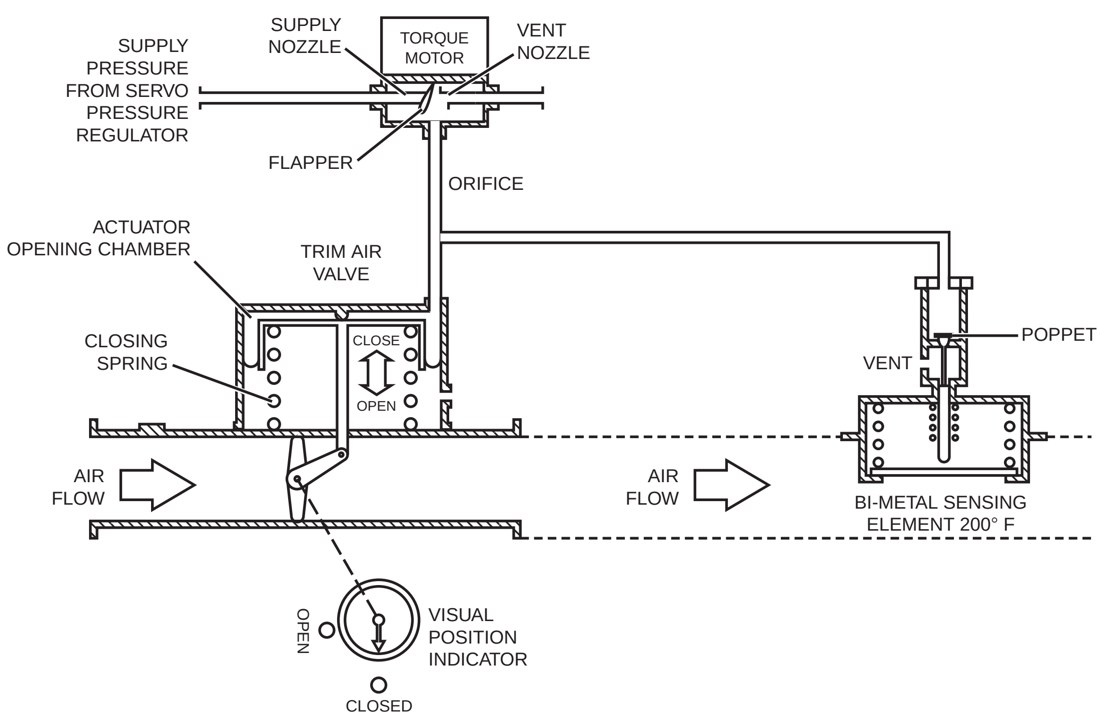


Figure 11-6. Trim-Air Valve

TEMPERATURE CONTROL SYSTEM

DESCRIPTION

The temperature controller provides a single temperature control system for both the cabin and flight compartment.

It is located on the left inside of the aft pressure bulkhead.

The cabin and flight compartment temperature is normally preset by positioning the TEMP CONTR knob, which consists of two potentiometers on a common shaft.

One potentiometer is used in the automatic temperature mode, and the other is used when operating in the manual temperature control mode.

The TEMP CONTR knob is on the ECS control panel at the top right side of the pedestal.

COMPONENTS

The temperature control system consists of the following components:

- Temperature controller
- 35°F cold-air low-limit sensor
- Turbine bypass valve
- Cabin temperature sensor
- Cabin temperature sensor fan
- Duct temperature sensor
- Trim air (hot air) valve
- TEMP CONTR temperature selector
- TEMP SELECT temperature selector

NOSE COMPARTMENT COOLING DESCRIPTION

The nose compartment has electronic equipment requiring ventilation on the ground and during flight.

In flight, the cooling is provided ram-air inlets on each side of the nose section of the aircraft.

On the ground, the nose compartment cooling consists of two cooling fans and two thermal switches. One switch is for cooling and one for warning. Circuit breakers on the overhead panel and relays in the forward relay panel provide electrical power and control of the fans.

A thermal switch controls both fans and begins operation when the nose compartment temperature exceeds 113°F (45°C) regardless of whether the aircraft is on the ground or in flight.

If the nose compartment temperature reaches 140°F (60°C), the second thermal switch sends a signal to the DCU, which illuminates the amber NOSE TEMP HI message on the EICAS.

The left and right cooling fans and relays receive electrical power from the left and right avionics and accessory busses respectively.

The circuit breakers are in the SERVICES section in the top left section of the overhead panel. The circuit breakers are labeled NOSE BLOWERS 1 and 2. These are 10-amp circuit breakers.

LIMITATIONS

The limitations contained in Section One of the *Airplane Flight Manual (AFM)* must be complied with regardless of the type of operation.

EMERGENCY AND ABNORMAL OPERATIONS

AIR CONDITIONING SYSTEM EICAS MESSAGES

The environmental system messages Warning, Caution, Advisory, and Status are displayed on the EICAS section of upper right portion of the primary page of the MFD (Figure 11-7). The messages associated with the air conditioning system are as follows:

WARNING Message

DUCT TEMP HI—Indicates the temperature in the ducts under the cabin floor is too high.

CAUTION Messages

CAB AUTO TEMP FAIL—The automatic cabin temperature system has malfunctioned.

NOSE TEMP HI—The temperature in the nose compartment has reached or exceeded 140°F (60°C).

STATUS Message

ECS BLEED OFF—The ECS selector switch is either in the RAM position, or the APU is selected and the APU is not running.

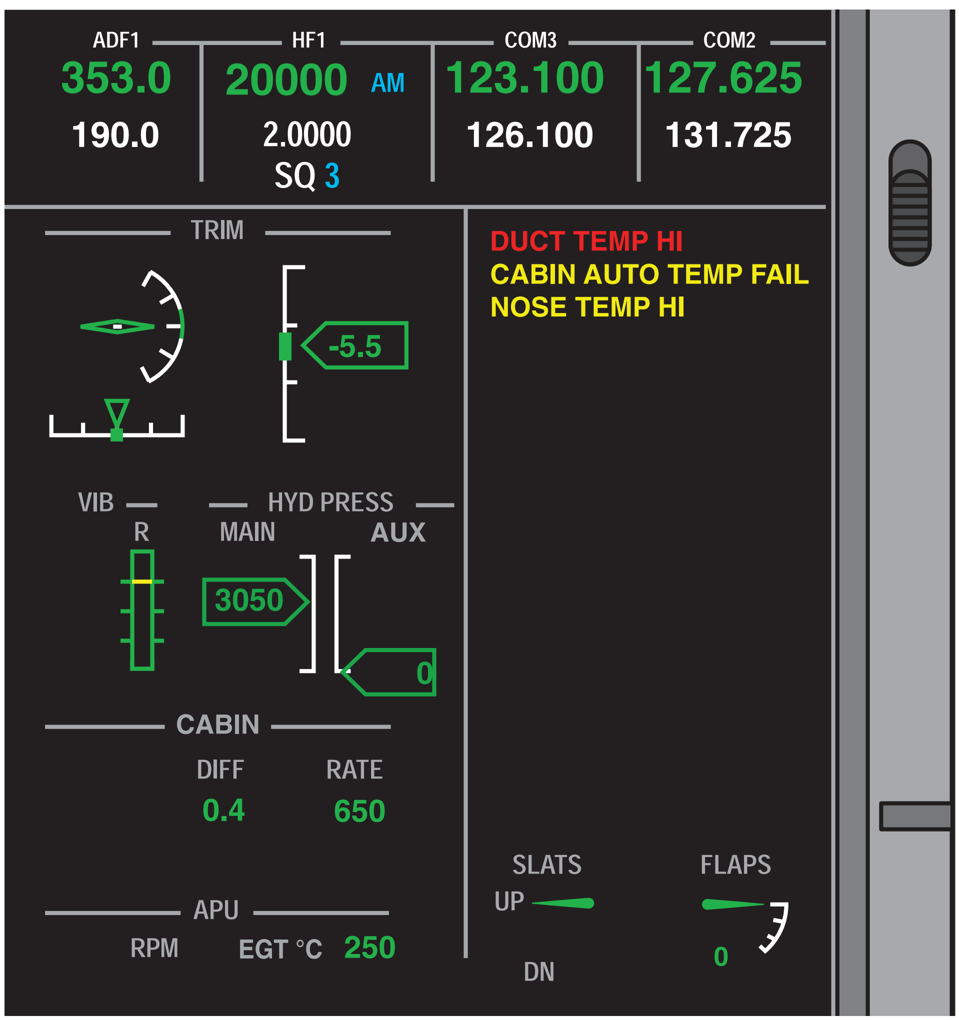


Figure 11-7. Air Conditioning System EICAS Messages

QUESTIONS

1. The temperature is controlled in emergency operation by:
   A. Manipulating right engine power
   B. Recirculating cabin air
   C. Ram air flow through the primary heat exchanger
   D. Manipulating left engine power

2. Automatic cabin temperature control is overridden when:
   A. Cabin pressurization is in MAN mode
   B. CABIN TEMP MAN is selected
   C. Either BLEED LEAK warning is displayed
   D. BLEED PRESS/TEMP HI warning is displayed

3. In order for the automatic temperature control system to work you must:
   A. Select the ground cooling button on
   B. Select APU, L ENG, BOTH ENGINES, OR R ENG on CABIN AIR selector
   C. Ensure that the CABIN TEMP MAN pushbutton is deselected
   D. Both B and C

4. The environmental control system (ECS) provides conditioned air for:
   A. Ventilation
   B. Cabin pressure
   C. Temperature control
   D. All of the above

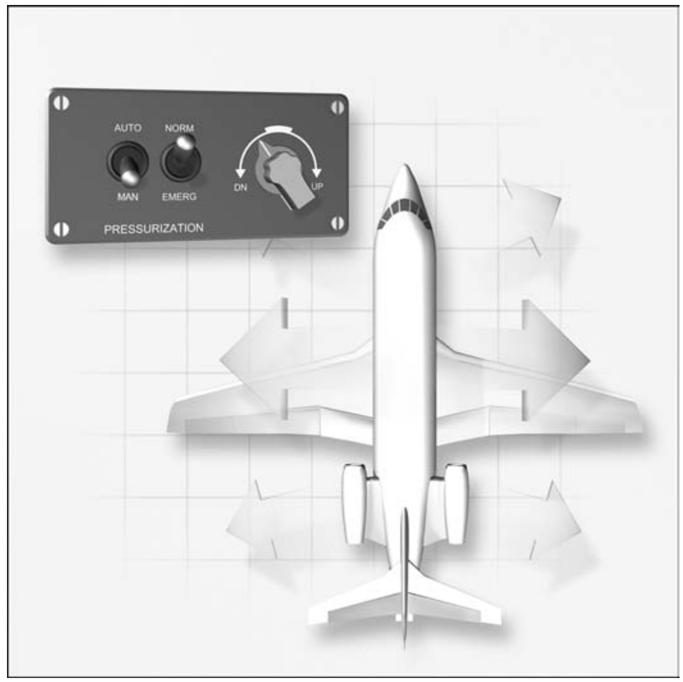
5. For ECS operation, bleed air is supplied from:
   A. LP bleed air from one or both engines
   B. LP bleed air augmented by HP bleed air from one or both engines
   C. HP bleed air from one or both engines
   D. All of the above

CHAPTER 12
PRESSURIZATION

CONTENTS

ILLUSTRATIONS

CHAPTER 12
PRESSURIZATION



INTRODUCTION

This chapter describes the pressurization system of the G150 series aircraft. The description includes the air sources, control, and distribution.

GENERAL

The pressurization system describes how the conditioned air is used to pressurize the aircraft.

PRESSURIZATION SYSTEM DESCRIPTION

The cabin pressure control system (CPCS) regulates conditioned air outflow from the cabin at the rates required to maintain cabin altitude pressure according to preset schedules. Figure 12-1 shows the component locations of the pressurization control system.

The pressurized space includes the flight compartment and passenger cabin. The system primarily operates in the automatic mode but may operate manually as the situation requires.

COMPONENTS

Digital Cabin Pressure Controller

The digital cabin pressure controller positions and monitors the operation of an electrical outflow valve (OFV) that controls the outflow of air from the cabin. The cabin pressure controller is below the left side panel. Access to this unit is through the left side console panel.

Outflow Valve

A dual DC electrical motor-operated outflow valve can be positioned anywhere between full open to full closed. The outflow valve operates in either the automatic or manual mode and is designed so that no one malfunction disables both modes.

The speed of the OFV from full open to full closed (90°) in automatic is approximately 12 seconds, while in the manual mode the speed from full open to full closed (90°) is approximately 55 ±1 seconds.

The outflow valve is on the lower right skin of the flight compartment, below the copilot seat.

Pneumatic Safety Valve

A pneumatic safety valve on the inside of the forward pressure bulkhead forward of the copilot rudder pedals releases cabin pressure to prevent the cabin from overpressurizing. The safety valve starts to open when the cabin pressure differential equals 8.95 psid. If the cabin pressure reaches 9.5 psid, the safety outflow valve is full open and allows no further buildup of cabin pressure.

CONTROLS AND INDICATIONS

A cabin pressure control panel is in the pedestal just above the flap/slat selector. (Figure 12-2).

The landing field elevation knob (FIELD ELEV) knob enables landing field altitude selection in the automatic (AUTO) mode. It has a range from –1,000 feet (–304.8 m) to +14,000 feet (4267.2 m). Rotating the FIELD ELEV knob clockwise increases the landing field altitude. Rotating the FIELD ELEV knob counterclockwise decreases the landing field altitude. Landing altitude is in 50 feet (15.2 m) detent increments.

The MODE SEL pushbutton enables selection of manual mode in both flight and ground operations. It illuminates the word MAN when selected to the manual mode.

The switchlight also indicates system faults during system built-in-test (BIT) functions. It illuminates the word FAULT if the system detects a system fault. When the system goes to manual, the outflow valve remains where it was until the manual CABIN ALT selector switch repositions it.

The CABIN ALT switch is spring-loaded to center and increases or decreases cabin altitude during manual mode operations.

The DUMP switchlight opens the outflow valve and dumps cabin pressure in AUTO and MAN modes to 13,500 feet ±1,500 feet. It illuminates the word DUMP when selected. This is a guarded switch.

The DITCH switchlight fully closes the outflow valve in AUTO or MAN modes. This switch closes the outflow valve prior to ditching the aircraft. It illuminates the word DITCH

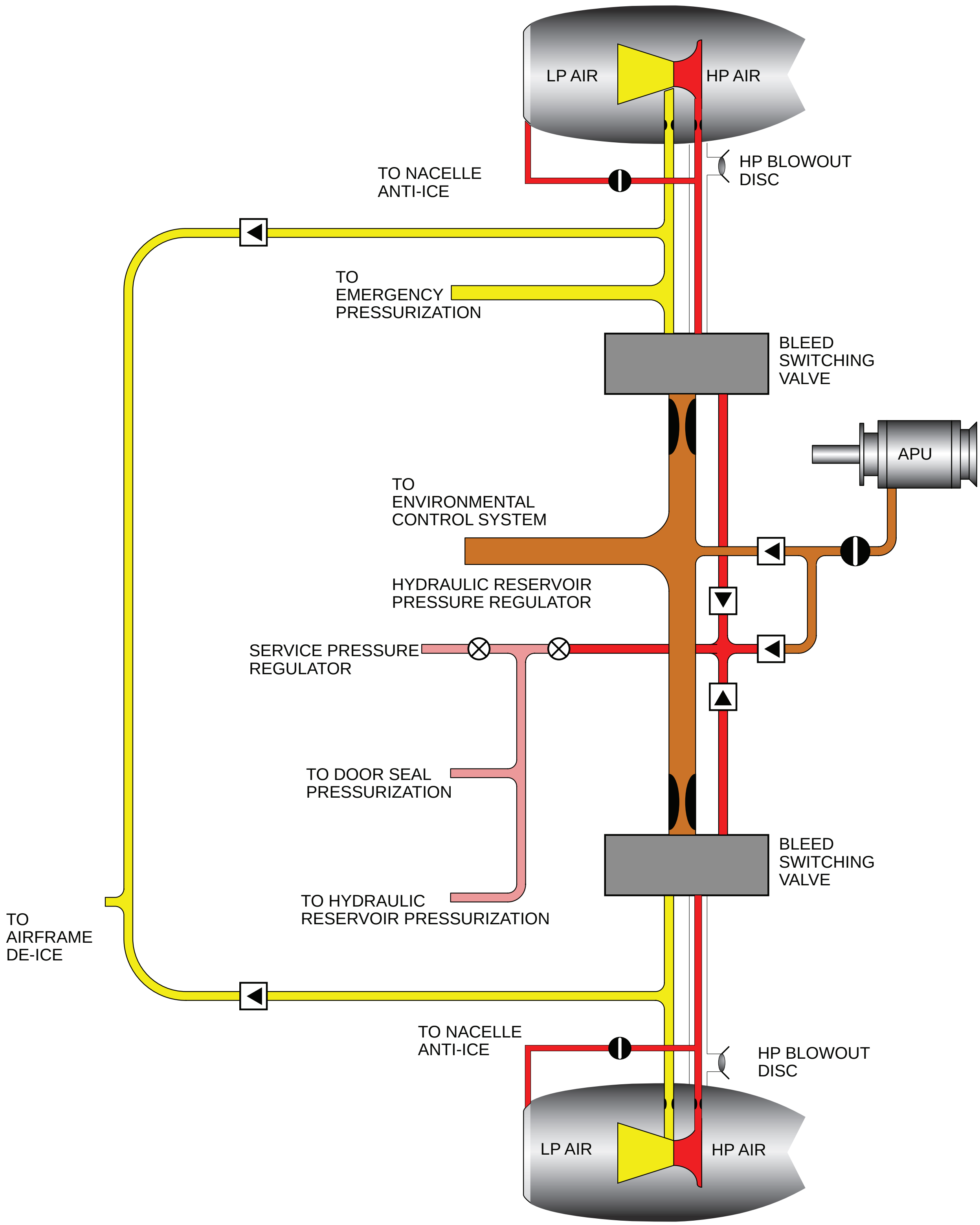


Figure 12-1. Pressurization Schematic

when selected. It is also a guarded switch.

OPERATION

Cabin Pressure Control System

Automatic Mode

The inputs and outputs of the cabin pressure controller are shown in Figure 12-2. Switches or selectors in the CPCS system supply discrete inputs and outputs.

The controller has three redundant channels and one internal aneroid: the automatic channel, the monitor channel and the manual channel. The three channels independently generate the following modes of operation. The channels convert the data and transmit it to the EICAS.

Ground Mode

When operating in the ground mode, the outflow valve is full open, which limits the maximum cabin differential. If the takeoff altitude is higher than sea level, the differential pressure may increase proportionally. The discrete signals to tell the cabin pressure controller to go to the ground mode comes from the right main landing gear WOW switch, left throttle switch at less than max cruise, air speed less than 100 kts, and aircraft altitude at or below 15,000 ft.

Takeoff Mode

When the left throttle is advanced for takeoff while operating on the ground, the system enters the takeoff (prepressure) mode. This mode eliminates any discernible pressure bumps during takeoff rotation by allowing the outflow valve to control cabin pressure prior to lift off. (Figure 12-3).

During the takeoff mode, the cabin descends at a maximum rate of 1,000 fpm to 250 ±50 feet below the takeoff field altitude. The cabin altitude remains in the prepressurized mode until the throttles are retarded, the aircraft becomes airborne, or the airspeed of the aircraft exceeds 100 knots.

Takeoff Abort

If the throttles are retarded prior to liftoff, the controller aborts the prepressurization process. In this condition, the cabin becomes unpressurized at the rate of 500 fpm for a maximum of 60 seconds. If after the 60 seconds the cabin is not fully depressurized, it continues to depressurize at 200 fpm.

Climb Mode

During normal takeoff conditions, the CPCS automatically switches from the takeoff mode to the climb mode. This occurs when the CPCS receives either an airborne signal from the right WOW switch or aircraft airspeed reaches or exceeds 100 kts.

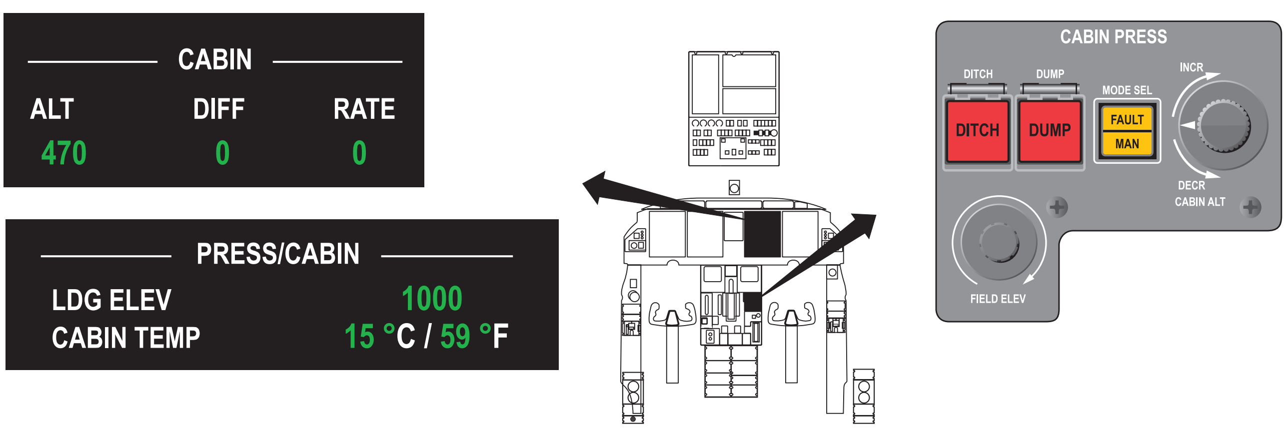


Figure 12-2. Pressurization System Component Location

The initial transition into the climb mode depends on the takeoff field elevation and the selected landing field elevation. Based on these conditions, the cabin either climbs or descends until it reaches the built-in climb schedule. After the cabin reaches the built-in climb schedule, it pressurizes the cabin as a function of aircraft altitude. With the aircraft at the maximum approved altitude of 45,000 feet, the maximum cabin altitude in either the climb or cruise mode is less than 8,000 feet.

Cruise Mode

When the aircraft altitude stabilizes and does not change more than ±100 feet for one minute, the CPCS automatically switches to the cruise mode, also known as the cabin clamp. The aircraft altitude must change more than ±200 feet from the altitude at which the cruise mode was entered before the cabin altitude is allowed to change. This prevents cabin pressure oscillations due to small changes in aircraft altitude and ensures a more comfortable flight.

Descent Mode

When the aircraft altitude descends 500 feet or more from maximum aircraft altitude that was achieved during flight, the CPCS enters the descent mode. Once the CPCS enters the descent mode it remains in that mode unless the aircraft climbs above the maximum aircraft altitude at which the cruise mode was started. At that point the CPCS returns to the climb mode.

In the descent mode the cabin altitude normally descends at the maximum rate of 300 fpm. When the aircraft descends to 25,000 feet of aircraft altitude the CPCS drives the cabin altitude to the selected landing field elevation. If the cabin altitude is already below the selected landing field altitude, it brings the cabin up to the selected landing field altitude at a rate of 500 fpm. If the cabin altitude is above the selected landing field altitude, it continues to descend the cabin altitude to the selected land-ing field altitude at a rate of 300 fpm.

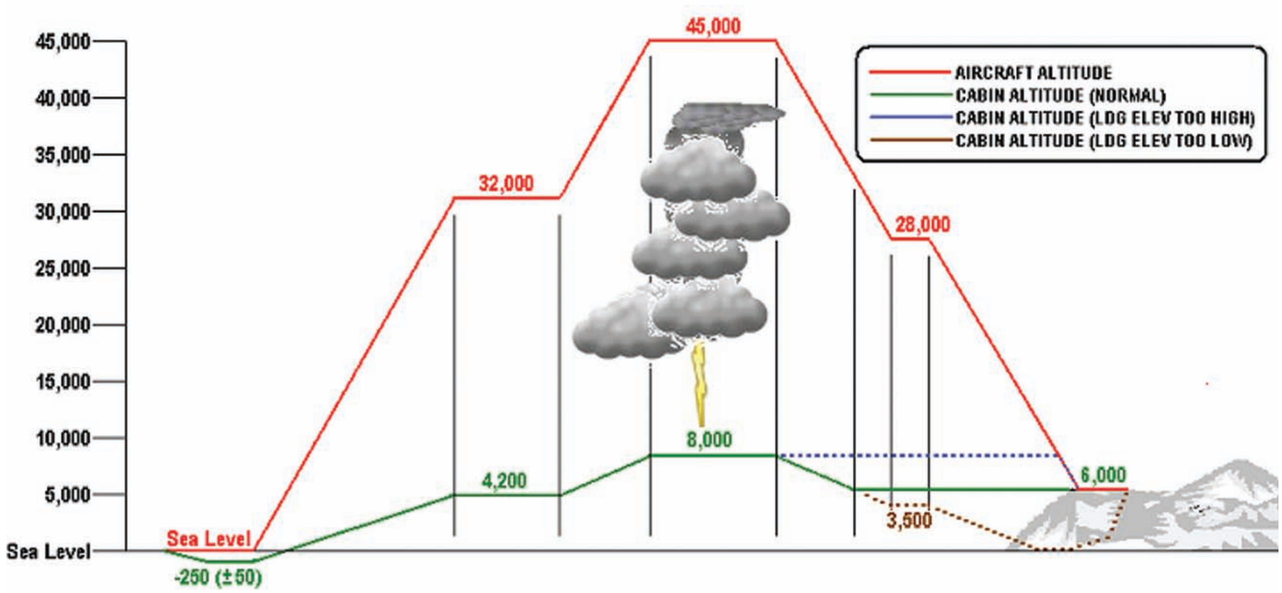


Figure 12-3. Cabin Altitude Profile

Landing Mode

Under normal conditions the CPCS enters the landing mode from the descent mode when the aircraft airspeed drops to 90 kts, the left throttle is reduced to below max cruise, or the right WOW switch detects weight-on-wheels. The outflow valve drives full open, putting the system in the ground mode.

If, on landing, the cabin altitude is different than the field elevation, the CPCS brings the cabin altitude to the field elevation at a controlled rate for 60 seconds. The maximum rate of cabin altitude decrease is 300 fpm, and the maximum rate of cabin altitude increase is 500 fpm. After the 60-second controlled rate, the CPCS goes into the ground mode where the outflow valve is driven full open. This ensures the cabin fully depressurizes within 60 seconds of landing.

The landing mode may also be entered from the climb mode where the 60 second of rate control applies the same as when entered from the descent mode.

Touch and Go

If at any time after touchdown but before ground mode initiation, the throttles are advanced to takeoff, the CPCS enters the takeoff (prepressure) mode as described above. If the aircraft becomes airborne during landing mode but before the throttles are advanced to takeoff, the CPCS enters the descent mode.

CPCS Output Functions

The CPCS also controls discrete functions such as cabin altitude warnings and signals to the oxygen system that activate the oxygen mask drop function. The CPCS allows a normal cabin differential pressure of 8.79 psid. This provides a cabin altitude of 7,960 feet at FL 450.

Positive differential pressure relief is a function of both the outflow valve and safety pressure-relief valve, resulting in redundant overpressurization protection. The maximum overpressurization electrically provided by the CPCS is from 8.806 to 8.892 psid. The maximum overpressurization provided pneumatically by the safety outflow valve is from 8.95 to 9.5 psid.

The CPCS provides the following outputs:

- Cabin altitude, cabin differential, and cabin rate of climb—These are displayed on page one of the EICAS The display is yellow dashes when there is invalid data. (Figure 12-4).
- ALT—Numbers under the ALT legend indicate the current cabin altitude. The yellow box M indicator after the numbers indicates the pressurization system is operating in the manual mode.
- Cabin altitude of less than –2,100 feet or greater than ±8,200 feet, the altitude indication is yellow. From –2,100 feet to +8,200 feet, the indication is green.

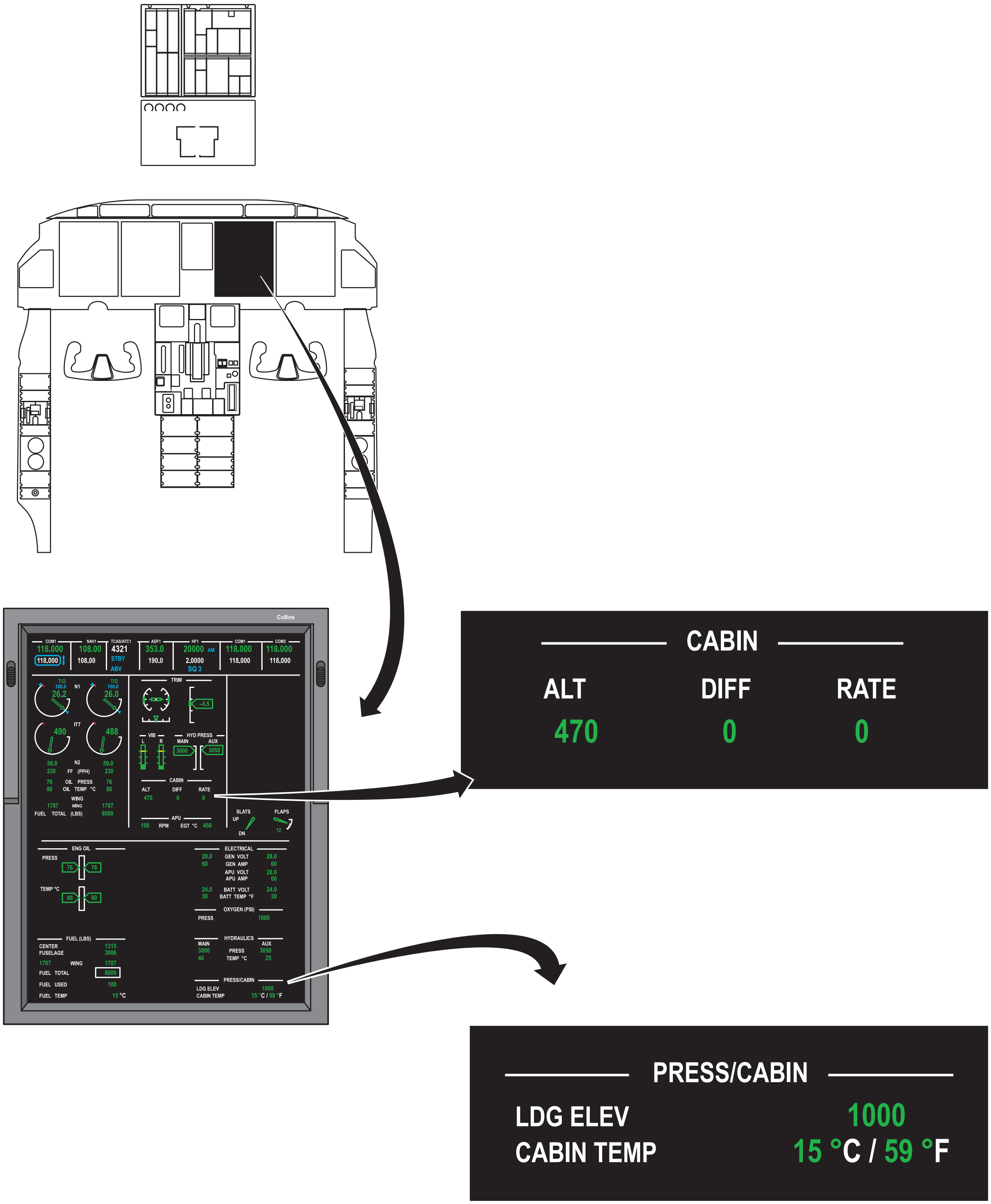


Figure 12-4. EICAS Display

- DIFF—The value shown under the DIFF represents the current cabin-to-ambient pressure differential.
- A cabin differential from –0.5 to 8.9 psid is green. From 8.9 to 9.0 psid, the indication is amber. An indication of less than –0.5 or 9.0 psid or greater is red.
- RATE—The current cabin altitude rate-of-change displays under the RATE legend. An arrow to the left of the value points up or down to indicate increasing or decreasing cabin altitude respectively. All cabin rates, positive or negative, display in green. Cabin rate is in the range of from 0 to 9950 fpm in 50 feet increments.
- Landing field elevation selection has a range from –1,000 to +14,000 feet. It is displayed in green under the PRESS/CABIN legend in the lower right corner of the EICAS secondary page.
- CABIN TEMP—The cabin temperature is displayed under the PRESS/CABIN legend below the LDG ELEV area in the lower right corner of the secondary page of the EICAS. The temperature is displayed °C next to the C/ and °F next to the F. The cabin temperature is displayed in green.
- The CPCS provides a cabin altitude limiting control that prevents the cabin altitude from exceeding 15,000 feet in the event of any reasonably probable malfunction in the pressurization system or when the dump switch activates. Electronically closing the outflow valve accomplishes this. If for any reason there is not enough air entering the cabin to maintain that altitude, the cabin altitude would climb based on the leak rate of the cabin.
- The CPCS provides a discreet output from both channels that automatically activates the passenger oxygen mask dropout at 13,500 ±500 feet. If the takeoff or landing field selection is greater than 8,000 feet and the aircraft is below 25,000 feet, the dropout occurs at 14,750 feet.
- Selection of the DITCH pushbutton signals the CPCS to electrically drive the outflow valve to the full closed posi-tion and illuminate the word DITCH in the switch.
- Selection of the DUMP pushbutton signals the CPCS to electrically drive the outflow valve full open and illuminate the word DUMP in the switch. This condition causes the cabin to depressurize up to the cabin altitude limit of 13,500 feet ±1,500 feet.
- If the CPCS receives signals from either of the two switches on the cabin door indicating that the door is not closed, it prevents the cabin from pressurizing if the aircraft is below 15,000 feet. If the aircraft altitude is above 15,000 feet, the CPCS ignores the door switches.

Manual Pressurization Mode

Manual mode is available for use during either flight or ground operations. To select manual mode press down on the MODE SEL switch light on the CABIN PRESS control panel. This illuminates MAN in the switch (Figure 12-5).

Selecting MAN stops the outflow valve in the last commanded position and allows the CABIN ALT switch, also on the CABIN PRESS control panel, to position the outflow valve. Cabin altitude rate of change depends on the changes in aircraft altitude and variations in cabin-air inflow if no adjustment is made to the outflow valve position.

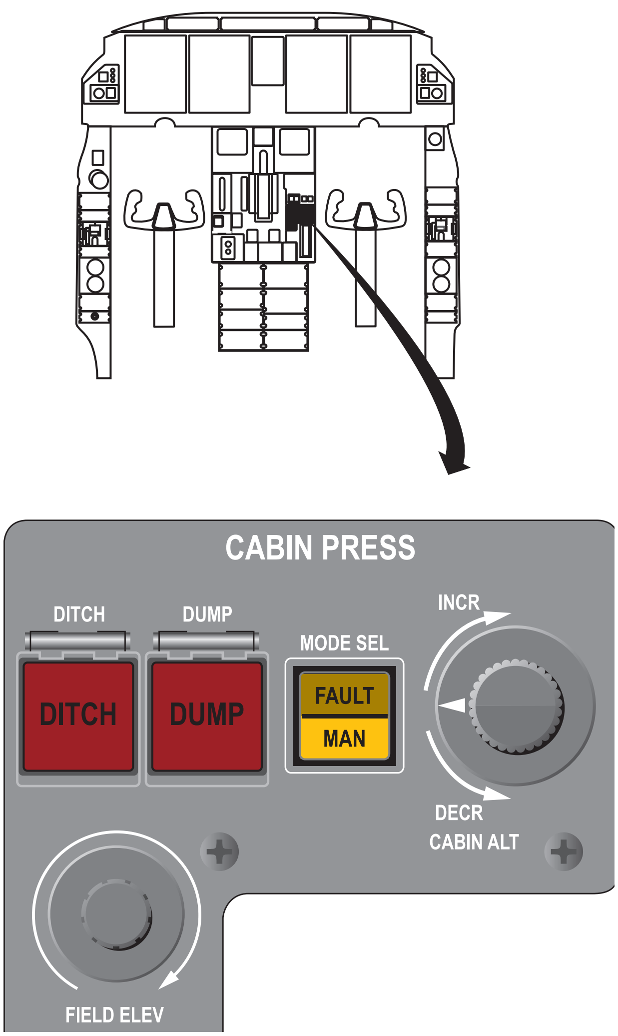


Figure 12-5. Cabin Pressure Control

Positioning the CABIN ALT switch to either INCR or DCRS uses the manual motor to position the outflow valve through a separate circuit in the pressurization controller. The switch is spring-loaded either way back to the center position. In the center position there is no command signal sent to the outflow valve; therefore, it remains in the last commanded position.

INCR drives the outflow valve open, causing the cabin altitude to climb. Selecting DCRS drives the outflow valve closed, causing the cabin altitude to descend.

As the CABIN ALT switch moves further from center, the speed of the manual motor increases the speed to open/close the outflow valve. The maximum speed of the outflow valve from full open to full closed is approximately 55 seconds.

As the CABIN ALT switch operates the outflow valve, it causes the MODE SEL switch light to pulse. Two pulses of the mode select pushbutton are equal to approximately 200 fpm cabin rate of change.

The altitude limiting circuit of the CPCS also operates in the manual mode. Therefore, the maximum cabin altitude while operating in the manual mode is 15,000 feet, assuming there is enough airflow entering the cabin to hold this altitude. The maximum cabin pressure differential in the manual mode is limited to 9.0 psid by the pneumatically operated safety outflow valve.

LIMITATIONS

CABIN PRESSURIZATION

Maximum cabin differential pressure is 8.9 psi.

Maximum cabin differential pressure for takeoff and landing is –0.2 psi.

EMERGENCY AND ABNORMAL OPERATIONS

PRESSURIZATION SYSTEM EICAS MESSAGES

The pressurization system EICAS messages are displayed on the upper right portion of the MFD (Figure 12-6).

WARNING Messages

CAB ALT 15000 —Cabin pressure controller malfunction and the cabin altitude is above 15,000 feet.

CAB ALT HI —Cabin altitude is above 10,000 feet.

CAB DOOR UNLOCKED—Main entrance cabin door is not properly locked, or the cabin door microswitch has failed.

CAUTION Messages

CAB AUTO PRESS FAIL—The automatic cabin pressurization system has malfunctioned.

DOOR SEAL PRESS LOW—Indicates that the main entrance cabin door seal is not inflated.

ADVISORY Messages

CAB PRESS TEST OK—The cabin pressurization system has successfully completed the verify mode test.

STATUS Messages

CAB PRESS IN TEST—The cabin pressure control is in the verify mode test mode.

CABIN PRESS MONITOR—The monitoring section of the cabin pressure controller has malfunctioned.

ECS BLEED OFF—The ESC selector switch is either in the RAM position, or the APU is selected and the APU is not running.

CAB DOOR UNLOCKED—Main entrance cabin door is not properly locked, or the cabin door microswitch has failed.

Figure 12-6. Pressurization System EICAS Indications

QUESTIONS

1. The air source used for pressurization may be from the:
   A. LP compressors
   B. LP and HP compressors
   C. HP compressors
   D. Any of the above sources

2. The normal differential pressure is controlled by the:
   A. Altitude limiters on each outflow valve
   B. Isobaric valve on the secondary outflow valve
   C. Automatic controller logic
   D. Differential limiters on each outflow valve.

3. The ground unpressurized mode of operation is determined by:
   A. Selecting an altitude below field elevation
   B. Selecting an altitude above field elevation
   C. A dump solenoid opened by the ground contact switch
   D. Ground contact switch and power lever position

4. In normal operations the automatic pressurization system is tested by:
   A. Momentarily turning the airflow off
   B. Initial application of electrical power to the system
   C. Increasing or decreasing the selected altitude
   D. A constantly run self test with a built in test

5. If the preselected altitude is higher than the destination field, the:
   A. The normal and safety valves open prior to landing
   B. Aircraft will land pressurized
   C. Maximum differential will be exceeded
   D. The aircraft will land unpressurized

CHAPTER 13
HYDRAULIC POWER SYSTEM

CONTENTS

ILLUSTRATIONS

CHAPTER 13
HYDRAULIC POWER SYSTEM

INTRODUCTION

This chapter describes general information about the G150 hydraulic system.

GENERAL

The G150 has two hydraulic systems (Figure 13-1), referred to as the main and auxiliary systems. Hydraulic power from the two systems is used for actuation of the landing gear retraction and extension system, nose wheel steering, wheel brakes, airbrakes, thrust reversers and aileron servoactuators that assist in aileron control.

MAIN HYDRAULIC SYSTEM

The main hydraulic system is used to provide hydraulic pressure for the following:

- Landing gear retraction and extension
- Wheel brakes
- Nose wheel steering
- Flight and ground airbrakes
- Aileron servoactuators

If the main hydraulic system fails, compressed nitrogen is used to provide the necessary pressure to extend the landing gear.

The main hydraulic system is pressurized by two engine-driven hydraulic pumps. Either of the pumps can supply adequate pressure and volume to operate all of the accessories that are operated by the system.

AUXILIARY HYDRAULIC SYSTEM

The auxiliary hydraulic system provides hydraulic pressure for the following items:

- Thrust reversers
- Aileron servoactuators
- Parking brake
- Emergency Brake system

A single hydraulic pump that is powered by an electric motor pressurizes the auxiliary system.

Control circuit logic regulates the operation of the pump. Operating criteria for the pump is determined by the position of a three-position switch in the cockpit and the availability of electrical power. Normally, the pump will operate whenever the landing gear is not up and locked. This provides continuous auxiliary system pressure during takeoff and landing. However, there are other events that will cause the pump to operate, discussed later in the text.

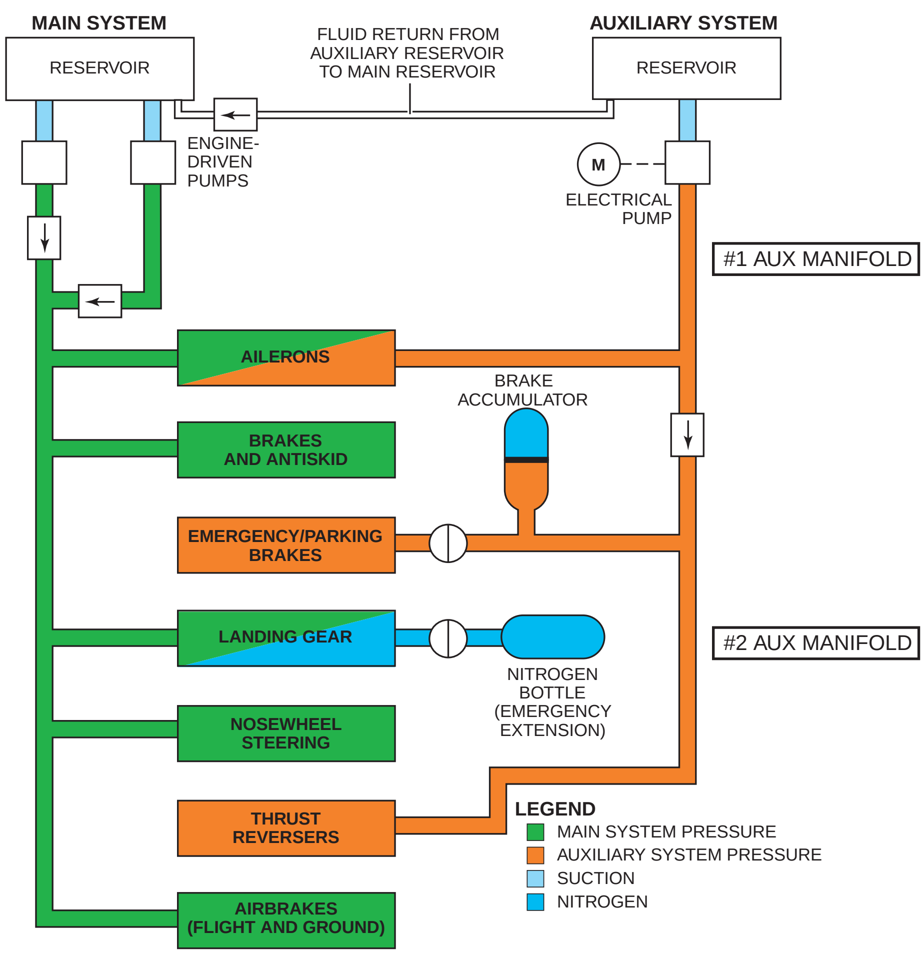


Figure 13-1. Hydraulic Pressure Distribution

The auxiliary system is equipped with an accumulator. The accumulator is provided to store a small amount of hydraulic fluid under operating pressure so that it may be used in case the auxiliary system hydraulic pump fails. This will allow the emergency brake system to operate and will keep the parking brake applied for some time after the aircraft is shut down.

HYDRAULIC FLUID

The type of hydraulic fluid that is used to service the hydraulic systems is critical. The preferred fluid is SKYDROL LDIV.

COMPONENTS

MAIN HYDRAULIC SYSTEM

The main hydraulic system components (Figure 13-2) except for the engine driven pumps, are installed on the left side of the service compartment and consist of the following:

- Two engine-driven hydraulic pumps (EDPs)
- Hydraulic reservoir assembly
- Two hydraulic fire shutoff valves
- Two filters in the high pressure lines
- One filter in the return line
- Hydraulic manifold assembly
- Fluid temperature sensor

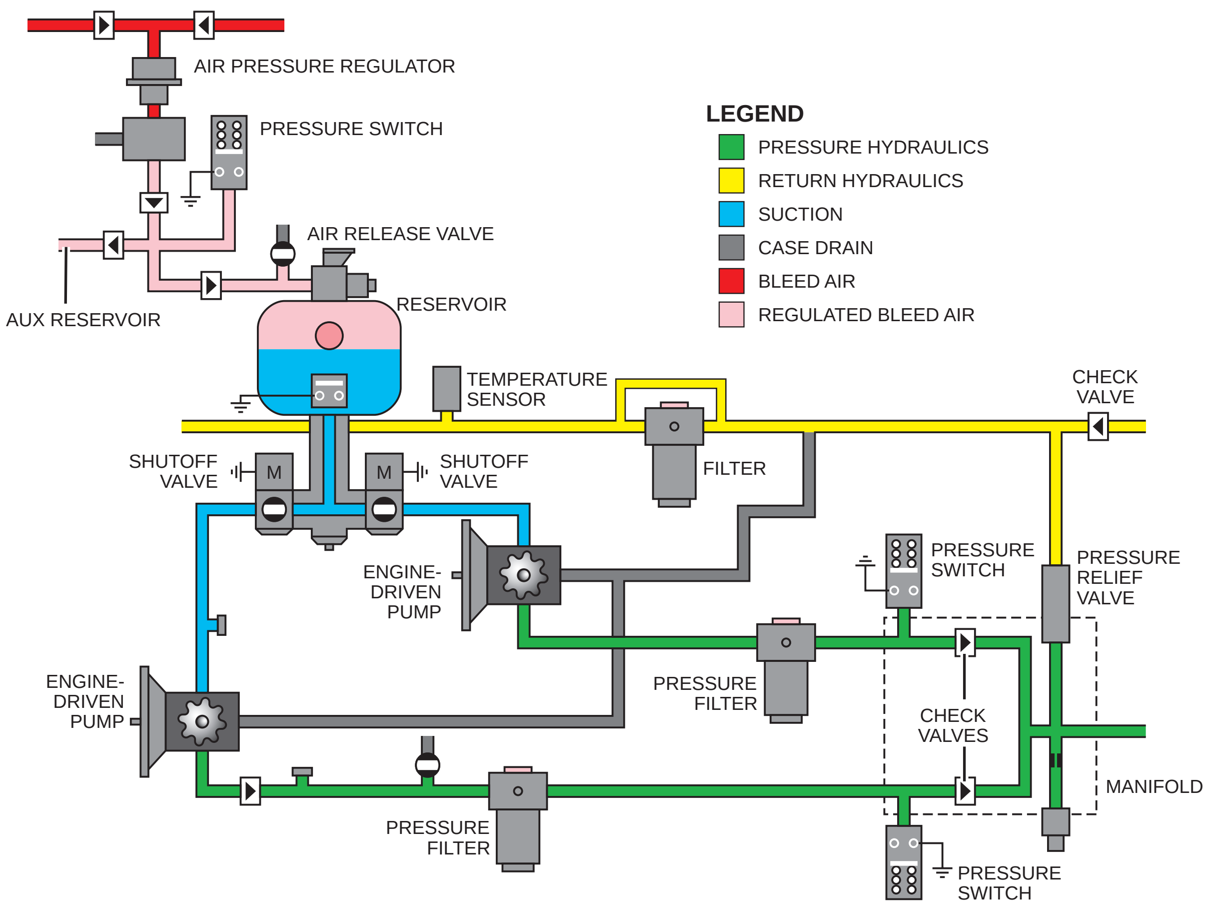


Figure 13-2. Main Hydraulic System

Engine-Driven Hydraulic Pumps (EDPs)

A single hydraulic pump(Figure 13-3) is driven by the accessory gearbox of each engine. The pumps are variable displacement, constant-pressure type pumps. The pumps utilize a series of pistons and cams, rather than intermeshed gears to produce hydraulic pressure. They internally regulate their operating pressure, nominally producing 3,000 psi. As system demand increases, the volume of fluid pumped will increase and the pressure of the fluid will decrease slightly. Maximum pump output is approximately 6.8 gpm.

The pumps operate whenever the high-pressure (N_2) spool of the associated engine is operating. A small amount of fluid is continually passed through each pump, to facilitate cooling and lubrication.

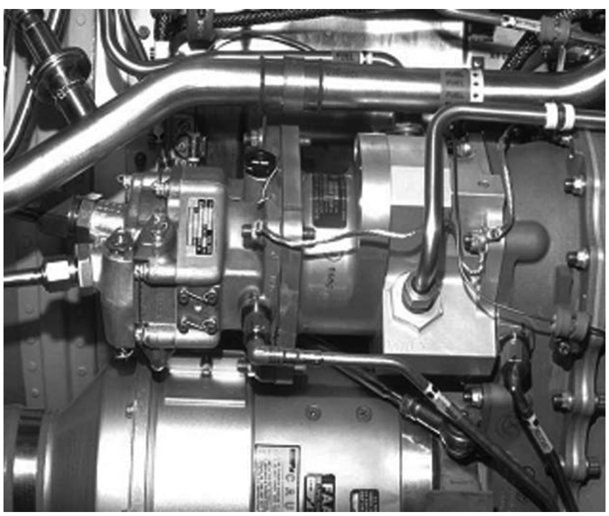


Figure 13-3. Engine Driven Hydraulic Pumps

Main Hydraulic System Reservoir

The main hydraulic system utilizes a reservoir (Figure 13-4) located on the left side of the fuselage in the service compartment. The service compartment is just aft of the baggage compartment. The reservoir holds approximately one-half gallon of hydraulic fluid. The reservoir is designed so that a momentary negative-g loading of the aircraft will not cause a loss of fluid supply to the hydraulic pumps.

The reservoir features a non-vented, locking filling cap (Figure 13-4). This is designed to prevent fluid leakage when air pressure is applied to the reservoir, as described later in the text. A glass sight gauge that is visible from outside the aircraft is located on the side of the reservoir to enable the flight crew to determine hydraulic fluid quantity prior to flight. The proper fluid level is indicated by the surface of the fluid at the midpoint of the sight gauge. Other features include a low-level switch, an over-pressure relief valve and a magnetic drain plug. The plug is magnetized so that certain metal particles will attach to it and their sources can be identified by maintenance. For servicing, the reservoir is also equipped with an air pressure release valve.

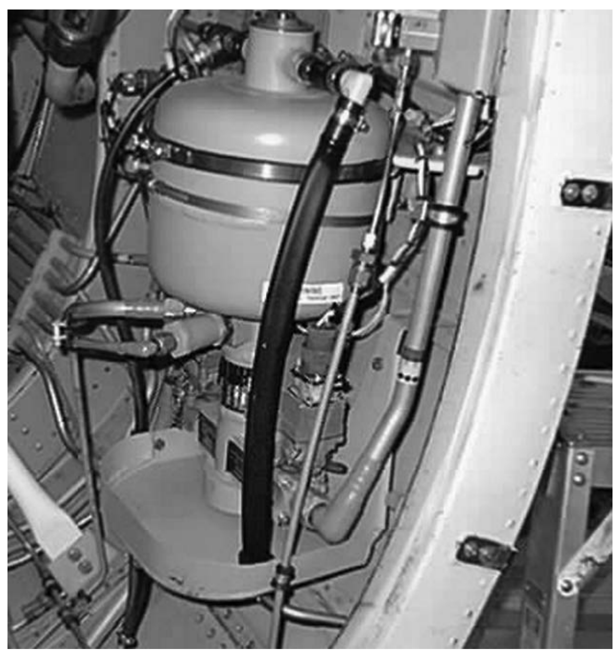


Figure 13-4. Main Hydraulic System Reservoir

Fire (Hydraulic) Shutoff Valves

Two fire (hydraulic) shutoff valves (Figure 13-5) are located at the base of the reservoir, one for each engine-driven pump. These are used to stop hydraulic fluid from fueling an engine fire and are controlled by the FIRE/OVERHT switchlights (Figure 13-5) in the cockpit. The left switch controls the valve for the left engine, the right switch controls the valve for the right engine.

