# GENERAL

Jet engines produce thrust by accelerating air. It is the product of the mass of the air times the increase in velocity that determines thrust output. To generate a given amount of thrust, a small volume of air can be accelerated to a very high velocity, or a relatively large amount can be accelerated to a lower velocity.

In a turbojet engine, incoming air is compressed, mixed with fuel, combusted and exhausted at a high velocity. In a turbofan engine, only a portion of incoming air is combusted. The hot air then drives the fan which accelerates a large volume of air at a lower velocity. This air is bypassed around the engine and is not mixed with fuel or combusted. The relation of the total mass of bypassed air, to the amount of air going through the combustion section, is known as the bypass ratio. The bypass ratio of the Citation XLS engine is 3.8 to 1.

The PW545B, developed by Pratt and Whitney Canada Inc., is a turbofan engine rated at 4095 pounds static thrust each for takeoff, at an ambient temperature of 59°F. A concentric shaft system supports the fan and turbine rotors. The inner shaft connects the fan ($N_1$) and the axial boost stage of the low pressure compressor at the front of the engine to the three rear low pressure turbines. The outer shaft connects the 2 axial and 1 centrifugal high pressure compressors, ($N_2$) and the forward high pressure turbine.

All intake air passes through the fan. Immediately aft of the fan the airflow is divided by a concentric duct. About 80% of the total airflow is bypassed around the engine through the outer duct and is exhausted at the rear. Air entering the inner duct passes through guide vanes to the axial boost compressor stage, then through a second set of guide vanes and is compressed by two more axial compressors and the centrifugal compressor. The high pressure air then passes through a diffuser assembly and moves aft to the combustion section.

The combustion chamber is of a reverse flow design to save space and reduce engine size. A portion of the air entering the chamber is mixed with fuel and ignited. The remainder enters the chamber liner downstream for cooling.

Fuel is introduced by eleven hybrid nozzles supplied by a dual manifold. The mixture is ignited initially by two spark igniters which extend into the combustion chamber at the four and eight o'clock positions. After start, combustion becomes self-sustaining. The hot gases expand, reverse direction and pass through a set of turbine guide vanes to the high pressure turbine. The power generated by this turbine is transmitted by the outer shaft to turn the high pressure compressor.

Only a small part of the energy available in the hot, high pressure air is absorbed by the high pressure turbine. As the expanding gases move rearward, they pass through another set of guide vanes and enter the three-stage, low pressure turbine. A greater portion of the remaining energy is extracted there and transmitted by the inner shaft to the forward mounted fan and boost rotor. The hot gases then exhaust into the atmosphere.

The turbofan is in effect two interrelated power plants. One section is designed to produce energy in the form of high velocity, hot air. The other utilizes some of this air to provide the power to drive the fan. The fan of the PW545B, pumping a high volume of cool low velocity air, produces over 50% of the total thrust.

# ENGINE AIRFLOW AND CROSS SECTION

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**FLANGES**

A NACELLE TO FAN CASE
B FAN CASE TO INTERMEDIATE CASE
C INTERMEDIATE CASE TO OUTER BYPASS DUCT
D OUTER BYPASS DUCT TO REAR BYPASS DUCT
E REAR BYPASS DUCT TO AIRFRAME SUPPLIED BYPASS DUCT

CORE AIR FLOW
BYPASS AIR FLOW

**STATIONS**

| | |
|---|---|
| 0 | AMBIENT |
| 1 | FAN CASE INLET (ID) |
| 1.1 | FAN CASE INLET (OD) |
| 1.2 | FAN BYPASS INLET |
| 1.3 | FAN BYPASS OUTLET |
| 1.6 | BYPASS EXHAUST |
| 2 | FAN CORE INLET |
| 2.1 | FAN CORE OUTLET |
| 2.2 | HP COMRESSOR AXIAL INLET |
| 2.5 | HP COMPRESSOR INTERSTAGE |
| 2.8 | HP COMPRESSOR IMPELLER INLET |
| 3 | COMBUSTION CHAMBER INLET |
| 4 | HP TURBINE INLET |
| 4.5 | INTERTUBINE |
| 5 | LP TURBINE OUTLET |
| 6 | CORE EXHAUST |

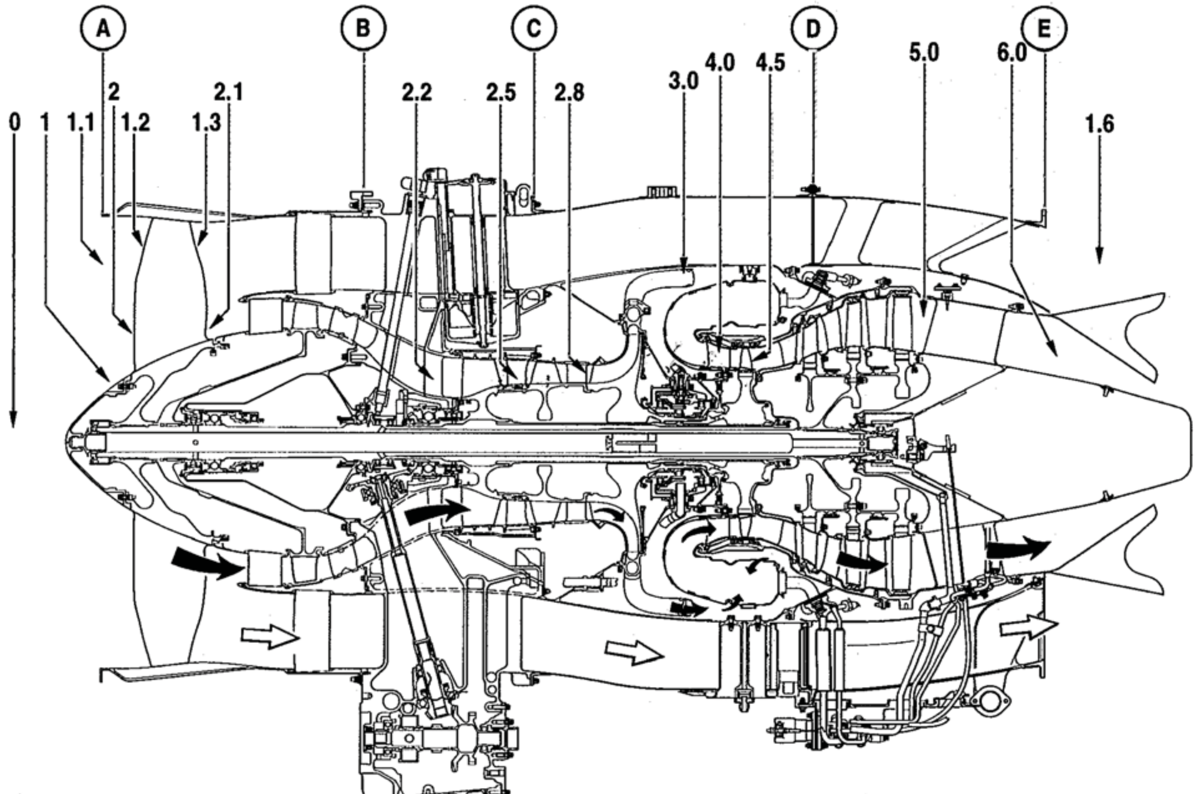


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Figure 2-1

# ENGINE CONTROL SYSTEM

The primary function of the Electronic Engine Control (EEC) system is to control the engine low rotor speed ($N_1$) and thereby the engine thrust as requested by the pilot's throttle position and the existing ambient conditions. The engine control system, which is a single channel, microprocessor based controller, provides two main modes of operation: AUTO mode and MANUAL (MAN) mode. MANUAL mode will automatically be entered in the case of an EEC major fault or may be selected by the pilot by placing the EEC switch, located on the lower left of the instrument panel, in the MAN position.

In AUTO mode the EEC provides the following functions in response to the Thrust Lever Angle (TLA) signal:

- Detented throttle, automatic thrust setting ($N_1$ governing).
- Idle governing ($N_2$ governing) at ground idle and flight idle.
- Acceleration and deceleration limiting.
- $N_1$ and $N_2$ speed limiting.
- Closed loop bleed valve (BOV) control.
- Engine diagnostic system (EDS) functions.
- Overspeed protection ($N_2$).
- $N_1$ or $N_2$ synchronization.