Each valve features a reversible electric motor that positions the valve either OPEN or CLOSED.

There are no intermediate positions on these valves and the motors are controlled by internal limit switches that stop operation when the valves reach the desired position. It takes approximately one second for a valve to open or close. Power is supplied from the respective No. 1 or No. 2 distribution bus. Generator power is not required to operate the valves.

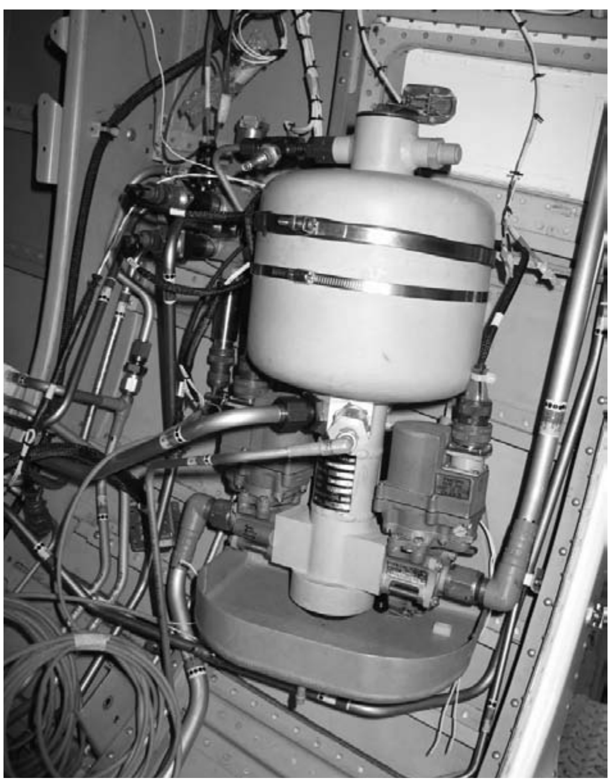


Figure 13-5. Fire (Hydraulic) Shutoff Valves

High-Pressure Filters

A high-pressure filter (Figure 13-6) is located in the hydraulic line that comes from each EDP. These filters are installed to prevent any debris from a failed pump to contaminate the remainder of the hydraulic system. To perform this function properly, the filters must pose a complete barrier to foreign material that exceeds a certain size. Therefore, there is no provision for fluid to bypass these filters in case the filter element becomes blocked.

A differential pressure (pop-out) indicator is located at the top of each of these filters to indicate if an excessive pressure difference has occurred across the filter element. The indicator will extend from the top of the filter housing if the element becomes contaminated and a certain threshold of pressure is exceeded.

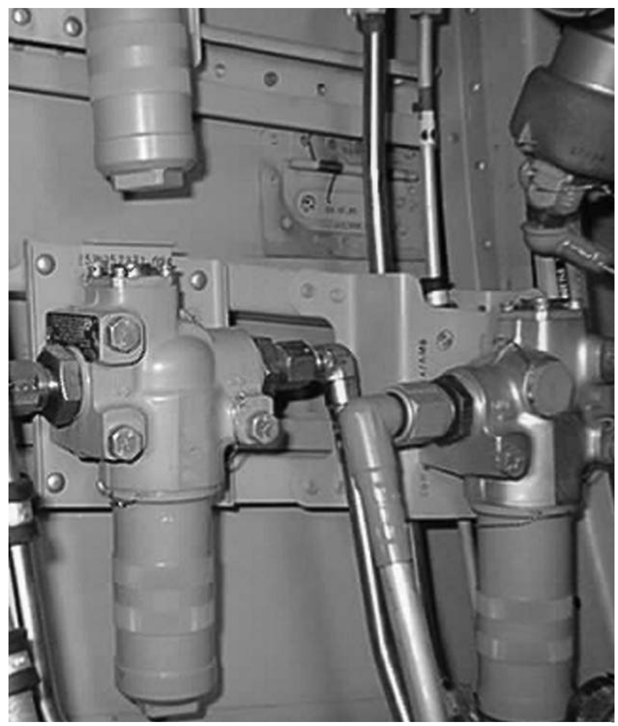


Figure 13-6. High-Pressure Filters

Low-Pressure Filter

The main system is also equipped with a hydraulic filter to remove contaminants from the hydraulic fluid after it has been utilized in the aircraft accessories. It is located in the common low-pressure return line to the hydraulic fluid reservoir. This filter is also equipped with a differential pressure indicator that will extend if filter contamination occurs.

The low-pressure filter has the capability of bypassing fluid around the filter element in case the element becomes completely blocked. This is to allow the main hydraulic system to continue operation until a safe landing can be conducted. If the filter could not bypass, the hydraulic system would fail very quickly after a return filter blockage because of fluid starvation of the hydraulic pumps. However, if the bypass mechanism is being used, any contaminants in the fluid will be introduced to the hydraulic reservoir and then be able to reach the remainder of the hydraulic system.

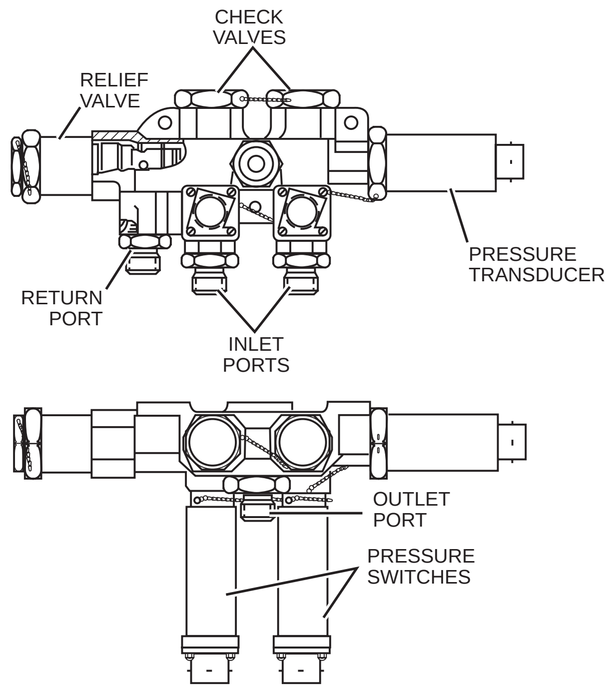


Figure 13-7. Hydraulic Manifold Components

Hydraulic Manifold

The main hydraulic system is equipped with a single hydraulic manifold (Figure 13-7) to distribute hydraulic pressure to its associated accessories. The manifold is used to connect the various components of the hydraulic system to each other. It is located in the service compartment on the left side of the fuselage, just aft of the main reservoir.

The manifold contains two check valves, two pressure switches, a pressure transmitter and a pressure relief valve. The check valves allow fluid to pass in only one direction, each associated with one of the engine-driven pumps. If a pump or engine should fail, the associated check valve will prevent fluid from backflowing through the pump. This will enable the remaining pump to maintain pressure in the hydraulic system. Each of the hydraulic pumps is associated with one of the pressure switches, as well. If a pump fails or a pressure filter becomes blocked, an EICAS caution message (L/R) HYD PUMP PRESS LOW will be presented. If both of these switches are activated simultaneously, system logic will cause the auxiliary hydraulic system pump to operate.

The pressure transmitter is used to signal system pressure to the primary and secondary EICAS pages. Normal indications are presented in green. Abnormal pressure values are presented in amber. If the pressure within the system becomes excessive, the MAIN HYD PRESS HI caution message will appear on the MFD.

The pressure relief valve is installed to prevent a failure of a hydraulic pump pressure regulator from causing an over-pressurization of the main hydraulic system. Its function is automatic. If the pressure limit is exceeded, the valve will open. It will re-close when the pressure returns to normal.

Fluid Temperature Sensor

The temperature of the hydraulic fluid in the system must be maintained within limits. A temperature sensor located in the low-pressure return hydraulic line to the reservoir. The temperature of the hydraulic fluid is continuously monitored and is available on the secondary EICAS page. If the temperature reaches 85°C or higher, a MAIN HYD TEMP HI caution message will be displayed.

AUXILIARY HYDRAULIC SYSTEM

The auxiliary hydraulic system (Figure 13-8) components are located primarily on the right side of the aft fuselage and include the following:

- Electrically powered hydraulic pump
- Hydraulic fluid reservoir
- Hydraulic accumulator
- Two manifolds
- High-pressure filter
- Low-pressure filter
- Fluid temperature sensor
- Control switch

The auxiliary hydraulic system does not share hydraulic fluid with the main system during normal operations. They are separate systems.

Hydraulic Pump

The auxiliary system is pressurized by an electrically powered hydraulic pump (Figure 13-9). The motor that drives the pump operates on 28 VDC and has an integral cooling fan. The pump operates continuously under certain conditions and can be powered by engine or APU generators or by battery power. Pump operating time with battery power only should be limited because of the high electrical draw of the pump. The pump is a variable-displacement type pump with a nominal operating pressure of approximately 3000 psi. and operationally similar to the main hydraulic system pumps. Volume of fluid pumped is

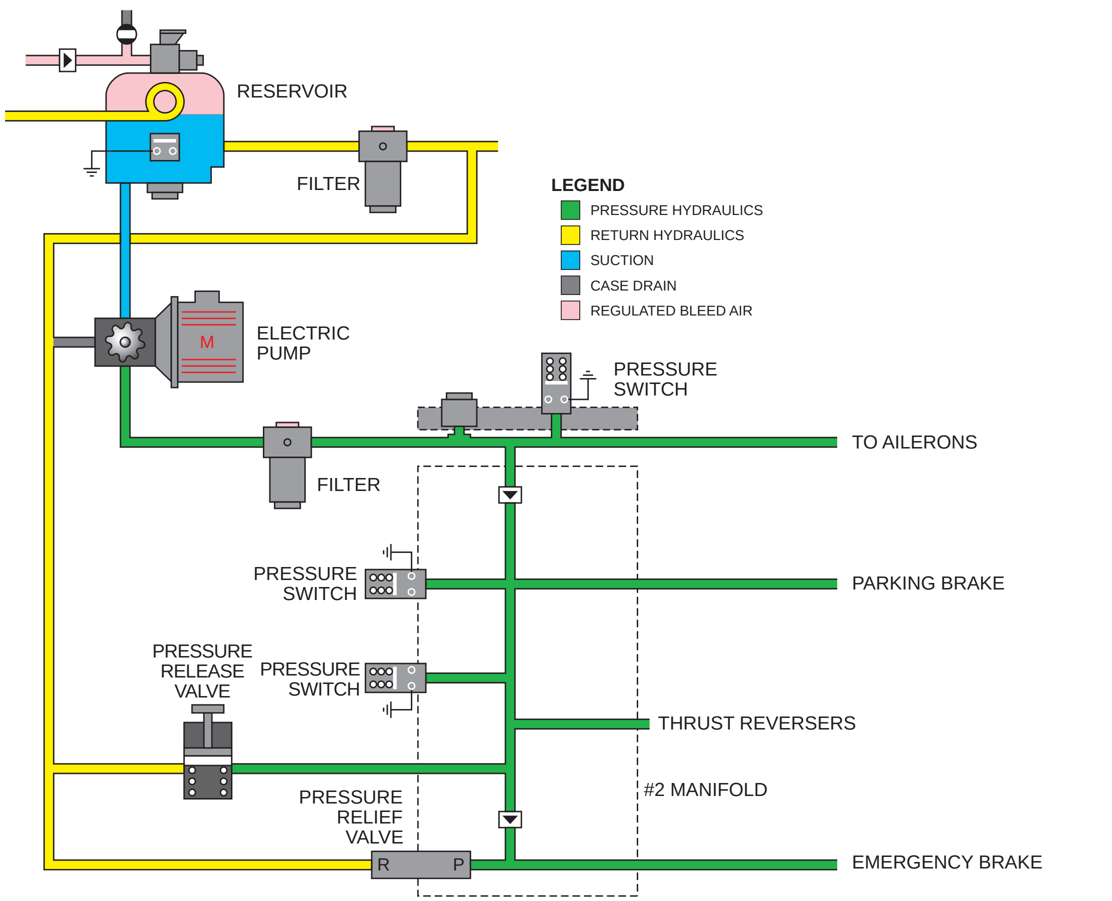


Figure 13-8. Auxiliary Hydraulic System

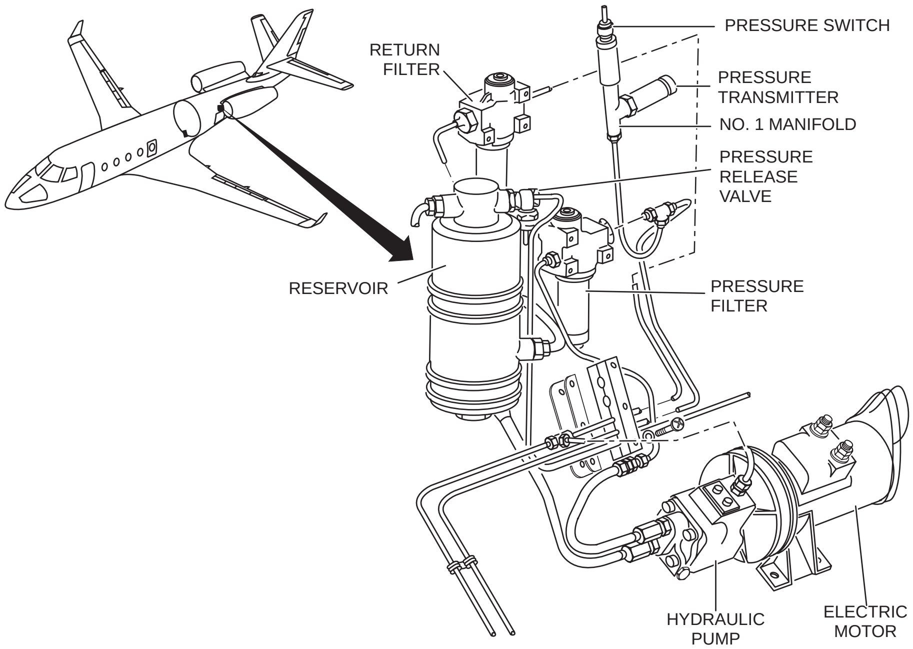


Figure 13-9. Auxiliary Pump

proportionate to system demand. The pump has a maximum rated flow of 2.9 GPM. It is controlled by the 2,000 psi pressure switch in manifold #2. Just as occurs in the engine-driven pumps, a small amount of hydraulic fluid is continuously pumped through the case drain and back to the reservoir to facilitate cooling and lubrication of the hydraulic pump. The pump and motor assembly are located on the right side of the service compartment below and aft of the auxiliary hydraulic reservoir.

Auxiliary Hydraulic Reservoir

The auxiliary hydraulic reservoir (Figure 13-10) is located on the right side of the fuselage in the service compartment. It is similar to the reservoir of the main system, but has a smaller capacity. It features a locking cap, fluid level sight gauge, air pressure relief and release valves and a low-level switch. The reservoir is designed to maintain positive fluid flow to the hydraulic pump under a negative-g load for a short period of time. The reservoir is designed to cause returned fluid to circulate within it in order to facilitate cooling.

Figure 13-10. Auxiliary Hydraulic Reservoir

The reservoir cap and sight gauge are accessible through a hinged door on the right side of the aft fuselage. Fluid level should be maintained to approximately the mid-point of the sight gauge. The low-level switch will cause the AUX HYD LEVEL LOW caution message to be displayed on the EICAS.

Auxiliary Hydraulic Accumulator

The auxiliary hydraulic system is equipped with an accumulator to store a small amount of hydraulic fluid under normal operating pressure. This is to allow the auxiliary system to be used for emergency or parking brakes or for thrust reverser deployment in case the auxiliary hydraulic pump fails. It is also used to maintain pressure on the parking brake when the auxiliary pump is not operating.

The accumulator is pressurized with nitrogen to approximately 1,500 psi., depending on the ambient temperature. This is referred to as the precharge. When the auxiliary hydraulic pump is activated, hydraulic pressure of approximately 3,000 psi is applied to the accumulator. The hydraulic fluid under pressure enters the accumulator and acts against the nitrogen, which undergoes additional compression. The compressed nitrogen then acts like a spring to keep the fluid under pressure. The pressure gauge in the pressure refueling compartment will indicate normal hydraulic system pressure of approximately 3,000 psi. under these conditions.

The number of emergency brake or parking brake applications available depends on the state of charge of the accumulator. Each time a brake is applied and released, fluid is drawn from the accumulator and sent to the reservoir. Once this occurs, this hydraulic fluid is no longer under pressure and is unavailable for use. A fully-charged accumulator can give a maximum of ten brake applications or one thrust reverser deployment. If the precharge is insufficient, there will be fewer available operations of these systems.

During the exterior inspection, any fluid that is under pressure in the accumulator must be depleted in order to check the pre-charge. This is accomplished by manually depressing the pressure release valve (Figure 13-11) located in the pressure refueling compartment. This will release hydraulic pressure from the accumulator by sending the pressurized fluid back to the auxiliary reservoir. The precharge can then be read directly from the pressure gauge.

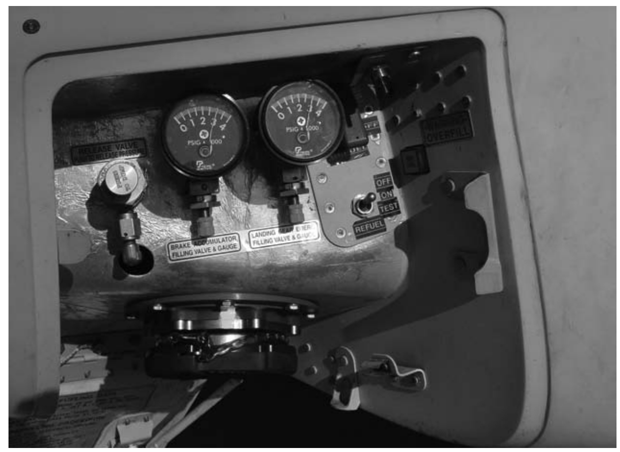


Figure 13-11. Pressure Refueling Compartment

Auxiliary Hydraulic Manifolds

There are two manifolds (Figure 13-12) in the auxiliary hydraulic system, referred to as the No. 1 and No. 2 manifolds. The No. 1 manifold includes a pressure transmitter and a pressure switch. The pressure transmitter provides auxiliary hydraulic system pressure indication on the primary and secondary pages of the EICAS. The pressure switch activates the AUX HYD PRESS LOW caution message on the EICAS if aileron system pressure falls to 900 psi for 2.5 seconds and the auxiliary hydraulic pump should be running. Manifold No. 1 is located on the right side of the service compartment, just at of the auxiliary hydraulic system reservoir.

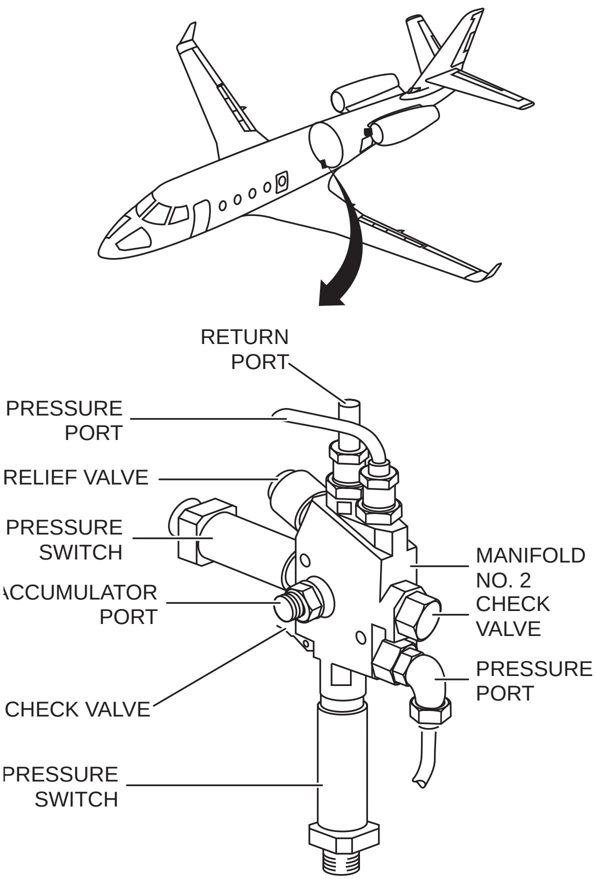


Figure 13-12. No. 2 Manifold Assembly

The No. 2 manifold directs system pressure to the thrust reversers and the emergency brake system and is located just aft of the main landing gear wheel well. It contains two check valves, two pressure switches, and a relief valve. One check valve prevents pressure loss from the auxiliary accumulator and the thrust reverser system; the other check valve prevents pressure loss from the emergency brake system.

The two pressure switches have different pressure settings. One will close when the pressure in the auxiliary hydraulic system accumulator falls below 2,000 psi to signal the logic circuitry to switch on the auxiliary pump. The second switch will close when the pressure in the accumulator falls below 1,200 psi to illuminate the AUX HYD PRESS LOW caution message on the EICAS.

The relief valve opens to the return line if auxiliary system pressure becomes excessive. It will automatically close again when pressure returns to normal.

High-Pressure Filter

The auxiliary hydraulic system is equipped with a single high-pressure filter to remove contaminants that may originate within the hydraulic pump. Its construction and operation are similar to those of the high-pressure filters in the main system, described earlier in the text. The filter features a differential pressure indicator to provide visual evidence of element blockage. There is no fluid bypass provision on the high-pressure filter.

Low-Pressure Filter

A single low-pressure filter removes suspended particulates from the hydraulic fluid before it is returned to the auxiliary hydraulic reservoir. It is also equipped with a differential pressure indicator and has bypass capability.

Fluid Temperature Sensor

A temperature sensor is installed in the return line to the auxiliary hydraulic reservoir to provide temperature information to the EICAS. Hydraulic fluid temperature is continuously available on the secondary page of the EICAS. The AUX HYD TEMP HI caution message on the EICAS will appear if the fluid temperature exceeds 85°C.

Auxiliary Hydraulic Pump Control Switch

Operation of the auxiliary hydraulic pump is controlled by logic circuitry and depends on the selection of the three-position control switch (Figure 13-13) that is located on the pedestal, near the thrust levers. The switch positions are OFF, AUTO and OVRRD. Pump operation in each of these switch positions is discussed in the Operations section of this chapter.

OTHER COMPONENTS

Reservoir Air Pressurization System

Both of the hydraulic reservoirs share a common air pressurization system (Figure 13-14). This system is necessary to prevent the formation of bubbles within the fluid of the hydraulic reservoirs at high altitude or at high operating temperatures. If bubbles are allowed to form in the reservoir, fluid flow to the hydraulic pumps would be interrupted and loss of hydraulic pressure could result.

To pressurize the reservoirs, bleed air is extracted from the engines and APU. A common manifold is shared by these three sources. Check valves are installed to allow any one of the sources to maintain required pressure. Pressurization of the reservoirs is independent of the cabin air selector setting.

An air pressure regulator within the system maintains the air pressure between 25 and 30 psi. If pressure drops significantly below this value, a pressure switch on the regulator will cause the HYD TANK PRESS LOW caution message to be displayed on the EICAS. Check valves are installed in the system to prevent this message from appearing unnecessarily at low engine power settings. The regulator also

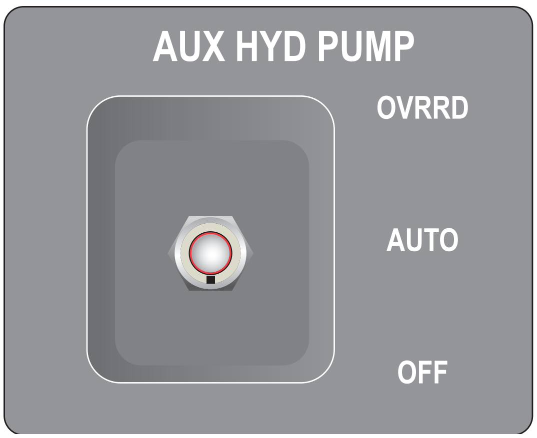


Figure 13-13. AUX HYD PUMP Switch

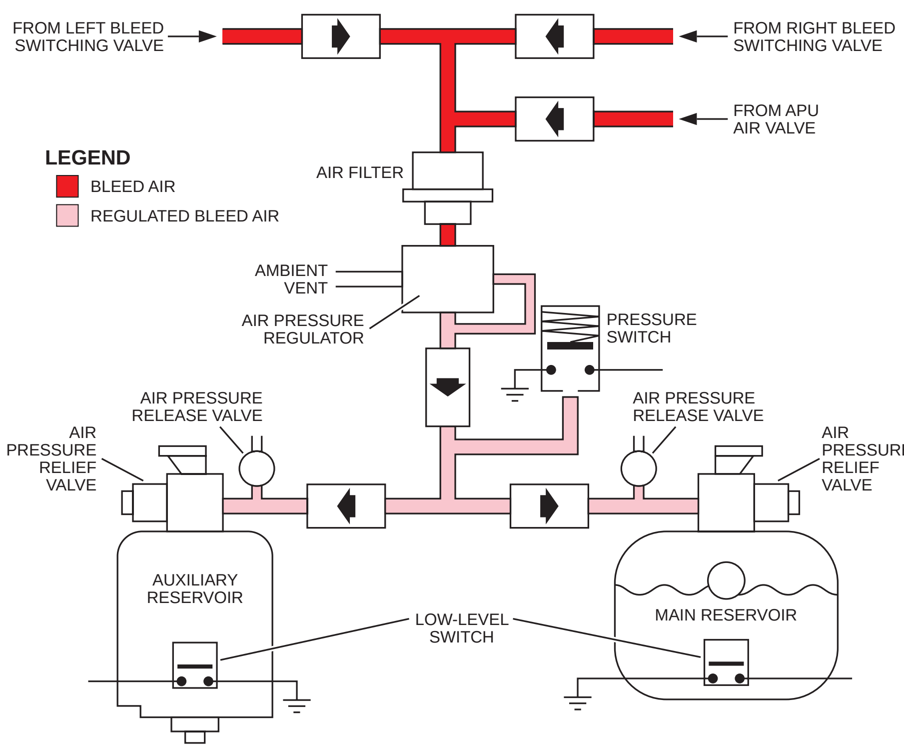


Figure 13-14. Reservoir Air Pressurization System

features a vacuum relief valve to prevent a negative pressure from developing within the reservoirs.

The air that pressurizes the reservoirs is filtered to remove any foreign material. To prevent over-pressurization of the system, each reservoir is fitted with a pressure relief valve. If excessive pressure builds in the system, the valves open automatically and re-close when the pressure is reduced to within limits.

Each reservoir is equipped with a pressure release valve, allowing maintenance personnel to deplete the air pressure in the reservoirs prior to opening the caps for servicing the hydraulic system. There is also a provision to allow the system to be pressurized with compressed shop air for maintenance purposes.

Reservoir Crossover Plumbing

When normal brakes are applied, fluid from the main hydraulic system is pushed into the wheel brake assemblies. If the aircraft is stopped and the parking brake is subsequently applied, only a small amount of auxiliary hydraulic system fluid is capable of entering the brake assemblies. If the normal brakes are held firmly while the parking brake is selected off, the fluids will tend to remain in their respective systems. However, if the normal brakes are not held during the release of the parking brake, fluid in the brake assemblies from both systems will be returned to the auxiliary hydraulic system reservoir. This can cause the auxiliary reservoir to exceed its fluid capacity, resulting in the excess fluid being vented overboard.

To keep the fluid from being lost from the aircraft as a result of this process, a crossover plumbing system is installed to return fluid from the over-serviced auxiliary reservoir to the main reservoir. This is composed of a tube that connects the auxiliary reservoir and the main reservoir with a one-way check valve. Fluid can only flow from the auxiliary reservoir to the main reservoir.

OPERATION

MAIN HYDRAULIC SYSTEM

The main hydraulic system becomes operational as soon as either engine is started. As soon as bleed air pressure becomes sufficient, the HYD TANK PRESS LOW caution message will extinguish. Bleed air for reservoir pressurization can also be provided by the APU.

The operating engine's hydraulic pump will draw fluid from the main reservoir through its respective open fire (hydraulic) shut-off valve. Fluid is pressurized to its operating pressure by the pump and its internal pressure regulating mechanism. The pressurized fluid is then passed through one of the high-pressure filters and to the main system manifold. If the filter is blocked, the fluid has no ability to bypass it and will not reach the manifold.

Within the manifold, the pressure of the fluid opens one of the pressure switches which causes the (L/R) HYD PUMP PRESS LOW caution message to extinguish on the EICAS. The fluid then flows past a check valve to enter the common plumbing of the main hydraulic system. The check valve functions to prevent back flow into a failed pump or the pump of an engine that is not operating. At this point, the pressure transmitter signals the actual system operating pressure to the EICAS. If the pressure is low, the indication will appear in amber. If normal, the pressure indication will be green. If the pressure regulating function of the pump has failed and an excessive pressure is applied to the system, the indication will turn amber again and the EICAS will generate a MAIN HYD PRESS HI caution message. The system pressure relief valve will also open, returning fluid to the main reservoir and reducing the pressure to an acceptable level.

When the second engine is started, the process is repeated. The other pressure switch will be opened, extinguishing the other HYD PUMP PRESS LOW caution message. Only one pump is necessary to operate the system.

Once the fluid from either pump passes through the main system manifold, it is available for use in the aircraft accessories. Main system pressure is used to operate the nose wheel steering, normal braking system, landing gear extension and retraction mechanisms, air brakes and part of each aileron servoactuator.

Once the main system fluid has been utilized in the aircraft, it passes through the low-pressure filter before being returned to the main reservoir. Should this filter become completely blocked during operation, fluid can bypass the filter and return to the reservoir. This prevents the main system from failing due to fluid starvation. The temperature of the fluid is continuously monitored as it returns to the reservoir and is indicated on the secondary EICAS page. A MAIN HYD TEMP HI caution message will appear on the EICAS if the returned fluid exceeds 85ºC.

If a leak develops in the system, the low-level switch within the main system reservoir will close. This will cause the MAIN HYD LEVEL LOW caution message to appear on the EICAS. It will also cause activation of the auxiliary hydraulic pump if the control switch is selected to AUTO.

If an engine fails or if one of the hydraulic pumps fail, the associated pressure switch within the main system manifold will close, activating the appropriate HYD PUMP PRESS LOW caution message on the EICAS. The check valve in the manifold will prevent pressure from being lost through the failed pump so that the system can continue to operate. If pressure from both pumps is lost, the other pressure switch will close. The other HYD PUMP PRESS LOW caution message will appear. System logic will cause the auxiliary hydraulic pump to operate when both of these pressure switches are closed and the control switch is set to AUTO.

If reservoir air pressurization is lost or drops below limits, the HYD TANK PRESS LOW caution message will appear. There is only one message because both reservoirs share a common system.

Because many of the aircraft systems require hydraulic pressure to operate, a provision is made so that the main system can operate with an external hydraulic pressure source. It is connected to the left engine hydraulic pump assembly to allow maintenance personnel to operationally check these systems.

AUXILIARY HYDRAULIC SYSTEM

The auxiliary hydraulic system is designed for part-time operation. It is the source of pressure for systems that are normally used during takeoff, landing and ground operations; such as thrust reversers and the parking brake. It also serves as a back-up source of hydraulic pressure for the brake system and for the aileron servoactuators. As a result, continuous operation of the system is not necessary except during takeoff, landing, ground operations, and during abnormal system conditions that affect the operation of the main hydraulic system.

When the auxiliary hydraulic pump is operating, fluid is drawn from the auxiliary hydraulic reservoir by the pump. After the pump pressurizes the fluid, it is sent through a high-pressure filter.

If this filter becomes blocked, there is no provision to allow fluid to bypass it. Once the fluid leaves the filter, it proceeds to the first of two hydraulic manifolds.

The No.1 manifold contains a pressure transmitter and a pressure switch. It has attachments for hydraulic lines that lead to the aileron servoactuators and to the No. 2 manifold, discussed shortly.

The pressure transmitter will send pressure information to the EICAS primary and secondary pages.

Display logic is consistent with the main system, with green being a normal indication. When the landing gear is retracted and all systems are functioning normally, an auxiliary pressure indication of zero is presented in green because this is a normal operating condition. If the pressure regulating mechanism within the pump fails and excessive pressure is produced, the AUX HYD PRESS HI caution message will be displayed on the EICAS.

The pressure switch in the No. 1 manifold is used to generate the EICAS message AUX HYD PRESS LOW if the pressure is below limits and the pump should be operating according to system logic. This switch incorporates a 2.5 second delay to prevent the unnecessary appearance of this message due to momentary fluctuations in system pressure.

The No.2 manifold has fittings for hydraulic lines to the thrust reversers, accumulator and emergency/parking brake valve. It contains two check valves, has a pressure relief valve and two pressure switches.

As fluid enters the No. 2 manifold from the No. 1 manifold, it passes through a check valve. The purpose of this valve is to keep pressure in the accumulator during periods of pump inactivity. Pressurized fluid is available for the thrust reversers after passing the check valve. The accumulator is pressurized and both pressure switches are opened. These switches have two different settings. One switch will activate the auxiliary hydraulic pump if it senses that the accumulator has lost a portion of its pressure. The other switch activates at a lower pressure and will cause the AUX HYD PRESS LOW caution message to be displayed on the EICAS.

Pressurized fluid then proceeds, as needed, through the second check valve to the emergency/parking brake valve and the pressure relief valve. The check valve is located so that parking brake pressure will be maintained after the accumulator has been depleted for the purpose of checking the pre-charge. The pressure release valve utilized in this procedure is also connected to the No. 2 manifold by system plumbing and located in the pressure refueling compartment near the pressure gauge.

The pressure relief valve serves the same function as the one on the main system. If an excessive pressure develops within the system, this valve will open and return fluid to the auxiliary reservoir until the pressure is within limits.

Once the auxiliary system hydraulic pressure has been utilized in the aircraft, the fluid is returned to the auxiliary reservoir. Before reaching the reservoir, the fluid is passed through a low-pressure filter to remove any contaminants. Fluid may bypass this filter, in case the element is blocked, to prevent fluid starvation of the pump. The temperature of the fluid is also continuously monitored prior to entering the reservoir. The temperature is available on the secondary EICAS page. The AUX HYD TEMP HI caution message will be displayed if the temperature exceeds 85ºC.

Operation of the auxiliary hydraulic pump is controlled by the position of the AUX HYD PUMP control switch in the cockpit and by the occurrence of certain system conditions within the aircraft. This switch has three positions: OFF, AUTO and OVRRD. During normal operations, the switch is in AUTO.

If the switch is in OFF, the pump will not normally operate. There is one exception. Each aileron servoactuator is equipped with a microswitch that will be closed if there is a mechanical failure within that servoactuator. If this occurs, the auxiliary hydraulic pump will be activated regardless of the position of the control switch. To deactivate the pump, DC power must be removed from the pump circuit.

If the switch is in AUTO, the pump will operate if any of the following conditions exist:

- Either main landing gear not up and locked.
- Activation of both main system low pressure switches.
- Low fluid level in the main system reservoir.
- Depletion of auxiliary system accumulator hydraulic pressure.
- Loss of hydraulic pressure to an aileron servoactuator or mechanical failure within a servoactuator.

The main landing gear uplock switch logic ensures that the pump will operate as soon as the landing gear is extended for landing and will remain operating for all ground operations. After take-off, the pump will continue to operate for approximately 10 seconds after the gear is up and locked in order to ensure full pressurization of the accumulator.

If both of the pressure switches in the main system manifold are activated, insufficient pressure is available from the main system. This could have resulted from both engine-driven pumps failing, both main system high-pressure filters becoming blocked, or a loss of main system fluid. The auxiliary pump is activated to ensure that there is pressure available for aileron servoactuator operation.

When the pump has been activated as a result of one of the above conditions, the pump will continue to operate as long as that condition exists. If the condition that activated the pump is corrected, pump operation will cease unless it has been caused by one of the servoactuator microswitches. If the pump has been activated because of a loss of main system hydraulic pressure within one of the servoactuators, the pump will remain operating if hydraulic pressure is restored. A holding relay will keep power applied to the pump until the control switch is selected OFF. If the pump has been activated by a mechanical failure within one of the servoactuators, the only method of de-activating the pump is by removing power from the aircraft or opening the appropriate circuit breaker. In this case, power to the pump bypasses the switch logic circuit. This ensures that hydraulic assistance will be available to aid in aileron control.

With the switch in OVRRD, the pump operates continuously regardless of other conditions.

If the auxiliary hydraulic pump fails or auxiliary system pressure is lost, the accumulator is charged with hydraulic pressure to allow a limited amount of braking or thrust reverser operation. A fully charged accumulator provides one deployment of the thrust reversers or 10 emergency brake applications after loss of the auxiliary system pressure. There is no way to directly assess the state of charge of the accumulator from the cockpit.

LIMITATIONS

The limitations contained in Section One of the *Airplane Flight Manual (AFM)* must be complied with regardless of the type of operation.

APPROVED HYDRAULIC FLUIDS

Preferred fluid: SKYDROL LD IV—Monsanto Co.

Alternative fluids:

- Type II SKYDROL 500B—Monsanto Co.
- Type II CHEVRON HYJET W—Chevron International Oil Co. Inc.
- Type III AERO SAFE 2300 W—Stauffer Chemical Co.
- Type III CHEVRON HYJET III—Chevron International Oil Co. Inc.

The above mentioned fluids may be used and mixed without restriction.

EMERGENCY AND ABNORMAL OPERATIONS

The EICAS displays both the main and auxiliary system pressure on both the primary and secondary pages of the MFD (Figure 13-15).

Caution messages display as amber EICAS messages on the MFD, and are accompanied by a MASTER CAUTION light. There are no WARNING messages associated with Hydrualics (Figure13-15).

AUX HYD LEVEL LOW—This indicates that the fluid level in the auxiliary reservoir has dropped to approximately 0.14 gallons or less (530 cc) remaining in the reservoir.

AUX HYD PRESS HI—This indicates the pressure in the auxiliary hydraulic system is above 3,500 psi. This signal comes from the AUX PRESS transmitter.

AUX HYD PRESS LOW—This indicates the pressure in the auxiliary accumulator is below 1,200 psi. as sensed by pressure switch 43C in the No.2 manifold of the auxiliary system. This message may also be illuminated if the auxiliary system pressure to the ailerons is below 900 psi for more than 2.5 seconds and the auxiliary pump should be operating.

AUX HYD TEMP HI—The temperature in the return line to the auxiliary reservoir is above 85°C (185°F).

L/R HYD PUMP PRESS LOW—This indicates that the pressure output of this pump has dropped to 1,500 ±100 psi as sensed by switch in the main system pressure manifold. When the pressure from the pump rises to 1,800 psi or more this message extinguishes

HYD TANK PRESS LOW—Output from the hydraulic reservoir air pressure regulator has dropped to 8.5 ±1 psi. When the pressure rises to 10.5 ±1 psi this message extinguishes.

MAIN HYD LEVEL LOW—Level in the main hydraulic level has dropped to 0.079 (300 cc). When the level rises to 0.147 gallons (556 cc) the message extinguishes.

MAIN HYD PRESS HI—Pressure transmitter senses that the main hydraulic pressure is above 3,500 psi. When the main hydraulic pressure decreases to 3,200 psi this message extinguishes.

MAIN HYD TEMP HI—Fluid temperature at the return line to the main hydraulic reservoir exceeds 85°C (185°F).

If both systems are operating normally, there will be no messages displayed on the EICAS.

Figure 13-15. Hydraulic EICAS Indications

QUESTIONS

1. The pumps in the main hydraulic system are:

A. Electrically driven, variable pressure
B. Engine driven, constant pressure
C. Engine driven, variable pressure
D. Electrically driven, constant pressure

2. With the AUX HYD PUMP switch in AUTO, the auxiliary pump operates if:

A. There is a complete loss of main system pressure
B. The auxiliary system accumulator pressure is low
C. The main system reservoir fluid level is low
D. Any of the above

3. The AUX HYD PUMP automatically stops operation if all failures are corrected except after:

A. HYD PUMP PRESS LOW L/R
B. MAIN HYD LEVEL LOW
C. Brake accumulator pressure loss
D. Aileron failure

4. In the event the auxiliary pump fails or pressure is lost, a fully charged accumulator provides:

A. Up to ten emergency brake applications
B. One deployment of the thrust reversers
C. Eight emergency brake applications
D. Up to ten emergency brake applications or one deployment of thrust reversers

5. The preferred hydraulic fluid for use in the system is:

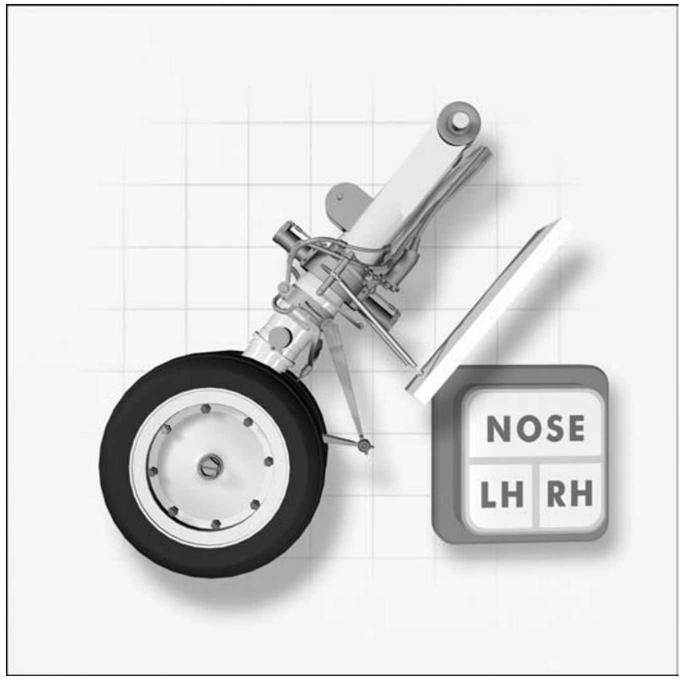
A. SKYDROL 500B
B. SKYDROL LD IV
C. AERO SAFE 2300 W
D. CHEVRON HYJET III

CHAPTER 14
LANDING GEAR AND BRAKES

CONTENTS

ILLUSTRATION

CHAPTER 14
LANDING GEAR AND BRAKES



INTRODUCTION

This chapter covers general information about the G150 landing gear and brake systems.

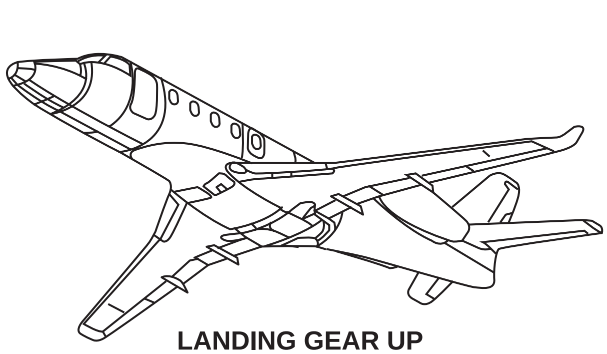
GENERAL

The G150 has a mechanically controlled, hydraulically operated fully retractable tricycle landing gear system (Figure 14-1). Features of the system include:

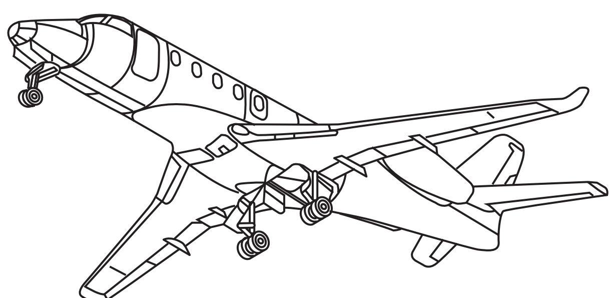
- A steerable dual-wheeled nose gear
- Two dual-wheeled main gear
- Mechanically actuated gear doors
- Hydraulic retraction and extension
- Emergency gear extension subsystem
- Position indicating and warning subsystems

The hydraulic wheel brake system incorporates the following:

- Hydraulic wheel brake assemblies for each main landing gear wheel
- Parking brake
- Normal brakes with antiskid protection
- Emergency brakes without antiskid protection



LANDING GEAR UP



LANDING GEAR DOWN

Figure 14-1. Landing Gear

SYSTEM DESCRIPTION

Normal landing gear extension and retraction is hydraulic and controlled by the landing gear control lever on the center instrument panel. The main hydraulic system supplies pressure.

Emergency extension is pneumatic and controlled by the emergency extension control handle on the left side of the center console. Pneumatic pressure for emergency gear extension is supplied by stored compressed nitrogen.

The nose gear has an oleo-pneumatic shock strut made of forged aluminum with a steel piston. A hydraulic actuator extends and retracts the nose landing gear through operation of the drag brace. The nose landing gear is held in the down position by a spring-loaded lock within the drag brace.

A spring-loaded lock within the nose gear well, in addition to hydraulic pressure within the actuator, retains the nose gear in the retracted position. The locks are released by hydraulic pressure during retraction or extension, as appropriate. The uplock pneumatically releases during emergency gear extension.

Four doors fully enclose the nose gear when it is retracted. During gear extension, gear door sequence is as follows:

- The two forward doors open and re-close during gear extension.
- The aft two doors remain open when the gear is extended.

All of the nose gear doors are actuated by mechanical linkage to the drag brace.

Each main gear consists of an aluminum forged strut longitudinally hinged to the wing structure for inboard retraction. Two doors enclose each main gear assembly when retracted. Inboard door is operated by mechanical linkage to the hydraulic landing gear actuator. It opens fully during gear extension and remains partially open when gear is down and locked. The outboard door is mechanically linked to the support strut and moves to the fully open position when gear extends.

The main landing gear downlocks are located within each hydraulic actuator and are spring-loaded in a locked position. During retraction, hydraulic pressure releases the locks and allows the actuators to operate. Uplocks located on each inboard gear door maintain the main gear in the retracted position if hydraulic pressure is lost.

Nosewheel steering is provided by two hydraulic actuators located at the top of the nose landing gear assembly. Pressure to these hydraulic cylinders is controlled by an electronic control unit that receives input from a tiller in the cockpit. The unit also receives electrical signals from the rudder pedals, allowing either pilot to have a limited amount of nosewheel steering through the pedals. Hydraulic pressure for steering is supplied by the main hydraulic system. There is no mechanical linkage from the cockpit to the nosewheel steering system.

Position and warning systems consist of visual position indicators, CAS message and aural warning.

The brake system consists of normal and emergency operating subsystems, parking brake, and anti-skid subsystem.

LANDING GEAR

NOSE LANDING GEAR

The nose landing gear consists of the following components (Figure 14-2):

- Shock Strut Assembly, including wheels and tires
- Drag Brace and Trunion
- Actuator Assembly
- Doors

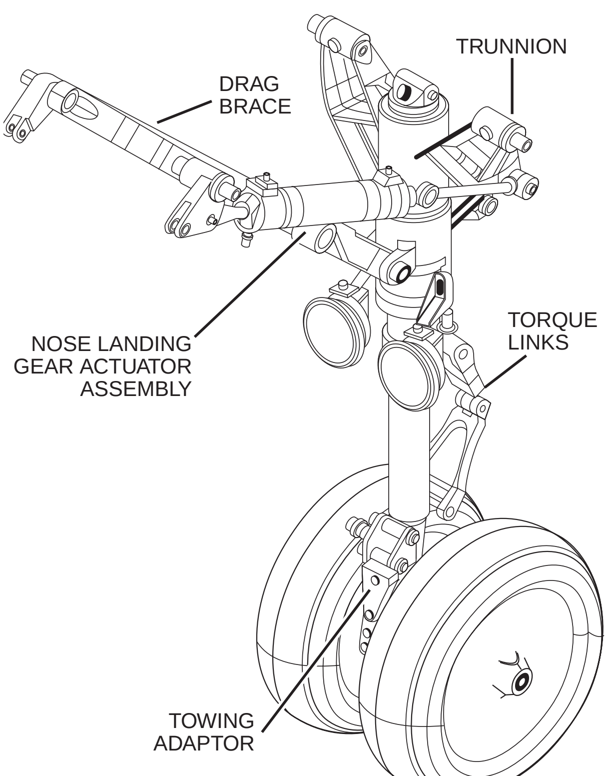


Figure 14-2. Nose Landing Gear

Shock Strut Assembly

The nose landing gear shock strut assembly is a dual-wheel system that retracts forward into the nose gear well. It is comprised of an oleo-pneumatic shock strut and torque link assembly. A spring-loaded cam on the forward part of the shock strut engages the uplock in the nosewheel well when the gear is retracted.

The torque link (scissor) assembly transmits steering forces to the nosewheel axle. A ball lock pin connects the upper and lower arms of the torque links and is removable to permit unlimited nosewheel steering angles during towing. A towing adapter assembly is installed on the nosewheel axle torsion shaft for aircraft towing.

NOTE

Towing with the nose gear torque links connected is prohibited

Drag Brace and Trunnion

The nose landing gear drag brace and trunnion provide structural support to the nose landing gear. The trunnion is mounted to the fuselage on a pivot axis permitting retraction of nose strut forward and up into the nosewheel well.

The drag brace is a two piece assembly. The upper arm contains a spring-loaded downlock that engages the lower arm to create a rigid brace that holds the nose gear in the down position. The lock automatically engages when the landing gear is fully extended and is mechanically released by the hydraulic downlock actuator during gear retraction. The downlock actuator is located within the drag brace and receives hydraulic pressure whenever the landing gear handle is selected UP and main hydraulic system pressure is available. This retracts the downlock and allows the drag brace to fold in response to actuator force.

Actuator Assembly

The nose landing gear actuator assembly is a conventional two-port hydraulic cylinder that provides the force required for nose landing gear extension and retraction. Once the nose gear downlock is released by hydraulic pressure, the actuator applies force to the drag brace on one end and the trunnion on the other.

This causes the drag brace to fold upward, resulting in gear retraction. Once the gear is up, hydraulic pressure is maintained continuously on the actuator and the nose gear uplock engages by spring pressure. These events keep the nose gear in the retracted position. During extension, hydraulic pressure (or pneumatic pressure in event of an emergency extension) is applied to the uplock to release it and to the actuator to extend the gear. The downlock is engaged once the gear is extended and hydraulic pressure is continuously applied to the actuator.

Doors

The nose landing gear is enclosed by four doors;

- Two forward
- Two aft

During landing gear extension all four doors open with the two forward doors returning to the closed position and the two aft doors remaining open. All four doors fully open for landing gear retraction, then close over the retracted nose gear. (Figure 14-3).

This leaves a smooth contour over the wheel well in flight. Both the forward and aft doors are actuated mechanically by control rods connected to the nose landing gear drag brace.

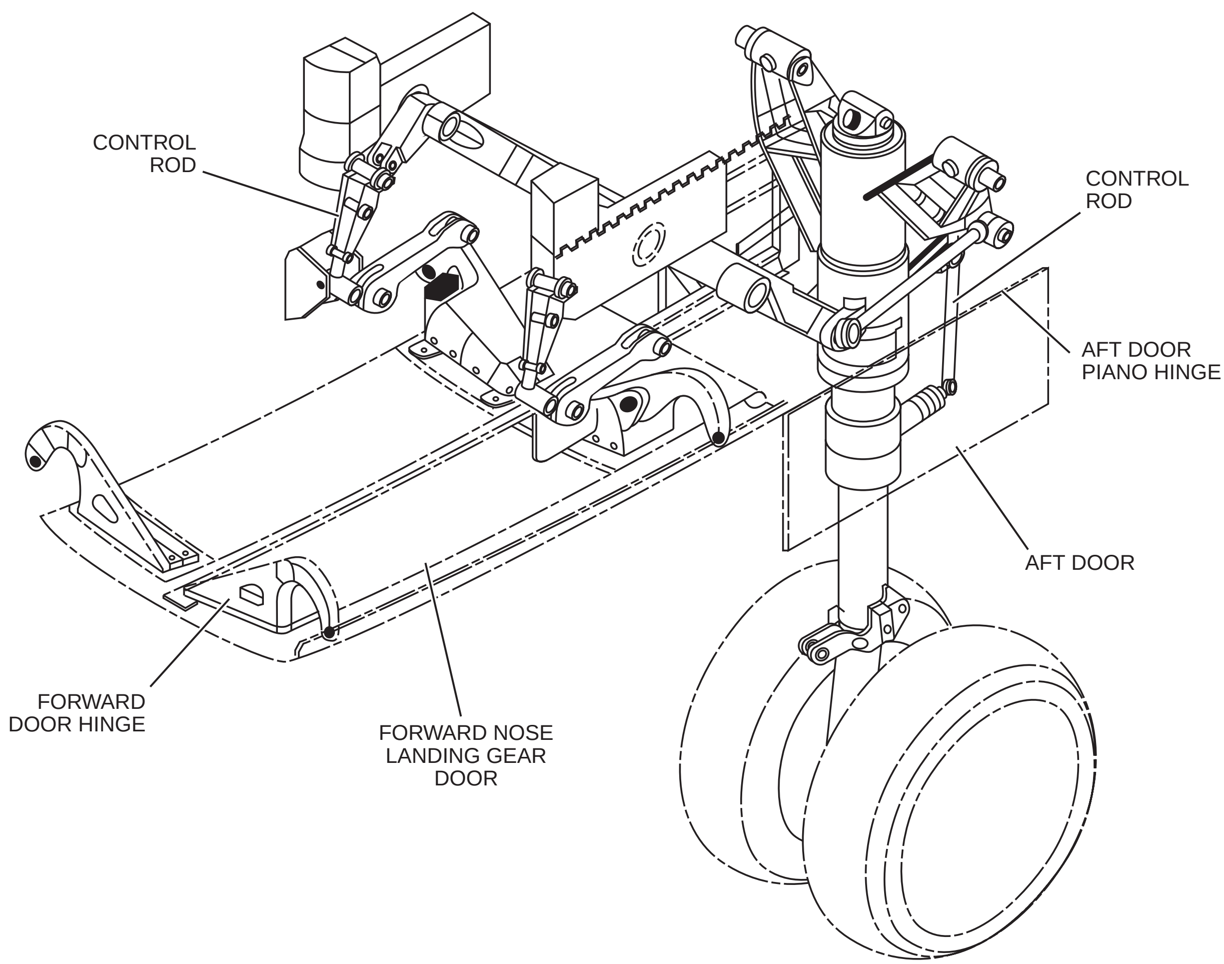


Figure 14-3. Nose Landing Gear Door Attachment

MAIN LANDING GEAR

The main landing gear consists of the following components:

- Shock strut assembly, including wheels and brakes
- Actuator assembly
- Doors

Shock Strut Assembly

The main landing gear shock strut assembly has three main components (Figure 14-4):

- Hydraulic shock absorber
- Support strut
- Trailing arm

The air-oil hydraulic shock absorber compresses in response to loads applied during taxiing, takeoff and landing. It is a steel piston riding in a cylinder that utilizes ring seals to prevent fluid leakage between the piston and the outer cylinder. The shock absorber is mounted on the aft side of the support strut and attaches to the upper end of the support strut and the trailing arm.

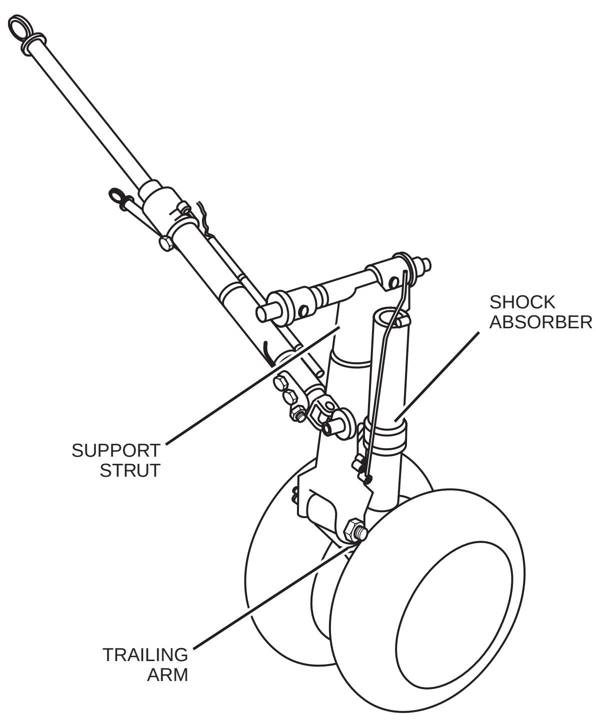


Figure 14-4. Main Landing Gear Shock Strut Assembly

The noncompressible support strut assembly is trunnion mounted to the wing structure. It is longitudinally hinged to allow inboard retraction of the main gear. The lower end attaches to the pivoting trailing arm with a hinge pin that is referred to as the knee pin.

The forward end of the trailing arm is pin-mounted to the lower end of the support strut. The aft end of the trailing arm attaches to the shock absorber and contains the axle assembly, brakes and wheels. The shock absorber is compressed when the trailing arm is deflected upward, pivoting about its forward mounting pin. The trailing arm is sometimes referred to as the drag link.

An electrical harness and hydraulic lines are attached to the gear structure. The electrical harness provides a means for wheel speed to be transmitted for the antiskid system and for ground contact to be signaled to the Integrated avionics processing system (IAPS). Hydraulic lines provide pressure to the hydraulic brake assemblies on each wheel.

Actuator Assembly

The main landing gear side brace actuator provides structural support to the wing as well as hydraulic extension and retraction of the main landing gear (Figure 14-5). The side brace actuator consists of a double acting hydraulic cylinder with an internal downlock mechanism. The actuator also contains the electrical switch that signals that the gear is down and locked. It operates in conjunction with the downlock mechanism.

The downlock mechanism contains key locks and a locking ram that engage whenever the actuator reaches its fully extended position. Spring pressure within the locking mechanism operates together with hydraulic pressure (or pneumatic pressure during an emergency extension) to automatically engage the lock. The locking mechanism maintains the gear in the locked position when all hydraulic pressure is removed .

During retraction, hydraulic pressure is applied to the retract side of the actuator. This pressure overcomes the spring force in the downlock mechanism, disengaging it. The hydraulic ram is then able to move into the cylinder and the gear is retracted by hydraulic pressure within the actuator. Hydraulic pressure is always required to unlock the downlock mechanism.

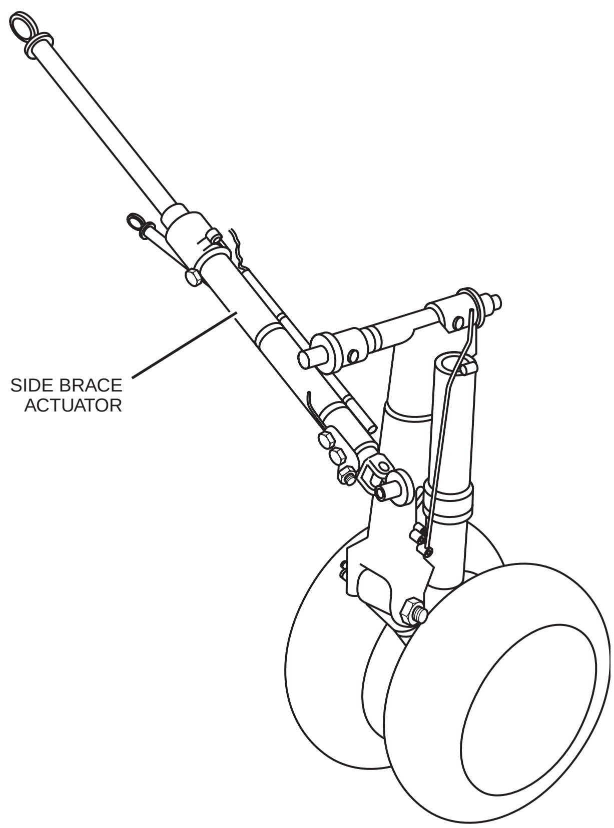


Figure 14-5. Main Landing Gear Actuator Assembly

Gear Doors

Each main landing gear has mechanically actuated outboard and inboard wheel well fairing doors. The doors fully enclose the wheel wells when the landing gear is retracted and form a smooth, flush fit with the contours of the airframe. Seals around the doors seal the wheel wells during flight.

Each outboard door is hinged to the wing structure and is operated by mechanical linkage connected to the support strut assembly.

Each inboard door is attached to the fuselage structure by dual hinges and is mechanically operated when the main landing gear is retracting or extending. Linkage to the side brace actuator operates the door. Because of the geometry of this system, the inboard gear doors are partially open when the gear is down and locked (Figure 14-6).

Two spring-loaded latches are located on each inboard door. These engage uplocks within the wheel well when the gear is retracted. By acting through the door actuating mechanism, the uplocks on the inboard gear door will keep the main gear fully retracted if hydraulic pressure is lost. These locks are hydraulically released for normal gear extension and pneumatically released for emergency gear extension.

Figure 14-6. Main Gear Doors

OTHER COMPONENTS

Selector Valve

The four-port, two-position selector valve is mechanically operated by the landing gear selector handle on the forward instrument panel via cable linkage. The landing gear selector valve routes hydraulic pressure from the main hydraulic system to the landing gear actuators and locking mechanisms. The valve is a rotary type with two positions; one for gear extension and one for gear retraction (there is no intermediate position).

Selector Panel

The landing gear position is selected by the landing gear control handle located on the landing gear selector panel (Figure 14-7). Landing gear position is indicated by three green lights on the panel. A red light is located in the control handle.

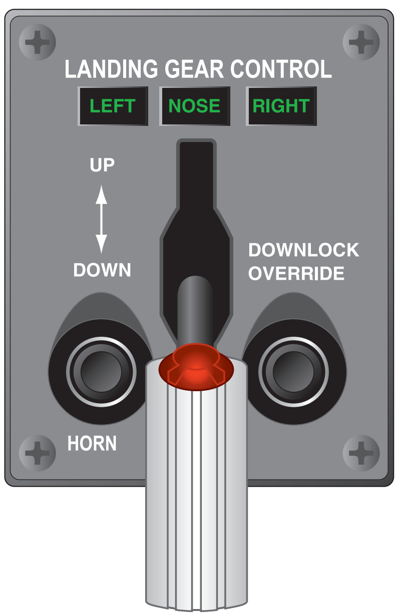


Figure 14-7. Landing Gear Control Handle

A plunger prevents inadvertent landing gear retraction when the aircraft is on the ground. The plunger is operated by a solenoid which is activated when aircraft weight is off the left main landing gear. The solenoid is overridden by pressing the DOWNLOCK OVERRIDE switch on the landing gear control panel.

A button is located on the selector panel to allow the landing gear warning system to be tested. It is labeled HORN and will cause an aural warning GEAR to be sounded and an EICAS warning message GEAR NOT DOWN to be displayed.

Control Cable

The landing gear control cable is a push-pull cable that provides the mechanical interface between the landing gear control handle on the selector panel and the landing gear selector valve.

Uplock Actuators

A single nose landing gear uplock actuator is mounted in the forward portion of the nose gear wheel well. Dual main gear uplock actuators are located in each main gear well. They are all functionally identical and spring loaded in the engaged position and engage automatically without hydraulic assistance during gear retraction (Figure 14-8). When landing gear extension is selected, hydraulic pressure is applied to the uplock actuators which causes the plunger within the actuators to extend and push the corresponding latches off the rollers. The uplock actuators are pressurized whenever the gear selector handle is selected DOWN and main hydraulic pressure is available.

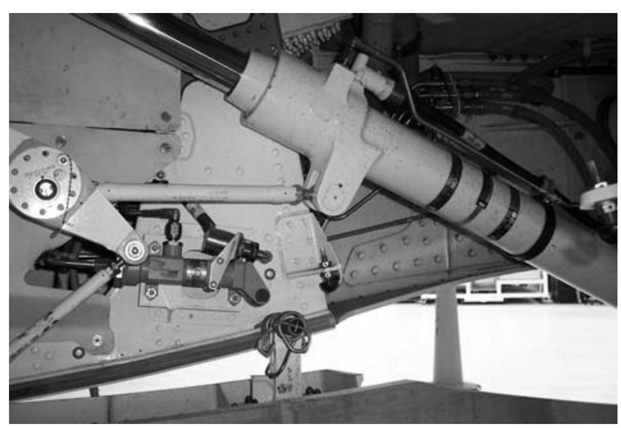


Figure 14-8. Main Landing Gear Uplock Assembly

Uplock Rollers and Uplock Latches

The landing gear uplock rollers are located on the ends of the uplock actuator assemblies. When the gear is retracted, the uplock rollers engage the spring-loaded uplock latches on the nose gear shock strut and the main gear inboard doors to retain the landing gear in the up and locked position. The rollers reduce friction between the actuators and latches to ease engagement and disengagement of the uplocks

Hydraulic Flow Regulator

The hydraulic flow regulator is a hydraulic restrictor that controls the actuation speed of the landing gear.

It limits the rate of hydraulic fluid that enters the actuators, keeping operation of the gear from becoming excessively abrupt.

EMERGENCY LANDING GEAR EXTENSION SYSTEM

In the event of a main hydraulic system malfunction, the aircraft is equipped with a landing gear emergency extension system. Compressed nitrogen gas is used to provide the pressure necessary to lower the gear. The emergency extension system is comprised of a pressure vessel, pressure gauge, filler valve, dump valve, and a landing gear emergency control handle with a cable connecting it to a landing gear emergency extension valve.

The emergency landing gear extension system is primarily composed of the following components:

- Emergency selector valve
- Emergency handle/cable
- Emergency extension nitrogen blowdown bottle
- Emergency extension dump valve

Emergency Selector Valve

The emergency selector valve has three ports and two positions. During normal operations, the port to the nitrogen supply bottle is closed. An open port is vented to prevent hydraulic pressure from building in the emergency gear extension system due to small amounts of hydraulic fluid that normally seeps around various components of the landing gear actuating system. This port is also used to depressurize the system by maintenance after the emergency extension system has been used.

This valve is operated by a cable that connects to the emergency gear extension handle in the cockpit. Once selected for emergency extension, the valve will open the port to the nitrogen supply bottle which allows nitrogen pressure to release all of the uplocks and pressurize the extension side of each of the gear actuators. Nitrogen pressure is also applied to the emergency extension dump valve, discussed later.

Emergency Handle/Cable

The emergency landing gear extension handle is located on the left side of the pedestal, near the pilot's right knee. It is connected to the emergency selector valve by a cable that transmits the mechanical motion of the handle to the valve. A spring-loaded latch must be released in order to operate the handle. (Figure14-9).

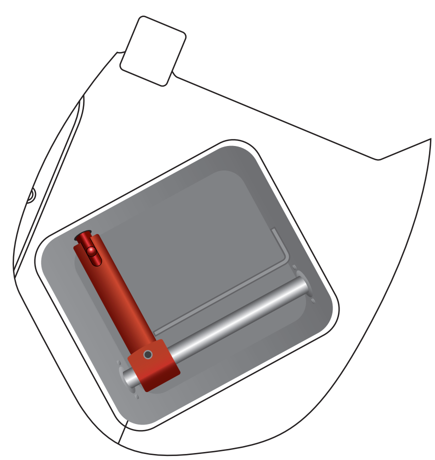


Figure 14-9. Emergency Gear Extension Handle

Emergency Extension Nitrogen Blow-down Bottle

A high-pressure bottle located in the aft fuselage is used to store compressed nitrogen that is used for emergency gear extension (Figure 14-10). The bottle is normally charged to approximately 3,000 psi, depending on ambient temperature. The pressure within this bottle is indicated directly on a gauge located in the pressure refueling compartment and is checked by the flight crew during the exterior inspection. A blowout disk is fitted to the bottle. If a significant over-pressurization of the bottle occurs, the disk will be blown from its port. This will also be evident during the exterior inspection.

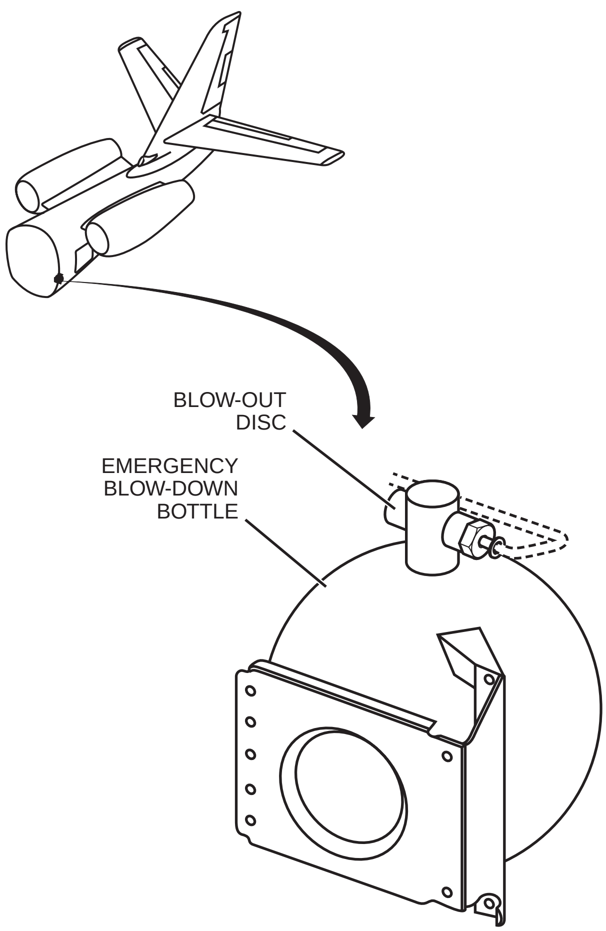


Figure 14-10. Landing Gear Emergency Extension Blow-Down Bottle

Emergency Extension Dump Valve

The dump valve is pressure operated by the emergency extension system. There is no mechanical linkage between the dump valve and the emergency gear extension handle. When nitrogen pressure enters the dump valve during an emergency extension, normal hydraulic pressure is shut off and a return path for hydraulic fluid within the actuators is opened to the main hydraulic fluid reservoir. This action prevents a hydraulic lock in the landing gear extension system.

POSITION AND WARNING

The landing gear position and warning subsystem provides landing gear position sensing and indication. This system also provides weight-on-wheels (WOW) signals that are used by various other aircraft systems.

Landing gear position is indicated visually on the flight deck by lights on the selector panel. Basic indications are either UP, DOWN, or UNSAFE. Under certain circumstances, an UNSAFE indication will also include aural and EICAS warnings.

The landing gear position and warning subsystem includes various switches and associated electrical circuitry. When contacted, each switch provides a path to electrically ground its associated circuit, causing the illumination of the various annunciators and the transmission of electrical signals to various systems.

Nose Landing Gear Uplock Sensing Switch

The nose landing gear uplock sensing switch is located in the upper part of the nose gear well, near the nose gear uplock. When contacted by the nose gear strut, the nose landing gear uplock switch provides an indication in the cockpit that the nose landing gear is up and locked.

Nose Landing Gear Downlock Sensing Switch

When the nose landing gear downlock engages, the nose landing gear downlock switch provides an indication in the cockpit that the nose landing gear is down and locked. This switch is located in the lower arm of the drag brace. This switch also completes the circuit to the taxi lights, extinguishing them prior to gear retraction.

Main Landing Gear Uplock Sensing Switches

One main landing gear uplock sensing switch is located in each main gear wheel well, near the forward inboard door uplock. When contacted by the uplock latch, the switch will give a cockpit indication that the main gear is locked in the retracted position. When both main gear uplock switches have been contacted, the logic circuit of the auxiliary hydraulic system will terminate hydraulic pump operation after 10 seconds.

Main Landing Gear Downlock Sensing Switches

When contacted by the internal mechanisms on the side brace actuator, the main landing gear downlock switches provide an indication in the cockpit when the main landing gear is down and locked. One switch is located within each side-brace actuator and operates in response to downlock engagement.

Weight-on-Wheels Sensing Switches

Two weight-on-wheels sensing switches are located on the nose gear and one on each main gear. They provide various systems of the aircraft with a signal that it is either in flight or on the ground. These include the thrust reverser system, nosewheel steering, airbrakes, and many others.

Gear Indicator Lights

The lights are controlled by downlock microswitches on the nose landing gear drag brace and main landing gear sidebrace actuators. If the red light in the gear handle is illuminated, it indicates that the gear is in neither a down and locked nor an up and locked position. It is referred to as an unsafe light and sometimes as a disagreement light. This is because the landing gear is not in a safe condition for landing and the handle position disagrees with the locked gear position.

At the beginning of the gear retraction cycle the three green lights in the gear selector panel will be illuminated. They are labeled LEFT, NOSE and RIGHT and indicate that the gear is down and locked. These lights are illuminated by the completion of electrical circuits when the downlock switches are contacted by the downlocks (Figure 14-11),

After take off, the weight-on-wheels switches will signal that the aircraft is in flight and a solenoid located within the landing gear selector panel will be energized, allowing the handle to be placed in the UP position. If the handle cannot be moved and gear retraction is necessary, the DOWNLOCK OVERRIDE button may be pressed to energize this solenoid. The signal that the aircraft is in flight will also deactivate the nose wheel steering so that the nose gear will automatically center prior to retraction.

When the gear handle is raised, the red light within it (the unsafe light) will illuminate, indicating that the handle is in the UP position with the gear not yet up and locked. (Figure 14-12). At approximately the same time, the three green lights in the landing gear selector panel will extinguish, indicating that the downlocks have been released. Once the gear reaches its fully retracted position, the uplocks will engage and the uplock switches will close. When all three of the landing gear are retracted and locked in the UP position, the light in the gear handle will extinguish (Figure Figure 14-13). In keeping with the concept of having no indicator lights illuminated in cruise

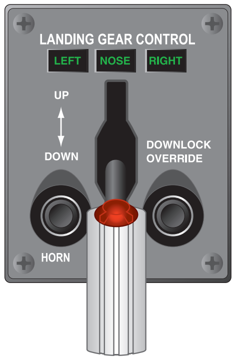


Figure 14-11. Gear Down and Locked

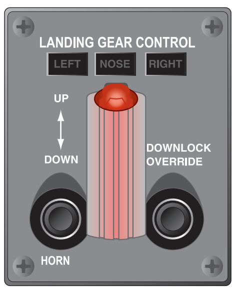


Figure 14-12. Gear in Transit

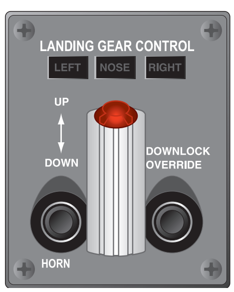


Figure 14-13. Gear Up and Locked

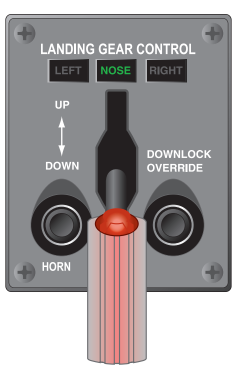


Figure 14-14. Main Gear Not Down

flight under normal circumstances, all of the landing gear indicators are extinguished if the gear is up and locked.

Ten seconds after the main gear are retracted and the uplock switches are contacted, the auxiliary hydraulic pump will stop operating. The time delay is established to fully pressurize the auxiliary hydraulic system accumulator before the pump is deenergized. Auxiliary hydraulic system pressure will indicate zero and will be presented in green on the EICAS because this is a normal flight condition.

Under certain conditions, a warning that the gear is unsafe for landing will be presented if all three of the gear are not down and locked. These warnings will occur whether the gear was inadvertently not extended or if the gear extension was unsuccessful (Figure 14-14). The landing gear position warning is tested by pressing the HORN button on the landing Gear selector panel.

The first condition occurs at any altitude on the radio altimeter if flaps are selected to 40º (anything more than 30º). The visual and aural warnings will be presented as described above.

The second condition occurs if the radio altimeter reaches an indicated altitude of 400 feet or below and either thrust lever is retarded. All of the warnings will be presented as above. The aural warning cannot be silenced, however.

The third condition occurs when either thrust lever is retarded and the airspeed decreases to below 158 KIAS (radio altimeter inoperative). The red light in the gear selector handle will illuminate, a red warning message GEAR NOT DOWN will appear on the EICAS, and an aural warning GEAR, GEAR will be announced. In this situation, the aural warning is silenced by pressing either CAUTION/WARNING switchlight on the glareshield panel. The visual warnings, however, will remain.

WHEELS AND BRAKES

The nose and main wheels provide the ability of movement for the aircraft while on the ground. The landing gear incorporates dual wheels on the nose landing gear and on each main landing gear.

The nose wheels are free to rotate with no braking system installed. The main wheels are fitted with a hydromechanical braking system controlled by the cockpit rudder pedal tips. The pedals are mechanically connected to valves that apply hydraulic pressure to the brakes in direct proportion to the brake pedal pressure.

The brake assemblies provide for normal braking, emergency braking and locked wheel parking. An antiskid system provides antiskid control, locked wheel protection, touchdown/hydroplane protection and landing gear retraction braking.

The wheel and brake system is composed of the following:

- Wheel and tire assemblies for nose and main gear
- Wheel brake assemblies on the main gear
- The normal brake system
- The antiskid system
- The emergency/parking brake system

Nose Wheels and Tires

The nose wheels and tires support the nose of the aircraft during takeoff, landing, taxiing and parking. (Figure 14-15). They are mounted on the axle fitting assembly. The aluminum alloy nose wheels are mounted with 10-ply rated high-pressure tires.

The nose wheel assembly and the tire are balanced as a unit to correct out-of-balance conditions.

The nose wheels each rotate on two tapered roller bearings. The bearings are protected against dirt, moisture, contamination and loss of lubricant by rubber bearing seals. The bearings are held in place by axle nuts and washers.

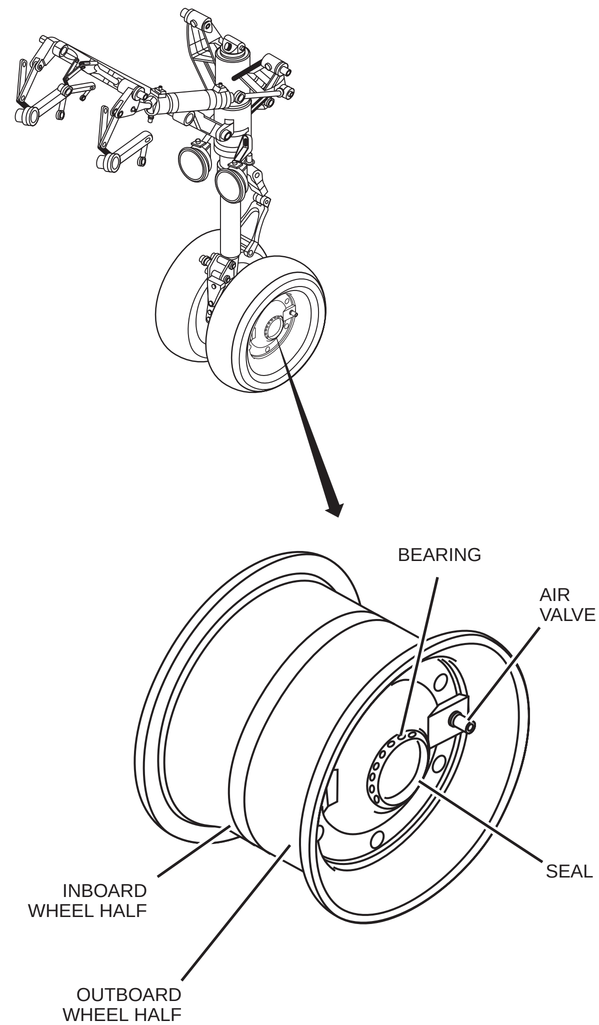


Figure 14-15. Nosewheels and Tires

Main Wheels and Tires

The main wheels and tires support the midsection of the aircraft during takeoff, landing, taxiing and parking (Figure 14-16). They are mounted on the axle fitting assemblies. Each main wheel is in two halves for tire removal and installation.

The wheel halves may be assembled in any position relative to each other and may be interchanged without rebalancing. If severe braking causes the wheel to overheat so that a tire blowout could occur, fusible plugs in the inboard wheel half melt and release tire pressure. Each main wheel assembly is fitted with a 12-ply rated, high-pressure tire.

Main Wheel Brake Assemblies

The main wheel brake assemblies are bolted to the axle fitting assemblies. The braking system has a normal hydraulic subsystem, parking/ emergency brake subsystem and an antiskid subsystem. The normal hydraulic brakes operate with main hydraulic system pressure through valves within the power brake valve assembly that are actuated by the rudder pedals. Hydraulic pressure is admitted through antiskid valves, hydraulic fuses and shuttle valves to the multi-disk brake assemblies.

Each brake consists of a housing, a one piece back plate and torque tube, a pressure plate, three rotating disks, two stationary disks, six pistons, hydraulic fittings, seals , bolts, washers and self-locking nuts. assembly (Figure14-17).

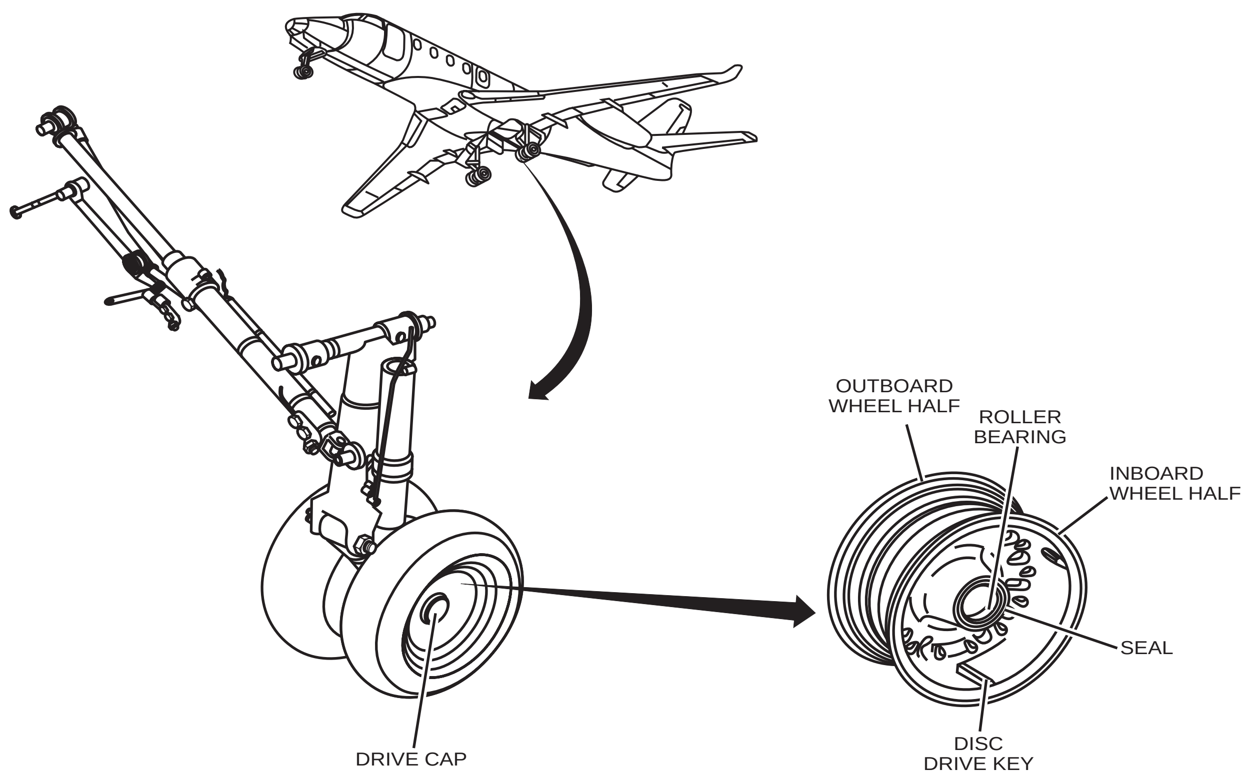


Figure 14-16. Main Wheels and Tires

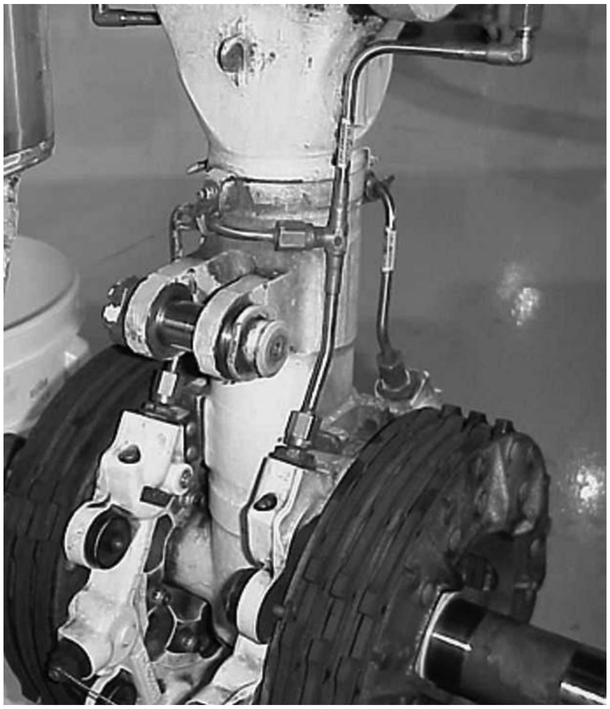


Figure 14-17. Main Wheel Brake Assembly

The brake housing contains two separate hydraulic systems. Each system activates three of the six pistons incorporated into the housing through passages interconnecting the three pistons. Each system has one inlet port and one bleeder port, which permit the brake assembly to be used interchangeably for either right or left installation.

Hydraulic pressure applied to the brake causes the pistons to contact the pressure plate and compress the disk stack against the back plate. The resisting torque caused by friction created between the rotating and stationary parts in the disk stack is transmitted to the wheel by the rotating disks. When the brake pressure is released, four return springs pull the pressure plate from the stack, forcing the pistons into the cylinders and allowing the rotating disks and wheel to turn freely.

Self-adjusting mechanisms compensate for wear in the brakes. Four self-adjusting mechanisms are installed in openings around the perimeter of the housing. Each mechanism consists of a pin, spring, retaining ring, swage tube, spring guide and spring housing. Application of the brake causes compression of the (return) spring by the pin and the swage tube. The movement of the swage in the tube compensates for brake wear. When brake pressure is released, the pressure plate is pulled back a distance equal to the preset brake clearance. The preset clearance is maintained throughout the life of the brake. Brake wear is checked during the Exterior Inspection by the flight crew. It is indicated by the position of the pins within the self-adjusting mechanisms (Figure 14-18).

Each wheel brake assembly is hydraulically divided into two sections that normally supply pressure to the brake simultaneously. However, either system is capable of supplying adequate pressure to the brake independently of the other, providing a dual system.

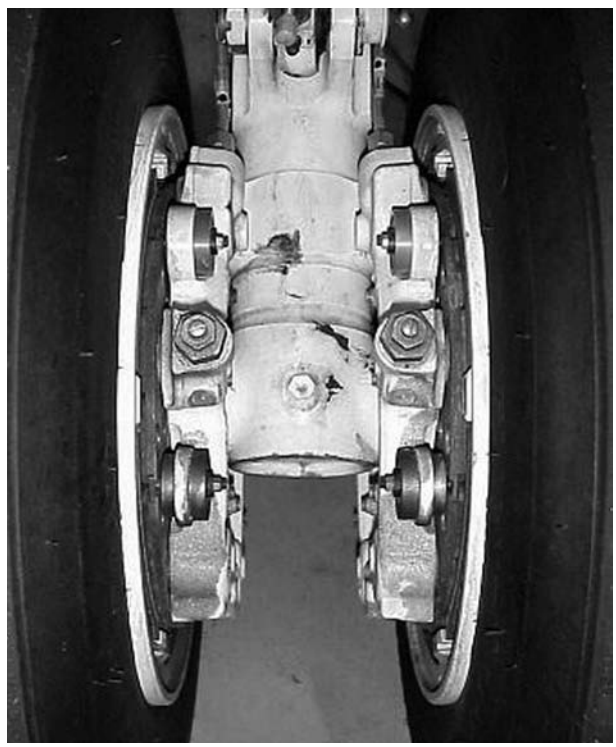


Figure 14-18. Brake Wear Indicator Pins

NORMAL BRAKE SYSTEM

The normal brake system utilizes main hydraulic system pressure to apply force to the disks in the main gear brake assemblies. An antiskid system operates in conjunction with the normal brake system to prevent wheel lockup and allow maximum braking under varied conditions.

Some components of the normal brake system are (Figure 14-19):

- Power brake valve
- Shutle valves
- Hydraulic fuses

As described below, some of these items function in the parking/emergency brake system as well.

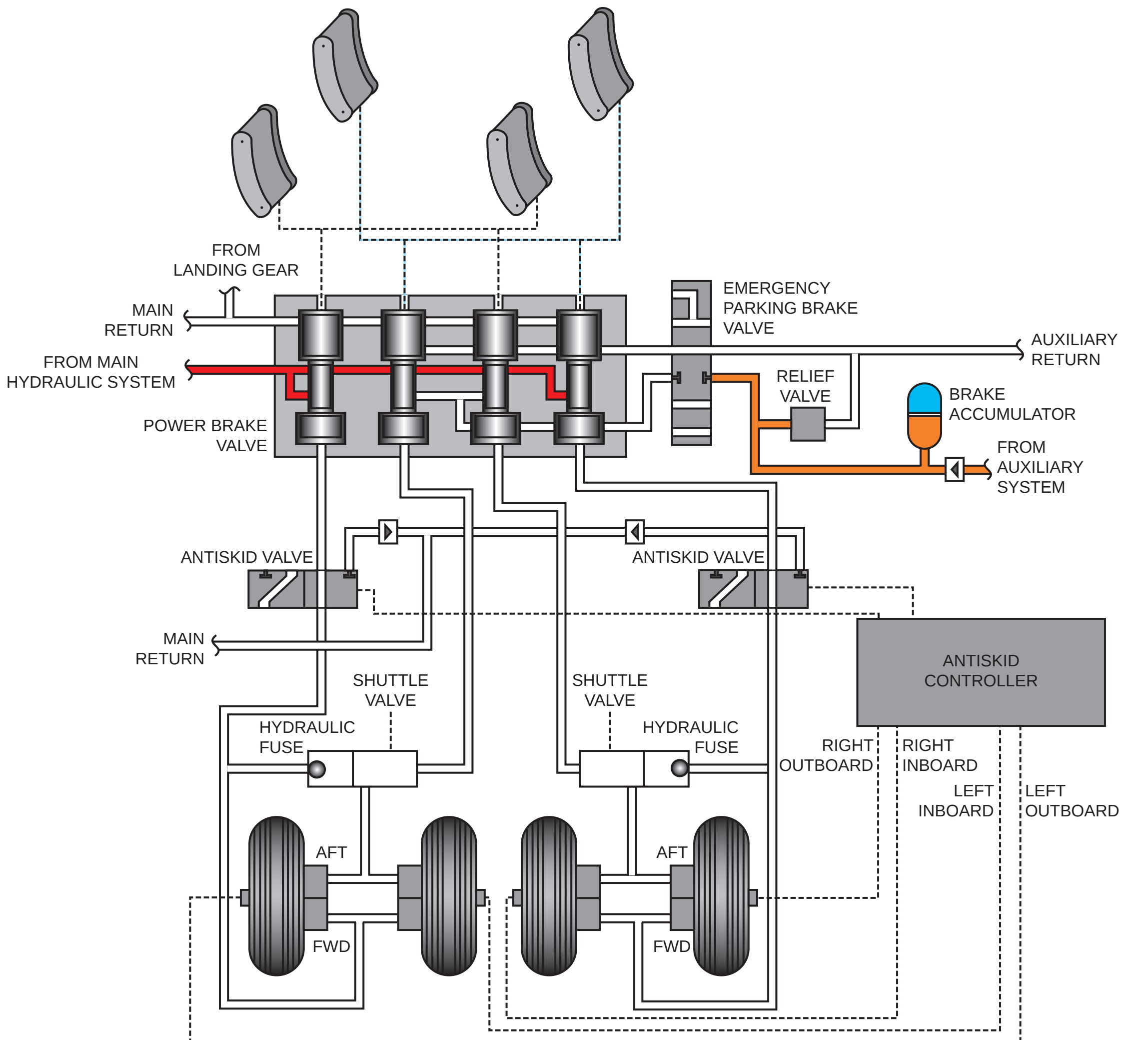


Figure 14-19. Brake System Components

Power Brake Valve

The power brake valve assembly contains valves that are manually actuated by deflection of the rudder pedal tips, metering hydraulic pressure to the brake assemblies in direct proportion to pedal deflection. Releasing toe pressure on the pedal returns the valves to neutral, releasing the brakes by venting braking pressure and returning the hydraulic fluid to the reservoir.

Shuttle Valves

Two shuttle valves separate normal and parking/emergency brake systems. The valves are automatic in operation, directing either normal or emergency pressure to the brakes, depending on which pressure is applied to it.

Hydraulic Fuses

Two hydraulic fuses, installed near the shuttle valves in the normal brake system, sense quantity flow during brake application. If flow exceeds a predetermined value, the fuse closes. This prevents excessive fluid loss in the event of a ruptured brake line or seal on a wheel brake assembly. The fuse will automatically reset when the brakes are released. These fuses are only installed on the aft hydraulic lines of each brake assembly, as the shuttle valves ensure brake availability in case of a leak in a forward line or brake.

PARKING/EMERGENCY BRAKES

The parking/emergency brake system (Figure 14-20) consists of a parking/emergency brake selector, control cable, parking/emergency brake valve, power brake valve and parking/emergency brake shuttle valve.

Brake Selector

The parking/emergency brake selector is located on the left side of the pedestal and has three operating positions: PARK, OFF and EMERG. The parking brake is set when the control handle is positioned in PARK. With the control handle in the OFF position, normal aircraft braking is selected. Emergency braking is engaged by setting the control handle to EMERG.

Control Cable

The parking/emergency brake cable provides the mechanical connection between the parking/emergency brake selector and the parking/emergency brake valve.

Brake Valve

The parking/emergency brake valve is controlled by the parking/emergency brake selector handle on the pedestal through a control cable. If the parking/emergency brake handle is selected to OFF, the parking/emergency brake valve will not allow auxiliary hydraulic system pressure to enter the power brake valve. The brakes will only respond to main system hydraulic pressure as modulated by pedal depression through the power brake valve.

If the parking/emergency brake handle is set to EMERG, auxiliary hydraulic system pressure is admitted to the power brake valve by the parking/emergency brake valve. Pressure and return pathways are opened to the auxiliary hydraulic system to allow it to substitute for the main hydraulic system for brake application. Depression of the pedals will modulate the power brake valve and control brake pressure. Because of the design of the power brake valve and associated plumbing, only the aft three pistons of each brake assembly will receive pressure during brake application.

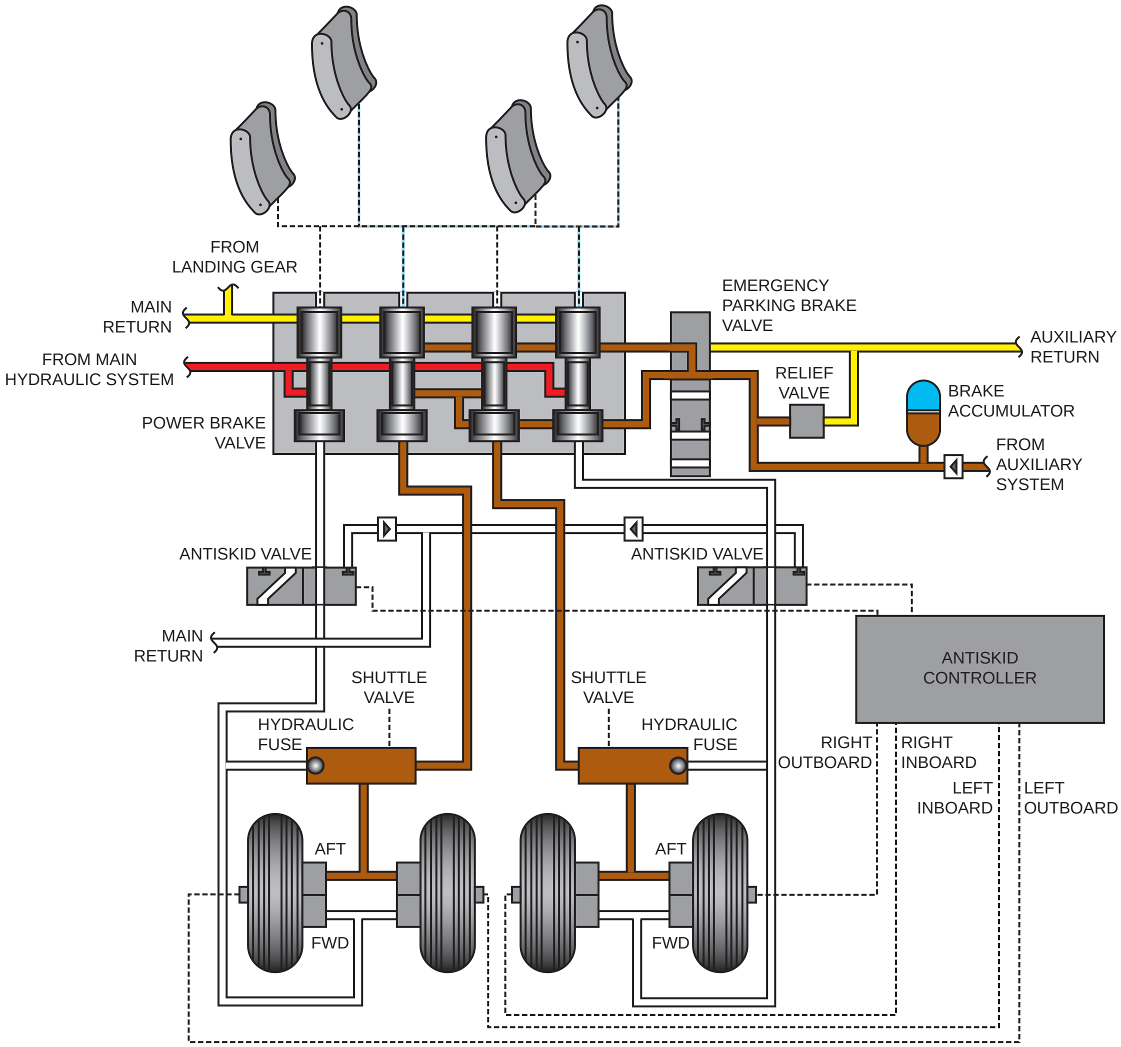


Figure 14-20. Parking Brake Operation

When the parking/emergency brake handle is selected to PARK, both the supply and return ports of the power brake valve are supplied with auxiliary hydraulic system pressure by the parking/emergency brake valve. This eliminates the modulating function of the power brake valve and supplies full auxiliary hydraulic system pressure the aft three pistons of each brake assembly.

Parking/Emergency Brake Shuttle Valves

In the event of a main hydraulic system failure and with the parking/emergency brakehandle in the EMERG or PARK positions, auxiliary hydraulic pressure is directed through shuttle valves into two of the four hydraulic blocks on the brakes on each main landing gear. This allows pressure to be applied to three of the six brake pistons on each wheel brake. Operation of the shuttle valves occurs automatically with selection of the emergency brake system.

ANTISKID SYSTEM

Maximum braking effectiveness is obtained when all wheels decelerate at a maximum rate without skidding. The antiskid system prevents wheel skids by automatically limiting application of pressure to the brakes, resulting in a shorter landing roll with minimum tire wear.

In flight and until ground contact is established, the antiskid system is inoperative (Figure 14-21). With both main gear down and locked and an ANTI SKID switchlight pressed ON, the antiskid system is operable. Electrical power for the system is from the No. 1 and No. 2 distribution buses through the left and right ANTI SKID circuit breakers. The system is controlled by two ANTI SKID switchlights on the glareshield, one for each pilot. The last switch that has been pressed has control over the system. If the pilot's switch has been pressed to deactivate the system, the pilot's switch will have to be pressed in order to reactivate it. The copilot switch will not affect the system in this case. All ANTISKID switchlights and the left and right OFF annunciators are illuminated when either switchlight has been pressed off. If the antiskid system is operating normally, there will be no lights illuminated.

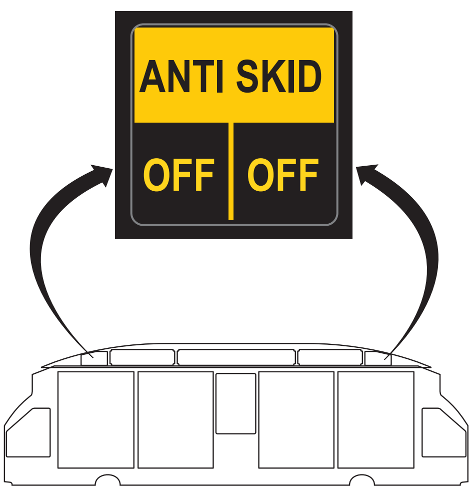


Figure 14-21. ANTISKID Switchlight

The antiskid system includes the following:

- Antiskid control unit
- Antiskid valves
- Wheel speed sensors

Antiskid Control Unit

The antiskid control unit compares wheel speed information and regulates brake application with the antiskid valves. Inside the control unit are two identical plug-in circuit boards. One board contains the control and monitoring circuitry for the two left gear wheels while the other board serves the two right gear wheels.

Each board has its own power source. A loss of power to one side does not disable the other side. The left side receives power from the No. 1 distribution bus, and the right side is powered from the No. 2 distribution bus. Either bus can power both OFF circuits, so a bus failure does not disable the warning circuits.

Antiskid Valves

Two antiskid valves are incorporated into the brake system plumbing upstream of the hydraulic fuses, one for each main gear. The valves are controlled by electrical signals from the anti-skid control unit, which in turn receives signals from speed sensors in each wheel. During normal braking, the valves have no function. During antiskid braking, the valves modulate, reducing brake pressure to prevent wheel skidding. When an antiskid valve operates, it ports hydraulic pressure away from the brake assemblies and sends the return fluid back to the hydraulic reservoir.

Wheel Speed Sensors

Four wheel speed sensors, one for each wheel, provide wheel speed information to the anti-skid control unit.

A wheel speed sensor in the axle of each main gear wheel is driven by a cap mounted on the wheel hub. The sensor generates electrical signals relative to wheel speed, providing that information to the antiskid control unit for modulation of the antiskid valves as required.

NOSEWHEEL STEERING

Nosewheel steering is controlled by the tiller and/or the rudder pedal and powered by two hydraulic actuators which are controlled and monitored by an electro-hydraulic servo system. The system is controlled by electrical signals only, with no mechanical interface. The mechanical commands are converted to electrical signals and processed through the electronic control unit (ECU).

The nose wheel steering system is composed of the following:

- Tiller assembly
- Toggle switch
- Rudder pedal transducer
- Steering control valve
- Steering actuators
- Electronic control unit

Tiller Assembly

Steering control inputs originate from the tiller assembly or from either set of rudder pedals. The tiller assembly allows the pilot to control the steering angle 60° left or right of center while the rudder pedals provide a steering angle of up to 3° either direction from the centerline of the aircraft.

Pressing the PEDALS DISC button in the center of the tiller assembly disables rudder pedal steering capability. When not in active steering mode, the tiller is returned to the neutral or straight ahead position by a centering spring and cam when it is released.

Toggle Switch

When the nose wheel steering toggle switch is placed in the NWS CONNECT position, the steer-by-wire system uses inputs from the tiller assembly to allow the pilot nose wheel steering capability. Selecting DISCONNECT the nose wheel return passive mode, responding to the tracking forces of differential braking or engine power settings. Shimmy damping is provided by trapped hydraulic fluid in the system and hydraulic flow restrictors when the system is not powered.

Rudder Pedal Transducer

Steering the nose wheel with rudder pedal commands is functionally identical to operation of the tiller. A rudder pedal transducer is located on a connector rod beneath the cockpit floor on the copilot side. Displacement of the rudder pedals left or right prompts the transducer to signal the steering command to the ECU.

Steering Control Valve

The steering control valve is an electro-hydraulic unit. It controls steering by directing hydraulic pressure to the hydraulic actuators in the required direction. The steering control valve is mounted on the strut and is controlled from the flight compartment by a steering tiller and the rudder pedals through an ECU to provide powered directional control of the aircraft.

Steering Actuators

The nose landing gear is steered on the ground by means of two hydraulic actuators that push or pull the steering collar on top of the nose landing gear strut.

Electronic Control Unit

The nose wheel steering system is an electro-hydraulic servo system controlled by the ECU. The ECU receives electrical power if the nose landing gear is down and locked and the nose wheel steering toggle switch is in the NWS CONNECT position. The ECU consists of three main independent functional circuits:

- Two identical control channel circuits
- One monitoring channel circuit

The control channels perform closed loop steering control. The monitoring channel performs continuous system protection. The monitoring function is automatically self-tested whenever the nose wheel steering is switched on, by means of a built-in-test (BIT). Nose wheel steering ECU control and monitoring sections are totally separate circuits and they do not have common components. No single point or common mode failure will lead to failure of the functional circuits along with its protecting system, which guarantees fail-safe operation.