In MANUAL mode, the Fuel Control Unit (FCU) takes over full control of the engine speed in response to the throttle position. In MANUAL mode the throttle directly controls the FCU by means of a mechanical linkage. MANUAL mode provides the following functions:

- Pilot adjustable power setting ($N_2$ governing).
- Idle governing ($N_2$ governing) at flight idle and anti-ice idle.
- Acceleration and deceleration limiting (ratio unit control).
- $N_2$ speed limiting.
- Closed Loop Bleed Valve (BOV) control.
- Limited engine diagnostic system functions (EDS)

The Engine Diagnostic System (EDS) provides troubleshooting tools to resolve engine and airframe related EEC system problems.

# CONTROL SYSTEM SCHEMATIC

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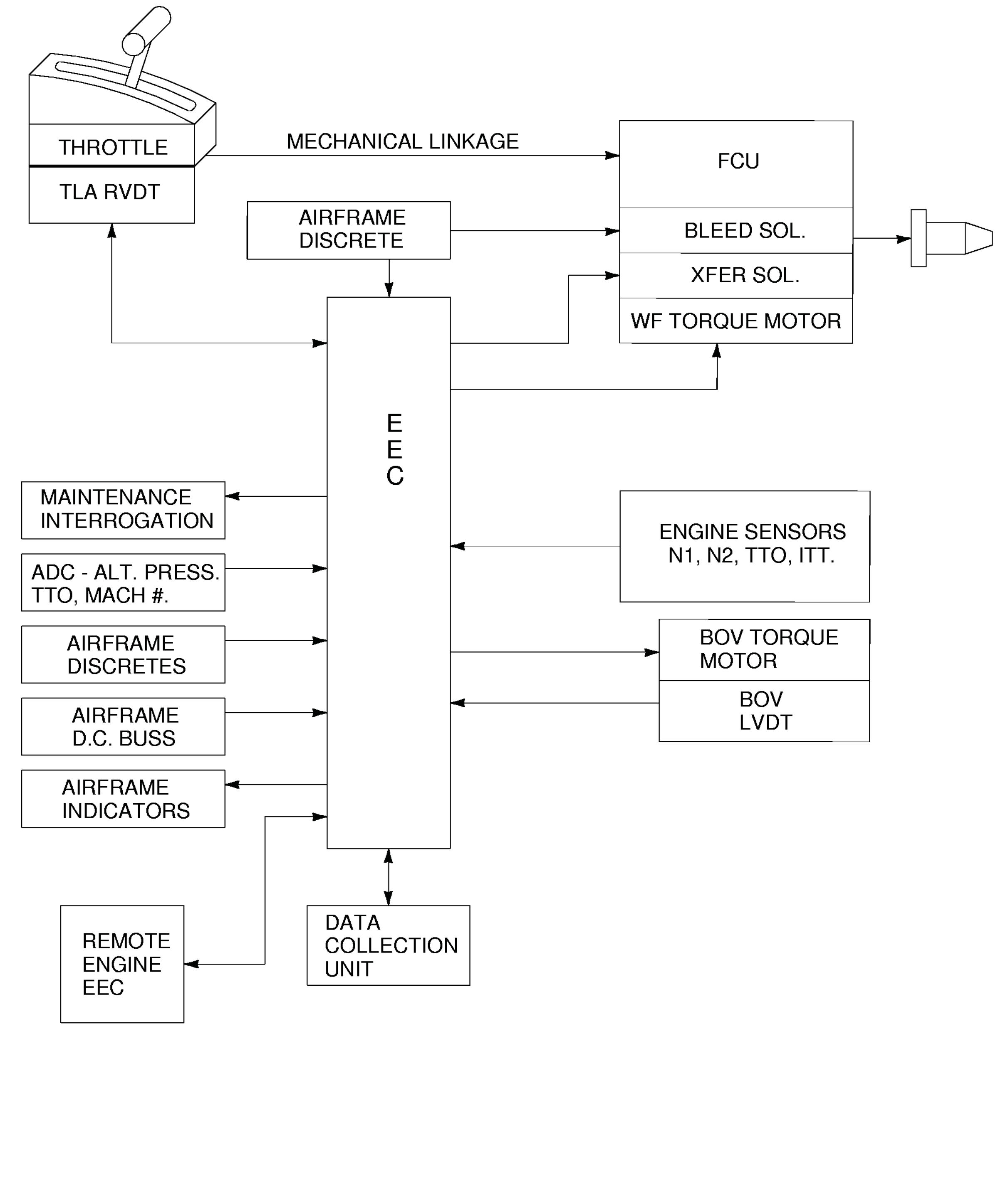


Figure 2-2

# GROUND IDLE

The Citation XLS is equipped with a ground idle system which automatically allows the engines to decelerate to an idle speed eight seconds after the landing gear squat switches have sensed a landing. The slower idle speed allows better taxiing control at lighter weights and in very cold temperatures, when the normal flight idle speed of 54.4% $N_2$ (at sea level) would require more use of the brakes, resulting in reduced brake life. The ground idle function is controlled automatically by the EEC. A GND IDLE annunciator is located on the annunciator panel. The annunciator will illuminate when the airplane is on the ground and the engine has assumed the slower idle speed, or will assume it when the throttle is reduced to idle.

# ENGINE SYNCHRONIZER

An engine synchronizer system provides automatic $N_1$ (fan) or $N_2$ (turbine) RPM matching of the right (slave) engine to the left (master) engine. The synchronizer will continuously monitor the engine speeds and adjust the slave engine speed setting as required. The synchronizer system is controlled by the EEC and has a range capability of 4.5% of fan RPM.

A rotary FAN-OFF-TURB switch on the pedestal actuates the engine synchronizer system. The FAN position synchronizes $N_1$ RPM. The TURB position synchronizes $N_2$ RPM. The OFF position deactivates the system. An indicator light adjacent to the synchronizer switch comes on when the system is turned on. A turbine out-of-sync condition is generally more noticeable in the cockpit and a fan out-of-sync condition is usually more noticeable in the area of the rear seats. Synchronization may not be achieved in some cases with the engines operating near idle due to limitations in the synchronization logic. Automatic synchronization is not available with the EEC's in manual mode.

# IGNITION SYSTEM

Each engine incorporates dual exciter units and two igniters. The exciter units convert battery or generator input to high voltage Direct Current (DC), store it momentarily until a given energy level is reached, and allow it to discharge in spark form through the igniters. System wiring is such that malfunction of one igniter or exciter will not affect normal operation of the other.

Cockpit control consists of two-position R and L ignition switches. In NORM, function is automatic during start and with engine anti-ice selected. Moving the throttle to IDLE after depressing the start button activates ignition until it is terminated automatically at approximately 38% turbine RPM ($N_2$). Continuous ignition occurs any time the respective engine anti-ice or ignition switch is ON.

A small green IGN annunciation adjacent to the ITT indication illuminates when that engine's exciter is receiving electrical power. If one ignitor should fail, ignition will still be available from the remaining ignitor. If the ignition annunciator does not illuminate when ignition is selected, or should be automatically provided, check the applicable ignition system circuit breaker on the left circuit breaker panel, or circuit breakers in the aft power junction box.

## ACCESSORY GEARBOX

The starter/generator, fuel control, hydraulic pump, oil pump, $N_2$ monopole speed sensor and an AC generator for the windshield anti-ice are driven by the accessory gearbox mounted below the engine. Power to drive the gearbox is transmitted from the $N_2$ section through the tower shaft and a series of bevel gears. Lubrication is provided by the engine oil system.

## OIL SYSTEM

The oil system is a self contained system installed on each engine. Each engine has a nominal oil capacity of 7.49 U.S. quarts, of which 0.60 quarts are unusable. Oil level is to be checked using the outboard sight gage on each engine ten minutes after the engine has been shut down, per the engine maintenance manual. Oil is added to the oil tank via the filler neck located next to the sight gage. The filler neck is equipped with a check valve to prevent loss of oil in the event the filler cap is not properly installed.