OPERATION

LANDING GEAR NORMAL OPERATION

The landing gear normal extension and retraction system uses main system hydraulic pressure for movement of the landing gear to either the up or down position. The emergency extension system provides nitrogen pressure for landing gear extension in the event of hydraulic pressure failure. The landing gear cannot be retracted if hydraulic pressure is lost.

Retraction

During retraction, when the landing gear selector is set to UP, it actuates the landing gear selector valve to admit pressure from the main hydraulic system to the main and nose landing gear actuators and the nose landing gear downlock cylinder.

Mechanical downlocks in the main gear actuators and in the nose gear drag brace are hydraulically disengaged. This hydraulic pressure is maintained within the downlocks whenever the gear is selected UP and main system hydraulic pressure is available. All three landing gear assemblies then retract and are held in place by constantly applied hydraulic pressure within the actuators. The uplocks (one for the nose gear and two for each main gear) automatically engage by spring force when each gear assembly reaches its fully retracted position. These uplocks keep the gear retracted in the event of main hydraulic system pressure loss.

During the retraction process, all of the gear doors will open as necessary to allow the landing gear to retract.

As the gear is fully retracted, all of the doors close over their respective gear assemblies. Each door is mechanically actuated by linkage to the respective gear retraction mechanism. Return hydraulic pressure is also applied to the brakes to stop wheel rotation during retraction.

Extension

In normal hydraulic extension when the landing gear selector is set to DOWN, it actuates the selector valve to admit pressure to the main and nose uplock cylinders and to the main and nose gear actuators. The uplock cylinders release the main landing gear inboard doors and nose landing gear strut and the landing gear actuators extend the landing gear to the down and locked position. Hydraulic pressure is continuously applied to the uplock actuators whenever the gear is selected DOWN and main hydraulic system pressure is available. The downlocks within each main gear actuator and the nose gear drag brace automatically engage when the gear is fully extended (Figure 14-22).

During extension, the two outer main gear doors will open in conjunction with the extension of the main gear assemblies. They remain fully open with the gear down. The main gear inner doors will open as the actuators move the gear assemblies to their extended position. As the tires and wheels pass the open doors, the inner doors will start to reclose. When the main gear is fully extended, the inner doors will remain partially open.

The nose gear doors will operate in response to movement of the drag brace. The forward doors will open for gear extension and then fully close once the gear is down. The aft doors will open fully as the gear extends and will remain open once the gear is down.

LANDING GEAR EMERGENCY EXTENSION

When emergency extension is required, the landing gear selector should be set in the DOWN position. Actuation of the emergency landing gear extension handle causes the emergency gear selector valve to open. This distributes the stored high pressure nitrogen to the four main landing gear uplock actuators, two main landing gear side-brace actuators, nose landing gear uplock actuator, nose landing gear actuator and dump valve.

Nitrogen is directed to the dump valve which allows it to provide a return path for the hydraulic fluid from the gear actuators to the reservoir. This return flow bypasses the main selector valve, allowing emergency extension even if the gear control handle is jammed or the main selector valve has malfunctioned. However, if possible, the handle should be placed in the DOWN position and remain there

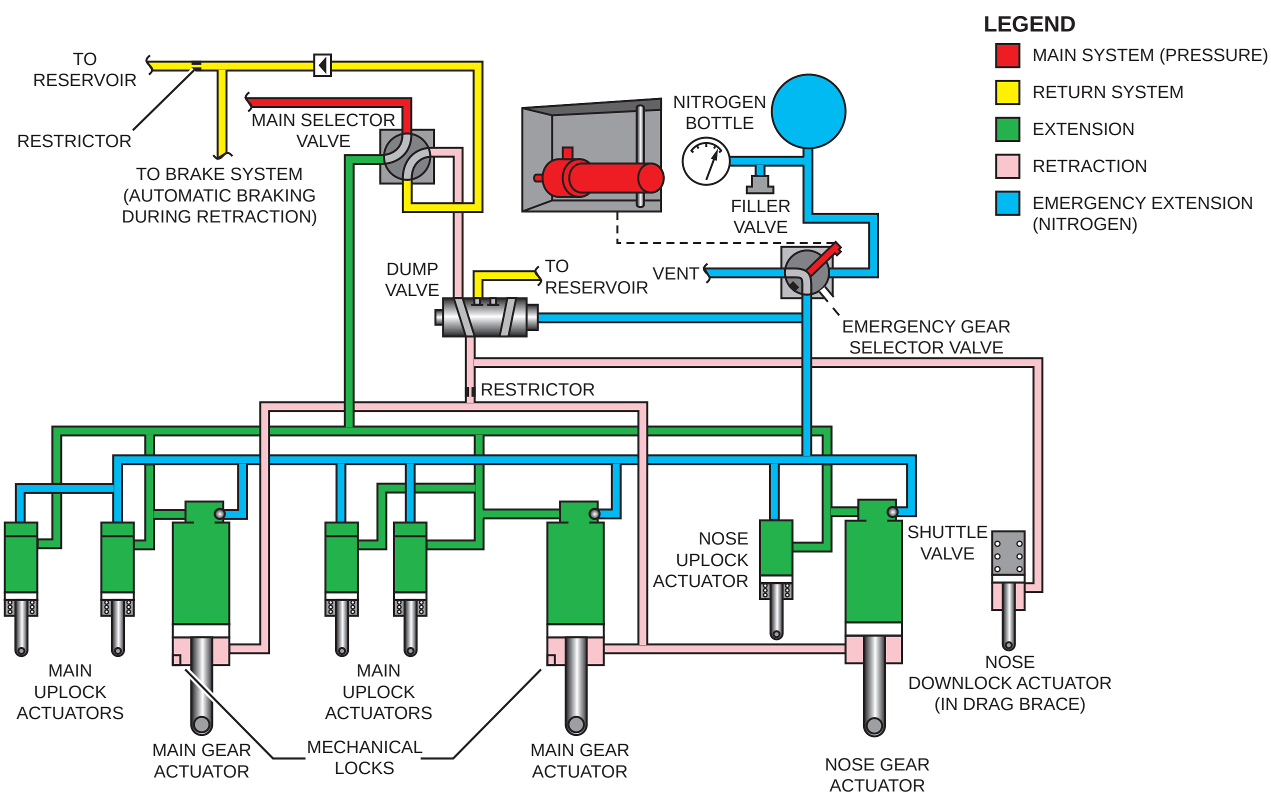


Figure 14-22. Landing Gear Extension

to prevent an inadvertent disengagement of the downlocks after the extension is complete.

Shuttle valves and floating pistons are used in the emergency extension system to keep the hydraulic system and nitrogen blowdown system separated. The uplock actuators are equipped with floating pistons. These act upon hydraulic pistons within the actuators when the nitrogen pressure is applied, which releases the uplocks. Shuttle valves for each gear actuator allow nitrogen pressure to extend the gear while also preventing it from escaping through the hydraulic system.

LANDING GEAR POSITION AND WARNING

On the ground, with main hydraulic system pressure applied, an electrical circuit is completed through a pressure switch and the nose gear ground contact switch to energize the bypass valve closed. (Figure14-23).

Command signals from the pedal and tiller transducers are sent by the control channels to the ECU. The resultant command is compared with the signal from the feedback potentiometer installed on the steering actuator assembly that is being received through the monitoring channel. If the command signal differs from the feedback signal, hydraulic pressure will be applied to the appropriate steering actuator until the signals are in agreement.

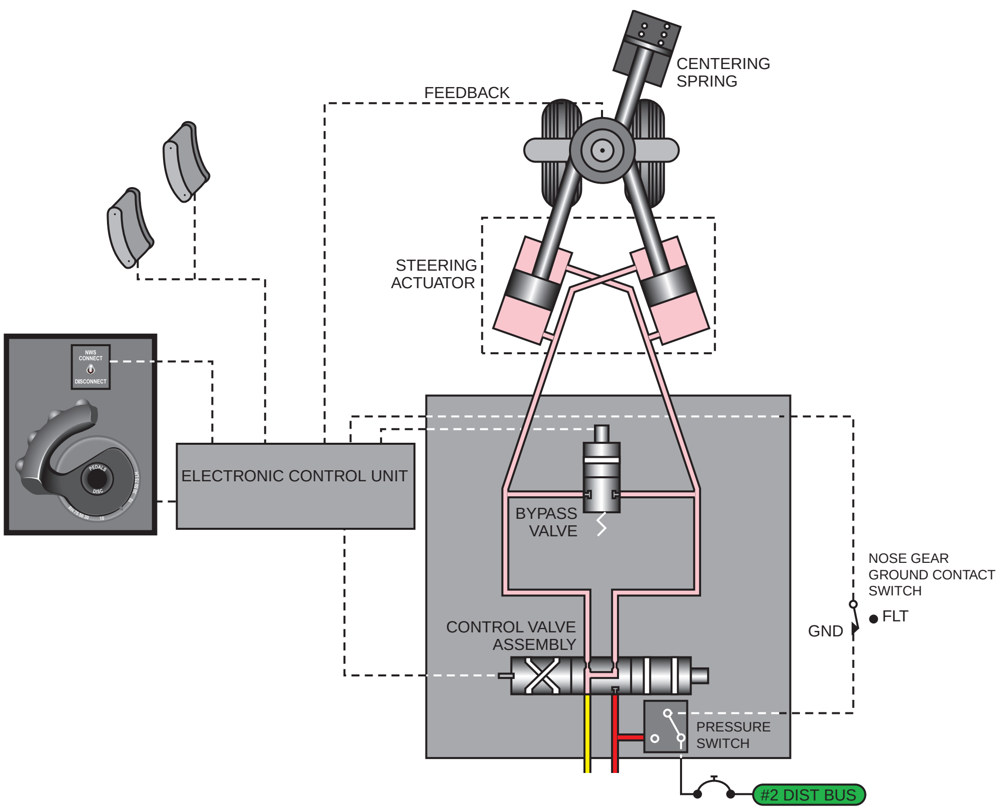


Figure 14-23. Nosewheel Steering System

The rate of steering is controlled by the amount of rotation force applied to the tiller. Low rotation force produces slower steering than high rotation force. When steering is not being actuated, flow through restrictors in the control valve damps nose wheel shimmy.

During touchdown and taxi, the nose landing gear WOW switches trigger the time circuit of the ECU which generates a three second time delay. During this period the nose wheel is centered. After the three second time delay, the closed loop system will start to operate in its active steering mode, enabling the nose wheel steering system.

During takeoff the nose wheel steering control system operation is similar to the landing process, but in the opposite sequence. The active steering mode is switched off and the nose gear automatically centers immediately after lift-off, as signaled by the nose landing gear WOW switches.

Loss of main hydraulic system pressure actuates a pressure switch, opening the circuit to the bypass valve, which spring-loads open. This hydraulically connects both steering actuators. The aircraft can then be steered by differential braking.

To extend the gear, the selector handle is placed in the DOWN position. The red light in the handle will once again illuminate. When the gear is down and locked, the associated green annunciator on the landing gear Selector Panel will illuminate, indicating LEFT, NOSE or RIGHT. The red light in the handle will extinguish when gear position matches handle position. If an emergency landing gear extension is performed successfully, the indications will remain the same.

Once the aircraft lands, the weight-on-wheels switches will signal that the aircraft is on the ground. This will allow the nose wheel steering, ground airbrakes, thrust reversers and several other systems to be enabled.

The solenoid in the landing gear selector panel will also deenergize, allowing spring pressure to return the locking mechanism to its locked position.

NORMAL BRAKING—ANTISKID

When main hydraulic system pressure is available, this pressure is metered through the power brake valve to the wheel brake assemblies The hydraulic pressure is routed through the anti-skid valves, hydraulic fuses and shuttle valves before reaching the wheel brakes (Figure14-24).

At wheel spin-up during landing or take off, each circuit board inside the antiskid control unit receives an electrical signal from the two wheel speed sensors on the main wheels it is controlling. At a wheel speed above 35 knots, signals from the wheel-driven speed sensors are matched with a reference rate in the antiskid control unit.

If sufficient brake pressure is applied to induce a tire skid, a difference in rotation speed develops between wheels. The difference is sensed by the antiskid control unit which then signals the applicable antiskid control valve to reduce brake pressure to slightly below the skid level. This is accomplished by porting some of the hydraulic pressure away from the brake assemblies and returning the hydraulic

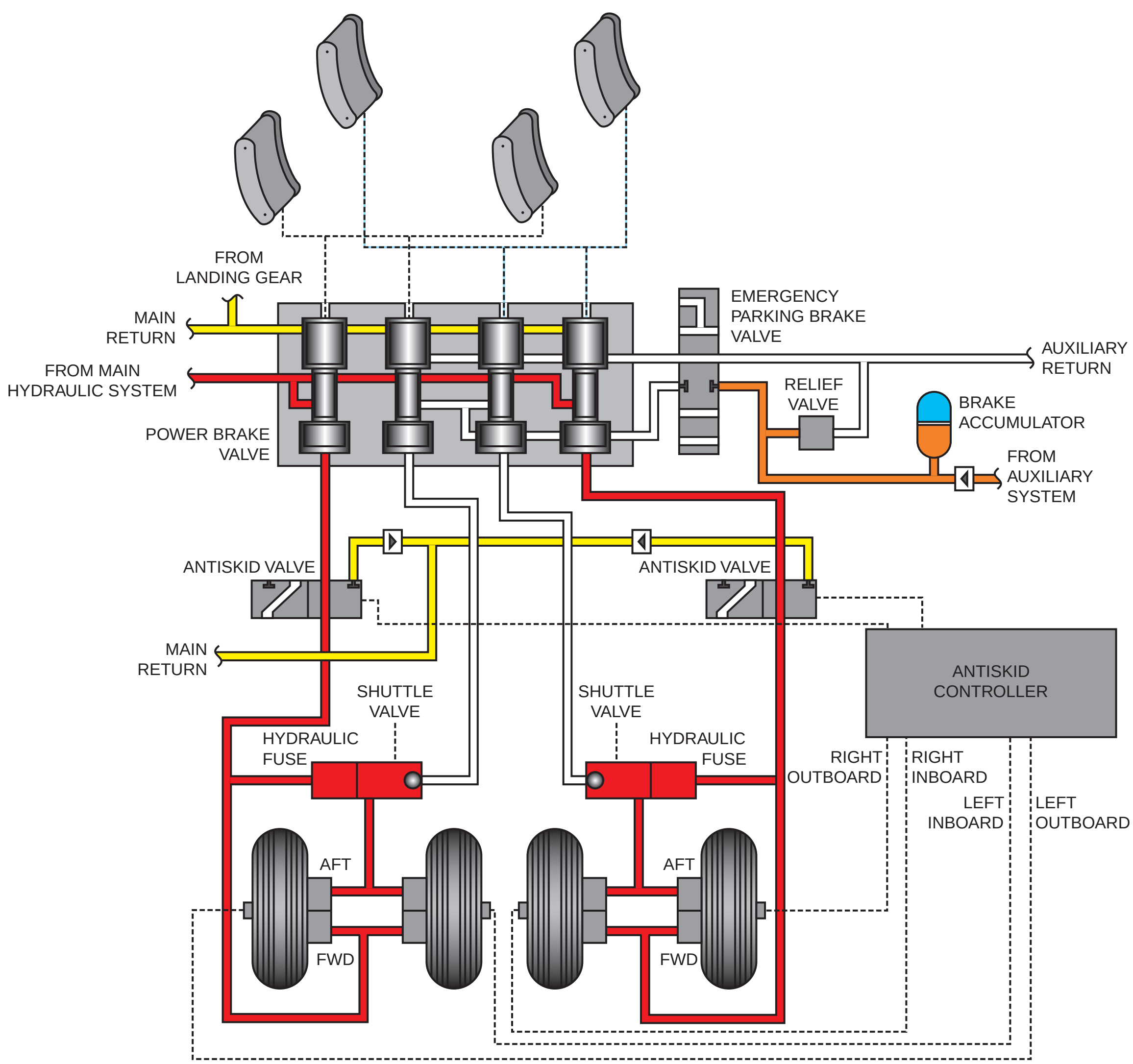


Figure 14-24. Normal Brake Operation

fluid to the main reservoir. There is one antiskid valve for each main landing gear. Therefore, if one wheel enters a skid, brake pressure to both wheels on that main gear will be reduced simultaneously. Once wheel rotation speed is within the parameters of the antiskid controller logic, the antiskid valves will cycle back to normal, allowing brake pressure to increase once again.

Brake pressure is cyclic when the antiskid system is utilized, reducing to slightly below skid level, then building again to incipient skid level, and then reducing again. In this way, maximum effective braking is maintained under all runway conditions.

Locked Wheel Protection

Unusual runway conditions such as hydroplaning can sometimes cause deep wheel skids to occur. To compound the problem, wheel spin-up forces may be virtually nonexistent, so the affected wheel will spin up very slowly even with all the braking pressure removed. In extreme skidding situations, when most wheel rotation is gone, the control unit will cause the control valves to dump all hydraulic pressure to release the brakes. Locked wheel protection removes brake pressure on both paired wheels when the wheel speed on one or both of the wheels drops below 30% of the aircraft groundspeed (such as in tire burst). This protection is available when the ground speed is 30 knots or greater.

Low-Speed Dropout

The system is designed to remove antiskid control when the wheel velocity is below 10 knots. Since this is a normal feature of the system, no INOP lights illuminate. When wheel speed increases above 10 knots, the antiskid system is automatically armed if it has been selected to the ON position.

Touchdown Protection

To prevent landing with the brakes applied, the system incorporates a touchdown feature that removes all brake pressure until the aircraft is firmly on the ground. The touch-down mode is terminated 2.5 to 3.5 seconds after making ground contact with either main gear or when wheel spin-up above 35 knots occurs. This system will not prevent wheel lockup if brakes are applied by the emergency brake or parking brake systems prior to touchdown.

OPERATION—WITHOUT ANTISKID

If the antiskid system is inoperative or deactivated, the brake system works in the normal manner except that no automatic reduction in brake pressure will occur if a skid occurs. Brake pressure will be proportional to the amount that the brake pedals are depressed. Differential braking is available to assist in directional control, if necessary.

If antiskid protection is not available, careful application of the wheel brakes by the flight crew is required in order to prevent a wheel skid. If a skid or lockup occurs, the pilot or copilot must reduce the amount of toe pressure that is applied to the pedals to release brake pressure and allow the wheels to turn at an appropriate speed.

PARKING/EMERGENCY BRAKE OPERATION

Parking Brakes

The parking brakes are set by moving the parking/emergency brake handle to PARK. This positions the parking/emergency brake valve to apply auxiliary hydraulic system pressure, either from the hydraulic pump or the accumulator, through the power brake valve to the shuttle valves and three of the six pistons in each brake assembly. Pedals do not have to be applied to engage the parking brake. The parking brake is either fully engaged or disengaged, brake pressure is not modulated within the parking brake system (see Figure 14-20).

Returning the parking/emergency brake handle to OFF permits return flow through the power brake valve and the parking/emergency brake valve to the auxiliary hydraulic system return, releasing the brakes.

Emergency Braking

If the main hydraulic system fails, emergency braking is accomplished using pressure from the auxiliary hydraulic system and is initiated by moving the parking/emergency brake handle to the EMERG position. This causes the parking/emergency brake valve to direct auxiliary hydraulic system pressure to the power brake valve, as described above. Depressing the brake pedals then actuates the Power brake valve to meter pressure, in proportion to pedal deflection, through shuttle valves to three pistons in each brake assembly. Reduced braking efficiency should be anticipated. The hydraulic fuses are not in the flow circuit during emergency braking. As the pedals are relaxed, pressure from the brakes returns through the shuttle valves and the power brake valve to return, releasing the brakes (Figure14-25).

If the auxiliary hydraulic pump is inoperative and the accumulator is being used for emergency braking, braking should be accomplished with one steady, constant application of the pedals. Pumping depletes the accumulator because hydraulic fluid is returned to the auxiliary hydraulic reservoir, not the accumulator.

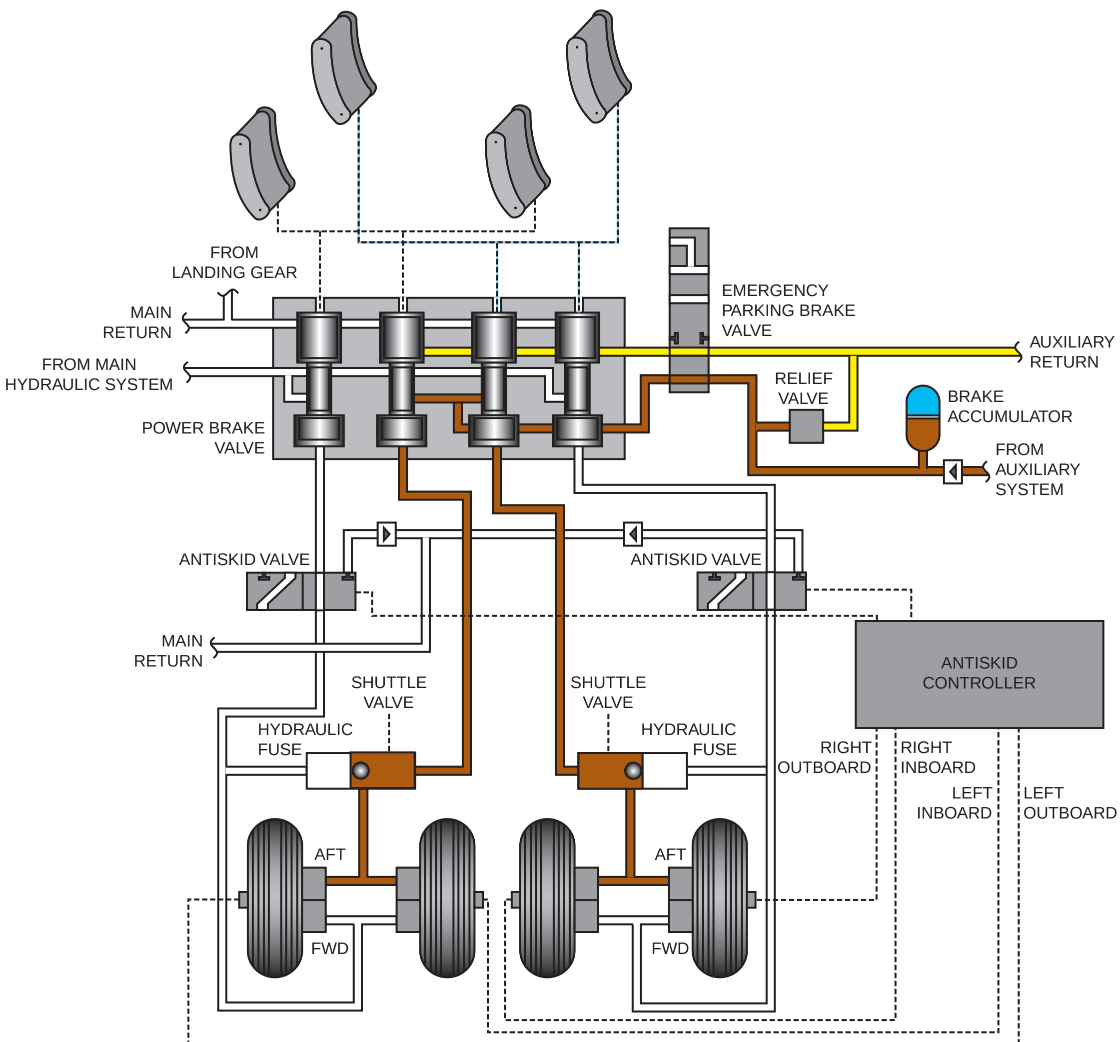


Figure 14-25. Emergency Brake Operation

The antiskid system does not function during emergency braking. Although the antiskid valves will modulate, they do not reduce brake pressure because the valves are only installed in the normal brake system hydraulic lines. However, differential and portional braking are available to assist with directional and speed control.

LIMITATIONS

The limitations contained in section one of the *Airplane Flight Manual (AFM)* must be complied with regardless of the type of operation.

The maximum speed which is safe to operate the landing gear or fly with the landing gear extended (V_{LO}/V_{LE}) is 180 KIAS.

NOTE

Do not operate landing gear above 20,000 feet.

EMERGENCY AND ABNORMAL PROCEDURES

EICAS INDICATIONS

Following is a ready-reference guide which briefly outlines the EICAS message meanings and corrective action for associated messages.

Messages having 🔈 symbol are accompanied by an aural alert.

Warning Messages

CONFIG PARKING 🔈 —The aircraft is on the ground with both engines beyond 70% N_1 thrust and the parking brake is engaged.

In order to extinguish the message, configure aircraft as required.

GEAR NOT DOWN 🔈 —The landing gear is not down and locked with a radio altimeter altitude less than 400 ft and one thrust lever at or below maximum cruise or flaps position more than 30º.

Verify landing gear position and configure the aircraft.

Caution Messages

Caution messages display as amber EICAS messages on the MFD, and are accompanied by a MASTER CAUTION light (Figure 14-26).

GND BRK WOW MISCOMP—Ground brakes weight on wheels switches miscompare.

NWS INOP—The nosewheel is down and locked and the nosewheel steering system is either off or not aligned. To correct the problem check and/or reset the NWS CONNECT/DISCONNECT switch. If system stays off, use differential braking to steer the aircraft.

Advisory Messages

PARKING BRAKE ON—If the message is green, the aux hydraulic system is supplying pressure to the brake system and adequate pressure is sensed at the brake.

Figure 14-26. Landing Gear and Brakes EICAS Indications

QUESTIONS

1. The main landing gear downlock is:
A. Mechanically engaged, hydraulically disengaged
B. Hydraulically engaged, mechanically disengaged
C. Hydraulically engaged, and disengaged
D. Mechanically engaged and disengaged

2. The nose landing gear uplock is:
A. Hydraulically engaged and disengaged
B. Hydraulically engaged, mechanically disengaged
C. Mechanically engaged, hydraulically disengaged
D. Hydraulically engaged, pneumatically disengaged

3. With the aircraft fully loaded, the minimum visible chromed piston on the struts should be:
A. 2 inches, all struts
B. 1.6 inches, all struts
C. 1.6 inches (nose), 2 inches (mains)
D. 2 inches (nose), 1.6 inches (mains)

4. The landing gear aural warning GEAR NOT DOWN sounds when:
A. One or more gear are not down and locked
B. One or both power levers are retarded to IDLE
C. Radio altitude is below 400 feet
D. All of the above

5. During landing gear emergency extension, pneumatic pressure from the gear blow down bottle:
A. Releases the uplocks
B. Extends the landing gear
C. Actuates the dump valve
D. All of the above

CHAPTER 15
FLIGHT CONTROLS

CONTENTS

ILLUSTRATIONS

CHAPTER 15
FLIGHT CONTROLS

INTRODUCTION

This chapter covers general information about the G150 flight control system.

GENERAL

The Gulfstream G150 flight control system consists of primary and secondary flight controls. Primary flight control of the aircraft is provided by aileron, elevator and rudder control surfaces. The elevator and rudder control surfaces are mechanically operated. The aileron control surfaces are hydraulically boosted with mechanical inputs.

Secondary flight control of the aircraft is provided by wing trailing edge flaps, leading edge slats and airbrakes (two inboard and two outboard per wing) (Figure 15-1).

SYSTEM DESCRIPTION

PRIMARY FLIGHT CONTROLS

Control Columns

Each control column mounts a control wheel that transmits the pilot input forces for operation of the elevators and ailerons. Mechanical linkage transmits the input forces to the appropriate control surfaces.

Rotation of either control wheel moves the aileron control surfaces to provide roll control. Hydraulic assistance reduces control force. Forward or aft movement of the control column moves the elevator control surfaces to provide pitch control.

Rudder Pedals

The left and right rudder pedal assemblies are interconnected by mechanical linkage that transfers pilot inputs to the rudder control system. Force applied to a rudder pedal is transmitted through the linkage, causing the rudder to deflect. An independent rudder pedal adjustment mechanism is provided at each pedal assembly to facilitate fore and aft adjustment for each pilot.

Ailerons

Two ailerons are installed for roll control. Each aileron is mounted on two outer hinges and one double center hinge attached to the wing rear spar. The center hinge is also the attachment for the aileron servoactuator.

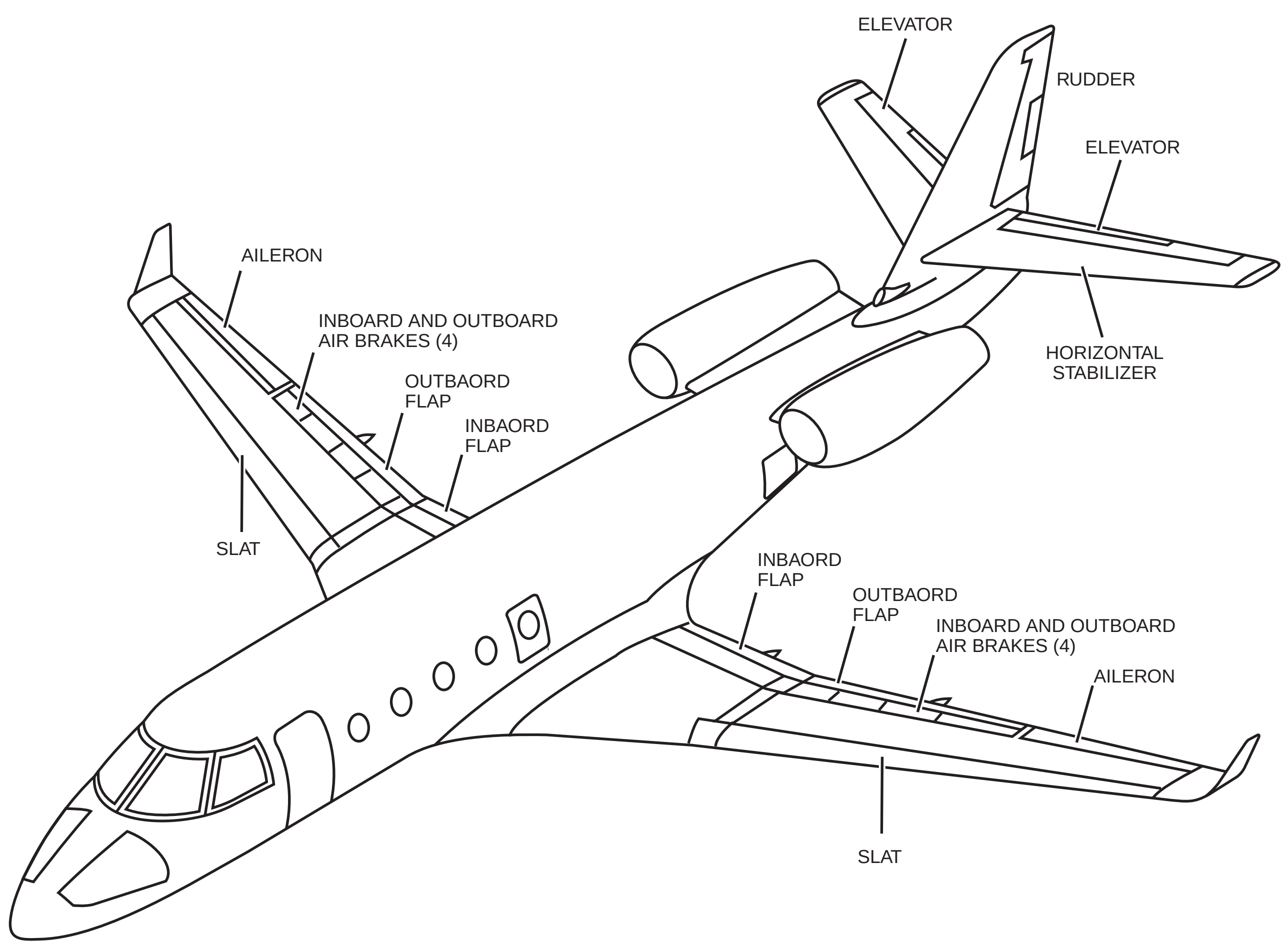


Figure 15-1. Flight Controls

Travel limits are set by mechanical stops on the wing structure. The aileron control linkage (Figure 15-2) incorporates an autopilot actuator with disengaging clutch to prevent the system from jamming during an autopilot actuator failure, as well as an artificial feel unit and an aileron trim actuator.

Servoactuators

The aileron servoactuators (Figure 15-3) provide hydraulic assistance to reduce the amount of control wheel force necessary for roll control of the aircraft. The servoactuators are separately and simultaneously pressurized by both the main and the auxiliary hydraulic systems.

Each servoactuator is a dual hydro-mechanical moving-body type and is comprised of:

Two cylinder actuators—Each actuator incorporates a piston rod (travel limit) striker. The cylinder installed on the inboard side includes two mechanical stops to limit movement of the input lever mechanism.

Two spool main control valves (MCVs)—The valves control the hydraulic flow in each respective cylinder. The actuating levers of the control valves are mechanically interconnected and actuated by the pilot through the control linkage.

The MCVs are linked by a lever to an external centering spring. This causes centering of the aileron surface in case crew control input is disconnected.

Each MCV spool is centered by an internal spring. In the centered position, the valve allows a restricted interflow between the two sides of the piston, thus preventing jamming of the actuator if one spool separates from the input lever.

A microswitch is installed between the two ends of the piston rods of each servoactuator. This activates an AILERON FAIL caution message on the EICAS display and causes the auxiliary hydraulic pump to operate whenever there is unequal movement of the piston rods. In this case, auxiliary hydraulic pump operation is independent of pump switch position.

If there is a total loss of hydraulic pressure to the actuator, bypass valves bridge the pistons and the actuator functions as a rigid part of the flight control mechanism operation system, enabling direct manual operation of the aileron.

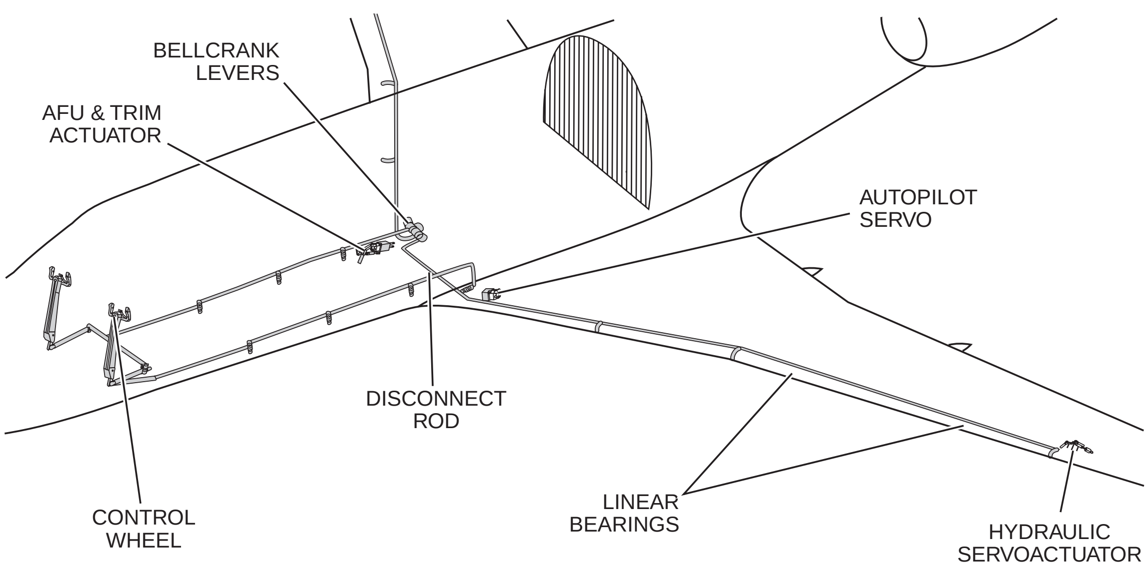


Figure 15-2. Aileron System

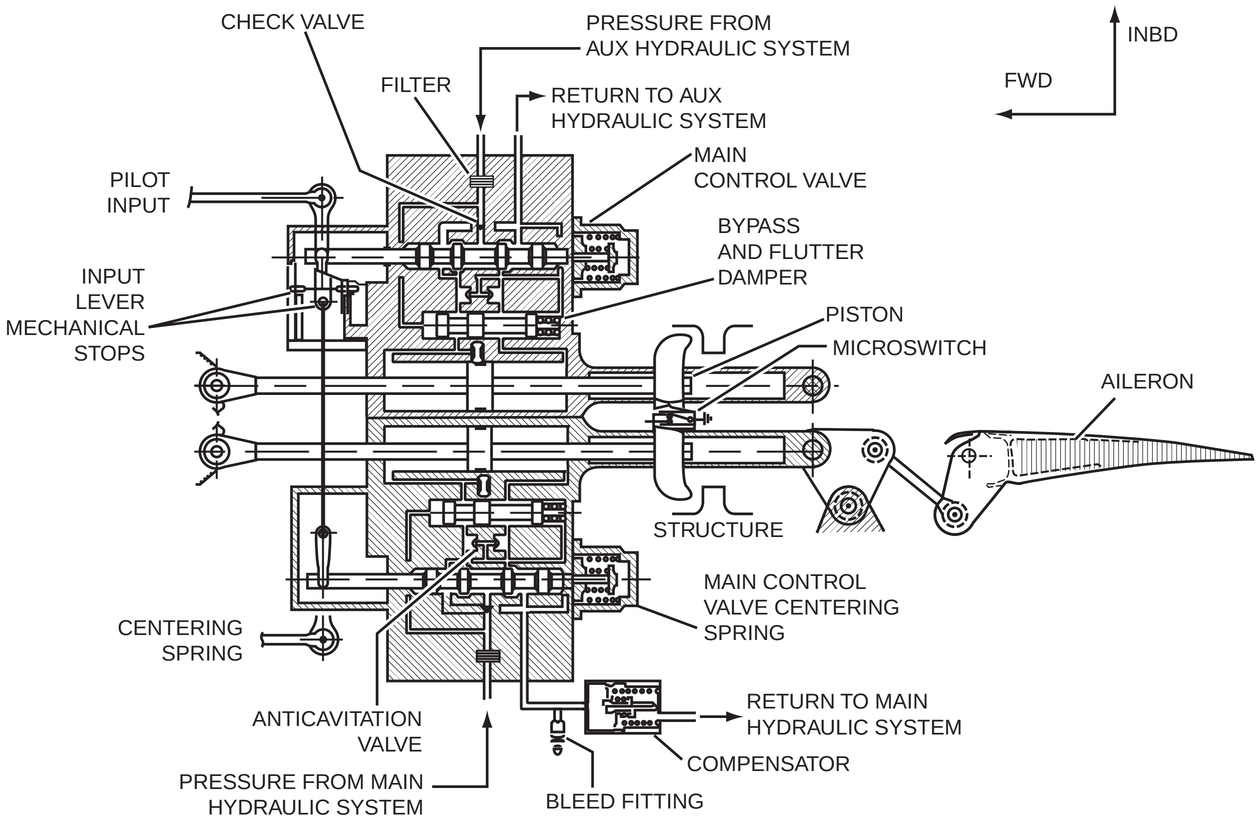


Figure 15-3. Aileron Servoactuator

Artificial Feel Unit

Because of the hydraulic servoactuators, aerodynamic forces are not transmitted back to the control wheels from the ailerons to provide the pilot with tactile reference as to how much aileron deflection is occurring.

The artificial feel unit (AFU) provides this feedback, using physical tension and compression of springs to simulate aerodynamic forces. The AFU is installed in the control linkage to the right aileron assembly in the lower part of the fuselage.

Aileron Trim Actuator

The aileron trim actuator relieves control loads by shifting the neutral position of the artificial feel unit. This is accomplished through the use of an electric motor that is controlled by a three-position electric trim switch in the cockpit.

The aileron trim actuator features limit switches to stop motor operation when it reaches either end of its operating travel. An internal brake is applied whenever the actuator control switch is released.

Interconnect Rod

The interconnect rod connects the left and right assemblies of the aileron control system. If one aileron control becomes jammed, the connection between left and right aileron systems can be broken by the pilot and copilot simultaneously rotating their respective control wheels inboard, applying sufficient pressure on the system to break the interconnecting rod. The unaffected aileron can then operate independently. If the left aileron is operable, the autopilot is available. If the right aileron is operable, the artificial feel and aileron trim are available.

Rudder

Rudder control is mechanical, operated by either pilot through the rudder pedals (Figure 15-4). Control rods, bellcrank levers, linkages and torque shafts convey mechanical inputs from the rudder pedals to the rudder. The rudder control linkage incorporates an autopilot actuator with disengaging clutch to prevent the system from jamming during actuator failure.

The rudder control surface incorporates a single tab (Figure 15-5) which operates as both a servo tab and a trim tab. When the rudder is deflected from centerline, the geometry of its attachment linkage causes the tab to deflect 1º in the opposite direction for every 3º of rudder movement. This aerodynamically assists the pilot and reduces the amount of pressure required at the rudder pedal to deflect the rudder.

In addition to its servo action, the tab can be displaced as a trim tab by two electric actuators interconnected by a flexible shaft. The actuators are installed in the upper and lower portions of the rudder structure and are attached at the forward end to the rear spar of the vertical stabilizer. The upper actuator incorporates a potentiometer to provide trim position information on the MFD.

Elevators

The elevator control system is an all-mechanical system (Figure 15-6). Control rods, bellcrank levers, linkages and torque shafts convey mechanical inputs from the pilot control columns to the elevators. Each elevator is provided with a servo tab to reduce control forces during flight.

When the elevators move up or down, the tabs deflect in the opposite direction and aerodynamically lighten the elevator control forces. The elevator linkage incorporates an autopilot actuator with disengaging clutch to prevent the elevators from jamming during an actuator failure.

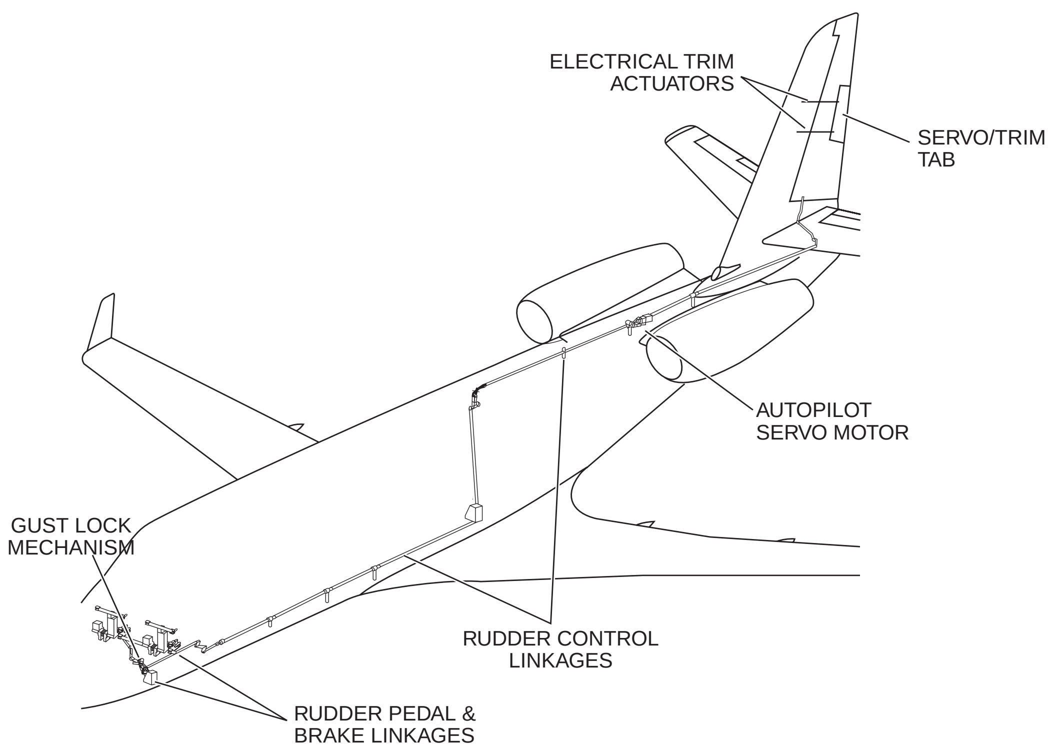


Figure 15-4. Rudder Control System

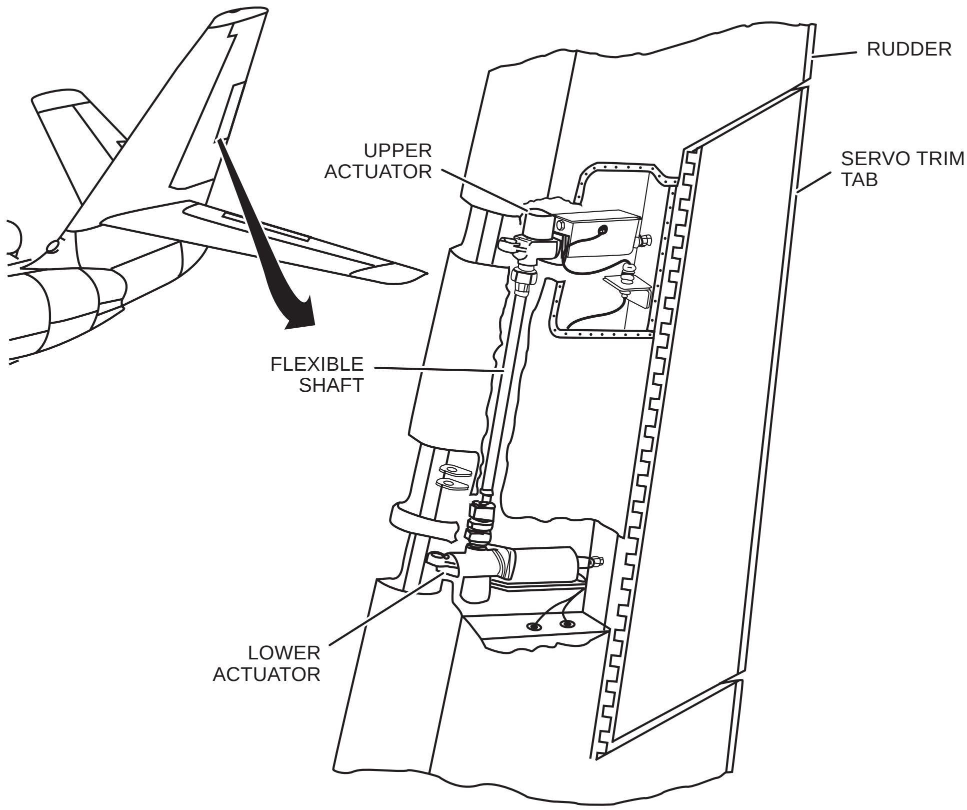


Figure 15-5. Rudder Trim Tab System

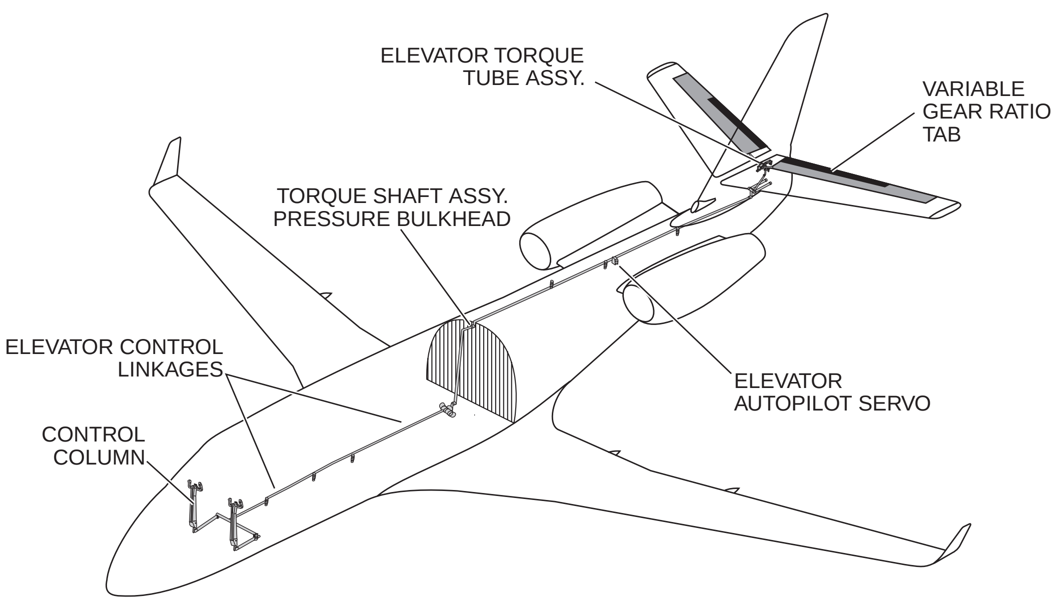


Figure 15-6. Elevator Control System

Horizontal Stabilizer

The horizontal stabilizer is located in the lower portion of the vertical stabilizer. The rear spar of the horizontal stabilizer is hinged to allow the stabilizer incidence angle to be changed for longitudinal trim adjustment. The front spar is attached to the horizontal stabilizer trim actuator. A scissor assembly provides additional strength while allowing flexibility for horizontal stabilizer movement.

Horizontal Stabilizer Trim Actuator

The horizontal stabilizer trim actuator (Figure 15-7) is an electromechanical unit used to position the horizontal stabilizer for pitch trim. The actuator has dual jackscrews and is located in the base of the vertical stabilizer.

The actuator incorporates three electric motors that are powered by different buses and operate through different switches, each driving both jackscrews. One motor is used during normal operation (two if the airspeed is below 250 KIAS), another during override mode, and all three during emergency pitch trim.

Limit switches in the normal and emergency systems prevent actuator damage due to overtravel. There are no limit switches in the override system. If the elevators become jammed or disconnected, all three motors operate in unison and the horizontal stabilizer serves in place of elevator for longitudinal control. A thermostatically controlled heating blanket protects the actuator from freezing at altitude. The heating element, is powered from the No. 1 main bus.

SECONDARY FLIGHT CONTROLS

Flap/Slat System

Two Fowler flap panels are located on each wing. Each flap section runs on three curved track rails and is supported on roller carriages. Linear actuators, which are powered by a centrally mounted actuator through flexible drive shafts, position the flaps as needed.

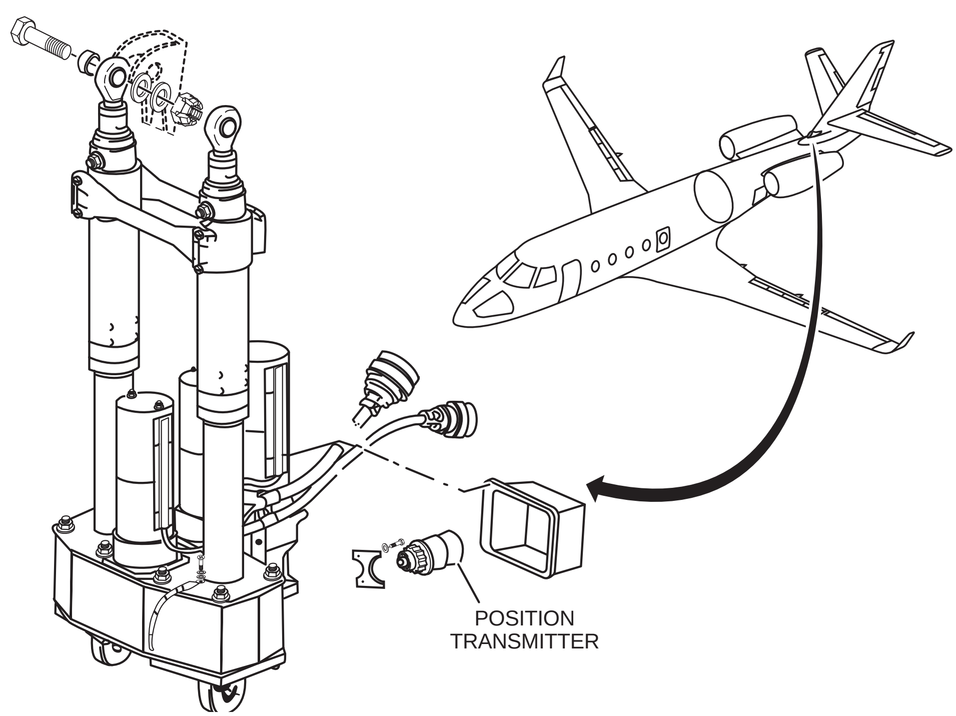


Figure 15-7. Horizontal Trim Actuator

The aircraft is equipped with leading-edge slats, which reduce aircraft stall speed and improve aircraft handling characteristics at high angles of attack. The slats deploy automatically at high angles of attack to provide stall protection for the aircraft. The slat installation and drive system are similar to those of the flaps.

Trailing Edge Flaps

The aircraft has one inboard and one out-board trailing-edge flap on each wing (Figure 15-8).

Figure 15-8. Wing Flaps

The flaps are driven by a single power drive unit (PDU) through a series of flexible drive shafts and linear actuators (Figure 15-9).

A link interconnecting the two flap panels on each wing provides secondary support and drive for the inboard flap. The flaps extend aft from the trailing edge of the wing as well as down when extended. The flaps can be selected to 0º, 12º, 20º or 40º. There are no intermediate positions that can be selected.

Leading Edge Slats

The leading edge slats provide high lift and low drag on takeoff and maximum lift on approach and landing but their main function is to improve the stall characteristics of the aircraft. The aircraft has one leading edge slat on each wing. Each slat section runs on six curved track rails by means of rollers (Figure 15-10). Linear actuators, which are powered by a centrally mounted power drive unit (PDU) through span-wise flexible drive shafts, position the slats along the track rails. The slats can be selected either UP or DN; equivalent to 0º or 25º.

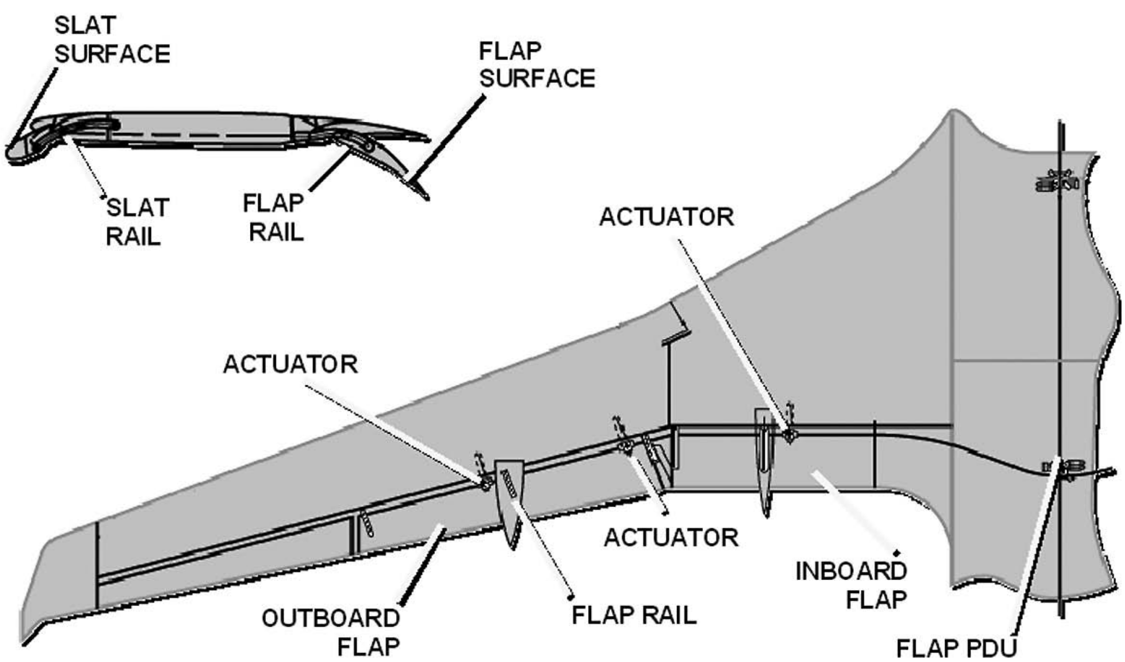


Figure 15-9. Flap Drive System

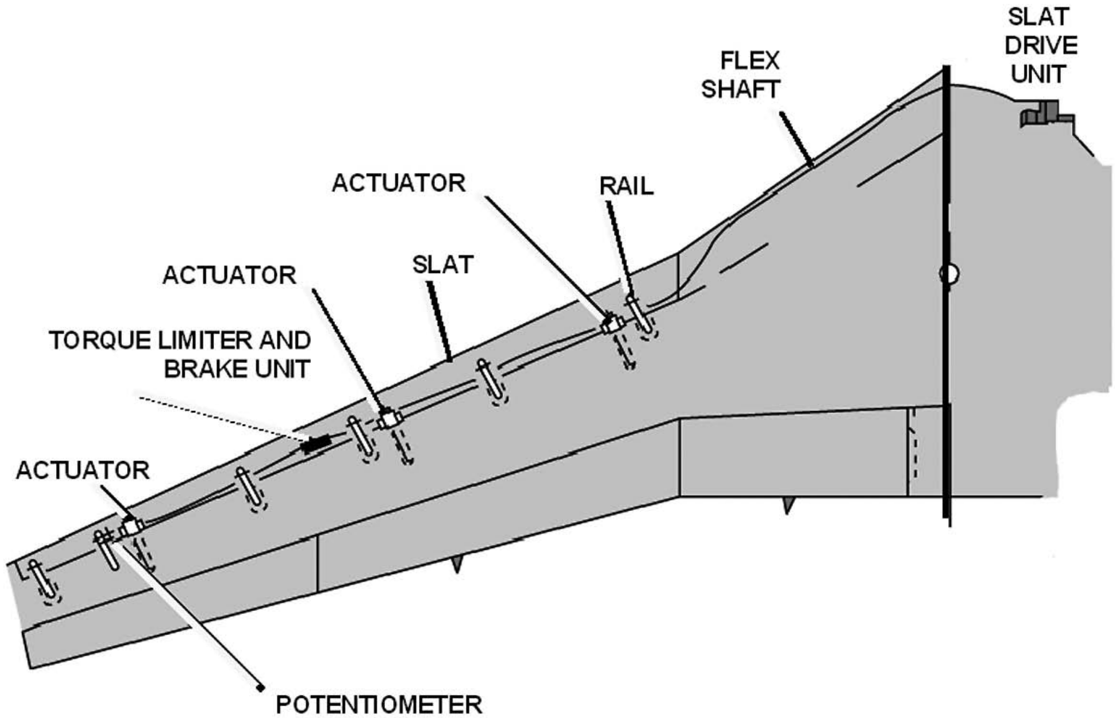


Figure 15-10. Slat Drive System

Flap/Slat Electronic Control Unit (FSECU)

The FSECU (Figure 15-11) is an electronic unit that governs and controls operation of the flap and slat mechanical drive systems. It receives signals from the flap/slat control lever, air data computer, and sensors to control and monitor all flap and slat operations.

The FSECU performs many functions. It processes the commands from the flap/slat control lever in the cockpit for flap and slat position and causes the power drive units (PDUs) to operate as needed. As it causes the flaps or slats to operate, it also provides flap and slat position information to the EICAS display in the cockpit.

The FSECU monitors the electrical current and temperature of the PDUs. It monitors the position of the flaps and slats and stops flap or slat operation if an asymmetry condition develops. It alerts the flight crew to the asymmetric condition via the EICAS. In the event of a slat system malfunction, the FSECU will allow the flaps to operate with the SLAT BYPASS function. It also provides an interface for maintenance troubleshooting of the flap and slat system.

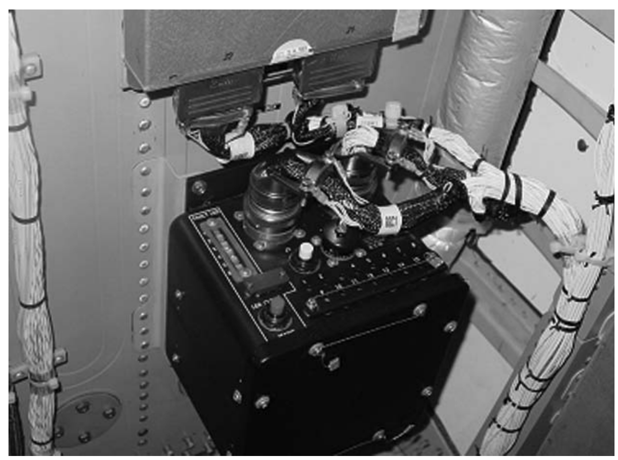


Figure 15-11. Flap/Slat Electronic Control Unit (FSECU)

The FSECU controls the automatic extension function of the slat system by monitoring airspeed, Mach number, angle-of-attack and

AOA probe heat function. Under certain conditions, it will cause the slats to extend without selection by the flight crew as discussed later in the text.

Flap Power Drive Unit (PDU)

A flap drive unit (Figure 15-11) is installed in the fuselage behind the left landing gear wheel well. The electrical unit comprises a single 28-VDC electric motor, clutch, brake, reduction gearbox, two torque limiters (each to an output shaft), and a switch for stopping the drive motor at the desired position. The torque limiters limit the torque available for each wing and prevent full actuator torque from overloading a single linear actuator or damaging the drive unit housing. The PDU also includes a potentiometer to provide position information to the FSECU.

Slat Power Drive Unit (PDU)

The slat drive unit (Figure 15-12), located in the fuselage, comprises a single 28-VDC electric motor, clutch, brake, reduction gearbox, two torque limiters (each to an output shaft), and a switch for stopping the drive motor at the desired position. The drive unit incorporates a potentiometer for transmitting slat angle information to the FSECU. The unit is operationally similar to the Flap PDU, as described above.

Flap/Slat Actuators

Both the flap and slat systems have three linear ballscrew actuators (Figure 15-13) on each wing that convert rotary input motion from the flex shafts into linear output to drive the respective flap or slat surface. The flaps or slats can only be positioned by action of the actuators through their rotary input shafts. Each actuator features a high-ratio worm gear arrangement that makes the actuators irreversible. They will not move in response to aerodynamic loading, even in the event of drive shaft failure. The most outboard actuators of the flap and slat systems feature internal brake mechanisms and potentiometers to detect asymmetry conditions. If an asymmetry develops, the electric brakes lock the slats or flaps as necessary.

Airbrake System

There are four airbrake panels installed on the upper surface of each wing upper surface, forward of the outboard flap (Figure 15-14).

Each panel is attached at its leading edge. They are electrically controlled and hydraulically actuated using main hydraulic system pressure. Individual hydraulic actuators rotate each panel upward at a 45°angle. The inboard pair of panels act as either flight or ground air brakes and the outboard panels act as ground air brakes only.

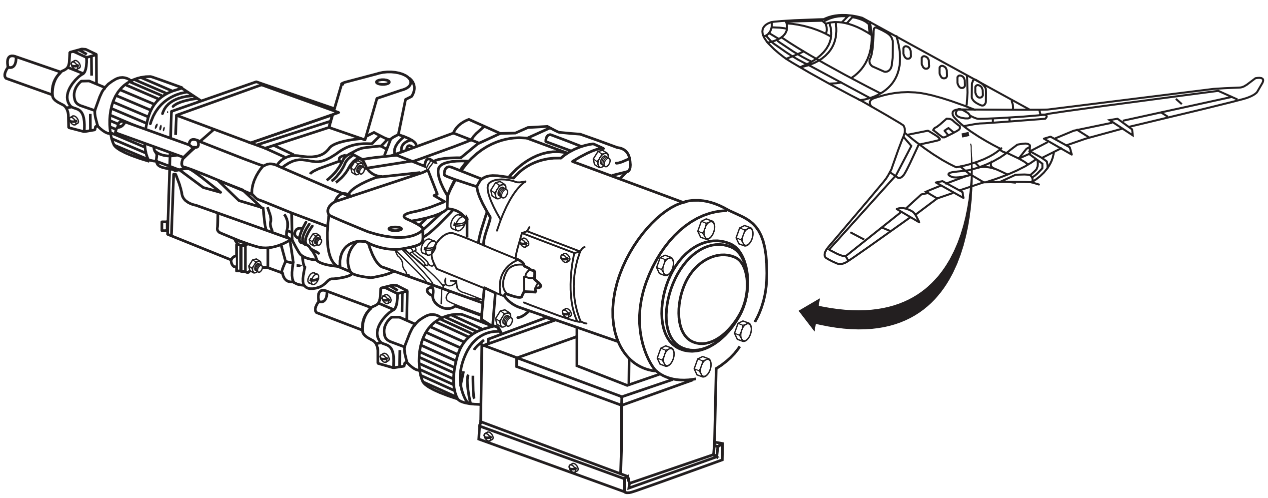


Figure 15-12. Slat Power Drive Unit (PDU)

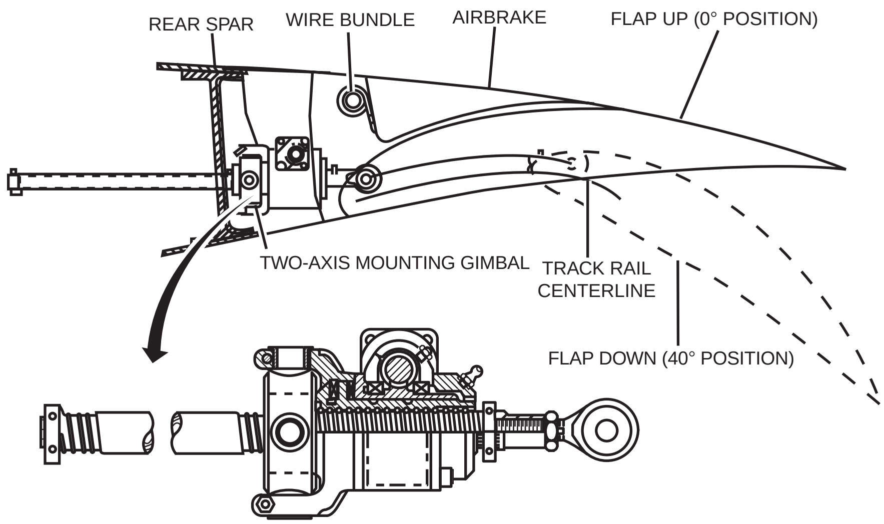


Figure 15-13. Ballscrew Actuator

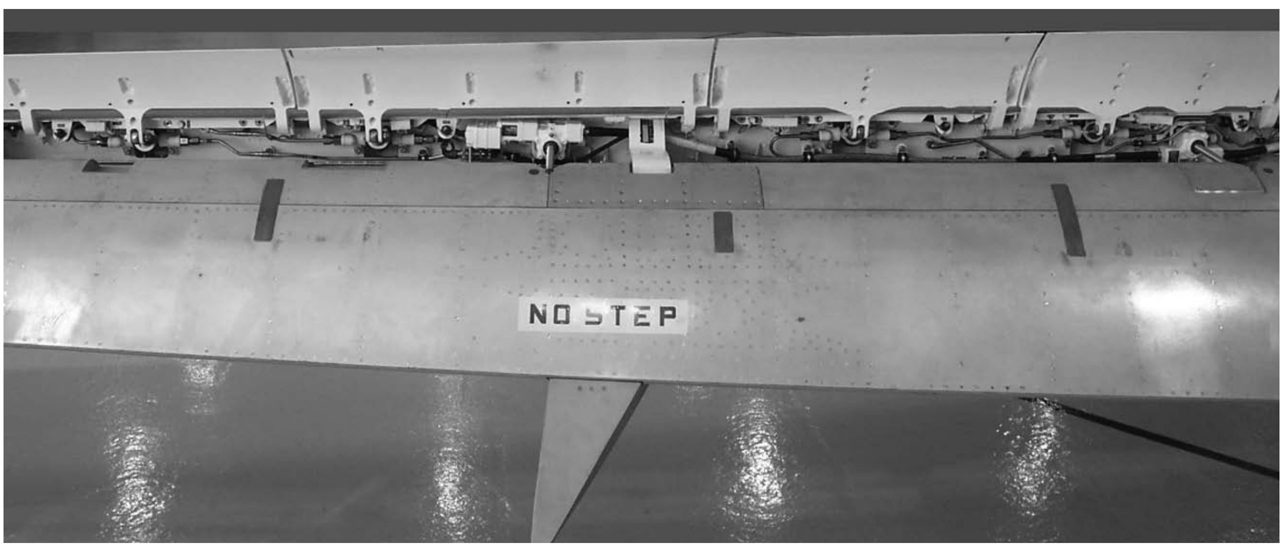


Figure 15-14. Airbrakes Deployed

Airbrake Actuators

Eight actuators operate the airbrakes, one for each panel. All actuators are double-acting, single-rod hydraulic cylinders with internal mechanical locking devices that lock the actuators in the retracted position. When hydraulic pressure is applied to the actuators, the locks will release and hydraulic pressure will extend the airbrakes. All the actuators have the same installation geometry, the cylinder body is trunnion mounted to the wing and the airbrake panel is moved by its attachment to the hydraulic piston.

Gust Lock

An internal gust lock mechanism (Figure 15-15) is installed, enabling ground locking (in neutral) of the elevator and rudder control systems, thus preventing damage to these systems when the aircraft is parked in strong wind conditions. An operating handle on the thrust lever quadrant is used to engage plunger assemblies in the control linkage below the cockpit floor. It is designed so that thrust cannot be increased above idle with the gust lock engaged. Gust locking for the aileron control system is not required, due to the damping action of the aileron hydraulic servo actuators.

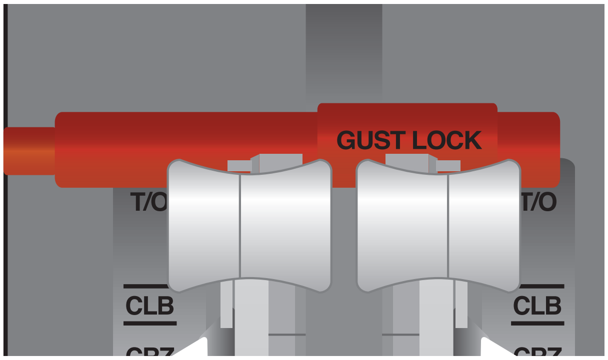
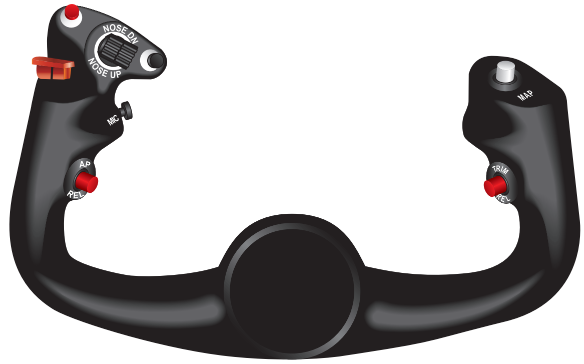
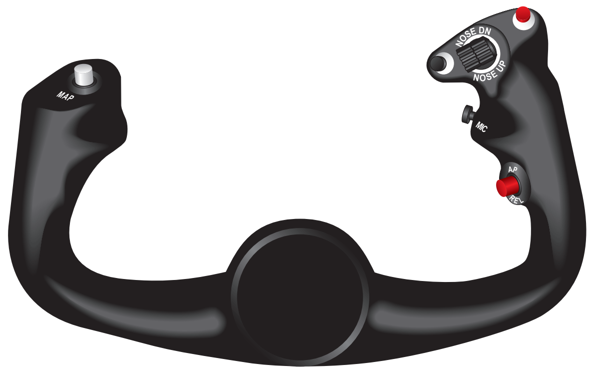


Figure 15-15. Gust Lock



LEFT CONTROL WHEEL



RIGHT CONTROL WHEEL

Figure 15-16. Control Wheels

OPERATIONS

PRIMARY FLIGHT CONTROLS

Ailerons

The ailerons of the G150 are actuated through control wheel (Figure 15-16) rotation by either pilot. Control linkage then transmits this force to the ailerons and is assisted by hydraulic pressure within the servoactuators that are attached to each aileron. An artificial feel unit is installed in the control linkage to provide tactile feedback to the flight crew as the ailerons are deflected from their neutral position. Each servoactuator can receive hydraulic pressure from the main and auxiliary hydraulic systems and receives pressure from both systems simultaneously under certain conditions. Hydraulic fluid from the main and auxiliary systems does not mix within the servoactuators. The servoactuators each contain two piston and valve assemblies within a single housing. Either hydraulic system can provide hydraulic assistance through the full range of aileron movement. If the pressure from one of the hydraulic systems is removed, a bypass valve within the servoactuator allows fluid to flow through the mechanism, preventing hydraulic lock. When all hydraulic pressure is removed, a compensator in the return hydraulic line provides hydraulic resistance to protect the ailerons from gust damage.

Each servoactuator has two microswitches that monitor hydraulic pressure and mechanical position of the internal mechanisms. If hydraulic pressure is lost from the main hydraulic system, the auxiliary hydraulic pump will be activated if the control switch is in AUTO. If there is a mechanical failure of the servoactuator, the auxiliary hydraulic pump will be activated regardless of control switch position. In either case, the AILERON FAIL warning message will be displayed on the EICAS.

If there has been a main system hydraulic failure, the auxiliary pump will continue to operate even if pressure is restored. The pump can be deenergized by selecting the control switch to OFF and then back to AUTO. If there has been a mechanical failure of a servoactuator, the only means of deactivating the auxiliary hydraulic pump is by removing power from its electrical circuit.

A centering spring within each servoactuator will position its respective aileron 3º up in case the control linkage to that servoactuator is disconnected. This prevents aerodynamic flutter.

If both hydraulic systems are lost, the ailerons can be actuated manually. The servoactuators become rigid and provide the required linkage to the control system.

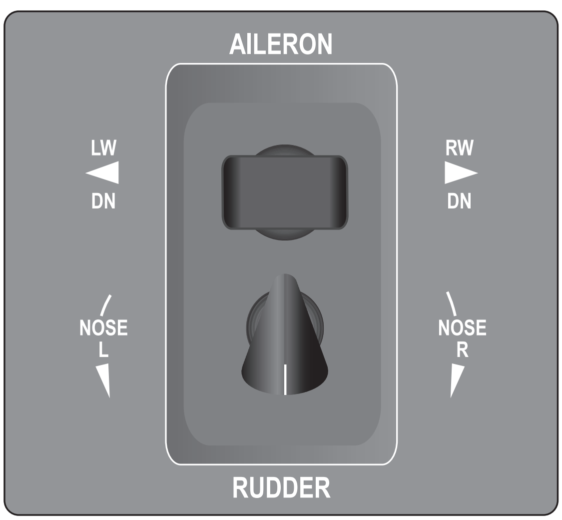
If one of the ailerons becomes jammed because of a mechanical failure or ice accumulation, the link between the left and right aileron controls can be broken. If sufficient force is applied by both pilots by rotating the control wheels inboard, the interconnect rod between the subsystems will be disconnected. This will allow the pilot with the operable aileron to assume control of the aircraft. If the pilot has an operable aileron, the autopilot will be available. However, the artificial feel and aileron trim will not be available. The opposite is true if the copilot has the operable aileron.

Aileron Trim

Aerodynamic forces are simulated at the control wheels by an artificial feel unit (AFU) consisting of a spring box attached to the aileron control linkage. As the aileron control system is deflected from its neutral position, one of the springs within the AFU is compressed. This causes an increase in the amount of force required by the pilot to deflect the ailerons. The electrically powered aileron trim actuator is connected to the AFU. Operation of the aileron trim actuator causes the neutral point of the AFU to be displaced to favor either left wing down or right wing down. The AFU has maximum authority of about 1/3 aileron displacement. The aileron trim setting is continuously displayed on the primary EICAS page. Green bands indicate the proper range of aileron trim for takeoff when the aircraft is on the ground.

The aileron trim actuator is controlled from the spring loaded AILERON trim switch on the lower left of the center pedestal (Figure 15-17).

Momentary displacement of the switch to the LW DN or RW DN position causes operation of the actuator to reset neutral feel in the left wing down or right wing down direction respectively. When released from either operating position, the switch returns to the neutral position and the trim actuator will stop operating at its present position.



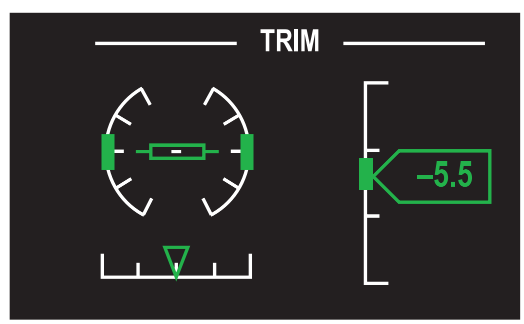


Figure 15-17. Aileron/Rudder Trim Switches and Indications

Rudder

The rudder is manually actuated by either the pilot or copilot through the rudder pedals. When force is applied to a rudder pedal by the pilot or copilot, the motion is transmitted by a series of pushrods and bellcranks to the rudder. The rudder trim tab also acts as a servo tab and aerodynamically relieves some of the control force necessary to deflect the rudder. This makes hydraulic assistance unnecessary.

Rudder Trim

The rudder trim is controlled from the spring loaded RUDDER trim switch knob on the lower left of the center pedestal (Figure15-17).

Turning the RUDDER trim switch knob to either the NOSE L or NOSE R position energizes both trim actuators, which are mechanically linked in the rudder, to position the trim tab. When released, the trim switch automatically returns to the neutral position and the actuators stop operating in its current position. Trim tab position is indicated on the primary EICAS page with a green band showing the proper range for takeoff while the aircraft is on the ground.

Elevators

During flight, the elevators provide aircraft pitch control. A series of bell cranks, torque tubes and control rods mechanically transmits the control column input to the elevators. Servo tabs attached to each elevator aerodynamically relieve some of the control force, making hydraulic assistance unnecessary. Elevator movement is in unison. There is no provision for independent operation if one of the elevators becomes jammed because pitch control can be regained through operation of the horizontal stabilizer trim system.

Horizontal Stabilizer Trim System

The variable incidence horizontal stabilizer provides longitudinal (pitch) trim. There are two electrically independent drive systems; with separate switches, wiring and three motors. All three motors are incorporated into a single drive unit that mechanically positions the stabilizer leading edge. Stabilizer trim position is continuously displayed on the primary EICAS page. During ground operations, a green band will appear on the trim indicator to show the proper range for takeoff, depending on the current flap setting. The horizontal stabilizer trim operates in one of three modes—NORMAL, OVRRD (override), and EMERG (emergency), as described below.

Normal Trim

Pitch trim switches, on the top of the outboard grips of the pilot and copilot control wheels, control normal operation. Each is a three position switch spring loaded to the center (OFF) position. The forward momentary position (NOSE DN) provides nose down trim, and the aft momentary position (NOSE UP) provides nose up trim. The autopilot also uses the normal system for pitch trim. The Mach trim function operates when the autopilot is disengaged in the flight regime in which trim adjustment for Mach effects is needed (0.79 M_I and above) as received via air data computer Mach data. At normal cruising airspeed, one electric motor within the actuator positions the stabilizer. To increase pitch trim response at low airspeed, the emergency motor is also activated if the airspeed is below 250 KIAS. The trim actuator incorporates limit switches that restrict stabilizer travel. A pitch trim tone sounds in the cockpit when the normal system operates and has a momentary delay to prevent the tone from sounding for small trim adjustments.

Mach Trim

Mach trim inputs are made by the integrated autopilot/flight instrument system. As speed increases, the autopilot uses the pitch trim system to adjust the horizontal stabilizer. When the autopilot is not engaged and the aircraft is within the .79 to .87 M speed range, input signals are generated to overcome a nose down tendency as the center of lift moves aft on the wing with increasing speed.

There are no inputs from the flight station into this system nor can the system be turned on or off. There is a single caution lever message generated by the system when it is not functioning properly. The only limitation associated with this system is to limit the speed of the aircraft to .78M with the mach trim and autopilot inoperative. Notify maintenance that the message occurred in flight.

Yaw Damper

The yaw damper is controlled by a single pushbutton on the flight guidance panel. The only control the pilot has is to turn the system on or off. A green YD is displayed on the Flight Mode Annunciator (FMA) on the Primary Flight Display (PFD) when the system is active. The yaw damper puts automatic signals into the rudder actuator servo to control Dutch roll tendencies exhibited by swept wing aircraft and to aid in turn coordination. The yaw damper will disengage with the AP disconnect switch when full flaps are selected. The yaw damper will also disengage when the YD/AP disconnect bar is lowered, when the nose gear touches down, or when a yaw damper failure is detected by internal monitors.

The yaw system is automatically turned on when the autopilot is engaged.

The limitations associated with the yaw damper system is that it must be turned off for takeoff and the maximum altitude is limited to 31,000 ft with both the yaw damper and autopilot inoperative. If the yaw damper is inoperative, the autopilot alone will provide satisfactory lateral-directional dampening.

Override Trim

If the normal system fails to operate, the pilot can regain pitch trim capability by pressing the PITCH TRIM REL switch located on the right (inboard) grip of the pilot control wheel. This deenergizes the normal (primary) pitch trim system and energizes the override system. The light in the OVRRD switchlight illuminates (Figure 15-18).

When the override system is energized, pitch trim adjustment is accomplished by actuating the three-position HORIZ TRIM switch on the pedestal. This switch operates in a similar manner to the normal trim switches on the control wheels. When this switch is activated, the override (secondary) trim motor is energized to reposition the stabilizer. This system does not incorporate electrical limit switches. The flight crew must monitor stabilizer trim position to prevent the actuator from reaching its mechanical limits of travel.

The flight crew can reengage the normal trim system by pressing the illuminated OVRRD switchlight. The light will extinguish and power is removed from the override system and restored to the normal system.

Figure 15-18. Override Trim Engaged

Emergency Trim

The emergency pitch trim system enables the crew to regain pitch control if the elevators become jammed or disconnected. It may also be used to restore pitch trim functionality if both the normal (primary) and override (secondary) pitch trim systems fail. The emergency trim system is energized by pressing the guarded EMERG ARM switchlight on the pedestal (Figure 15-19). The light in the switch illuminates and control of pitch (or pitch trim) is then accomplished through the EMERGENCY TRIM switch located on the left (outboard) grip of the pilot control wheel. This switch features a guard that must be lifted in order to operate it. Once this is accomplished, the switch operates like the normal trim switches.

Figure 15-19. Emergency Trim Engaged

The system operates three electric pitch trim motors simultaneously (normal, override, and a third emergency motor). The trim operates at three times the normal rate with limit switch protection. Emergency trim, however, should not be used for normal trim adjustment. Pressing EMERG ARM pushbutton once again deenergizes the emergency system. The light in the switch will extinguish, the emergency trim system is deenergized and power is restored to the normal trim system.

SECONDARY FLIGHT CONTROLS

Flaps and Slats

Flap and slat operation is consecutive. Slats extend first and retract last. The slats normally will be fully extended prior to flap extension and will retract only after full retraction of the flaps.

The slats and the flaps are controlled with the slats/flaps control lever (Figure 15-20). The lever has five positions: UP, DN, 12°, 20°, and 40°. If the lever is in the UP position, all flaps and slats are fully retracted. Moving the lever to the DN position fully extends the slats (25°), while leaving the flaps fully retracted. If the SLATS/FLAPS control lever is moved farther aft, the flaps will be extended to the selected position with the slats remain-

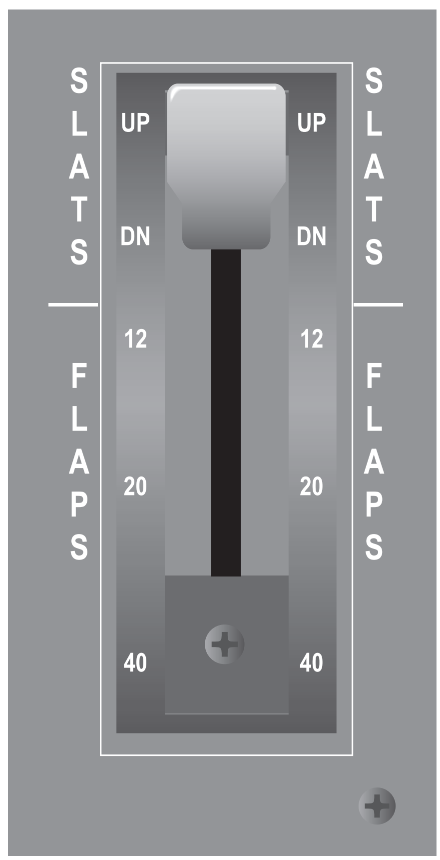


Figure 15-20. Slats/Flaps Control Lever

ing in the fully extended position. During retraction, the lever is used to select the desired flap or slat position. The slats will remain extended until the flaps are fully retracted. It takes approximately 30 seconds to fully extend the flaps and slats if they are retracted.

There is no requirement to stop at intermediate settings of the lever when extending or retracting the flaps and slats, provided that the airspeed is within limits. However, if reversing the direction of travel (selecting flaps UP from 12º and then reselecting 12º, for example), it is recommended that the flaps or slats be allowed to reach the selected position and stop before selecting the new position.

The slats extend automatically at high angles of attack to provide stall protection. The slats will automatically extend when all of the following conditions exist:

- Angle of attack at or above 0.82 normalized AOA
- Airspeed at or below 250 KIAS
- Mach number at or below 0.55 Mi

If the slats automatically extend for any of the above conditions, the AUTO SLATS EXTENDED caution message will be displayed on the EICAS and an aural warning will sound. The AUTO SLATS FAIL caution message appears if a failure in the automatic extension circuitry occurs, indicating that the system is inoperative. After automatic slat extension, retraction can be achieved by placing the lever DN and then UP, provided angle of attack is below 0.6° normalized AOA.

If 250 KIAS/0.55 MI is exceeded with slats extended, the V_{MO}/M_{MO} aural warning (clacker) sounds.

A slat asymmetry of 3° is indicated by illumination of an amber SLATS UNBALANCE EICAS message and a flap asymmetry of 3° by an amber FLAPS UNBALANCE EICAS message. In either case, surface actuation is automatically terminated.

If the flaps and slats do not extend when selected by the cockpit control, the SLAT BYPASS ARM switchlight can be pressed to allow flap extension with the slats retracted.

Flight Airbrakes

Moving the FLIGHT A/B switch to the OUTBD position energizes a solenoid-operated selector valve to direct main hydraulic system pressure to the two airbrake hydraulic actuators. The internal locking mechanisms are released, and the airbrakes extend (45°). The rate of extension is controlled by hydraulic flow restrictors. A green advisory FLT A/B OUTBD OUT message will appear on the EICAS as the airbrakes extend.

Moving the FLIGHT A/B switch to the INBD & OUTBD position energizes two solenoid-operated selector valves to direct hydraulic pressure to the four innermost hydraulic actuators. Green advisory FLT A/B OUTBD OUT and FLT A/B INBD OUT messages will appear on the EICAS as the airbrakes are extended. The flight airbrakes may be deployed at any airspeed.

Microswitches connected in series and actuated by airbrake panel movement detect airbrake asymmetry. If a panel does not extend, amber FLT A/B OUTBD FAIL and FLT A/B INBD FAIL EICAS caution messages will appear. Moving the FLIGHT A/B switch to the RETRACT position deenergizes the circuit, repositions the selector valve, allows all panels to retract and resets the system.

Moving the FLIGHT A/B switch to RETRACT deenergizes the selector valve to the retract position, directing pressure to the actuators for retraction. As the airbrakes lock in the retracted position, the EICAS messages disappear.

The airbrake control switches are shown in Figure 15-21

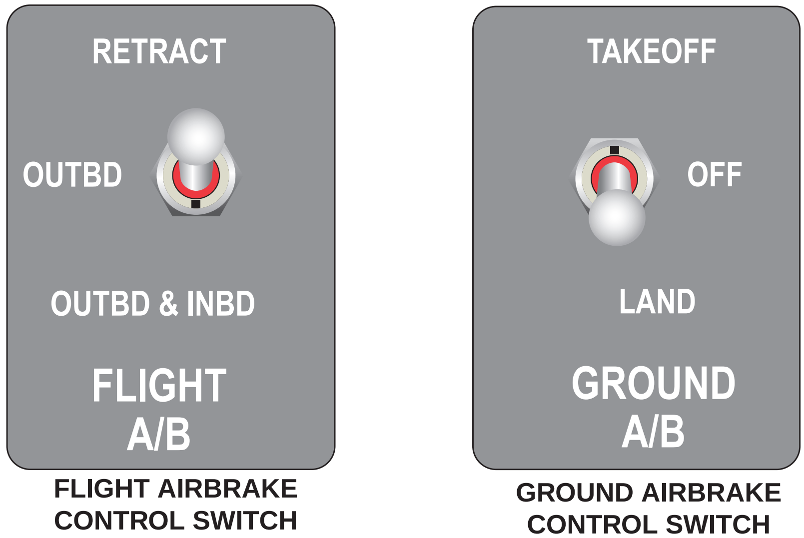


Figure 15-21. Airbrake Control Switches

Ground Airbrakes

Ground airbrakes only operate when the aircraft is on the ground. They dump excess lift from the wing when aborting a takeoff or during landing. The ground airbrakes are comprised of all four of the airbrake panels on each wing and also operate with main hydraulic system pressure. The ground brake extension rate is faster than that of the flight airbrakes because an additional selector valve is energized, bypassing the flow restrictor. To prevent erratic extension or retraction of ground airbrakes during a bounced landing, a holding relay bypasses the ground contact switches and maintains the selector valves in the extended position. Ground airbrake deployment is indicated by messages on the EICAS, depending on the specific condition of the system. When all of the panels are locked down, the messages are removed.

With the GROUND A/B switch in the TAKEOFF position, ground airbrake deployment will occur if the following conditions are met: nose weight on wheels (as signaled by both switches), both thrust levers at less than 80%, airspeed more than 80 KIAS and one main landing gear weight on wheels signal.

With the switch in the LAND position, ground airbrake deployment will occur if both thrust levers are at less than 80% and one main landing gear signals weight on wheels.

If the switch is in the OFF position, the ground airbrakes will not deploy.

Gust Lock

The elevator and rudder gust lock should be engaged as soon as both engines are shut down. Engaging the lock (Figure 15-22) places mechanical interference in the surface control linkage beneath the cockpit floor. Placing the control column and the rudder pedal assemblies in mid (neutral) position aligns the locking detents on the rudder and elevator bellcranks, with the spring loaded locking plungers in the gust lock system.

With both power levers in the CUT OFF position, the gust lock is engaged by pressing the button on the left of the operating handle and pulling the handle aft to the lock ENGAGED position. When the gust lock is engaged, engine thrust lever movement is automatically restricted to the IDLE power setting, preventing takeoff with flight controls locked.

The gust lock is disengaged by pressing the button on the operating handle and letting the internal spring move the handle fully forward, thus unlocking the rudder and elevator control surfaces. With the operating handle placed fully forward (controls unlocked), operation of the thrust levers is possible.

LIMITATIONS

The limitations contained in section one of the *Airplane Flight Manual (AFM)* must be complied with regardless of the type of operation.

SLATS/FLAPS

The maximum safe speed for operating/extended slats/flaps (V_{SE}/V_{FE}) is as follows:

Slats	250 KIAS
Flaps 12°	250 KIAS
Flaps 20°	225 KIAS
Flaps 40°	180 KIAS

Do not operate slats and flaps above 20,000 feet.

Auto slats operation and manual retraction are not altitude limited.

TRIM SYSTEM

Takeoff is prohibited with either normal, override, or emergency stabilizer trim system inoperative.

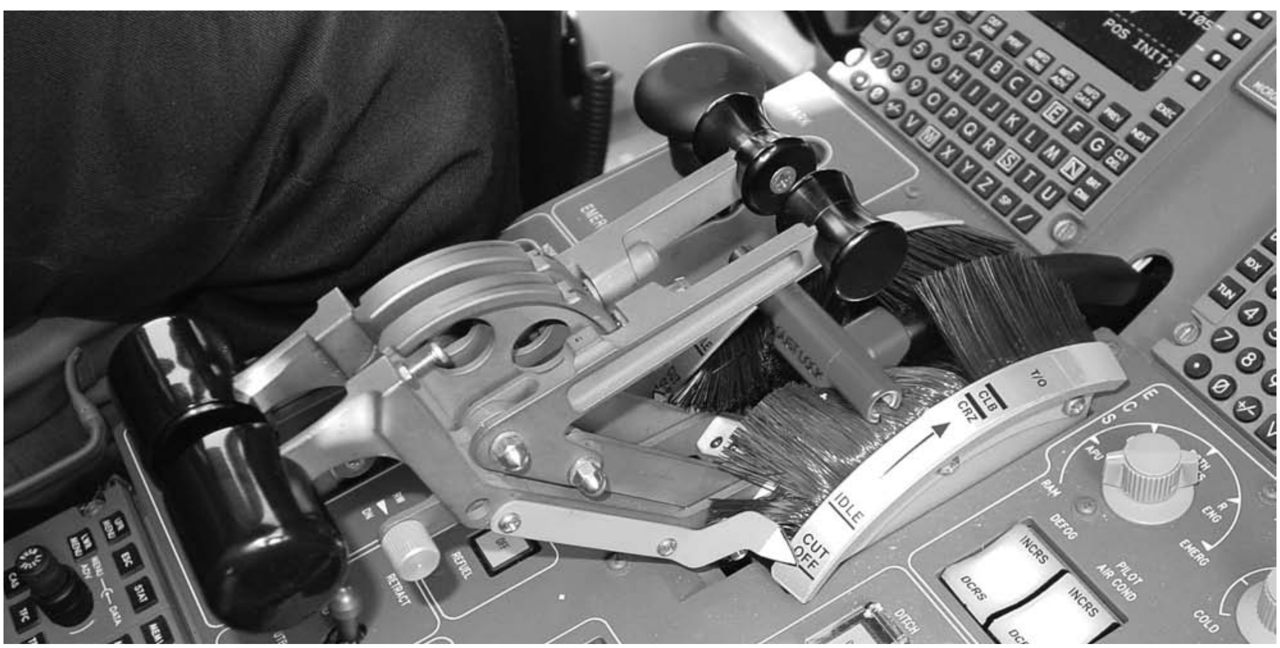


Figure 15-22. Gust Lock Engaged

EMERGENCY AND ABNORMAL PROCEDURES

EICAS INDICATIONS

The following is a ready-reference guide which briefly outlines the EICAS message meanings and corrective action for associated messages.

Messages having symbol are accompanied by an aural alert.

Warning Messages

Warning messages display as red EICAS messages on the MFD, and are accompanied by a MASTER WARNING light (Figure 15-23).

AILERON FAIL—Mechanical failure of one or both servo-actuators or main hydraulic pressure loss.

CONFIG AIRBRAKE —Aircraft is on the ground with both engines thrust beyond 70% N_1 and air brakes are unlocked.

CONFIG FLAPS —Aircraft on ground, both engines thrust beyond 70% N_1 and flaps position more than 22°.

CONFIG SLATS —Aircraft on ground, both engines thrust beyond 70% N_1 and slats position less than 23° with SLAT BYPASS pushbutton off.

CONFIG TRIM —Aircraft on ground, both thrust levers beyond 70% N_1 and horizontal stabilizer trim out of green band for the selected flaps setting and engines running.

STALL —Aircraft is approaching stall.

Caution Messages

Caution messages display as amber EICAS messages on the MFD, and are accompanied by a MASTER CAUTION light.

A/B T/O NOT ARMED—Ground air brakes not armed for takeoff.

AUTO SLATS EXTENDED—Slats automatic extension has been activated. Slats will extend automatically due to high AOA.

AUTO SLATS FAIL—Failure in slats automatic extension system.

FLAPS UNBALANCE—Failure of flap system or asymmetry between left and right flaps is more than 3°.

FLT A/B INBD FAIL—Inboard flight air brakes position differs from switch command. Outboard flight airbrakes should be used.

FLT A/B OUTBD FAIL—Outboard flight air brakes position differs from switch command. Inboard flight airbrakes should be used.

GND BRK WOW MISCOMP—Ground brakes weight on wheels switches miscompare.

MACH TRIM FAIL—Autopilot mach trim failure.

SLATS UNBALANCE—Failure of slat system or asymmetry between left and right slats is more than 3°.

STALL WARNING FAIL—Stall warning system has failed.

Advisory Messages

Advisory messages display as green EICAS messages on the MFD, and do not illuminate the MASTER CAUTION light.

CONFIG SLATS BYPASS—Illuminates due to take-off configuration setting of flaps –20° and SLATS BYPASS selected.

FLT A/B INBD OUT—Flight (inboard) airbrakes extended.

FLT A/B OUTBD OUT—Flight (outboard) airbrakes extended.

Figure 15-23. Flight Controls EICAS Indications

QUESTIONS

1. The primary flight control surfaces that are manually actuated are the:
A. Rudder and ailerons
B. Ailerons and horizontal stabilizer
C. Horizontal stabilizer and rudder
D. Rudder and elevators

2. The hydraulically boosted ailerons are pressurized by:
A. Main and auxiliary hydraulic pressure
B. Main hydraulic pressure only
C. Auxiliary hydraulic pressure only
D. Main or auxiliary hydraulic pressure, but never by both simultaneously

3. The horizontal stabilizer pitch trim system is:
A. Hydraulically powered and actuates in two modes
B. Electrically powered and actuates in one mode
C. Manually actuated with a handwheel in the cockpit
D. Electrically powered and actuates in three modes

4. The leading-edge slats:
A. Are hydraulically actuated
B. Are electrically actuated, extend 25°, and are sometimes extended automatically
C. Can be extended manually
D. Extend after flap extension and retract before flap retraction

5. If slats operation has been terminated by an asymmetric condition:
A. The SLATS UNBALANCE CAS message illuminates
B. The SLATS BYPASS will allow FLAP operation
C. The aircraft is still controllable
D. All of the above

CHAPTER 16
AVIONICS

ILLUSTRATION

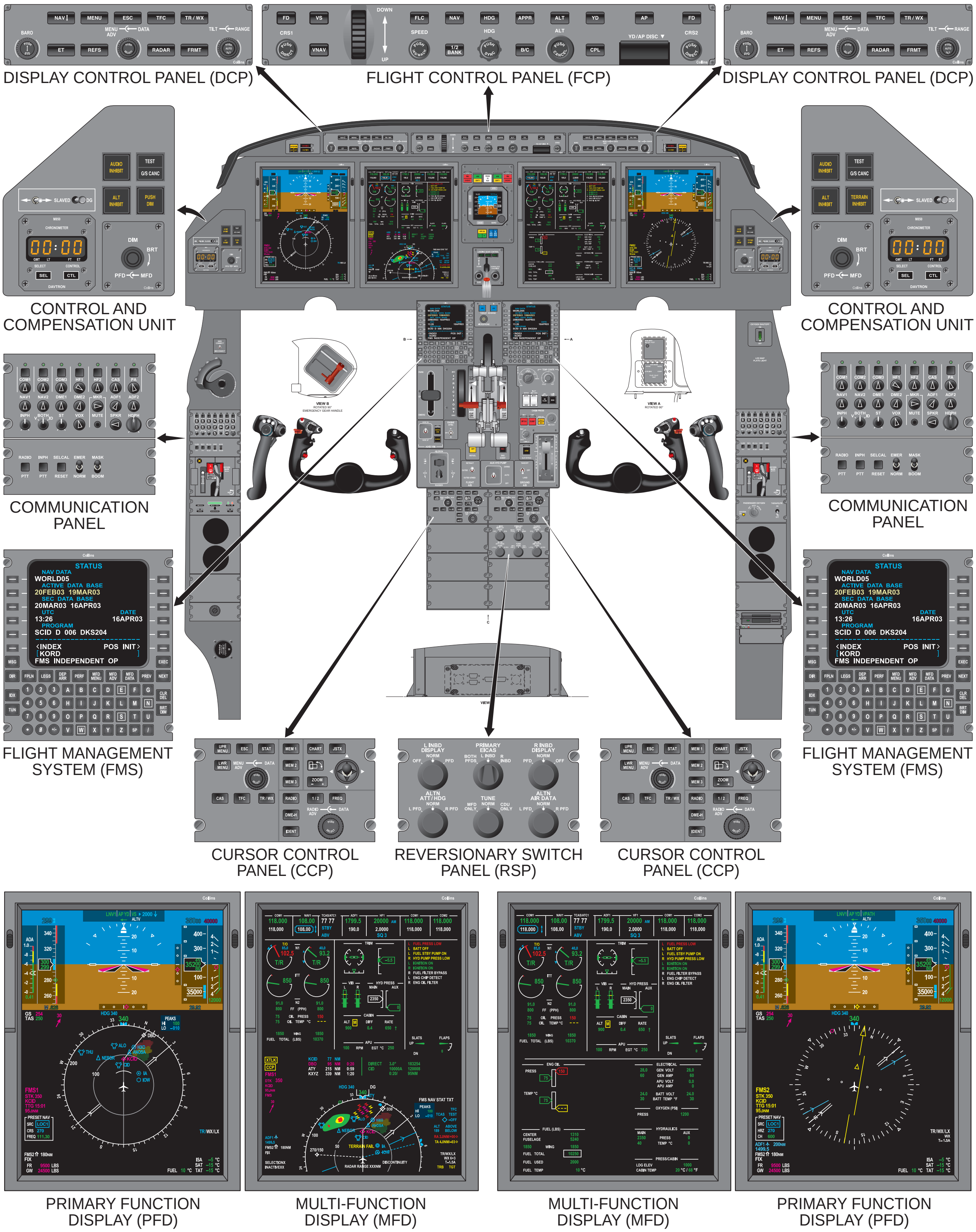


Figure 16-1. Cockpit Instruments and Displays

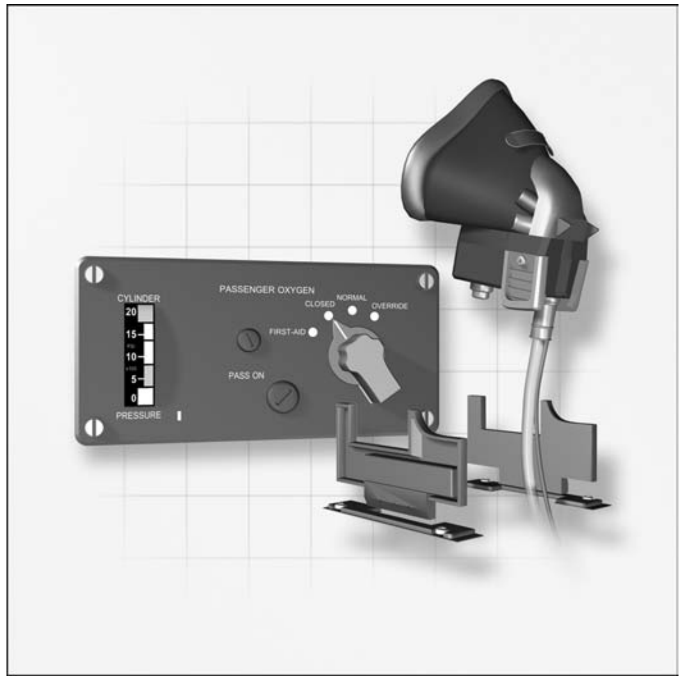
The information normally contained in this chapter can be found in the Pro Line 21 and FMS 6100 manuals.

CHAPTER 17
OXYGEN SYSTEM

CONTENTS

ILLUSTRATIONS

CHAPTER 17
OXYGEN SYSTEM



INTRODUCTION

The G150 oxygen system includes the crew, passenger, and therapeutic subsystems. The cabin oxygen supply is controlled by an oxygen supply valve.

OXYGEN SYSTEMS

GENERAL

The G150 aircraft is equipped with high-pressure and portable oxygen systems capable of supplying supplemental oxygen for all occupants in compliance with applicable Federal Air Regulations. Provisions are also made for a therapeutic oxygen system.

Oxygen is available to the crew masks under all operational conditions. The passenger oxygen supply is available under automatic or selective conditions. Therapeutic oxygen is available under selective conditions only.

SYSTEM DESCRIPTION

CREW OXYGEN SYSTEM

Crew oxygen is controlled by the OXYGEN SUPPLY toggle switch located on the forward right console in the flight compartment (Figure 17-1). This valve must be in the ON position for flight. Oxygen for the crew is continuously available at any altitude in this position.

The crew masks are pressure-demand, quick-donning with a mask-mounted regulator that allows one-hand donning within five seconds.

Each mask is located in a storage box in the left and right side consoles of the flight compartment. The mask supply line has an integral oxygen flow indicator.

The mask includes an inflatable and adjustable harness, which allows a comfortable fit to the head.