During engine operation, oil is drawn out of the oil tank past the chip collector and a screen to the oil pump inlet. After passing through the pump, oil flows past the pressure adjusting (relief) valve, which bypasses oil back to the tank if the system pressure is too high. Oil then flows through the oil filter and fuel-cooled oil cooler before going past the oil temperature bulb on the way to the engine bearings and accessory gearbox. Impending filter bypass is sensed by a pressure switch in the line which will illuminate an amber annunciator if the pressure drop through the filter becomes excessive.

The oil pressure indicating system measures the differential pressure between the #4 bearing scavenge line and a location just downstream of the oil cooler. A low oil pressure switch is installed which will cause the LOW OIL PRESS annunciator in the cockpit to illuminate if the differential pressure drops below 20 psid. The oil temperature indication system measures the temperature at the temperature bulb between the oil cooler and the engine bearings.

If the oil filter becomes clogged, the pressure drop across the filter will cause an impending bypass switch to activate which will illuminate the OIL FLTR BP annunciator in the cockpit indicating an impending bypass of the oil filter. Once the pressure drop across the filter becomes great enough, a bypass valve will open allowing lubrication to continue without filtration.

# ENGINE OIL SYSTEM SCHEMATIC

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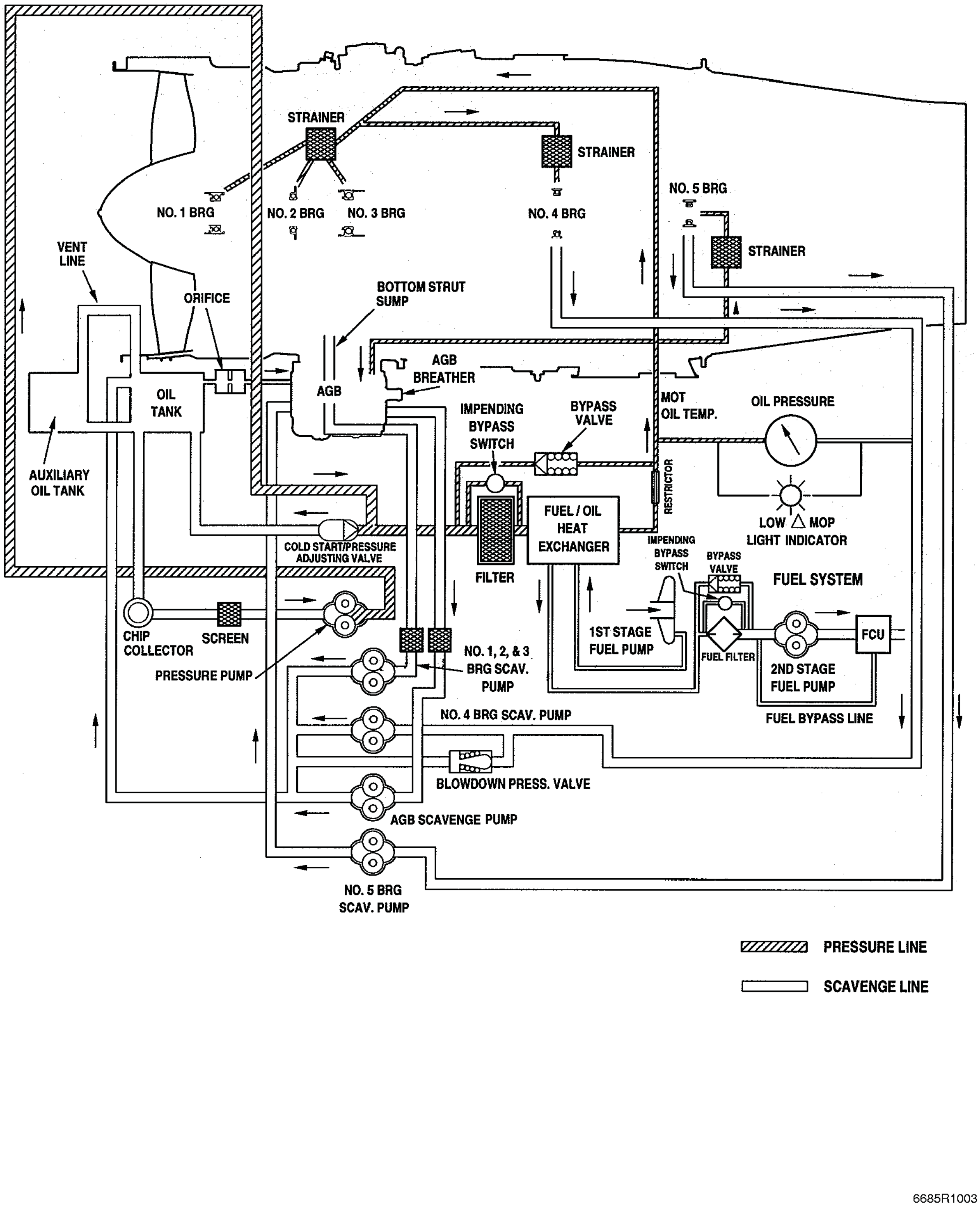