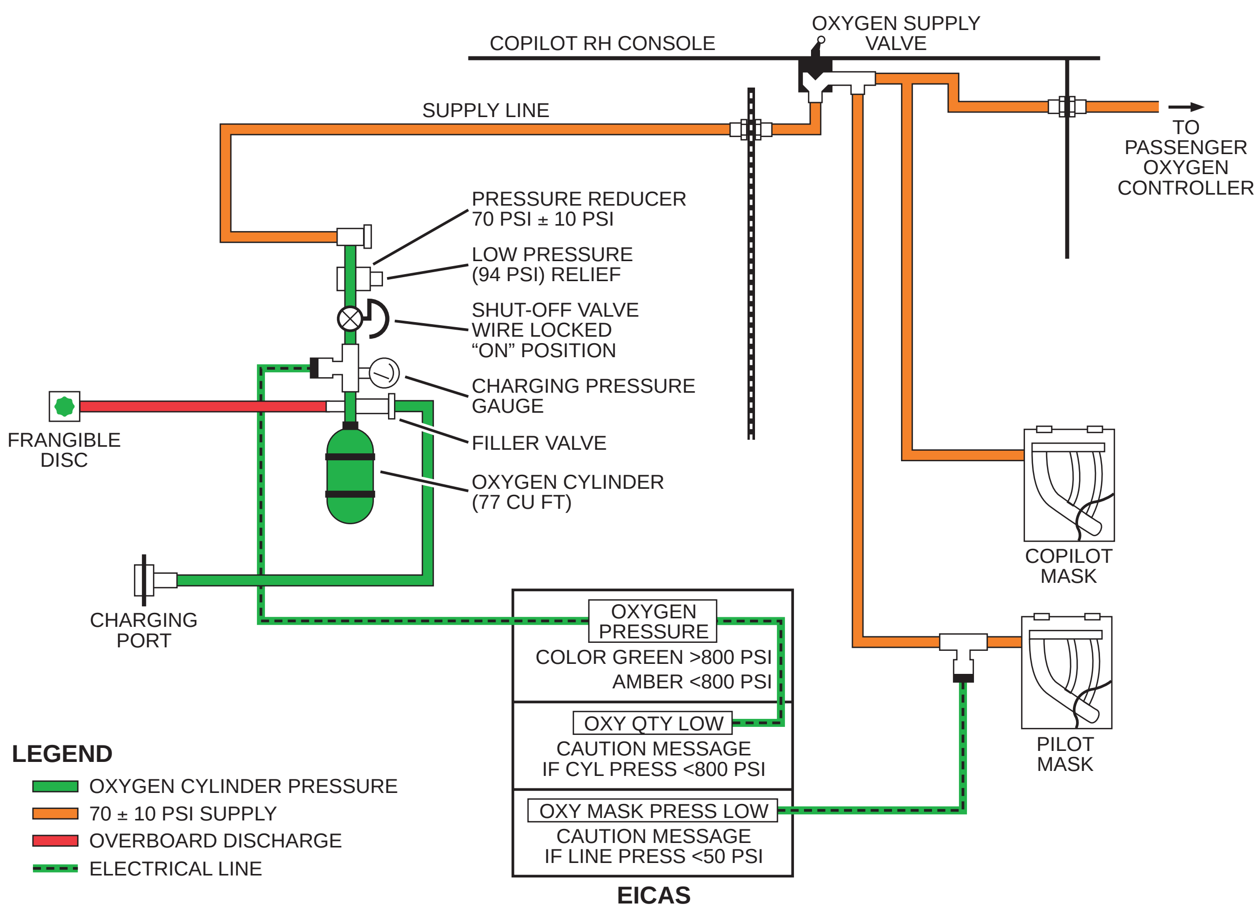


Figure 17-1. Oxygen System

Crew Mask Controls

Oxygen flow is available to the crew masks when the OXYGEN SUPPLY valve is selected to the ON position and the bayonet fitting in the crew mask supply line is connected.

The percentage of oxygen to cabin air is selectable with a diluter selector on the bottom of the regulator. The selector has two positions N (normal) and 100%.

The crew oxygen masks have three selectable modes as follows:

- Normal selection
  - During normal operation, the mask regulator supplies diluted oxygen, increasing oxygen ratio as cabin altitude increases
  - At 30,000 ft and above, the regulator automatically supplies 100% oxygen
  - At cabin altitudes of 35,000 to 40,000 ft, 100% oxygen is supplied with a positive pressure. Above 40,000 ft cabin altitude, the pressure significantly increases
- 100% selection
  - Provides 100% oxygen
- Emergency selection
  - A rotary knob on the bottom of the regulator selects emergency; turning the knob counterclockwise provides 100% oxygen and pressure breathing at any cabin altitude

TEST mode supplies pressurized oxygen to the mask to check the pressure breathing performance on the ground. This ensures there are no leaks to the mask.

Crew Mask Operation

1. Remove head set

2. Firmly grasp mask red tabs and completely pull mask from stowage box (Figure 17-2 Detail A).

3. Press red inflation tabs to fully inflate the harness

   - Allows donning in five seconds or less (Figure 17-2 Detail B and C).

4. Position mask over nose and mouth; release red tabs

   - Masks then deflates and tightens comfortably and securely around the head (Figure 17-2 Detail D).
   - Select crew comm switches to mask

Smoke goggles are stored beneath either pilot seat. In a pressurized mode, some of the oxygen is diverted to the goggles to prevent contamination from entering the goggles.

Detail A

Detail B

Detail C

Detail D

Figure 17-2. Crew Oxygen Mask

CREW OXYGEN MASK STOWAGE

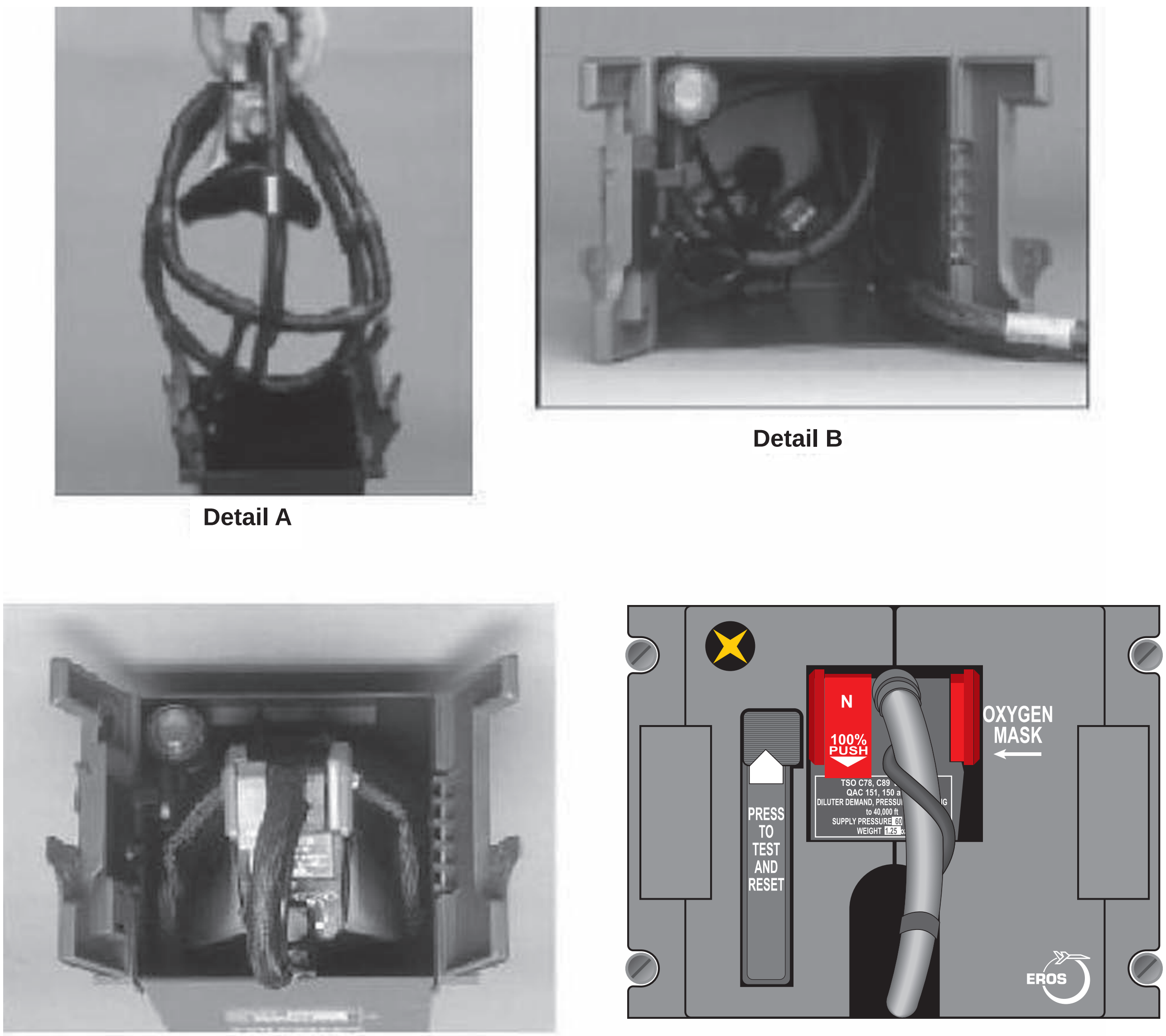
To stow the crew mask, ensure the pneumatic and electrical connections of the mask regulator are properly connected with the mating receptacles of the panel.

Ensure the harness is properly positioned behind the face piece (Figure 17-3, Detail A).

Coil the hose and position it on the bottom of the stowage box and position it to the middle (Figure 17-3, Detail B).

Fold the deflated harness into the face piece, and lower it into the stowage box. Make sure the mask is properly positioned when the doors are closed (Figure 17-3, Detail C).

Figure 17-3, Detail D shows the mask properly stowed and the doors of the crew oxygen storage box closed.



Detail A

Detail B

Detail C

Detail D

Figure 17-3. Crew Oxygen Mask Stowage

PASSENGER OXYGEN SYSTEM

The OXYGEN SUPPLY valve controls passenger oxygen.

With the OXYGEN SUPPLY in the ON position, low pressure oxygen (70 ±10 psi) is available and controlled by the passenger oxygen control panel.

The passenger oxygen control panel consists a selector knob, an oxygen control solenoid, OXYGEN ON indicator light, and a therapeutic selector switch (Figure 17-4).

The flow of oxygen to the passenger compartment masks is controlled by a normally closed solenoid within the passenger oxygen control panel. The solenoid receives electrical power from the No. 2 distribution bus.

COMPONENTS

Passenger Oxygen Control Panel

The passenger oxygen control panel contains a normally de-energized (closed) solenoid that controls the flow of oxygen to the passenger masks and is controlled by the OFF-AUTO–BYPASS selector knob (Figure 17-5).

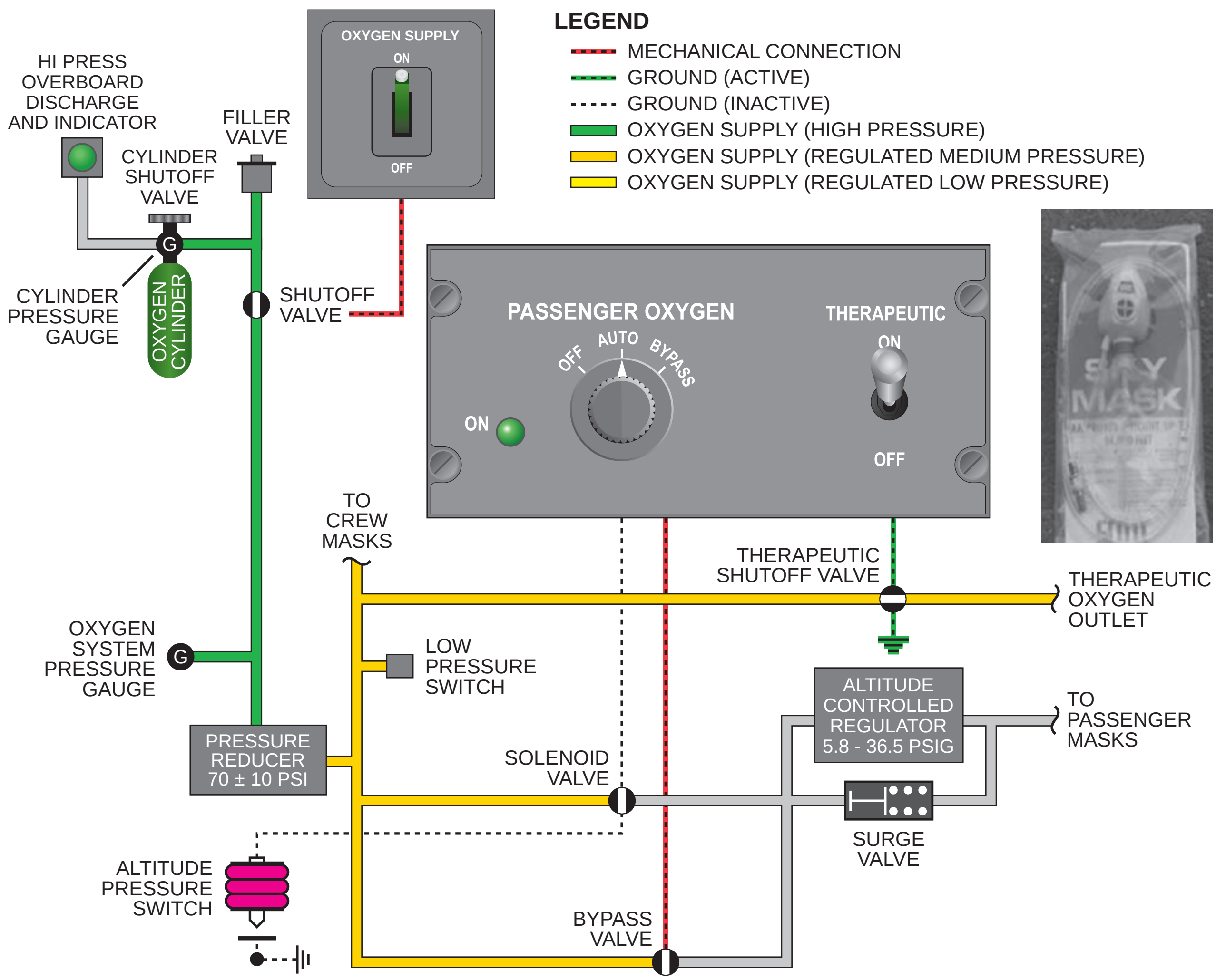


Figure 17-4. Passenger Oxygen Supply

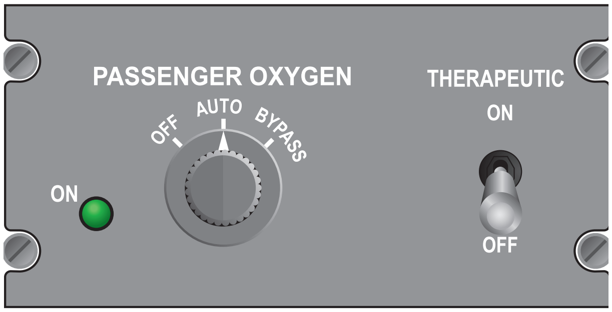


Figure 17-5. Passenger Oxygen Control Panel

The control knob on the passenger control panel has three positions as follows:

- OFF—Solenoid closes, stopping the flow of oxygen to the masks
- AUTO—Solenoid remains closed until it receives a signal from the cabin pressure control system (CPCS) as follows:
  - Landing field elevation below 8,000 ft—CPCS opens solenoid when cabin altitude reaches 13,500 ±250 ft
  - Landing field elevation 8,000 ft or above and aircraft altitude below 25,000 ft—CPSC opens solenoid when cabin altitude reaches 14,750 —±250 ft
- BYPASS—Oxygen bypasses the solenoid to flow directly to the passenger masks at any altitude in the event of a solenoid failure, loss of electrical power, or at cabin altitudes below 13,500 ft.

Surge Valve

All oxygen flows to cabin masks through a surge valve. It supplies oxygen to the drop out boxes at 70 psi for five seconds. This ensures that the masks deploy from their storage boxes. After five seconds the oxygen pressure is lowered to the normal pressure based on cabin altitude.

Oxygen Pressure Switch

When oxygen is sent to the passenger masks, it closes a pressure switch in the passenger oxygen control panel and illuminates the OXYGEN ON light in the lower inboard section of the passenger oxygen control panel.

CAUTION

Ensure the passenger oxygen control panel is in the OFF position when the OXYGEN SUPPLY switch selected, or it may cause an inadvertent deployment of the passenger oxygen masks.

After the masks have deployed, oxygen is available to the drop out boxes proportional to cabin altitude. Oxygen flow to the mask is initiated by pulling the lanyard attached to the mask.

The masks contain a bag that receives the oxygen and a check valve that allows breathing cabin air as well as the oxygen. Since the amount of oxygen available to the mask is proportional to cabin altitude the bag may not fully inflate. The mask, however; is receiving the correct amount of oxygen.

THERAPEUTIC OXYGEN

For passenger medical use, a mask calibrated for therapeutic use is located in the cabin. The therapeutic plug port is on the right upper side panel, just aft of the right emergency exit. It is clearly labeled THERAPEUTIC OXYGEN.

When the therapeutic oxygen toggle switch is selected to the ON position, it provides oxygen flow to a single point in the cabin next to the aft seats (Figure 17-5). Plugging the therapeutic mask into the outlet starts the flow of oxygen to the mask.

Therapeutic oxygen is a constant line pressure of 70 ±10 psi to a restrictor. The restrictor reduces the pressure further.

The therapeutic oxygen mask has a 98 inch flexible hose to accommodate any seat in the cabin. The therapeutic oxygen mask and flexible hose are stored in a sealed plastic bag (see Figure 17-4).

COMPONENTS

OXYGEN CYLINDER

Gaseous oxygen for the flight crew and passengers is stored in a 77 cu ft high pressure light-weight cylinder. An optional 115 cu ft cylinder may be installed.

The cylinder is on the right side of the nose compartment below the shelf. (Figure 17-6).

The oxygen cylinder is equipped with the following:

- Over pressure discharge valve
- Pressure transducer
- Pressure reducer
- Oxygen filler valve
- Charging pressure gauge
- Shutoff valve
- Pressure relief valve

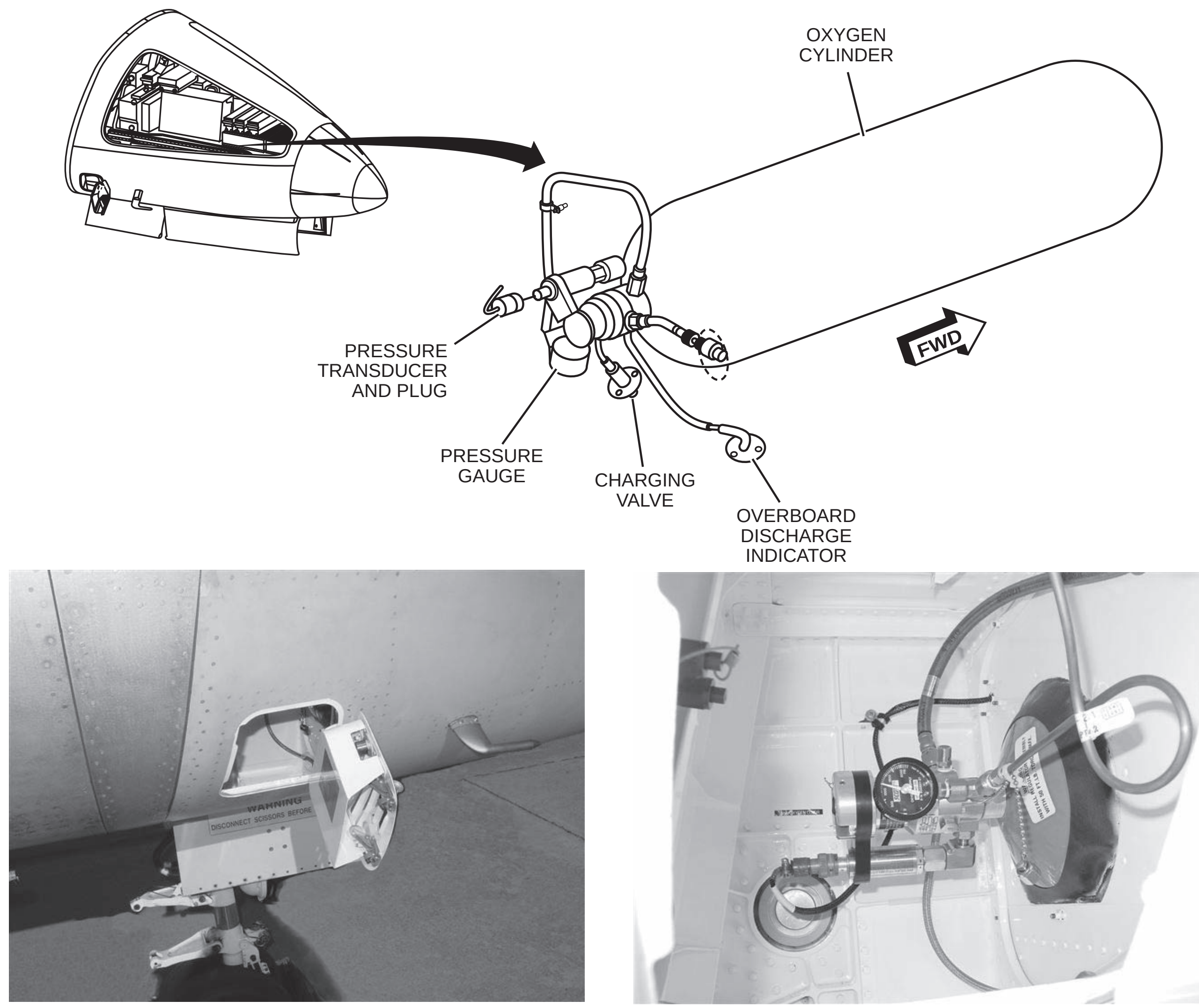


Figure 17-6. Oxygen Cylinder

OVERPRESSURE DISCHARGE VALVE

The overpressure discharge valve defaults to the OPEN position when the cylinder reaches a pre-determined pressure. The exit port for this valve is equipped with a green blowout disk to indicate cylinder overpressure discharge. This disk is on the lower right side of the nose compartment and should be checked during preflight inspection.

The valve is designed to open between 2,500 and 2,775 psi. If the discharge valve opens, it will completely empty the cylinder. If this occurs, the cylinder must be sent out to a FAA authorized service facility to be re-serviced.

PRESSURE TRANSDUCER

The pressure transducer sends electrical signals to the EICAS proportional to the cylinder oxygen pressure. Oxygen cylinder pressure is displayed on the lower right side of the secondary page of the MFD.

Oxygen cylinder pressure is displayed in green for pressures above 800 psi and changes to amber if the pressure drops to less than 800 psi.

PRESSURE REDUCER

The pressure reducer lowers the cylinder oxygen pressure to system pressure of approximately 70 ±10 psi.

The pressure reducer also incorporates a relief valve set to open at 94 psi to limit the oxygen pressure in the cabin to a safe value.

OXYGEN FILLER VALVE

The oxygen filler valve is located behind the door on the lower right side of the nose compartment. (Figure 17-6). This valve services the oxygen cylinder to 1,850 psi at 70°F (21°C).

CHARGING PRESSURE GAUGE

The pressure gauge shows cylinder pressure only. The charging pressure gauge eliminates the need to have electrical power on the aircraft while servicing the oxygen system.

SHUTOFF VALVE

The cylinder shutoff valve allows manual isolation from the rest of the system for maintenance. This valve must be safety wired in the OPEN position for flight. When the cylinder shutoff valve is closed, the low pressure line is vented to atmosphere.

PRESSURE RELIEF VALVE

The pressure relief valve prevents cylinder over pressure even with the shutoff valve closed.

This valve must be safety wired in the open position prior to flight.

The low pressure oxygen is routed from the pressure reducer to the SUPPLY valve.

A low pressure switch in the right side console illuminates the OXY MASKS PRESS LOW amber caution message on the EICAS. This switch is set to close when the system pressure drops to 50 psi or less.

CONTROL AND INDICATIONS

OXYGEN SUPPLY VALVE

The OXYGEN SUPPLY valve controls oxygen to the crew and passenger compartments. (Figure 17-7).

Figure 17-7. Oxygen Supply Valve

CREW OXYGEN MASK

- N—Normal position puts the mask into a diluter demand mode. In this mode the percentage of oxygen supplied to the crewmember is a function of cabin altitude up to a cabin altitude of 30,000 ft. Above 30,000 feet of cabin altitude the mask automatically switches to 100% oxygen.
- 100%—In this position, 100% oxygen is supplied to the mask supplies at any cabin altitude.
- N/100% diluter switch—Movable selector on the crew mask that selects the percentage of oxygen supplied to the mask (Figure 17-8).
  - When rotated counterclockwise supplies 100% oxygen and pressure breathing.

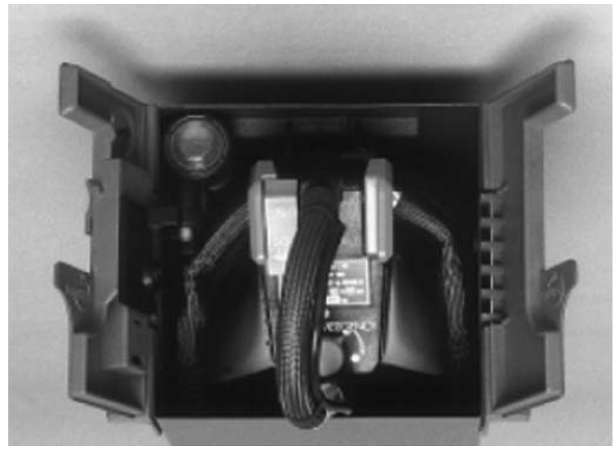


Figure 17-8. Crew Mask and Mask Mounted Regulator

- EMERGENCY/TEST button
  - When button depressed supplies pressurized oxygen to the mask. This checks the pressure breathing performance on the ground as well as checks for any loose or leaking connections.

VENT VALVE

This valve allows pressurized oxygen from the crew mask to enter the smoke goggles to alleviate vapor formation in the smoke goggles.

OXYGEN PRESSURE INDICATION

The oxygen cylinder pressure is displayed in the lower right side of the secondary page of the MFD (Figure 17-9).

The pressure transducer on the oxygen cylinder sends electrical signals to the DCU, which in turn, sends the information to the secondary page of the MFD.

The oxygen pressure is displayed in green for pressures above 800 psi. If the pressure in the cylinder drops, the indication turns amber.

Figure 17-9. Oxygen Pressure Indication

PASSENGER OXYGEN CONTROLLER

The passenger oxygen control panel has a three position rotary knob to control the flow of oxygen to the passenger masks:

- OFF—Solenoid closes, stopping the flow of oxygen to the masks
- AUTO—Solenoid remains closed until it receives a signal from the cabin pressure control system (CPCS) as follows:
  - Landing field elevation below 8,000 ft—CPCS opens solenoid when cabin altitude reaches 13,500 ±250 ft
  - Landing field elevation 8,000 ft or above and aircraft altitude below 25,000 ft—CPSC opens solenoid when cabin altitude reaches 14,750 ±250 ft
- BYPASS—Oxygen bypasses the solenoid to flow directly to the passenger masks at any altitude

OXYGEN ON LIGHT

The OXYGEN ON light on the passenger oxygen control panel illuminates any time oxygen is sent to the passenger masks.

THERAPEUTIC OXYGEN

This switch is an ON/OFF toggle switch that supplies oxygen to the therapeutic plug in the cabin when it is turned ON (see Figure 17-5).

LIMITATIONS

There are no specific limitations relating to the oxygen system.

EMERGENCY AND ABNORMAL OPERATIONS

EICAS MESSAGES

There are two EICAS messages associated with the oxygen system (Figure 17-10).

OXY MASKS PRESS LOW—Caution EICAS message illuminates when the pressure downstream of the OXYGEN SUPPLY valve drops to 55 psi.

OXY QTY LOW—Caution EICAS message illuminates when the oxygen pressure in the cylinder drops to less than 800 psi.

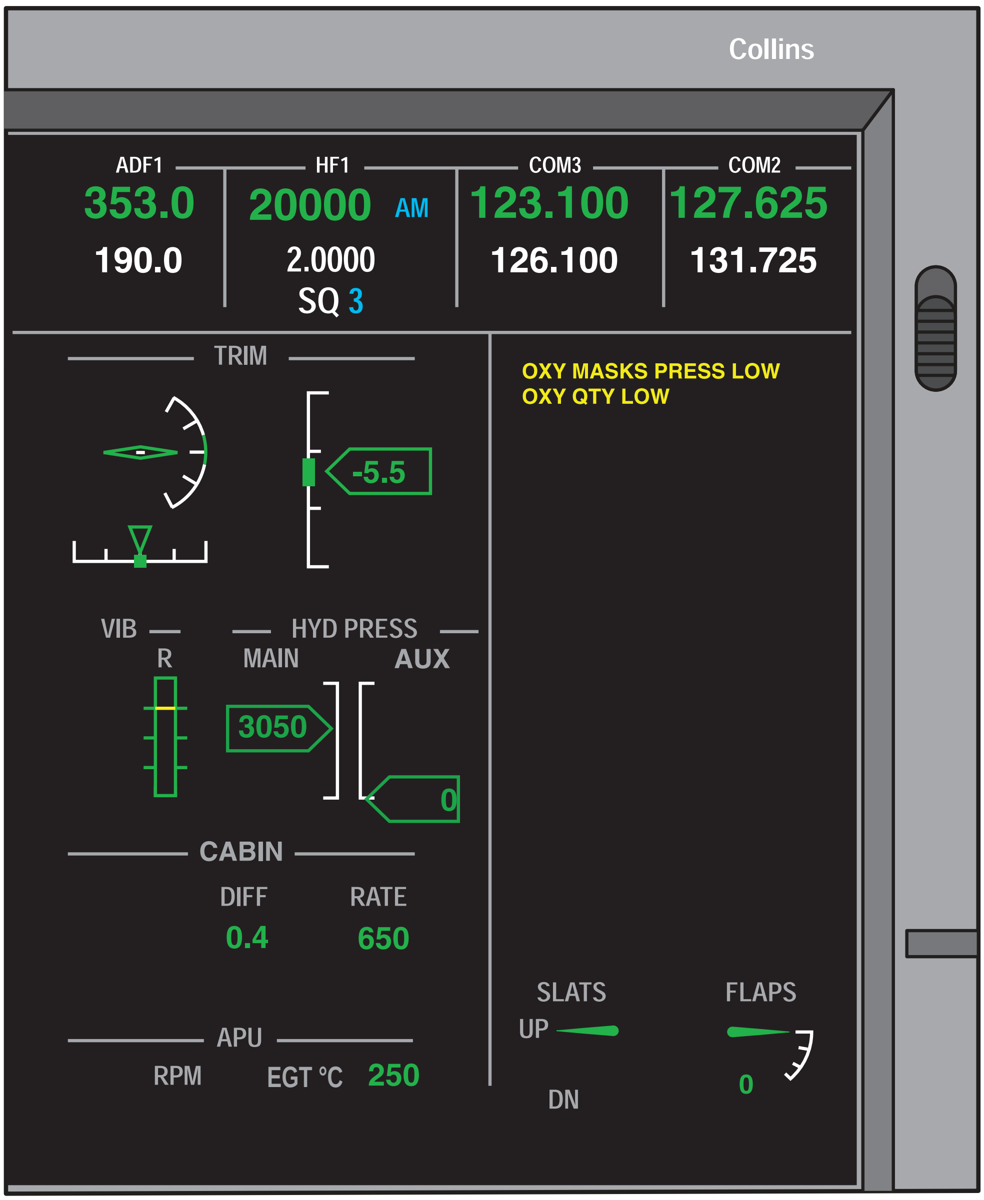


Figure 17-10. Oxygen EICAS Indications

QUESTIONS

1. Electrical power for automatic deployment of the passenger masks is supplied from:

A. No.1 distribution bus
B. The left and right main buses
C. No. 2 distribution bus
D. The priority bus

2. A normal oxygen cylinder pressure at 70°F is:

A. 1,780 psi
B. 1,870 psi
C. 1,700 psi
D. 1,850 psi

3. Following a decompression at 45,000 feet, an emergency descent to 20,000 feet is executed. The oxygen duration (77 ft cu cylinder) time for two crewmembers and four passengers if oxygen pressure on reaching 20,000 feet is 1,380 psi is:

A. 1 hour 50 minutes
B. 2 hours 25 minutes
C. 2 hours 13 minutes
D. 1 hour 27 minutes

4. An amber CAS message OXY MASKS PRESS LOW illuminates when:

A. Oxygen mask supply pressure is 60 psi or less
B. Oxygen mask supply pressure is 55 psi or less
C. Oxygen bottle pressure is below 800 psi
D. Oxygen bottle pressure is below 60 psi

5. The cockpit indication that passengers are receiving oxygen is:

A. OXYGEN ON status CAS message
B. Yellow OXYGEN ON indicator light located on the passenger oxygen panel
C. Green OXYGEN ON indicator light located on the passenger oxygen panel
D. An amber OXYGEN ON CAS message

MANEUVERS AND PROCEDURES

CONTENTS

ILLUSTRATIONS

TABLE

MANEUVERS AND PROCEDURES

BEFORE TAKEOFF

FLIGHT PLANNING

In the early stages of a client's training, emphasis should be placed on determination of takeoff and landing distance, balanced field length, second segment limitations, and V_1, V_R, V_2, and V_{FS} speeds. The client should understand the use of charts published in the Airplane Flight Manual (AFM) for this purpose and should understand the uses and limitations of quick reference charts developed by the manufacturer.

WEIGHT AND BALANCE

Although covered in general terms during ground school, weight and balance computations from the client's own AFM should be emphasized due to variations in aircraft. Proficiency in the use of the loading slide rule or other appropriate means of weight and balance computations will be required.

PREFLIGHT INSPECTION

Preflight inspections should be carried out using methods described in the approved AFM. The instructor should lead the client through the first few preflight inspections, and then observe them on subsequent flights.

The following items must be aboard the airplane before it may be used for flight training:

- Airplane Flight Manual
- Aircraft registration or pink copy of application for registration
- Airworthiness certificate
- FCC license
- First aid kit
- Flashlight with minimum of two size D cells
- Navigational enroute, terminal area, and approach and letdown charts for area in which flight is to be conducted
- Minimum of one headset

MANEUVERS

TAKEOFF PROFILE

Figure MAP-1 is a representation of the standard takeoff profile recommended by FlightSafety International.

NORMAL TAKEOFF

This maneuver provides practice in the procedures and techniques employed in a normal takeoff (Figure MAP-2).

Prior to takeoff, the pilot should ensure that all checklists are completed and the airplane is in a safe configuration for departure.

Pilot's Procedures During Takeoff

The pilot aligns the airplane with the centerline of the runway and holds the wheel brakes. The pilot's left hand should be on the nosewheel steering and the right on the thrust levers. The pilot then smoothly advances the thrust levers and sets the computed N_1 on both engines prior to brake release. The pilot smoothly releases the brakes and maintains centerline with nosewheel steering. As the rudder becomes effective, the pilot continues the takeoff roll with his hand monitoring nosewheel steering until the copilot calls "80 knots." The pilot crosschecks airspeed and acknowledges, moving his left hand from nosewheel steering to the yoke not later than 90 knots. As the pilot takes control of the yoke,

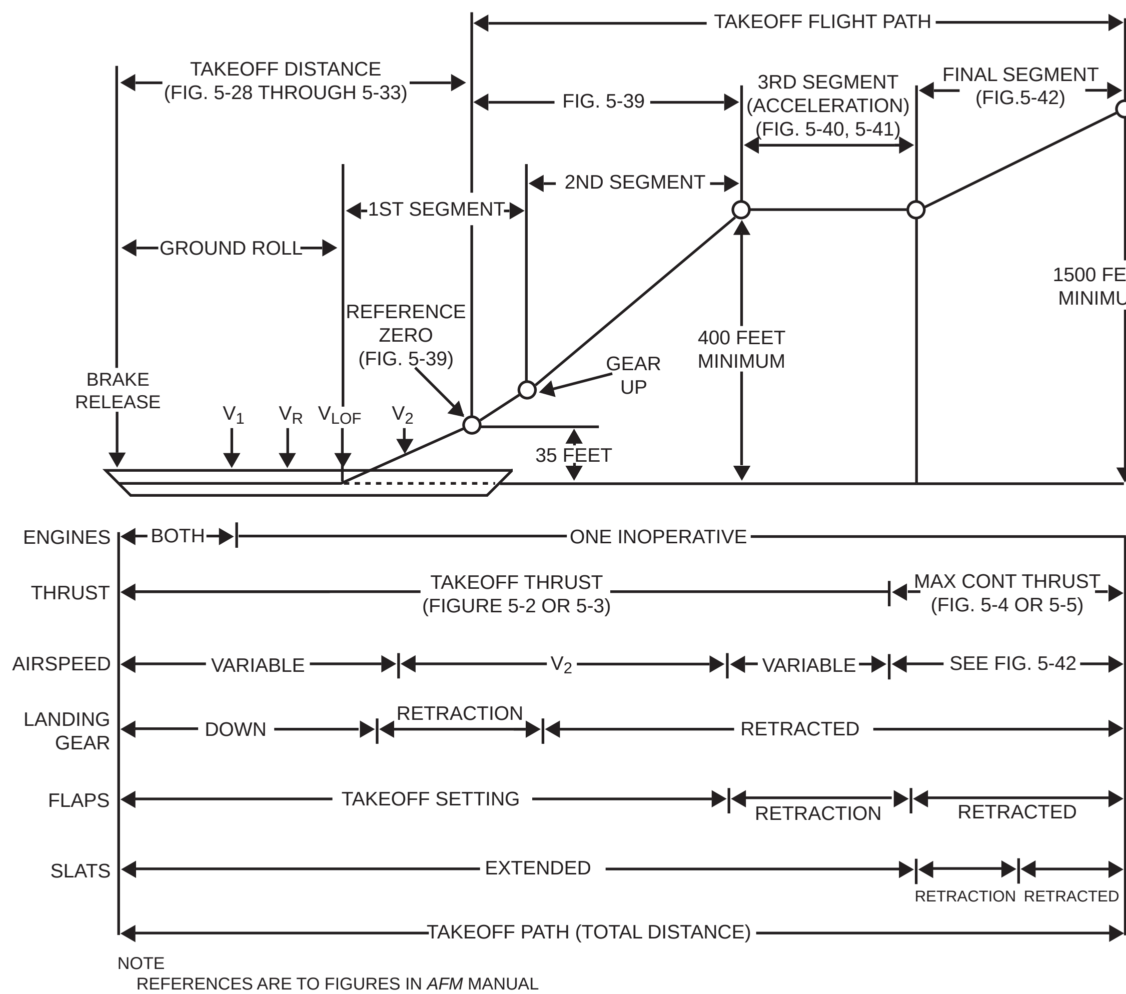


Figure MAP-1. Takeoff Path Profile (One Engine Inoperative)

he states "Pilot's yoke" and maintains forward pressure with cross wind correction as necessary. When the copilot calls "V_1" the pilot's right hand goes to the yoke.

When the copilot calls "rotate," the pilot smoothly rotates the airplane (not greater than 3° per second) toward a pitch attitude of 15° for a flaps 12° takeoff (14° for flaps 20°, 16° for flaps 0°).

As the airplane reaches the desired pitch attitude, the flight director command bar may be "pitch sync'd" to assist in maintaining that attitude. When the copilot calls "positive rate," the pilot will confirm the aircraft is airborne and call "gear up, thrust reversers off" to retract the landing gear. At a minimum of 400 feet (or obstacle clearance altitude) and an airspeed of V_2+10 knots minimum, direct the copilot to retract the flaps. Continue the climb, adjusting the power and pitch attitude. Retract the slats at V_{FS}, set climb power, then call for the "After Takeoff" checklist.

Above 3,000 feet AGL, with climb power set, accelerate to 250 knots.

NOTE

If an engine failure occurs above V_1, follow the "Engine Failure During Takeoff Above V_1" procedures.

CAUTION

Ensure that your feet are low enough on the rudder pedals to prevent unwanted application of brakes during takeoff, especially if an engine fails, and full rudder is required.

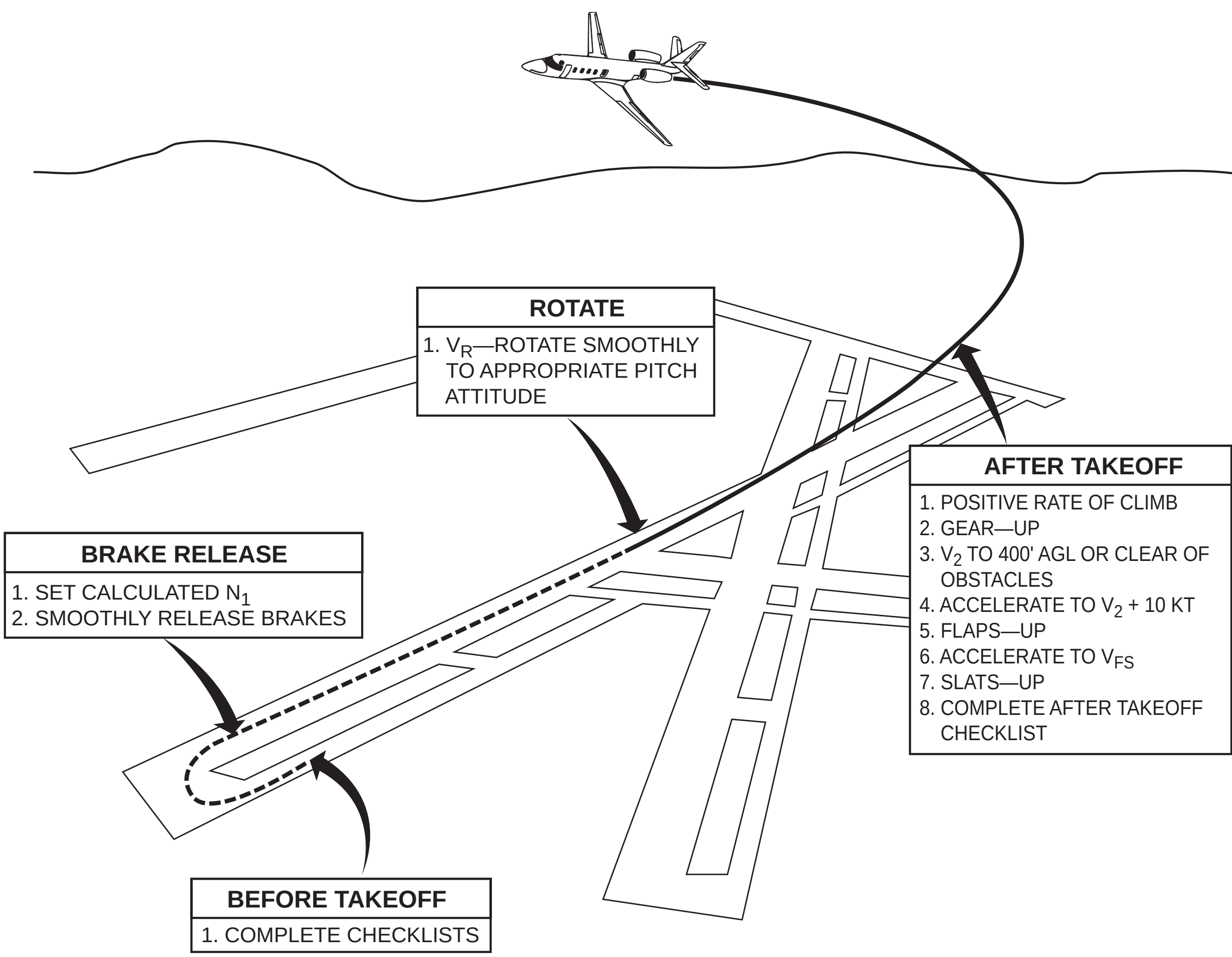


Figure MAP-2. Normal Takeoff

Copilot's Procedures During Takeoff

Once takeoff clearance is received, the copilot completes the lineup check. On takeoff roll, maintain forward pressure and crosswind correction on the yoke. Monitor engine instruments, annunciators, and airspeed. As the airspeed reaches 80 knots, the copilot calls "80 knots." When the pilot calls "pilot's yoke," the copilot relinquishes control and continues to monitor engine instruments, annunciators, and airspeed. When the airspeed reaches V_1, the copilot calls "V_1." When the airspeed reaches V_R, the copilot calls "rotate." Monitor the rotation, and when the VSI, altimeter, and visual indications show a positive rate of climb, call "positive rate." When the pilot directs "gear up, thrust reversers off" the copilot repeats the call "gear up, thrust reversers off" places the landing gear handle to the up position, and ensures the landing gear retracts. Monitor the initial climb profile and notify the pilot of any deviations. At 400 feet on the radar altimeter, the copilot calls "400 feet."

When the pilot calls for "flaps up," the copilot ensures that the airspeed is V_2 + 10 knots minimum and retracts the flaps. Ensure the flap indicator indicates flaps 0°. When V_{FS} is reached, the pilot calls for "slats up". The copilot ensures that airspeed is V_{FS} minimum and retracts the slats. Ensure the slat indicator indicates slats 0°. When the pilot directs, accomplish the "After Takeoff" checklist.

Acceptable Performance Guidelines

- V-speeds are considered as minimum and should be controlled within +10/–0 knots.

TAKEOFF WITH ENGINE FAILURE AT OR ABOVE V_1

This procedure provides practice in procedures and techniques employed in operating with an engine that fails at or above V_1 (Figure MAP-3).

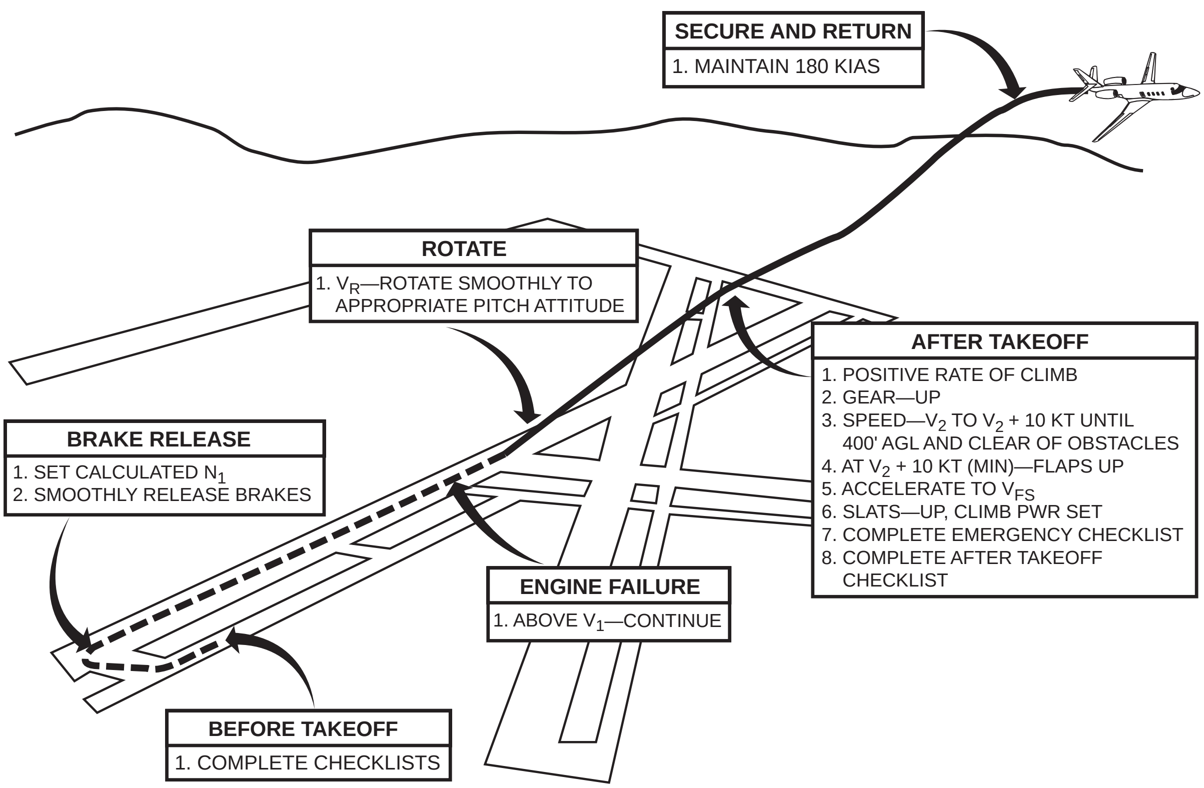


Figure MAP-3. Takeoff—Engine Failure At or Above V_1

Aircraft

In the aircraft, the instructor pilot will retard the thrust lever of either engine to idle at a minimum of 100 feet AGL.

Simulator

The simulator instructor will fail an engine at or above V_1.

Procedures and Techniques

Normal takeoff procedures will be followed until V_1. The engine failure may be announced by either pilot. The pilot flying (PF) will maintain directional control and forward pressure on the control column. The PF will ensure that the thrust levers are advanced to maximum thrust. Accelerate to V_R. When the pilot not flying (PNF) calls “rotate,” the PF will smoothly rotate the aircraft to an attitude of approximately 10° (G/A command bar may be used as a guide). Retract the landing gear when positively airborne. Accelerate to, and maintain V_2 minimum until 400 feet AGL and clear of obstacles.

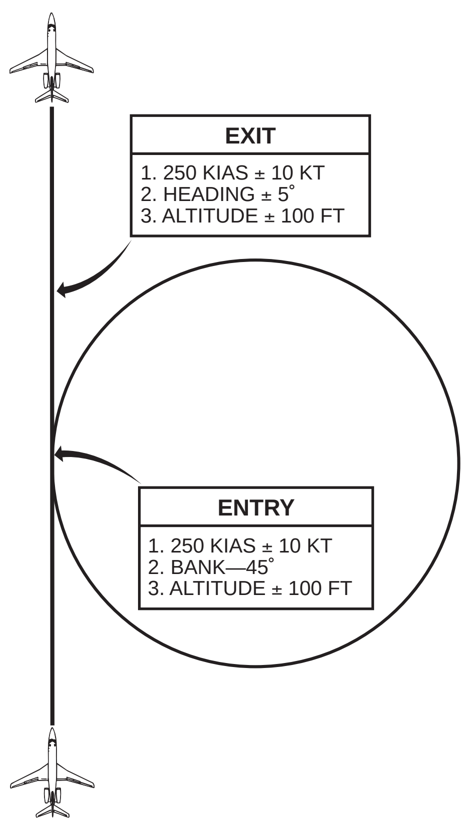


Figure MAP-4. Steep Turns

At 400 feet AGL minuimum, accelerate to V_2 + 10 knots and retract the flaps (see note below on this page). Continue acceleration to V_{FS} and retract the slats. With final segment climb established and climb power set above 400 AGL, identify the failed engine and accomplish the appropriate emergency checklist. After the engine is secured, accomplish the “After Takeoff” checklist.

Common Errors

- Rotating early, i.e: at V_1 instead of V_R
- Overrotation
- Applying brake with rudder while on the runway (feet too high on the rudder pedal)
- Not maintaining desired pitch attitude after takeoff, allowing the aircraft to descend

NOTE

If an engine failure occurs after V_1, but prior to V_2, maintain V_2, until 400 feet AGL or obstacle clearance altitude.

If an engine fails above V_2 but less than or equal to V_2 + 10 knots, maintain the airspeed at which the engine failed.

If an engine fails above V_2 + 10 knots, increase pitch to reduce speed and maintain V_2 + 10 knots.

Acceptable Performance Guidelines

- Maintain positive aircraft control.

STEEP TURNS

This maneuver affords practice in controlling the aircraft with greater than normal bank angles (Figure MAP-4). These are accomplished at all gross weights and altitudes above 5,000 feet AGL. Stabilize speed at 250 KIAS and make a smooth entry to a 45° bank. Turns should exceed 180°, but need not be greater than 360°. Use stabilizer trim to counteract pitch down moment to maintain altitude. The IVSI should be used in conjunction with the altimeter during this maneuver.

NOTE

Practice at other speeds may be utilized at the instructor pilot's discretion.

Acceptable Performance Guidelines

- Altitude ± 100 feet
- Airspeed ± 10 knots
- Heading ± 5°
- Bank Angle ± 5°

APPROACHES TO STALL

These procedures provide practice in the normal procedures and techniques used in stall recognition and recovery (Figures MAP-5 through MAP-7). Use of engine igniters is recommended during the maneuver.

Procedures and Techniques

In the aircraft, the approach-to-stall procedures should be practiced at a minimum altitude of 10,000 feet above terrain. The maneuver should begin insuring the autopilot is disconnected at 180 knots decelerating. Initiate recovery by advancing the throttles to maximum power. Approaches to stall should be practiced in the clean configuration, with partial flaps, and in the landing configuration, but not neccessarily in that order. They should also be practiced with wings level and during turns. Some altitude loss may be experienced during the stall buffet recovery; however, keep the loss to a minimum. In addition, keep increases in altitude to a minimum.

Clean Configuration

This maneuver should be started with the power at idle. Maintain a constant altitude and trim the aircraft as speed decreases. Trim may be used until stall buffet onset. Begin the stall recovery at the first indication of buffet onset by advancing the thrust levers to maximum power while allowing the pitch attitude to decrease not more than 1 or 2°, maintaining a minimum of 10° noseup. Do not release back pressure on the yoke. Roll the wings level, if applicable. This maneuver shall be considered complete when stabilized at the original altitude at not less than V_{FS} or greater than 180 knots.

Flaps 20°, Bank 15–30°

Configure the aircraft with flaps 20° and establish bank in either direction. Start this maneuver with the power reduced to approximately 50%-60% N_1. Maintain a constant altitude and trimmed aircraft as speed decreases. Trim may be used until stall buffet onset. Begin the stall recovery at the first indication of buffet onset by advancing the thrust levers to maximum power while allowing the pitch attitude to decrease not more than 1 or 2°, maintaining a minimum of 10° noseup. Do not release back pressure on the yoke. Simultaneously, roll the wings level. Confirm flaps at 20°. Establish a positive rate of climb. Once a positive rate is confirmed, accelerate to V_{REF}. Retract the flaps to 12°. Upon returning to the original altitude and accelerating to V_{REF} + 10 knots, retract the flaps to 0°. Accelerate to V_{FS} and retract the slats. This maneuver shall be considered complete when stabilized at the original altitude at not less than V_{FS} or greater than 180 knots.

Landing Configuration

Configure the aircraft with flaps 40° and gear down. Accomplish this maneuver straight ahead. Start this maneuver with the power reduced to approximately 60% N_1. Maintain a constant altitude and trim the aircraft as speed decreases. Use trim as required until stall buffet onset. Begin the stall recovery at the first

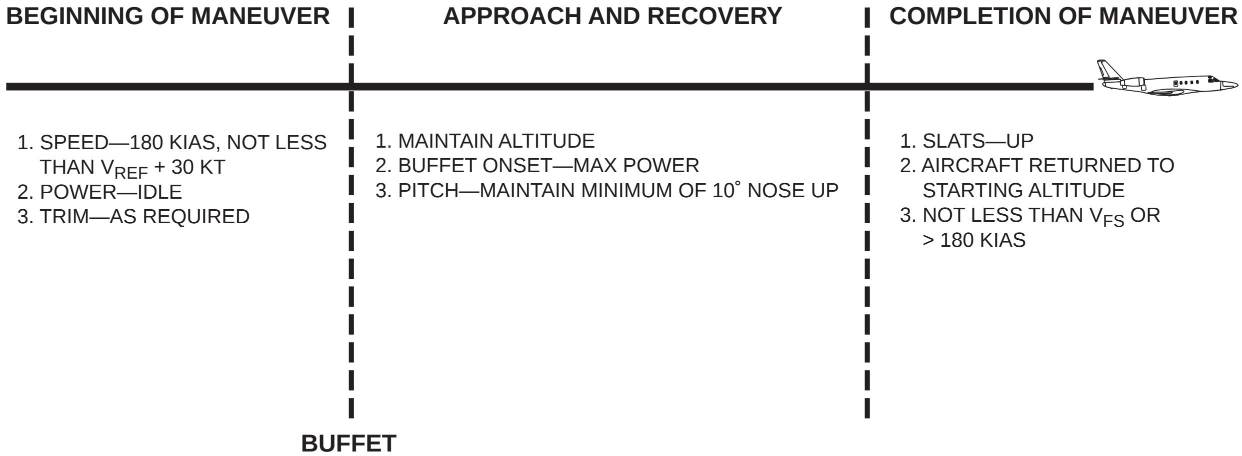


Figure MAP-5. Approach to Stall—Clean Configuration

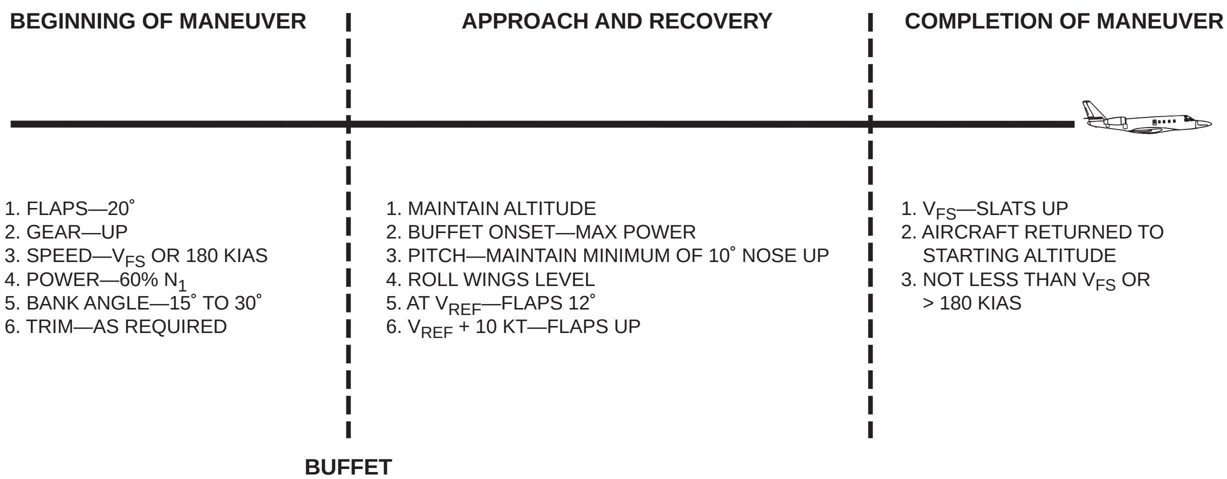


Figure MAP-6. Approach to Stall—Flaps 20°, 15-30° Bank

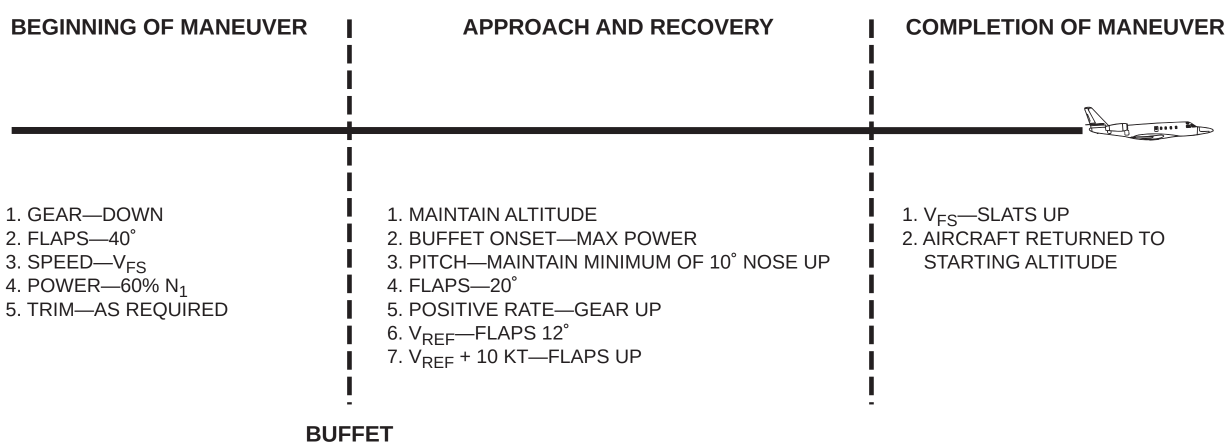


Figure MAP-7. Approach to Stall—Landing Configuration

indication of buffet onset by advancing the thrust levers to maximum power. Do not allow the pitch attitude to decrease more than 1 or 2°. Maintain a minimum of 10° noseup. Do not release back pressure on the yoke. Retract the flaps to 20°. Establish a positive rate of climb. Once a positive rate is confirmed, retract the gear and accelerate aircraft to V_{REF}. Retract the flaps to 12°. Return to the original altitude and accelerate to V_{REF} + 10 knots. Retract the flaps to 0°. Accelerate to V_{FS} and retract the slats. This maneuver shall be considered complete when stabilized at the original altitude at not less than V_{FS} or greater than 180 knots.

NOTE

The angle-of-attack indicator may be used as a visual reference for approaching stall speed.

Common Errors

- Not using noseup trim during deceleration
- Allowing pitch attitude to decrease excessively during recovery
- Excessive altitude gain during recovery

Acceptable Performance Guidelines

- Maintain positive control of the aircraft.

MANEUVERING AT MINIMUM SPEED (SLOW FLIGHT, FAA BALLET)

These maneuvers demonstrate the degree of controllability available while in close proximity to the prestall buffet. They provide the opportunity to practice control techniques which are beneficial in the low-speed regimes encountered during takeoffs and landings.

In the aircraft, these maneuvers should be accomplished at a minimum of 5,000 feet above the terrain at any gross weight. Various configurations and power ranges are also utilized. Table MAP-1 shows some recommended settings for practice purposes.

The following are recommended slow flight practice maneuvers which develop control touch and feel:

- Turns to specific headings, using bank angles of not more than 15°
- Climbing or descending turns to specified headings and altitudes
- Banks of 10 to 15° while turning alternately to a point 20° either side of a specified heading

Emphasis is placed upon coordinated flight and use of controls, as well as prompt, corrective action in response to undesired airspeed or altitude trends.

Table MAP-1. SLOW FLIGHT MANEUVERING SPEEDS

CONFIGURATION	AIRSPEED	POWER
CLEAN	V_{REF} + 40 Knots	As Necessary
SLATS	V_{REF} + 30 Knots	As Necessary
FLAPS 12°	V_{REF} + 20 Knots	As Necessary
FLAPS 20°	V_{REF} + 10 Knots	As Necessary
FLAPS 20°, Gear Down	V_{REF}	As Necessary
FLAPS 40°, Gear Down	V_{REF} – 10 Knots	As Necessary

Acceptable Performance Guidelines

- Maintain positive aircraft control
- Altitude ± 100 feet
- Airspeed +5 to –0 knots
- Heading ± 5°, when appropriate

MANEUVERING WITH ENGINE FAILURE

This maneuver provides practice in properly maintaining aircraft control during one of the more critical in-flight emergencies. It develops a knowledge of aircraft characteristics under adverse conditions, together with control applications necessary to achieve a maximum margin of safety.

This maneuver is practiced at a minimum altitude of 5,000 feet above the terrain until the pilot has demonstrated ability to apply rudder control forces during partial power maneuvering and has satisfactorily accomplished the "Engine Failure" checklist.

Engine failure is simulated by reducing power to flight idle. The pilot should request that the "Engine Failure" checklist be read and ensure that all items are accomplished.

The pilot demonstrates knowledge of the minimum aircraft speeds appropriate to the configuration of the aircraft.

Turns into and away from the dead engine are practiced to demonstrate proper control technique.

Acceptable Performance Guidelines

- Maintaining positive aircraft control
- Proper use of checklists

TRIM AND PITCH CHANGE CHARACTERISTICS

These maneuvers are practiced to acquaint the pilot with aircraft trim and pitch characteristics. They aid in understanding and properly anticipating trim changes made necessary by the pitch effects of thrust, landing gear movement, flaps, and airbrakes.

The aircraft should be trimmed for straight and level flight (hands off) at approximately 160 KIAS. This maneuver demonstrates the effects of:

- Increased and decreased thrust
- Extension and/or retraction of landing gear
- Extension and/or retraction of flaps
- Raising and lowering of airbrakes

For each of these conditions, the pilot inputs no control resistance and observes their effects on aircraft pitch. Subsequently, these same maneuvers are accomplished with the pilot maintaining a constant altitude so that control force changes may be experienced. The following should be noted:

- Rapid increase in power.......Nosedown
- Extension of landing gear.....No change
- Extension of the flaps............ Nose up
- Extension of airbrake (may be extended at speeds up to V_{MO}/M_{MO}......Nose up

Placement of the engines in a relatively high position with regard to the center of gravity gives significant nose up or nose down pitching moments. With a rapid decrease in power, there is a nose up tendency. Conversely, rapid increases in power induces nose down tendencies.

Longitudinal or stabilizer trim is explained and demonstrated, using both the normal and override systems.

Acceptable Performance Guidelines

No specific degree of proficiency is required.

UNUSUAL ATTITUDES (JET UPSET)

These procedures provide training in recognizing unusual attitudes, and in recovery procedures and techniques.

Procedures and Techniques

In the aircraft, unusual attitudes are accomplished at least 10,000 feet above the terrain, but not above 18,000 feet pressure altitude. The attitude should not be allowed to exceed 90° of bank. The airspeed should not be allowed to increase to more than V_{MO} minus 20 knots or decrease to less than V_{REF} . There will be no flight instrument failures or simulated failures before or during practice recoveries. In the aircraft, these restrictions are designed to protect the aircraft in the event of misapplication of control pressures or improper analysis. The aircraft should be placed in either a nose-high attitude toward an approaching stall with power reduced or a nose-low diving attitude with speed increasing and power increased. Turn as desired. In the simulator only, at least one recovery should be from a bank of greater than 90°.

Nose High

Upon confirming a nose-high decreasing airspeed unusual attitude using primary and performance instruments, the pilot should simultaneously increase power to maximum and roll the wings in the shortest direction toward a 60 to 90° bank. Do not use control pressures which would result in negative g-forces in excess of aircraft limits in order to prevent damage to the aircraft and reduce the possibility of engine flameout. Maintain the bank until the nose of the aircraft is approaching the horizon. Roll the wings level and allow the airspeed to build. Recovery is complete when the aircraft is returned to level flight at V_{REF} + 30 knots.

Nose Low

Upon confirming a nose-low increasing airspeed unusual attitude using primary and performance instruments, the pilot should simultaneously reduce power to idle and roll the aircraft in the shortest direction (towards the sky pointer) to establish upright, wings level flight. Deploy the airbrakes (if inverted, delay airbrake deployment until the wings are rolled through the 90° position). Increase the aircraft attitude until the nose of the aircraft is on the horizon. Do not use control pressures which would result in g-forces in excess of aircraft limits. Retract the airbrakes when the airspeed is safely reduced. Recovery is complete when the aircraft is returned to level flight and airspeed is reduced to V_{MO} or lower.

Acceptable Performance Guidelines

- Maintain positive control of the aircraft.
- Recovery should be sufficiently positive with minimum g-forces employed.
- Recovery should be quickly and safely executed.

HOLDING

This maneuver provides training in holding pattern procedures. Within three minutes prior to the holding fix, initiate a speed reduction to arrive at the fix at or below the maximum holding airspeed. One-minute legs will be flown at or below 14,000 feet, one-and-one-half minute legs above 14,000 feet. Maximum holding airspeeds for civil turbojet aircraft are contained in the Airman's Information Manual (AIM).

Recommended holding procedures and airspeeds:

- Maintain 180 knots or maximum endurance holding airspeed from the holding chart in the clean configuration

Letdown in the holding pattern while maintaining adequate power for anti-icing may be accomplished by lowering the landing gear and/or extending the airbrakes.

Acceptable Performance Guidelines

- Adhering to holding procedures as listed in the AIM
- Altitude ±100 feet while holding
- Airspeed ±10 knots

ILS APPROACH—TWO ENGINES

This maneuver provides practice in terminal area arrival using the ILS for the final approach portion (Figure MAP-8).

Front-Course ILS

Complete the "Before Landing" checklist down to "Landing Gear—Down." Conduct the approach briefing prior to entering the approach pattern.

The crew may be cleared for a front-course ILS approach from any specified position. Determine if the ILS approach is to be conducted using the FMS or raw data NAV, then configure accordingly.

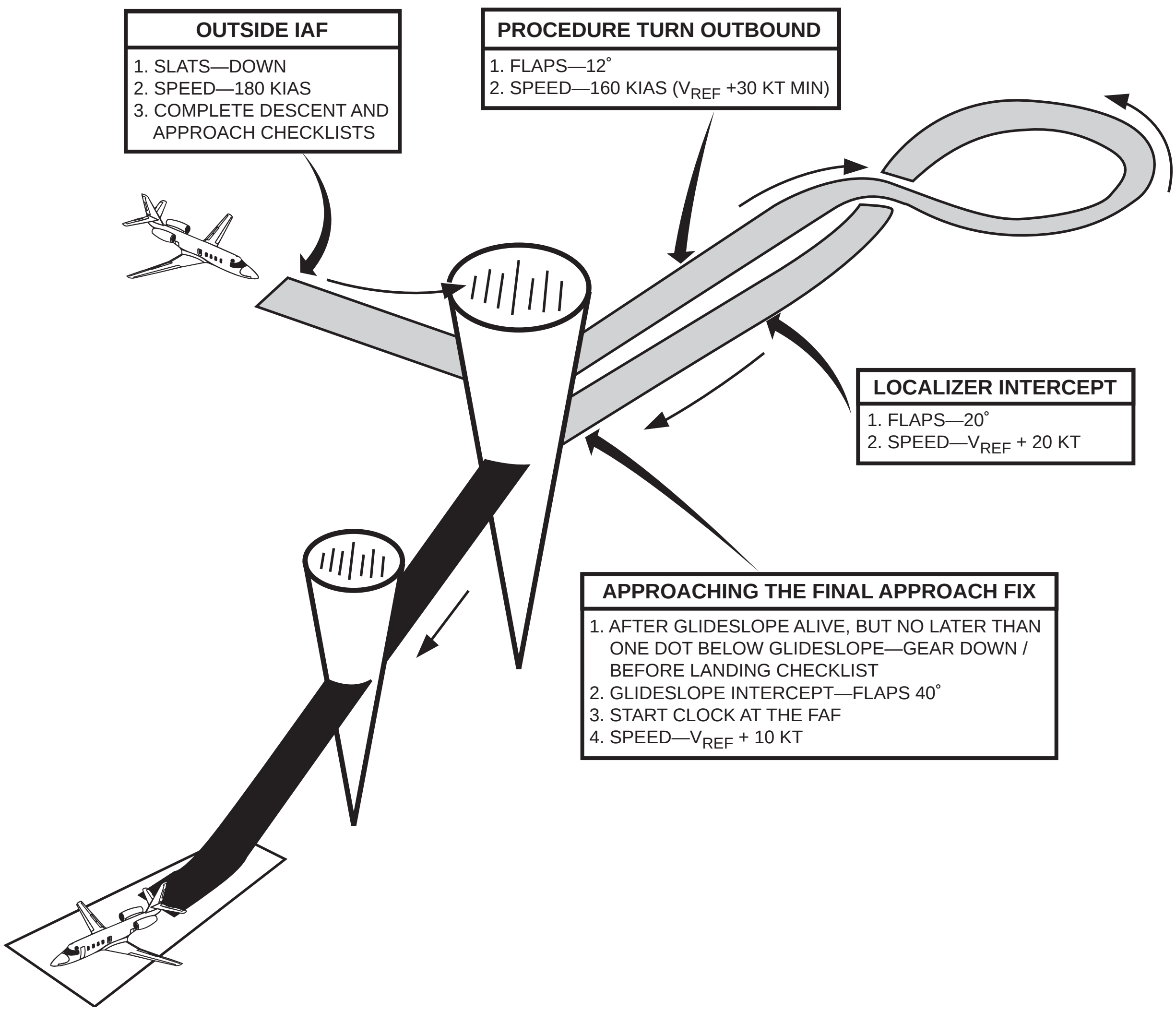


Figure MAP-8. ILS Approach—Two Engines

During the initial phase of the approach maintain 180 knots with slats extended. During the procedure turn, or base leg, extend the flaps to 12º and maintain 160 knots. When cleared for the approach and within 90º of the inbound course, select APPR on the flight fuidance panel.

After localizer interception, extend the flaps to 20º and reduce the airspeed to V_{REF} + 20 knots. After glide slope alive (but no later than one dot below the glide slope), extend the landing gear and continue the "Before Landing" checklist. At or before glide slope intercept, extend the flaps to 40º. Stabilize the airspeed at V_{REF} + 10 of the gust factor for the existing weight.

Upon leaving the final approach fix inbound, the pilot monitoring (PM) calls "Final approach fix and altimeters checked." At 500 ft. above decision altitude (DA), the PM calls "500 feet above DA, instrument check, no flags." At 100 ft above DA, the PM calls "Approaching minimums." At DA the PM calls either ".… In Sight" or "No Contact."

At decision altitude, the pilot flying (PF) responds "Landing" or "Go Around" and lands or executes a missed approach. If landing, the PM should continue radar altitude calls and airspeed deviations until touchdown. Raw data (without the flight director system) ILS approaches are introduced and practiced to simulate the loss of any component of the flight director system.

Acceptable Performance Guidelines

At decision altitude

- Airspeed ± 5 knots
- One quarter scale maximum deflection glide slope and localizer
- Manually fly the approach to a safe landing

Never go below the glide slope.

ILS APPROACH—SINGLE ENGINE

This maneuver provides single-engine practice in terminal area arrivals utilizing the ILS for the final approach portion (Figure MAP-9).

In the aircraft, prior to the procedure turn or on base leg if being vectored, the instructor pilot reduces thrust to idle on one of the engines, simulating an engine failure.

Front-Course ILS

Complete the "Before Descent" and "Before Landing" checklists down to "Landing Gear" and conduct approach briefing prior to entering the approach pattern.

The crew may be cleared for a Front-Course ILS Approach from any specified position. Determine if the ILS approach is to be conducted using the FMS or raw data NAV, then configure accordingly.

During the initial phase of the approach maintain 180 knots with slats extended. During the procedure turn, or on base leg, extend the flaps to 12° and maintain 160 knots. When cleared for the approach and within 90° of the inbound course, select APPR on the FCP.

After localizer interception, extend the flaps to 20° and reduce airspeed to V_{REF} + 20 knots. After glide slope alive, but not later than one dot below the glide slope (one dot above on the indicator) extend the landing gear and complete the "Before Landing" checklist. Fly the approach with flaps at 20°. As the glide slope is intercepted, stabilize airspeed at V_{REF} + 10 knots. When landing is assured, select flaps 40°. Cross the threshold at V_{REF} + 1/2 gust. Rudder trim may be neutralized before landing.

Upon leaving the final approach fix inbound, the pilot monitoring (PM) calls, "Final approach fix and altimeters checked." At 500 feet above decision altitude (DA), the PM calls, "500 feet above DA, instrument check, no flags." At 100 feet above DA, the PM calls, "approaching minimums", and at DA, calls either "in sight" or "no contact."

At decision altitude, the pilot flying (PF) responds "Landing" or "Go-Around" and lands or executes a missed approach as appropriate. The PM continues radar altitude calls and airspeed deviations until touch down.

Acceptable Performance Guidelines

At decision altitude

- Airspeed ± 5 knots
- One quarter scale maximum deflection glide slope and localizer

Never go below the glide slope.

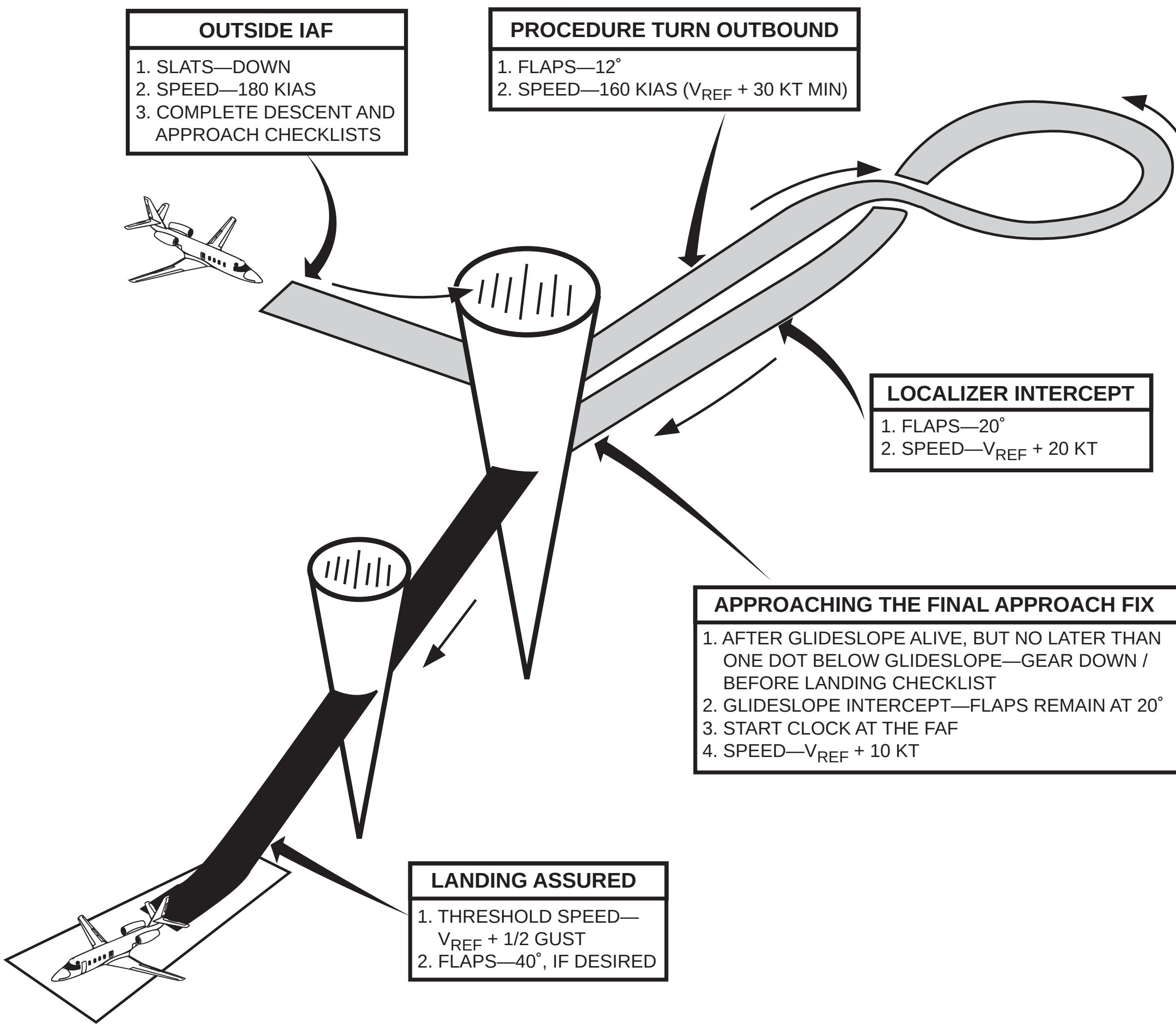


Figure MAP-9. ILS Approach—Single-Engine

NONPRECISION APPROACH— ONE OR TWO ENGINES

This maneuver provides practice in nonprecision terminal area arrivals. Emphasis is placed on the proper technique in tracking to or from a VOR/NDB or other approach aid (Figure MAP-10).

Procedures and Techniques

Complete the "Before Descent" checklist and the "Before Landing" checklist down to the "Landing Gear." Conduct the approach briefing prior to entering the approach pattern.

When beginning the approach, tune the correct frequencies. Set the course in the course windows. Identify all radio aids necessary for the approach. Check for warning flags. When flying an NDB approach in the Gulfstream G150 using the EHSI, it is not necessary to monitor the aural identification throughout the approach. Some older Astra models utilizing RMI equipment for approach require monitoring the aural identification throughout the approach. If the signal is lost, the "needle" is no longer visible in the instrument. Select HDG or NAV as appropriate on the FCP/MSP for the nonprecision approach in use. Place the mode selector of the flight director in heading mode (or NAV mode if using FMS Heading). During the initial

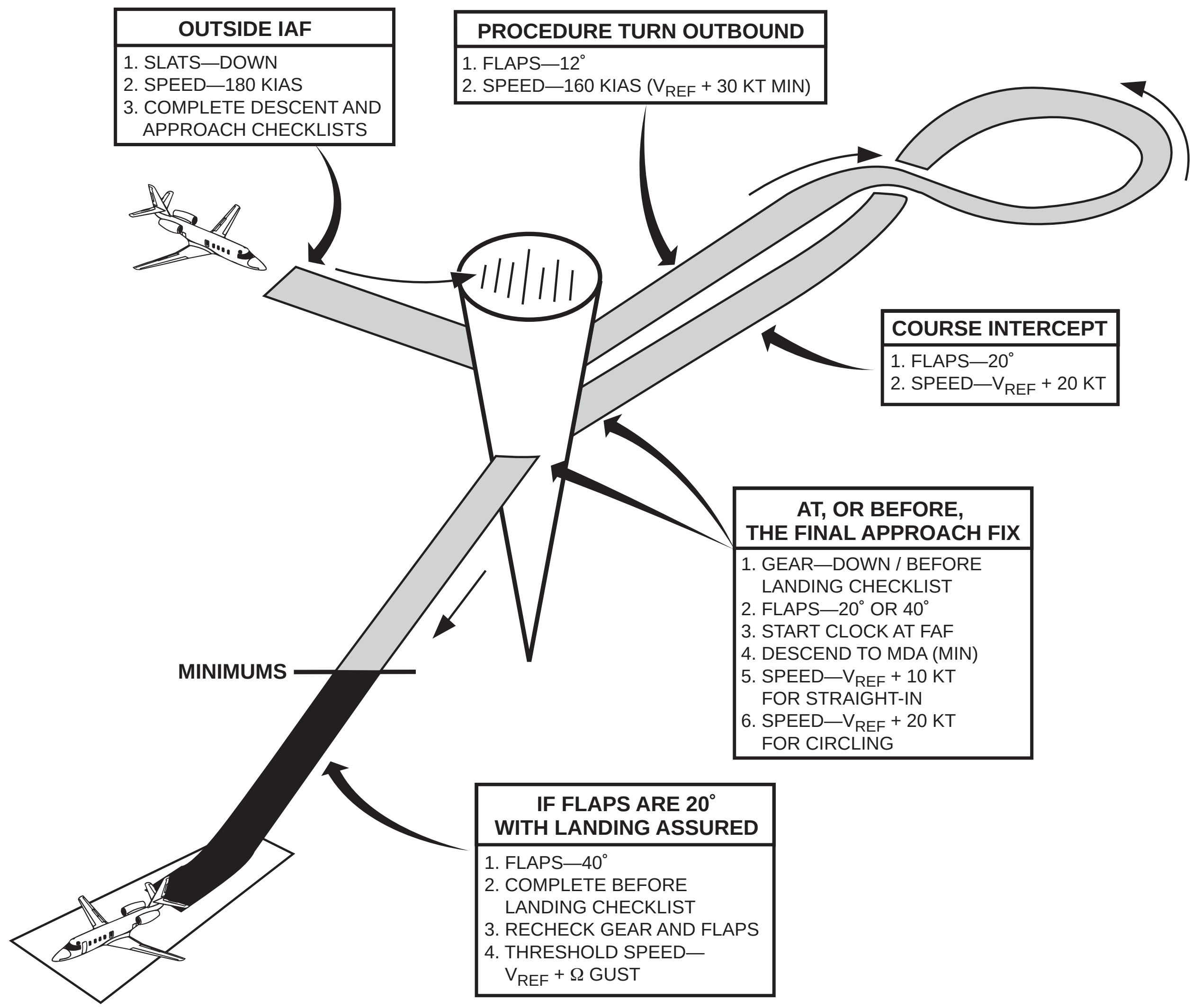


Figure MAP-10. Nonprecision Approach—One or Two Engines phase of the approach maintain 180 knots with the slats extended. When cleared for the approach, not later than within 90° of the inbound course, ensure HDG or NAV is selected on the FCP. During the procedure turn, or on base leg, extend the flaps to 12º and maintain 160 knots. When cleared for the approach and within 90º of the inbound course, select NAV mode on the flight director (if using FMS Heading, select “Intercept” on the FMS).

Intercept the final approach course inbound. Extend the flaps to 20º and reduce airspeed to V_{REF} + 20 knots. At or before the final approach fix extend the landing gear, maintain flaps 20º single engine, or circling two engine, and flaps 40º two engine landing straight in. Continue the “Before Landing” checklist. Upon leaving the final approach fix inbound, the pilot monitoring (PM) calls “Final approach fix, altimeters checked.” After the final approach fix inbound, stabilize airspeed at V_{REF} +10. Descend to the MDA. At 500 feet above MDA, the PM calls “500 feet above MDA, instrument check, no flags.” Track to the airport. At 100 feet above MDA, the PM calls “Approaching minimums,” and at the missed approach point (MAP) calls “… In Sight” or “No Contact”. At the MAP the PF responds, “Landing” or “Go-around” and either lands or executes a missed approach. When landing is assured, if flaps 20º, select flaps 40º. Cross the threshold at V_{REF} + 1/2 gust. If single engine, rudder trim may be neutralized before landing.

Refer frequently to the DME if DME information is a part of the approach procedure. If applicable, use time between the final approach fix and missed approach point at stabilized airspeed as noted on the approach chart. The FAA has approved the use of FMS overlay on nonprecision approaches. The PF may use the FMS approach mode provided the PM displays and monitors appropriate navaid raw data. Proper autopilot procedures must be used to avoid descending below MDA.

Acceptable Performance Guidelines

- Altitude ± 100 feet
- Airspeed ± 10 knots

At minimum descent altitude

- Altitude +50/–0 feet
- One quarter scale maximum deflection
- Deviation ± 5° RMI or HSI bearing pointer

CIRCLING APPROACHES

This maneuver is used to provide training in circling approach procedures with weather at or near circling approach minimums (Figure MAP-11). A circling approach may be made from various instrument approaches. When it is known that the aircraft has to be maneuvered through more than 30° of turn to align with the landing runway, a clearance for an instrument approach with a circle to land is issued.

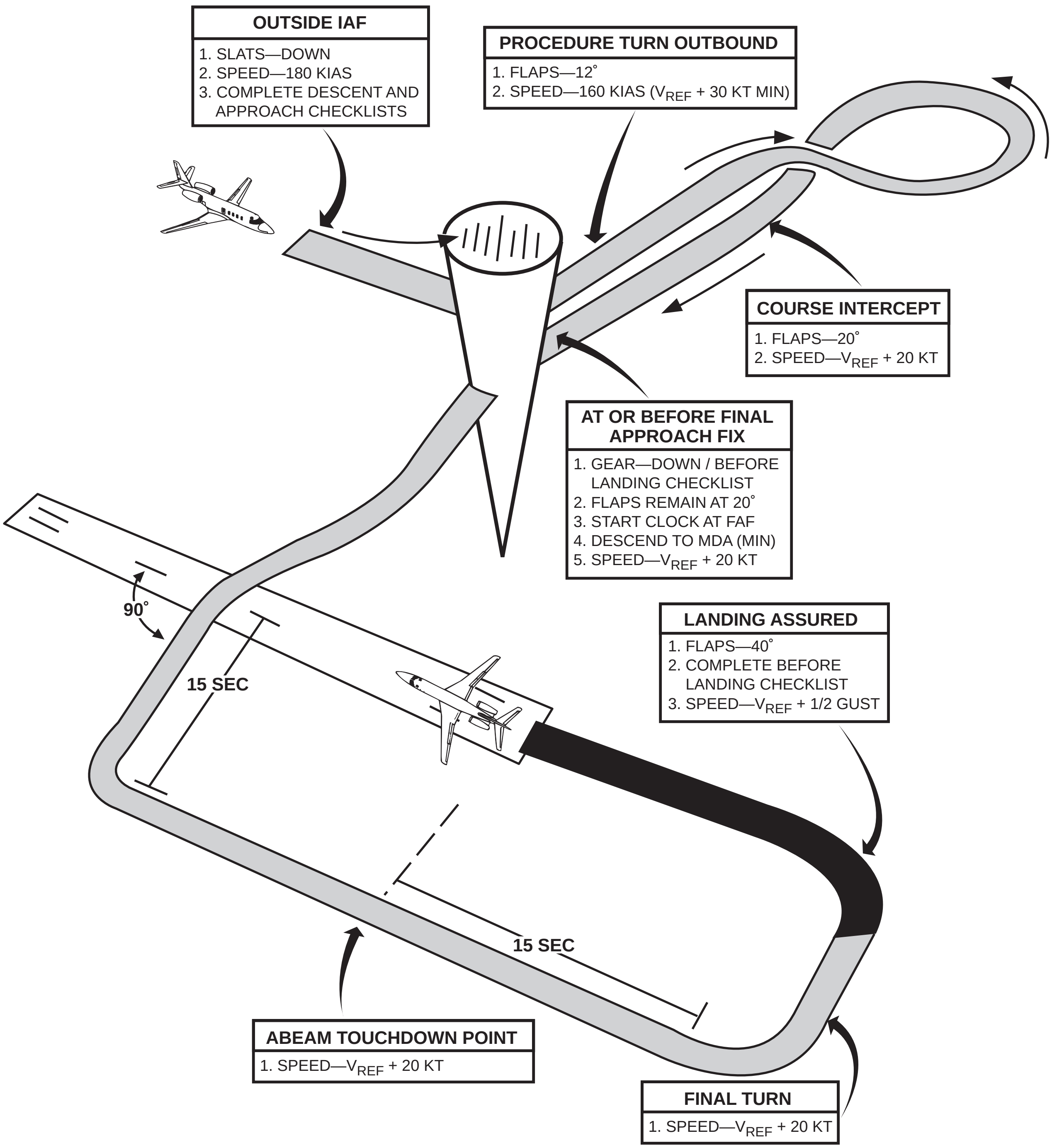


Figure MAP-11. Circling Approach—One or Two Engines

Procedures and Techniques

Complete "Before Descent" checklist and "Before Landing" checklist down to the line "Landing Gear Down" prior to entering the approach pattern.

Aircraft configuration and approach pattern airspeeds are the same as for a straight-in approach until the final approach fix. Extend the landing gear at or before the final approach fix and continue " Before Landing" checklist. Extend flaps 20° and maintain V_{REF} +20 knots throughout the circling maneuver until in a position to start a descent to the landing runway. Extend flaps to 40° when landing assured. Establish V_{REF} + 1/2 the gust factor and complete the "Before Landing" checklist. These procedures are the same for single engine operations.

The type of circling approach to be flown is normally left to the pilot. The aircraft must remain within the protected circling area. Maintain circling MDA until in position to descend for landing. Crew coordination should consist of, but not be limited to, the pilot flying (PF) the aircraft primarily by instruments and the pilot monitoring (PM) maintaining visual reference with the runway environment and calling any deviations in airspeed, angle of bank, and altitude.

NORMAL LANDING

This maneuver provides training in flying a normal traffic pattern and landing. It is used to develop proper techniques in power and control usage at relatively low speeds during the critical phases of final approach and touchdown (Figure MAP-12).

The "Before Descent" and "Before Landing" checklists (down to "Landing Gear Down") should be accomplished prior to entering the traffic pattern. The angle of bank should not exceed 30° while in the traffic pattern. Maintain 180 knots with the slats extended. Extend flaps to 12° and maintain 160 knots prior to downwind. Maintain V_{FR} traffic pattern altitude (normally1,500 feet above airport elevation). On downwind, extend the flaps to 20°, and decrease the airspeed to V_{REF} + 20. When abeam the touchdown point, extend the gear, and continue the "Before Landing"

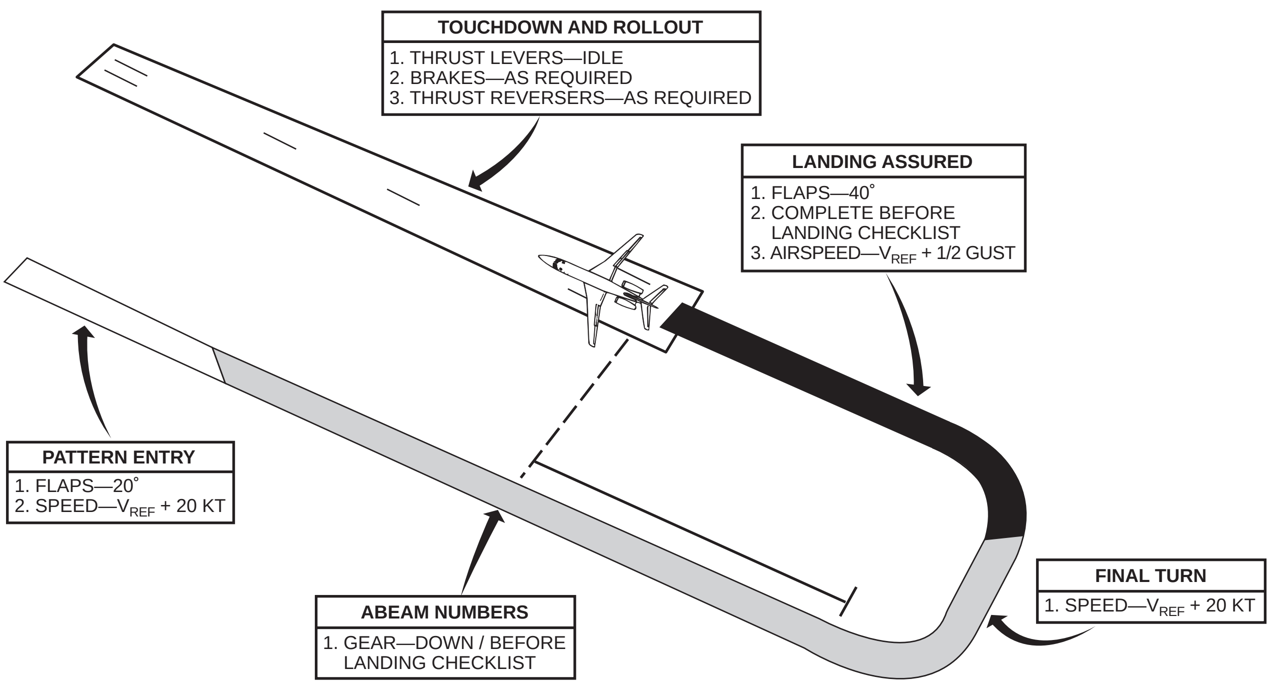


Figure MAP-12. Normal Landing

checklist.

The base leg may be flown as a straight path or continuing turn to final, depending on the position of the downwind leg. During the turn to final, make adjustments to establish the airplane on the proper glide angle as early as possible to avoid high rates of descent close to the ground. Vertical speed should be controlled with the elevator while airspeed is controlled with power.

When landing is assured, extend flaps to 40° and complete the "Before Landing" checklist. Maintain a constant rate of descent, and stabilize the airspeed at V_{REF} + 1/2 gust. Power is reduced to idle before touchdown. Stabilizer trim may be used during the landing flare. Lower the nosewheel to the runway before elevator control is lost. Use reverse thrust, ground air brakes, and brakes as necessary to bring the aircraft to a stop.

At approximately 80 knots, the pilot transfers control of the yoke to the copilot and positions the left hand to use the nosewheel steering when required. A verbal command and response by the pilot and copilot should be used to effect transfer of the yoke. Keep the aircraft straight with rudder until rudder effectiveness is lost. Thereafter, use small, smooth corrections with nosewheel steering. After taking control of the yoke, the copilot will hold the yoke forward to help provide positive nosewheel steering and maintain aileron deflection into crosswind, if required.

Acceptable Performance Guidelines

- Altitude + 100 feet on the downwind leg
- Airspeed +10/–0 knots

VISUAL APPROACH WITH NO-FLAP/PARTIAL-FLAP LANDING

This maneuver provides training in no-flap (FigureMAP-13) and partial-flap landings.

Because of the low drag and higher speed of the aircraft, the pattern should be made slightly larger than normal. A new V_{REF} will be determined from the "V_{REF} for Abnormal Flaps/Slats Configuration Chart" in the *AFM*. Set the airspeed bug for the new V_{REF}. Maintain V_{REF} + 20 knots while maneuvering

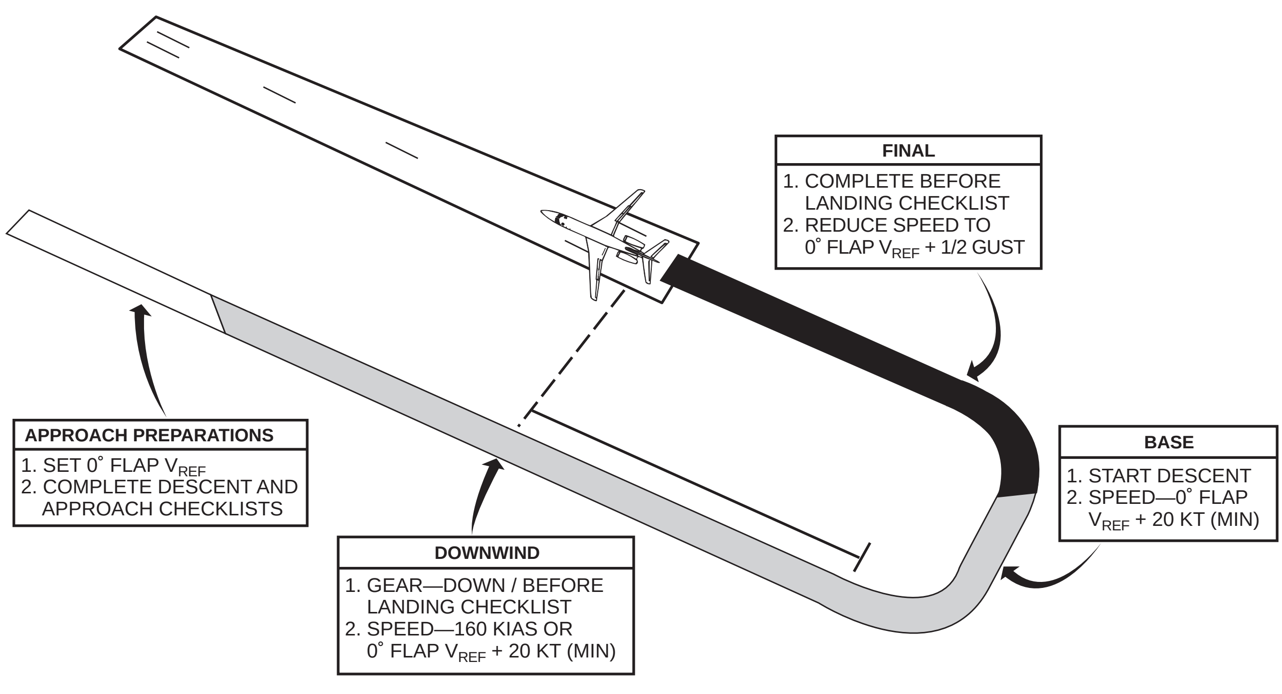


Figure MAP-13. Visual Approach—No Flaps

prior to final approach. The landing gear should be extended prior to turning base leg. On final approach, speed should be reduced to the new V_{REF} + 1/2 gust. Bank angles are limited to 15° while at the new V_{REF}. A higher deck angle and a slightly greater rate of descent will be required to maintain a normal glide path. Reduce power to idle approaching the threshold. Do not flare. Maintain pitch attitude, and allow the airplane to touch down. After touchdown, use thrust reversers and brakes as necessary.

Landing distances are approximately 10% (50% No flap/No slat) greater when making a no-flap landing due to higher touchdown speeds and the absence of drag. An instrument approach may be accomplished in conjunction with this maneuver.

CAUTION

Do not deploy thrust reversers until the nose gear is on the runway. Reversers tend to pitch the noseup after landing.

NOTE

Refer to the *AFM* “Abnormal Flaps/Slats Configuration Chart” to determine the new V_{REF} for any flaps/slats abnormality.

Acceptable Performance Guidelines

- Altitude + 100 feet on the downwind leg
- Airspeed +10/–0 knots on final
- Maximum bank angle of 15° on final

CROSSWIND TAKEOFFS AND LANDINGS

This maneuver provides training in controlling the aircraft during takeoff and landing with crosswind conditions.

Procedures and Techniques

Takeoff

During the initial phase of the takeoff, the pilot maintains directional control with rudder pedal nosewheel steering (and tiller control nosewheel steering if needed). The pilot not flying (PNF) should hold aileron into the wind until the pilot assumes control of the yoke at approximately 80 knots. The pilot maintains the aileron input into the wind for the remainder of the takeoff roll. As both aileron and rudder control become more effective during acceleration, adjust inputs as necessary. The control inputs must be maintained during rotation. Rotation should be sufficiently positive to prevent any skipping tendency. After becoming airborne, allow the aircraft to crab into the wind and maintain required drift correction.

Landing

During the early stages of a final approach, the aircraft may be turned or crabbed into the wind to compensate for the effects of crosswind. However, a transition to the wing-low method should be initiated so as to cross the threshold with a stabilized wing-low correction. Use bank angle into the crosswind to stop the drift and opposite rudder to maintain desired heading for runway alignment. The flight controls should be adjusted as necessary to compensate for any changes in wind direction or velocity.

As the flare and landing progress, the pilot may need additional control deflection as control effectiveness tends to diminish. The pilot should be alert for possible shifts in wind direction just above ground level which may require rapid flight control changes. After touchdown, required aileron deflection should be maintained by the pilot or copilot throughout the landing roll.

NOTE

Extreme caution should be exercised while landing on wet runways because of the tendency of this aircraft to hydroplane. This speed may be computed by the following formula:

Hydroplaning speed/knots = 9 x square root of tire pressure

NOTE

Touchdown should ideally be 1,000 feet down the runway. Minimum use of reverse thrust aids in maintaining aircraft control.

Acceptable Performance Guidelines

- Maintain positive aircraft control.
- Avoid side loads on landing gear.

LANDING WITH SIMULATED ENGINE FAILURE

This maneuver provides practice in controlling, maneuvering, and landing the aircraft with one engine inoperative.

In flight, engine failure is simulated by placing the throttle of the affected engine to idle.

The "Descent, Approach, and Before Landing" checklists should be completed at the appropriate times.

Aircraft configuration and airspeed are the same as for a two-engine approach until the final approach fix. The landing gear is lowered at the normal position, but flaps should not be extended beyond 20° until the landing is assured. Airspeed should be V_{REF} + 10 knots. No special techniques are required during approach and touchdown. Consideration should be given to landing with 20° flaps under adverse weather conditions (low ceilings, high crosswinds).

Acceptable Performance Guidelines

- Altitude + 100 feet on downwind
- Airspeed minimums controlled within +10/–0 knots
- Maintain positive aircraft control
- Positive directional control of aircraft

MISSED APPROACH PROCEDURES

These procedures provide training in executing missed approaches (Figure MAP-14).

Procedures and Techniques

When a missed approach is initiated, simultaneously apply takeoff thrust, select go-around mode on the flight director, increase the pitch attitude to the flight director command bar, and confirm that flaps are at 20°. When a positive rate of climb is established, retract the landing gear. At 400 feet above field elevation and clear of obstacles, accelerate to V_2 + 10 knots, retract flaps to 0°. Accelerate to V_{FS} and comply with published missed approach procedures. Complete the "Missed Approach" checklist. If diverting to an alternate airport, raise slats and accelerate to cruise speed.

Acceptable Performance Guidelines

- Maintain positive control of the aircraft.
- Aircraft is not permitted to descend to below MDA (nonprecision approach).

ABORTED TAKEOFF

This maneuver provides training in positive aircraft control and crew coordination if the takeoff is aborted (Figure MAP-15).

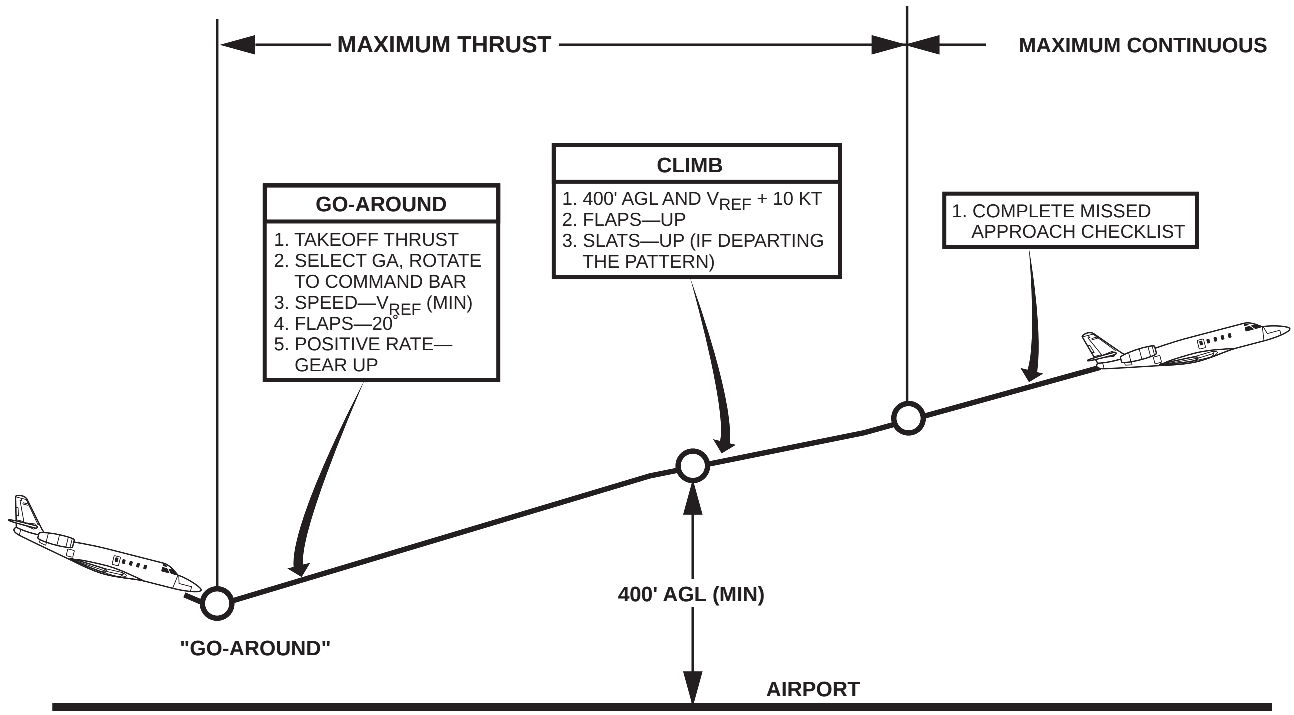


Figure MAP-14. Missed Approach—One or Two Engines

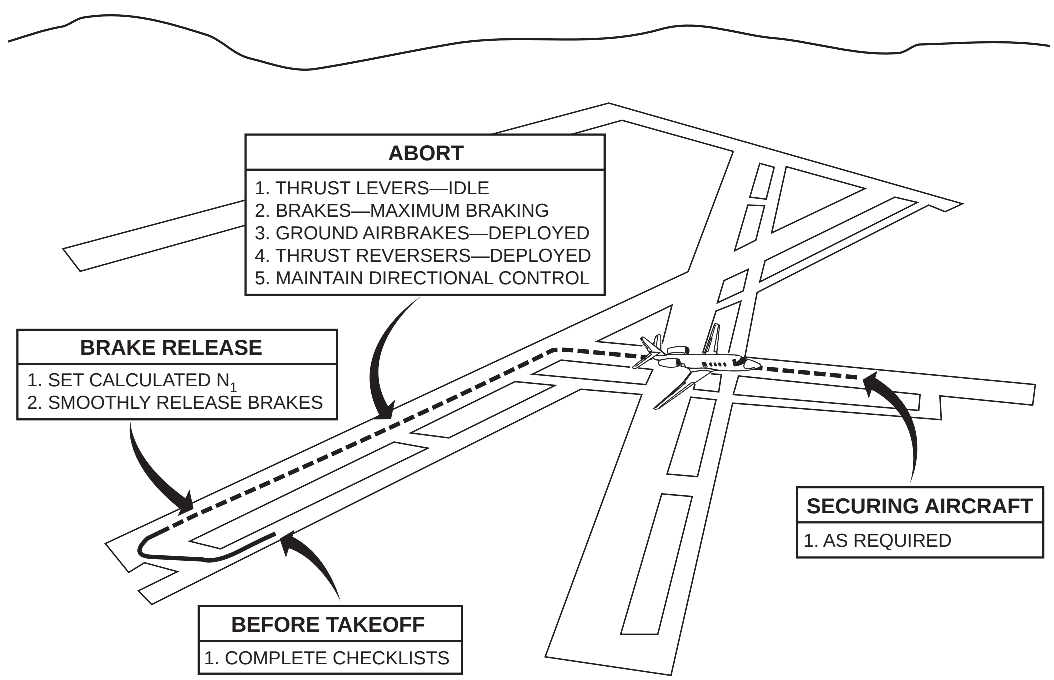


Figure MAP-15. Aborted Takeoff

The following applies to abort training in the aircraft:

- Practice aborted takeoffs will be executed at speeds of not more than 30% of VMCG (i.e., approximately 26 knots).
- Symmetrical thrust only will be used.
- Abort will be executed on command of the instructor pilot.

Procedures and Techniques

To abort the takeoff, reduce the thrust levers immediately to idle, apply maximum antiskid braking, and maintain directional control. Use reverse thrust as required. The pilot not flying (PNF) deploys the ground airbrakes on hearing the command “abort,” and advises tower or other traffic if at an uncontrolled airport.

NOTE

Takeoff will not be aborted at airspeeds above V_1.

Acceptable Performance Guidelines

- Use of proper procedures
- Positive directional control of aircraft

SPECIAL MANEUVER CONSIDERATIONS

RUNAWAY PITCH TRIM

This procedure is practiced to train the pilot in stopping runaway pitch trim, and return the aircraft to an in-trim condition.

In the aircraft, the instructor pilot shall initiate a runaway stabilizer by holding the stabilizer trim switch in either the noseup or nosedown position.

Procedures and Techniques

The pilot takes immediate action to control the pitch attitude by pressure on the control column. Then, the pilot deactivates the primary system by depressing the pitch trim release button located on the inboard side of the right handle of the pilot’s yoke. Secondary trim is now available by operating the pitch trim override switch up or down as required.

NOTE

Primary trim may be restored by depressing the lighted red trim reset button located to the right of the override switch.

Acceptable Performance Guidelines

- The pilot must recognize runaway pitch trim and take proper corrective action.
- Maintain positive aircraft control.

EMERGENCY DESCENT

This maneuver is used during emergency situations which require the highest allowable rate of descent.

In the aircraft, emergency descent should be practiced in day V_{FR} conditions at a minimum altitude of 10,000 feet AGL under ATC radar control. Without radar control, practice day V_{FR} at or above 12,000 feet. Do not exceed 20° nosedown pitch or .78M/350 knots in the descent training.

NOTE

During flight training, as soon as the aircraft is configured properly and stabilized in the descent and all prescribed procedures are accomplished, this maneuver is considered completed.

Procedures and Techniques

If fire, smoke, or rapid decompression occurs, the crew must immediately don oxygen masks. After the crew oxygen masks are on, the pilot retards the thrust levers to idle while initiating a maximum 60° bank turn, in either direction. While establishing the turn, the nose should be lowered, the air brakes extended, and a pitch angle (8–10° nosedown) established which will quickly attain a rapid descent at V_{MO}/M_{MO}. Both crewmembers should ensure communications are promptly established. In the case of a rapid decompression, the pilot not flying (PNF) should place the cabin air selector to emergency, turn on the oxygen system bypass valve, and ensure the passengers are receiving oxygen. Complete the "Rapid Decompression/Emergency Descent" checklist as applicable.

After the required pitch has been established and 45 to 90° of turn has been completed, the wings may be rolled level. The purpose of the 60° bank is twofold: it allows lowering the nose rapidly without the loss of positive g-forces and initiates a turn off the airway to minimize the possibility of collision with other aircraft.

At 2,000 feet above the desired level-off altitude, decrease the nosedown pitch to 5°.

At the desired altitude, complete a normal level-off and retract the airbrakes.

NOTE

In the event structural damage to the aircraft is suspected, descend in such a manner as to create the least amount of vibration.

FLIGHT IN ADVERSE WEATHER CONDITIONS

Turbulent Air Penetration

Flight through severe turbulence should be avoided. When flying at 30,000 feet or higher, it is not advisable to avoid a turbulent area by climbing unless it is obvious that it can be overflown well in the clear. For turbulence of the same intensity, greater buffet margins are achieved by flying the recommended speeds at lower altitudes.

- PENETRATION WITH AUTOPILOT ENGAGED:
  Altitude hold..............................OFF
- PENETRATION WITH AUTOPILOT DISENGAGED:
  Yaw damper.....................ENGAGED
- RECOMMENDED AIRSPEED VA minus 10 knots. (See "Maneuvering Speed", AFM Section 1.) Do not chase airspeed.
- ALTITUDE
  1. Extreme altitude changes may occur. Do not chase altitude.
  2. If terrain clearance permits, sacrifice altitude to maintain desired airspeed and attitude.
- ATTITUDE
  1. Maintain wings level and desired pitch attitude, using flight director indicator as primary instrument for maintaining constant attitude.
  2. Trim for penetration airspeed and do not change pitch trim to control pitch. Maintain control with elevators.
  3. Avoid sudden or extreme control movements.
- THRUST

  Set thrust for desired penetration airspeed. Change thrust only in case of extreme airspeed changes.
- IGNITION

  Use continuous ignition during moderate or severe turbulence.

Flight In Icing Conditions

NOTE

Before flight into known icing conditions, turn on continuous ignition prior to activating engine/nacelle anti-ice system. Use both continuously during flight in expected icing conditions. If ice has already accumulated, deice one engine at a time due to possibility of engine flameout resulting from ice ingestion. Activation of anti-ice system at high thrust settings may require thrust lever adjustments to maintain ITT within limits. At least 80% N_2 is required to illuminate NAC light.

With visible moisture and freezing temperatures, ice will form on unheated parts of windshield, wing and empennage leading edges, air inlets, and engine nacelles. Stall speed may increase with ice accumulations and the speed margin between stall warning buffet onset and the stall may be eliminated. Resultant increase of weight and drag may reduce airspeed and increase thrust requirements with consequent reduction in range.

During icing conditions:

- Maintain comfortable margin above normal stall speed.
- Activate all ice-protection devices—pitot-static, AOA probe, windshield heat, engine/nacelle anti-ice, and deicing systems, as necessary.
- If necessary, change altitude rapidly by climb or descent (terrain clearance permitting) to exit layered stratus formations or vary course to exit vertical cumulus formations.
- Do not operate wing and empennage deicing system at ambient temperatures below –40°C (–40° F) to avoid cracking the boots.

BIRD STRIKE

Birds can be a significant hazard to flight operations. Large birds can cause structural damage, penetrate the windshield, or damage an engine. While the danger of large birds should not be minimized, it should be appreciated that small birds may cause a significantly greater hazard primarily due to their tendency to travel in flocks.

The following recommendations may help to avoid or minimize damage from bird strikes:

- Birds will try to avoid airplanes. Beacons, strobes, and landing and taxi lights will enhance the bird’s chance of seeing the airplane.
- Airplane radar appears to have an effect on birds and might cause them to avoid aircraft using it.
- Birds tend to avoid aircraft by diving. If time permits, try to fly over the birds.
- Avoid taking off into a rising or setting sun; the glare will make it difficult to see birds.
- Bird strikes are more likely after a long period of inactivity at an airport. Be especially vigilant on early morning takeoffs with low ceilings.
- At controlled airports, help may be obtained in clearing the runway of birds prior to takeoff. At uncontrolled airports, it may be a good idea to taxi or perform a runup on the active runway prior to takeoff to disperse the birds.
- If birds are in the vicinity, use the igniters for takeoff to help prevent engine flameout from bird ingestion.
- If a bird strike appears imminent after takeoff, do not take evasive action at low altitudes and airspeeds.
- If a bird appears to be coming through the windshield, duck your head under the glareshield. An airplane can be flown without a windshield, provided the pilot is not incapacitated.

- If an actual bird strike is suspected and the igniters are not on, turn them on. Carefully monitor the engine instruments, and if you observe abnormal temperatures or pressures, or feel any unusual vibrations, shut down the affected engine unless it is needed to maintain flight.
- If there is any question about the safety of takeoff due to bird hazards, either delay takeoff or change runways.

BLACK HOLE PHENOMENA

This procedure should acquaint the pilot with the problems involved in making night approaches when the only visual reference is the runway itself.

Procedures and Techniques

The black hole approach should be flown, if possible, as a normal, stabilized 3° glide-slope approach. The normal descent rate at the proper V_{REF} is approximately 700 fpm. If the runway threshold is kept in the bottom third of the windshield with this rate of descent at V_{REF}, the airplane should cross the threshold at 50 feet AGL. A descent rate less than this with the threshold in the bottom third of the windshield, would indicate that the airplane is low. A high descent rate would indicate that the airplane is high.

Pilots should use all aids available during a black hole approach. FMS data, DME, VASI, or an electronic glide slope can be used to establish a 3° glide slope. A 3° glide slope may be established by having the aircraft 300 feet above field elevation for every mile away from the landing runway.

Acceptable Performance Guidelines

- Altitude ±50 feet
- Airspeed +5/–0 knots
- Manually flying the approach with sufficient accuracy to effect a safe landing

APPENDIX

CONTENTS

TABLE

APPENDIX

LIST OF ABBREVIATIONS

°C—Degrees Celsius

°F—Degrees Fahrenheit

1VS—First-stage variable stators

A—ADF

A/B—Airbrake

A/C—Aircraft

AC—Alternating current

ACARS—Aircraft communication addressing and reporting system

ACC—Air-conditioning controller

ACM—Air cycle machine

ADC—Air data computer

ADF—Automatic direction finder

ADI—Attitude director indicator

ADS—Air data system

AECU—Antiskid electronic control unit

AFM—Airplane Flight Manual

AGB—Accessory gearbox

AHC—Attitude/heading computer

AHP—Auxiliary hydraulic pump

AHRS—Attitude heading reference system

AHS—Attitude heading source

AIU—Audio interface unit

ALT SEL—Altitude, preselect

ALT—Altitude

ALTS—Altitude select

ANN—Annunciator

ANT—Antenna

AOA—Angle of attack

A/P—Autopilot

APR—Automatic power reserve

APU—Auxiliary power unit

APU-ECU—Auxiliary power unit engine control unit

ARINC—Aeronautical radio, incorporated

ARP—Air data reference panel

ASR—Auxiliary start relay

ATC—Air traffic control **or** automatic trim coupler

ATCRBS—Air traffic control radar beacon system

ATU—Automatic test unit

AUTO—Automatic

AW—Aural warning

B/C—Back course

BAGG—Baggage

BAT—Battery

BATT—Battery

BCP—Best computed position

BIT—Built-in test

BLC—Battery line contactor

BRG—Bearing

BTC—Bus-tie contactor

BTLE—Bottle

CAS—Crew alerting system

CB—Circuit breaker

CBV—Cross-bleed valve

CCSM—Current controlled state modulation

CCU—Cockpit control unit

CCW—Counterclockwise

CD—Compact disc

CDC—Cabin display computer

CDI—Course deviation indicator

CDU—Control display unit

CG—Center of gravity

CHG—Change

COM—Communication

COMPRTR—Comparator

COND—Conditioner

CONTR—Control

COR—Cutoff relay

CPC—Cabin pressurization computer

CPCS—Cabin pressure control system

CPL—Couple

CPN—Collins part number

CRT—Cathode-ray tube

CS—Current sensor

CSMU—Crach survivable memory unit

CT—Current transformer

CTL—Control **or** COMM/NAV tuning unit

CTS—Carry-through structure

CVR—Cockpit voice recorder

CW—Clockwise

DAU—Data acquisition unit

db—Decibel

DBU—Database unit

DC—Direct current

DCP—Display control panel

DCRS—Decreases

DCU—Data concentration unit

DDCU—Dual data concentrator unit

DETECT—Detector

DG—Directional gyro

DH—Decision height

DLU—Download unit

DME—Distance measurement equipment

DMSU—Dual-mode select unit

DN—Down

DTK—Desire track

DTMF—Dual-tone multiple frequency

E—East

EBBFL—Emergency bus battery feed line

EBC—Emergency bus contactor

EBLC—Emergency battery line contactor

EBTC—Emergency bus-tie contactor

ECO—Electrical connection boxes

ECS—Environmental control system

ECTM—Engine condition trend monitoring

ECU—Environmental control unit

ED—Engine display **or** EICAS display

EDC—EICAS display control panel

EDP—Engine-driven pump

EDR—Emergency disconnect relay

EDS—Engine diagnostic system

EEC—Electronic engine control

EFCV—Ejector flow control valve

EFD—Electronic flight display

EFDS—Electronic flight display system

EFIS—Electronic flight instrument system

EGT—Exhaust gas temperature

EICAS—Engine indication and crew alerting system

ELD—Emergency light distribution unit

ELT—Emergency locator transmitter

EMERG—Emergency

EMI—Electromagnetic interface

ENG—Engine

EP—External power

EPC—External power contactor

EPM—External power monitor

EPR—External power relay

EPU—External power unit

ET—Elapsed time

ETA—Estimated time of arrival

EXT—Extension **or** Exernal

F—FMS

FAA—Federal Aviation Administration

FADEC—Full authority digital engine control

FCC—Flight control computer

FCP—Flight control panel

FCS—Flight control system

FCU—Fuel control unit

FCV—Flapper check valve

FDBK—Feedback

FDR—Flight data recorder **or** Feeder

FF—Fuel flow transmitter

FIFO—First in–first out

FLC—Flight level change

FMS—Flight management system

FPD—Flap panel display

FPDU—Flap power drive unit

FQMC—Fluid quantity management computer

FQMS—Fuel quantity management system

FR—Fuel remaining

FSECU—Flap/slat electrical control unit

FSVM—Fuse and shutoff valve module

ft—Feet

FT—Flight time

FTB—Flap transmission brake

g—Gravity acceleration

GBE—Ground based equipment

GCU—Generator control unit

GEN—Starter–generator

GF—Ground fault

GFR—Generator fault relay

GLC—Generator line contactor

GND—Ground

GPM—Gallons per minute

GPS—Global positioning system

GPWS—Ground proximity warning system

GS—Groundspeed

GSC—Generator start contractor

GST—Gust lock

GW—Gross weight

HCU—Hydraulic control unit

HDG—Heading

HF—High frequency

HFMU—Hydromechanical fuel metering unit

Hg—Mercury

HLS—High-level sensor

HNDWL—Handwheel

HMU—Hydromechanical metering unit

HP—High pressure

HPA—Hectopascals

HPRV—High-pressure relief valve

HPSOV—High-pressure shutoff valve

HSCU—Horizontal stabilizer trim control unit

HSI—Horizontal situation indicator

HYD—Hydraulic

IAI—Israel Aircraft Industries, Ltd

IAPS—Integrated avionics processor system

IAS—Indicated airspeed

I-BIT—Initiate built-in test

ICAO—International Civil Aviation Organization

ICU—Internal compensation unit

ID—Ident

IFR—Instrument flight rules

IGV—Inlet guide vane

ILS—Instrument landing system

in—Inches

IN. Hg—Inches of mercury

INCRS—Increases

INOP—Inoperative

IOC—Input/output concentrator

IRC—Infrared remote controller

IRD—Infrared remote detectors

IRS—Inertial reference system

IRU—Inertial reference unit

ISA—International Standard Atmosphere

ISDN—Integrated Services Digital Network

ITT—Interturbine temperature **or** interstage turbine temperature

K—Krueger flaps

KCAS—Knots calibrated airspeed

kg—Kilogram

KIAS—Knots indicated airspeed

kpm—Knots per hour

KTAS—Knots true airspeed

L—Left

lb—Pounds

LC—Line contactor

LCD—Liquid crystal display

LCK—Lock

LCV—Load control valve

LFES—Landing field elevation selector

LH—Left hand

LHS—Left hydraulic system

LLS—Low-level sensor

LLV—Low limit temperature control valve

LOC—Localizer

LP—Low pressure

LR—Line relay

LRM—Line replaceable module

LRU—Line replaceable unit

LT—Light

LTS—Lights

LVDT—Linear variable differential transformer

LW—Left wing

m—Meters

MAST C/W—Master caution warning

MBTC—Main bus-tie contactor

MAN—Manual mode

MAX—Maximum

MCR—Maximum cruise thrust rating

MCV—Mechanical control valve

MDA—Minimum descent altitude

MDC—Maintenance diagnostic **or** maintenance data computer

MFD—Multifunction display

MHz—Megahertz

MI—Indicated Mach

MIC—Microphone

MKR—Marker

MLG—Main landing gear

MM—Middle marker

mm—Millimeter

M_{MO}—Maximum operating speed

MOP—Main oil pressure

MPV—Minimum pressure valve

MSU—Mode selector unit

MUX—Multiplexer

MV—Metering valve

N—North

N_1—Fan speed **or** first-stage turbine speed

N_2—HP rotor shaft speed **or** second-stage turbine speed

NAASV—Nacelle anti-ice air supply valve

NAC—Nacelle

NACA—National Advisory Committee for Avionics

NAV—Navigation

NBAA—National Business Aircraft Association

NLG—Nose landing gear

nm—Nautical mile

NORM—Normal

NUFA—Noseup flight attitude

NVI—Nonvectored interrupt

NWOW—Nose weight on wheels

NWS—Nosewheel steering

NWSECU—Nosewheel steering electronic control unit

OFV—Outflow valve

OLR—Override load reduction

OM—Outer marker

OPT—Optional

OV—Overvoltage

OVRRD—Override

OXY—Oxygen

PA—Passenger address

P-BIT—Periodic built-in test

PDU—Power drive unit

PED—Pedal

PFD—Primary flight display

PGM—Program

PLCU—Pulse light control unit

PMA—Permanent magnet alternator

PO-BIT—Power-on built-in test

Ppm—Parts per million

PRESS—Pressure

PREV—Previous

PRSOV—Pressure regulator and shutoff valve

PSI—Pounds per square inch

PSID—Pounds per square inch, differential pressure

PSIG—Pounds per square inch, gage

PSTN—Public switched telephone network

PTT—Push to talk

PTU—Position transducer unit

PWR—Power

QAD—Quick attach detach

QTY—Quantity

R—Right

RA—Radio altimeter **or** radio altitude

RAC—Radio altitude converter

RAM—Random access memory

RAT—Ram-air temperature

RCB—Remote circuit breaker

RCCB—Remote control circuit breaker

RDP—Refuel–defuel panel

RDR—Radar

RF—Radio frequency

RH—Right hand

RHS—Right hydraulic system

RPM—Revolutions per minute

RSP—Reversionary select panel

RTA—Receiver/transmitter antenna

RTU—Radio tuning unit

RVDT—Rotary variable differential transducer

RVSM—Reduced vertical separation minimums

RW—Right wing

S—South

SAI—Standby attitude indicator

SAT—Static air temperature

SC—Starter contractor

SCV—Surge control valve

SED—Secondary EICAS display

SELCAL—Selective calling

SG—Starter–generator

SID—Standard instrument departure

SLFPM—Sea-level feet-per-minute

SOV—Shutoff valve

SP—Stall protection

SPC—Stall protection computer

SPDU—Slat power drive unit

SPPR—Single-point pressure refueling

SPQC—Stall protection and Q-feel computer

SPQS—Stick pusher and Q-feel system

SR—Start relay

SSFDR—Solid state flight data recorder

STAR—Standard terminal arrival

STD—Standard

STLB—Slat torque limiter brake

STO—Store

T—Track

T/O—Takeoff

TAS—True airspeed

TAT—Total air temperature

TBV—Turbine bypass valve

TC—Time circuit

TCAS—Traffic alert and collision avoidance system

TDR—Transponder

TDR—Time delay relay

TE—Trailing edge

TEMP—Temperature

TFC—Traffic

TICV—Turbine inlet temperature control valve

TL—Thrust lever

TLA—Throttle lever angle

TOW—Takeoff warning

TR—Thrust reverser

TRK—Track

TRS—Trim release unit

TRU—True north

TTG—Time to go

TTR—TCAS II transmitter receiver

TWR—Turbulence weather radar

UHF—Ultra-high frequency

ULB—Underwater locator beacon

UNB—Unbalanced

UNS—Universal navigation corporation

V—VOR

V_1—Go/no-go speed

V_2—Takeoff safety speed

V_A—Maneuvering speed

V_{BUG}—Flight director reference speed

VDC—Volts direct current

Vdc—Volts direct current

V_{FE}—Maximum flaps extended speed

VGV—Variable guide vane

VHF—Very high frequency

VIGV—Variable inlet guide vane

V_{LE}—Maximum landing gear extended speed

V_{LO}—Maximum landing gear operating speed

V_{MCA}—Air minimum control speed

V_{MCG}—Ground minimum control speed

V_{MO}—Maximum operating speed

VNAV—Vertical navigation

VOL—Volume

VOR—VHF omnidirectional range

V_R—Rotate speed

V_T—Target airspeed

VS—Vertical speed

VSI—Vertical speed indicator

W—Watt **or** west

WAS—Weather avoidance system

WOW—Weight on wheels

WS—Windshield

WX—Weather

WXP—Weather radar control panel

WX+T—Weather plus turbulence

XFR—Transfer

XMT—Transmit

YD—Yaw damper

Table APP-1. CENTIGRADE TO FAHRENHIET CONVERSION TABLE

C	F C	F	C	F C	F	C	F C	F
–62.2	–80	–112	126.7	260	500	843.3	1550	2822
–56.7	–70	–94	132.2	270	518	871.1	1600	2912
–51.1	–60	–76	137.6	280	536	898.8	1650	3002
–45.6	–50	–58	143.3	290	554	926.6	1700	3092
–40.0	–40	–40	148.9	300	572	954.4	1750	3182
–34.4	–30	–22						
–31.7	–25	–13	154.4	310	590	982.2	1800	3272
–28.9	–20	–4	160.0	320	608	1010.0	1850	3362
–26.1	–15	5	165.6	330	626	1037.7	1900	3452
–23.3	–10	14	171.1	340	644	1065.5	1950	3542
–20.6	–5	23	176.7	350	662	1093.3	2000	3632
–17.8	0	32	182.2	360	680	1121.1	2050	3722
–15.0	5	41	187.8	370	698	1148.8	2100	3812
–12.2	10	50	193.3	380	716	1176.6	2150	3902
–9.4	15	59	198.9	390	734	1204.4	2200	3992
–6.7	20	68	204.4	400	752	1232.2	2250	4082
–3.9	25	77	210.0	410	770	1260.0	2300	4172
–1.1	30	86	215.6	420	788	1287.7	2350	4262
1.1	35	95	221.1	430	806	1315.5	2400	4352
4.4	40	104	226.7	440	824	1343.3	2450	4442
7.2	45	113	232.2	450	842	1371.1	2500	4532
10.0	50	122	237.8	460	860	1398.8	2550	4622
12.8	55	131	243.3	470	878	1426.6	2600	4712
15.6	60	140	248.9	480	896	1454.4	2650	4802
18.3	65	149	254.4	490	914	1482.2	2700	4892
21.1	70	158	260.0	500	932	1510.0	2750	4982
23.9	75	167	256.6	510	950	1537.7	2800	5072
26.7	80	176	271.1	520	968	1565.5	2850	5162
29.4	85	185	276.7	530	986	1593.3	2900	5252
32.2	90	194	282.2	540	1004	1621.1	2950	5342
35.0	95	203	287.8	550	1022	1648.8	3000	5432
37.8	100	212						
40.6	105	221	293.3	560	1040	1676.6	3050	5522
43.3	110	230	298.9	570	1058	1704.4	3100	5612
46.1	115	239	304.4	580	1076	1732.2	3150	5702
48.9	120	248	310.0	590	1094	1760.0	3200	5792
51.7	125	257	315.6	600	1112	1787.7	3250	5882
54.4	130	266	326.7	620	1148	1815.5	3300	5972
57.2	135	275	337.8	640	1184	1843.3	3350	6060
60.0	140	284	348.9	660	1220	1871.1	3400	6152
62.8	145	293	360.0	680	1256	1898.8	3450	6242
65.6	150	302	371.1	700	1292	1926.6	3500	6332
68.3	155	311	382.2	720	1328	1954.4	3550	6422
71.1	160	320	393.3	740	1364	1982.2	3600	6512
73.9	165	329	404.4	760	1400	2010.0	3650	6602
76.7	170	338	415.6	780	1436	2037.7	3700	6692
79.4	175	347	426.7	800	1472	2065.5	3750	6782
82.2	180	356	437.8	820	1508	2093.3	3800	6872
85.0	185	365	454.4	850	1562	2121.1	3850	6962
87.8	190	374	482.2	900	1652	2148.8	3900	7052
90.6	195	383	510.0	950	1742	2176.6	3950	7142
93.3	200	392	537.7	1000	1832	2204.4	4000	7232
96.1	205	401	565.5	1050	1922			
98.9	210	410	592.2	1100	2012			
101.7	215	419	621.1	1150	2102			
104.4	220	428	648.8	1200	2192			
107.2	225	437	675.6	1250	2282			
110.0	230	446	704.4	1300	2372			
112.8	235	455	732.2	1350	2462			
115.6	240	464	760.0	1400	2552			
118.3	245	473	787.7	1450	2642			
121.1	250	482	815.5	1500	2732			

ANSWERS TO QUESTIONS

CHAPTER 2
1. D
2. D
3. C
4. C
5. D

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

CHAPTER 5
1. A
2. D
3. D
4. D
5. A

CHAPTER 6
1. D
2. D
3. D
4. A
5. B

CHAPTER 7
1. A
2. A
3. D
4. C
5. B

CHAPTER 8
1. B
2. D
3. C
4. B
5. B

CHAPTER 9
1. C
2. B
3. D
4. A
5. C

CHAPTER 10
1. A
2. D
3. C
4. D
5. C

CHAPTER 11
1. A
2. B
3. C
4. D
5. B

CHAPTER 12
1. B
2. C
3. D
4. B
5. D

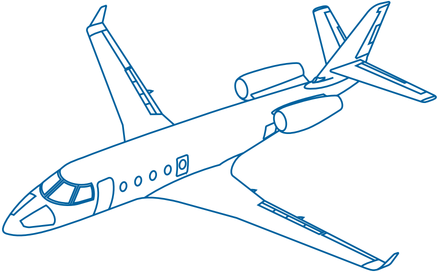
CHAPTER 13
1. B
2. D
3. D
4. D
5. B

CHAPTER 14
1. A
2. C
3. D
4. D
5. D

CHAPTER 15
1. D
2. A
3. D
4. B
5. D

CHAPTER 17

GULFSTREAM G150



Revision 2—April 2008

FLASH CARDS

INSERT LATEST REVISED CARDS, DESTROY SUPERSEDED CARDS

LIST OF EFFECTIVE CARDS

Dates of issue for original and changed cards are:

ORIGINAL.....0 ....AUGUST 2006
Revision ....1 ....October 2007
Revision ....1.01 ....December 2007
Revision ....2...................April 2008

TOTAL NUMBER OF CARDS IN THIS SET IS 65
CONSISTING OF THE FOLLOWING:

Card No.	*Revision No.	Card No.	*Revision No.
Title	2	SI-1 – SI-26A	2
ii	2	CAS-1 – CAS-9A	2
L-1 – L-28A	2		

*Zero in this column indicates an original card.

BAGGAGE COMPARTMENT

Capacity (Maximum) ....____ lbs

Floor Loading (Maximum) ....____ lbs

BAGGAGE COMPARTMENT

Capacity (Maximum) ....**1100** lbs

Floor Loading (Maximum) ....**105** lbs

BATTERY LIMITS

MIN VOLTAGE SAT above 0°C .... __ V

MIN VOLTAGE SAT below 0°C .... __ V

BATTERY LIMITS

MIN VOLTAGE SAT above 0°C **24**V

MIN VOLTAGE SAT below 0°C............................**23**V

BATTERY TEMP RANGES

BATT OVERHT CAS message illuminates at ___°F.

BATTERY TEMP RANGES

BATT OVERHT CAS message illuminates at **140**°F.

If illuminated, DO NOT TAKE OFF.

MAXIMUM CERTIFIED WEIGHTS

ZERO FUEL

LANDING

TAKEOFF

RAMP

MINIMUM WEIGHT

MAXIMUM CERTIFIED WEIGHTS

- ZERO FUEL — **17,500 lbs**
- LANDING — **21,700 lbs**
- TAKEOFF — **26,100 lbs**
- RAMP — **26,250 lbs**
- MINIMUM WEIGHT — **13,200 lbs**

FUEL TANK CAPACITY

Total Usable — ______________________

FUEL TANK CAPACITY

Total Usable — **1537 Gallons (10,300 lbs)**

MAXIMUM FUEL UNBALANCE

MAXIMUM FUEL UNBALANCE

400 lbs for Takeoff

600 lbs for Cruise and Landing

MAXIMUM N_1 and N_2

Transient

Overspeed

Shutdown

MAXIMUM N_1 and N_2

Transient – **N_1–100% to 100.8% for 10 seconds**
N_2–APR OFF 100.4% to 102.5% for 10 seconds
N_2–APR ON 101% to 102.5% for 10 seconds

Overspeed – **Red Indications**

Shutdown – **Operate 2 minutes at 38% N_1 or below before shutdown (including taxi time)**

MAXIMUM INTERSTAGE TURBINE TEMPERATURE

Starting

Takeoff

Takeoff with APR Activated

Maximum Continuous

Maximum Climb

Maximum Cruise

MAXIMUM INTERSTAGE TURBINE TEMPERATURE

Starting ....**990°C**

Takeoff ....**1004°C**

Takeoff with APR Activated........**1022°C** (5 minutes maximum)

Maximum Continuous...**990°C**

Maximum Climb....................................**974°C** (recommended)

Maximum Cruise ....**949°C** (recommended)

OIL PRESSURE

(AT NORMAL OPERATION TEMPERATURE)

ENGINE START

IDLE

TAKEOFF, CLIMB AND CRUISE

TRANSIENT

OIL PRESSURE
(AT NORMAL OPERATION TEMPERATURE)

ENGINE START – **Indication within 10 sec. after lightoff**

IDLE (Oil Temperature Below 30°C) – **50 to 150 psi minimum**

IDLE (Oil Temperature Above 30°C) – **62 to 83 psi**

TAKEOFF, CLIMB AND CRUISE – **62 to 83 psi**

TRANSIENT – **100 psi maximum** (3 min. maximum)

OIL TEMPERATURE

STARTING (Minimum Continuous Operation)

UP TO 30,000 FEET

ABOVE 30,000 FEET

TRANSIENT

OIL TEMPERATURE

STARTING — **At temperatures below –40°C for extended periods, preheat engine before attempting start. During cold temperature starts, oil pressure may exceed maximum allowable transients.**

UP TO 30,000 FEET – **30°C to 127°C**

ABOVE 30,000 FEET – **30°C to 140°C**

TRANSIENT – **149°C** (2 min. all operational altitudes)

DC STARTER / GENERATOR LIMITATIONS

MAXIMUM CONTINUOUS GROUND AND FLIGHT OPERATIONS

___ AMPS

DC STARTER / GENERATOR LIMITATIONS

MAXIMUM CONTINUOUS GROUND AND FLIGHT OPERATIONS

300 AMPS

CONSECUTIVE START ATTEMPTS

CONSECUTIVE START ATTEMPTS

15 seconds on

2 minutes off (for 3 cycles)

then 20 minutes off

START TIMES

Light-off (from initial fuel flow)

From light-off to idle

Starter-assist air starts (from initial fuel flow to 60% N_2)

Windmilling air start

START TIMES

Light-off (from initial fuel flow) – **10 seconds max**

From light-off to idle – **60 seconds max**

Starter-assist air starts – **45 seconds max**
(from initial fuel flow to 60% N_2)

Windmilling air start – **No time limit**

FlightSafety international

APU LIMITATIONS

Max. Altitude for Operation ____________

Max. Altitude for APU Start ____________

APU is approved for ____________ Operations.

Allow _ minutes cooldown periods between APU starts or between shutdown and next APU start.

APU LIMITATIONS

Max. Altitude for Operation **35,000 feet**

Max. Altitude for APU Start **20,000 feet**

APU is approved for **Unattended** Operations.

Allow **5** minutes cooldown periods between APU starts or between shutdown and next APU start.

FlightSafety international

MAXIMUM NORMAL OPERATION SPEEDS (V_{MO}/M_{MO}) ARE BASED ON:

1.

2.

MAXIMUM NORMAL OPERATION SPEEDS (V_{MO}/M_{MO}) ARE BASED ON:

1. **Altitude**
2. **Autopilot and Mach Trim Status**

MANEUVERING SPEED V_A

ALTITUDE (FT)

SEA LEVEL – 20,000

20,000 – 29,300

ABOVE 29,300

MANEUVERING SPEED V_A

ALTITUDE (FT) – **V_A (KIAS/MI)**

SEA LEVEL – 20,000 – **272 TO 287 KIAS**

20,000 – 29,300 – **287 TO 330 KIAS**

ABOVE 29,300 – **0.85 MI**

MAXIMUM SPEEDS

SLATS

FLAPS 12°

FLAPS 20°

FLAPS 40°

LANDING GEAR EXTENSION

MAXIMUM TIRE GROUND SPEED

MAXIMUM SPEEDS

SLATS	**250 KIAS**
FLAPS 12°	**250 KIAS**
FLAPS 20°	**225 KIAS**
FLAPS 40°	**180 KIAS**
LANDING GEAR EXTENSION	**180 KIAS**
MAXIMUM TIRE GROUND SPEED	**182 KIAS**

FLAPS 0°, 12° AND 20°
MINIMUM CONTROL SPEED/GROUND

FLAPS 0°, 12° AND 20° MINIMUM CONTROL SPEED/GROUND

103 KIAS

PROHIBITED MANEUVERS AND OPERATIONS

PROHIBITED MANEUVERS AND OPERATIONS

- **AEROBATIC MANEUVERS AND SPINS**
- **DO NOT APPLY SEQUENTIAL FULL OPPOSITE RUDDER INPUTS**
- **TAKE OFF WITH ANY STABILIZER TRIM SYSTEM INOPERATIVE**
- **SLATS / FLAPS OR LANDING GEAR ABOVE 20,000 FEET**

FlightSafety international

MAXIMUM CERTIFICATED ALTITUDE

MAXIMUM CERTIFICATED ALTITUDE

45,000 FEET

FlightSafety international

The Maximum Mach Number during RVSM Ops is __________.

The Maximum Mach Number during RVSM Ops is **0.84 Mach**.

During RVSM Ops, the maximum difference between the pilot and copilot altimeters is __ feet.

During RVSM Ops, the maximum difference between the pilot and copilot altimeters is **45** feet.

FlightSafety international

During gravity feed refueling, the total fuel quantity will be ___ pounds less.

During gravity feed refueling, the total fuel quantity will be **<u>655</u>** pounds less.

APU GENERATOR LOAD LIMITS

On the ground: ___ A.

In flight below 25,000 feet: ___ A.

In flight above 25,000 feet: ___ A.

APU GENERATOR LOAD LIMITS

On the ground: <u>**300**</u> A.

In flight below 25,000 feet: <u>**300**</u> A.

In flight above 25,000 feet: <u>**250**</u> A.

Do not use the APU if the ____ fire extinguisher has been activated.

Do not use the APU if the ____ standby fuel pump is inoperative.

Do not use the APU if the **right** fire extinguisher has been activated.

Do not use the APU if the **right** standby fuel pump is inoperative.

THRUST REVERSER

Below __ knots, do not exceed idle deploy thrust reverser.

Do Not deploy thrust reversers for more than _ minute(s) in a __ minute period.

THRUST REVERSER

Below **70** knots, do not exceed idle deploy thrust reverser.

Do Not deploy thrust reversers for more than **1** minute(s) in a **15** minute period.

AUTOPILOT LIMITATIONS

Takeoff – Do not engage the autopilot below ___ feet.

Minimum engaged height for precision approach – __ feet. (40° Flaps)

Minimum engaged height for non-precision approach – ___ feet AGL.

AUTOPILOT LIMITATIONS

Takeoff – Do not engage the autopilot below **200** feet.

Minimum engaged height for precision approach – **80** feet. (40° Flaps)

Minimum engaged height for non-precision approach – **400** feet AGL.

On the walk around inspection, the nose gear should have a minimum of _ inches extension.

On the walk around inspection, the main gear should have a minimum of ___ inches extension.

On the walk around inspection, the nose gear should have a minimum of **2** inches extension.

On the walk around inspection, the main gear should have a minimum of **1.6** inches extension.

When does the standby fuel pump operate?

1.

2.

3.

When does the standby fuel pump operate?

1. **Whenever the fuel pressure at the engine inlet falls below 6.0 psig if the switch is in AUTO position**

2. **When jettisoning fuel regardless of switch position**

3. **Whenever the switch is in the ON position**

Fuel Interconnect:

- Forward and aft interconnect valves open to ________________________ when the pilot positions the INTERCONNECT switch in the OPEN position.
- Balancing is accomplished by ______.

Fuel Interconnect:

- Forward and aft interconnect valves open to **laterally balance the wings** when the pilot positions the INTERCONNECT switch in the OPEN position.
- Balancing is accomplished by **gravity**.

List the order of tank depletion of fuel system

1.
2.
3.
4.
5.

List the order of tank depletion of fuel system

1. **Fuselage Tank (Above Standpipe)**
2. **Center Tank**
3. **Fuselage Tank (Below Standpipe)**
4. **Wing Tanks**
5. **Collector Boxes**

Collector Box Capacity

Collector Box Capacity

52.5 LBS

When does the auxiliary hydraulic pump run?

a.

b.

c.

d.

e.

f.

When does the auxiliary hydraulic pump run?

a. **Loss of main system pressure**

b. **Landing gear not up and locked**

c. **Low auxiliary system accumulator pressure**

d. **Low main system reservoir fluid level**

e. **Low hydraulic pressure or mechanical failure in either aileron actuator**

f. **Switch in OVRRD position**

FlightSafety international

The main hydraulic system controls:

1.
2.
3.
4.
5.

The main hydraulic system controls:

1. **Aileron boost**
2. **Brakes**
3. **Landing gear**
4. **Nosewheel steering**
5. **Airbrakes**

FlightSafety international

The auxiliary hydraulic system controls:

1.
2.
3.
4.

The auxiliary hydraulic system controls:

1. **Aileron boost backup**
2. **Emergency brakes**
3. **Parking brake**
4. **Thrust Reversers**

Antiskid is inoperative below __ knots

Antiskid switches are located _________________.

Antiskid is inoperative below **10** knots.

Antiskid switches are located **on the glareshield**.

Tiller Steering allows __° left or right deflection.

Pedal Steering allows _° left or right deflection.

Tiller Steering allows **<u>60</u>°** left or right deflection.

Pedal Steering allows **<u>3</u>**° left or right deflection from centerline.

When will the gear aural warning sound?

If any gear is not down and locked and:

1.

2.

3.

When will the gear aural warning sound?

If any gear is not down and locked and:

1. One thrust lever is retarded at an airspeed less than 158 KIAS (Radio Altimeter Inoperative)
2. One thrust lever is retarded with radio altitude less than 400 feet
3. Flaps are extended beyond 30 degrees

The automatic controller limits differential pressure to a normal value of ___ psid.

The safety valve allows a maximum differential of ___ to __ psid.

The automatic controller limits differential pressure to a normal value of **8.79** psid.

The safety valve allows a maximum differential of **8.95** to **9.5** psid.

FlightSafety international

Emergency Pressurization when selected on the CABIN AIR comes from:

1.

2.

3.

Emergency Pressurization when selected on the CABIN AIR selector comes from:

1. **Right Engine Only**
2. **Low Pressure Bleed**
3. **Separate Duct**

Restorable load reduction items:

1.

2.

3.

Restorable load reduction items:

1. **Windshield heats**
2. **Baggage heat**
3. **Galley equipment**

When the EPU is plugged in and the External Power switch is selected to the ON position, the EPU will have priority over

____________________.

When the EPU is plugged in and the External Power switch is selected to the ON position, the EPU will have priority over
all other generators.

In the manual mode, the temperature selector directly controls the ____________.

In the manual mode, the temperature selector directly controls the **trim air valve**.

Illumination of **NOSE TEMP HI** CAS message indicates ______________________________ .

Illumination of the NOSE TEMP HI CAS message indicates **excessive temperature in the nose compartment, and malfunction of nose compartment blowers (on the ground only)**.

When is the APR triggered?

N_2 differential of __%

When is the APR triggered?

N_2 differential of **15**%

FlightSafety international

Emergency lights will illuminate when:

1. Emergency Lights rocker switch in “ARM” and:
    -
2. Emergency Lights rocker switch is in “ON”
    -

Emergency lights will illuminate when:

1. Emergency Lights rocker switch in “ARM” and:
   - **both Distribution Buses fail**
2. Emergency Lights rocker switch is in “ON”
   - **regardless of DC power they will illuminate**

When are P_2/T_2 probes heated?

When are P_2/T_2 probes heated?

When nacelle anti-ice is on and the system senses engine oil pressure to confirm engine is running.

Deice cycle: Total time per cycle = ______________

a. Wing leading edge and slat leading edge = _ seconds

b. Horizontal stabilizer = _ seconds

c. __ seconds dwell off period, then cycle repeats

Deice cycle: Total time per cycle = **60 seconds**

a. Wing leading edge and slat leading edge = **6** seconds

b. Horizontal stabilizer = **4** seconds

c. **50** seconds dwell off period, then cycle repeats

The “FIRE/OVERHT” switchlight:

The “FIRE” light will illuminate when a ____ or _______ is detected in zone _.

The “OVERHT” light lens will illuminate when a ___ or _______ is detected in zone _.

The “FIRE/OVERHT” switchlight:

The “FIRE” light will illuminate when a **<u>fire</u>** or **<u>overheat</u>** is detected in zone **<u>1</u>**.

The “OVERHT” light lens will illuminate when a **<u>fire</u>** or **<u>overheat</u>** is detected in zone **<u>2</u>**.

FlightSafety international

What indication do you get from the PRESS TO TEST?

a. Both ______ lights illuminate to indicate that all 4 sensor tubes are intact and that none of the averaging gas has leaked.

b. Both ______ lights illuminate to indicate that the bulbs are working.

c. In the corners of the PRESS TO TEST, the four white indicator lights illuminate to indicate the ____________.

What indication do you get from the PRESS TO TEST?

a. Both **FIRE-OVERHT** lights illuminate to indicate that all 4 sensor tubes are intact and that none of the averaging gas has leaked.

b. Both **ARM-EMPTY** lights illuminate to indicate that the bulbs are working.

c. In the corners of the PRESS TO TEST, the four white indicator lights illuminate to indicate the **electrical continuity of the discharge circuits through the fire control panel to the respective discharge cartridges on the fire bottles.**

FlightSafety international

Is Zone 2 protected by the fire-extinguishing system?

Is Zone 1 protected by the fire-extinguishing system?

May either bottle be used to extinguish a fire in either engine?

Is Zone 2 protected by the fire-extinguishing system? **No**

Is Zone 1 protected by the fire-extinguishing system?

Yes, when the respective ARM/EMPTY switchlight is pushed, the discharge line is routed only to zone 1

May either bottle be used to extinguish a fire in either engine?

Yes, either or both bottles may be used on either engine

Masks will drop when the cabin pressurization reaches ________ feet.

Above ________ feet crew oxygen will be 100% automatically.

Masks will drop when the cabin pressurization reaches **13,500** feet.

Above **30,000** feet crew oxygen will be 100% automatically.

Length –

Height –

Width (Wingspan) –

Maximum passengers –

Length – **56 ft 9 in (16.94 m)**

Height – **18 ft 5 in (5.54 m)**

Width (Wingspan) – **55 ft 7 in (16.64 m)**

Maximum passengers – **6–9 passengers**

Certified up to ______________________________

Certified for FAR _______ (up through Amendment 54)

Maximum landing altitude: ________

Maximum altitude for autopilot and yaw damper inoperative: ________

Certified up to **Maximum Operating Altitude**

Certified for FAR **Part 25** (up through Amendment 54)

Maximum landing altitude: **14,000 ft.**

Maximum altitude for autopilot and yaw damper inoperative: **31,000 ft.**

An **APU OVERSPEED** WARNING message means ____________ and ______________________.

An **APU OVERSPEED** STATUS message means ____________ and ______________________.

An **APU OVERSPEED** WARNING message means **<u>APU RPM too high</u>** and **<u>APU did not enter automatic shutdown sequence</u>**.

An **APU OVERSPEED** STATUS message means **<u>APU RPM too high</u>** and APU **<u>entered automatic shutdown sequence</u>**.

An **APU OVERTEMP** WARNING message means
______________ and ___________________________.

An **APU OVERTEMP** STATUS message means
______________ and ___________________________.

An APU OVERTEMP WARNING message means **APU excessive temperature** and **APU did not enter automatic shutdown sequence**.

An APU OVERTEMP STATUS message means **APU excessive temperature** and **APU entered automatic shutdown sequence**.

In addition to each respective configuration discrepancy, what two conditions trigger the following WARNING messages? CONFIG AIRBRAKE, CONFIG FLAPS, CONFIG PARKING, CONFIG SLATS, and CONFIG TRIM

____

____

In addition to each respective configuration discrepancy, what two conditions trigger the following WARNING messages? **CONFIG AIRBRAKE**, **CONFIG FLAPS**, **CONFIG PARKING**, **CONFIG SLATS**, and **CONFIG TRIM**

Aircraft on ground

Both engines thrust beyond 70% N_1

FUEL STBY PUMP ON (L/R) indicates ________________.

FUEL STBY PUMP ON (L/R) indicates **respective standby pump is operating**.

The fuel standby pump will activate when any of the following occur:

The fuel standby pump will activate when any of the following occur:

Fuel pressure drops below 6 psi

STBY PUMP switch is selected to the ON position

FUEL JETTISON pushbutton is pressed

The **CAB ALT HI** WARNING message appears when the cabin altitude exceeds ______ feet.

The **CAB ALT HI** WARNING message appears when the cabin altitude exceeds **10,000** feet.

The **BATT DISCHARGE** message on indicates that ______________________ when ________________ and ________________________.

The **BATT DISCHARGE** message indicates that **both battery voltages are below 25V** when **at least one generator is active** and **one battery is connected**.

T/R will appear within the EICAS N_1 display when

____.

T/R will appear within the EICAS N_1 display when

____.

T/R will appear within the EICAS N_1 display when

____.

T/R will appear within the EICAS N_1 display when **the respective thrust reverser is armed**.

T/R will appear within the EICAS N_1 display when **there is a valid weight on wheels signal and the respective thrust reverser is deployed**.

T/R will appear within the EICAS N_1 display when **the respective T/R FAIL (L/R) message appears**.

The coloration of an EICAS message will match the coloration of the applicable parameter.

TRUE/FALSE

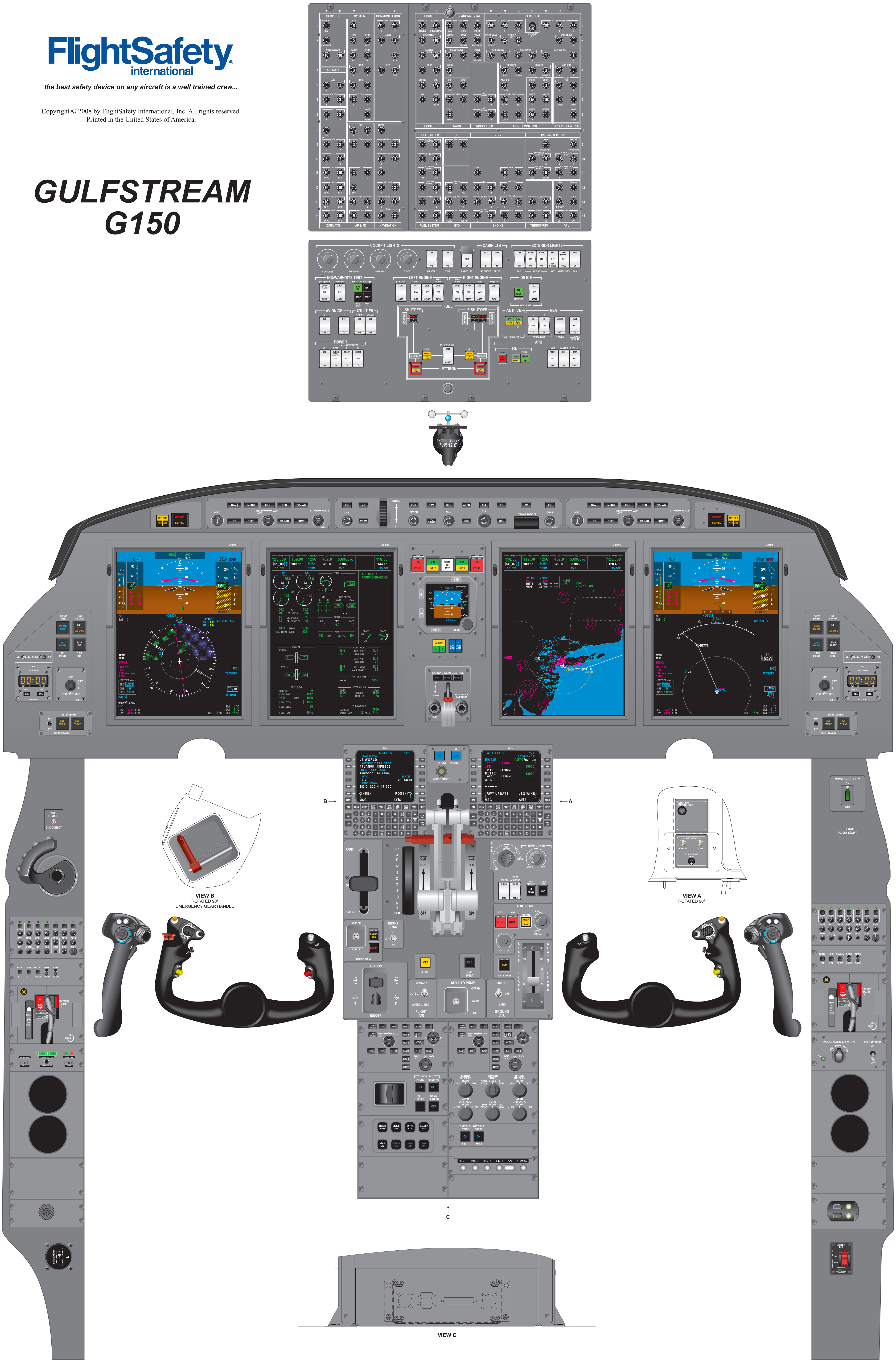
The coloration of an EICAS message will match the coloration of the applicable parameter.

TRUE

FlightSafety international

the best safety device on any aircraft is a well trained crew...

GULFSTREAM G150



042908

FOR TRAINING PURPOSES ONLY