Figure 2-3

# THRUST REVERSER SYSTEM

## DESCRIPTION AND OPERATION

The thrust reversers are of the external target type employing two vertically oriented doors or buckets, which, when deployed, direct exhaust gases forward to provide a deceleration force for ground braking. When stowed, the reversers fair into external airplane contours to form the aft portion of the nacelle. The reversers are mounted to the engine fan nozzle through an aluminum support casting and four interconnecting links per door.

## NORMAL OPERATION

The reverser system is designed for two-position operation: stowed during takeoff and flight and deployed during landing ground roll. The reversers are activated by pilot operation of the thrust reverser throttle levers and deployed by hydraulic pressure supplied by an engine-driven pump and directed to the drive actuators. The actuators are connected to a slider mechanism which is in turn connected to the reverser doors by a four-bar linkage system. The system, by design, incorporates an overcenter feature in the linkage which locks the reverser in the stowed position.

Hydraulic actuators are mounted to the support casting on each side of the reverser. The airplane hydraulic system provides pressure to these actuators which in turn operate the linkage system along a sliding track in the support casting to deploy and stow the reversers.

Control of the individual thrust reverser is through the reverse thrust lever mounted on each of the engine throttles. The reversers can only be deployed when the primary throttle levers are in the idle thrust position and the airplane is on the ground as sensed by either of the main gear squat switches. The reverse thrust lever also controls engine thrust during reverse thrust operation.

An automatic system is incorporated in the installation to reduce engine power approximately to idle if an inadvertent deployment, or stowage, of the thrust reverser should occur. In the event of an inadvertent thrust reverser deployment, an automatic throttle retarding device will bring the throttle to approximately idle thrust depending on the amount of throttle friction that has been applied. After this device has activated, the throttle lever can be advanced resulting in corresponding reverse thrust.

**WARNING**

- **DO NOT USE THROTTLE FRICTION OR MANUALLY RESTRAIN THE THROTTLE LEVERS DURING TAKEOFF. SHOULD AN INADVERTENT THRUST REVERSER DEPLOYMENT OCCUR, THIS COULD RESULT IN A DANGEROUS ASYMMETRICAL THRUST CONDITION.**
- **SHOULD AN INADVERTENT THRUST REVERSER DEPLOYMENT OCCUR, THE PILOT MUST ENSURE THAT THE THROTTLE LEVER IS IN THE IDLE POSITION.**

Moving the reverse thrust lever from the STOWED to the IDLE REVERSE position actuates the deploy cycle. This electrically opens the isolation valve, moves the reverser control to deploy and pressurizes the airplane hydraulic system. The isolation valve allows the airplane hydraulic system to pressurize the thrust reverser system. The amber ARM light indicates hydraulic pressure to the reverser control valve as sensed by a pressure switch.

During thrust reverser deployment, the initial movement of the actuators activates the unlocked switches. Either switch will cause the amber UNLOCK light to illuminate. Further movement of the actuator unlocks the reverser through the overcenter linkage. The remaining travel of the actuators deploys the reverser doors.

At full deployment of the reverser, the deploy switch is activated which in turn illuminates the white DEPLOY light and unlocks the pedestal-mounted throttle lock-out cam. The purpose of the lock-out cam is to prevent increasing engine thrust, once reverser deployment has been selected, until the reversers have fully deployed.

Three reverser indicator lights for each reverser are mounted on the cockpit glare shield for monitoring reverse functions: ARM, UNLOCK and DEPLOY.

**NOTE**

The DEPLOY light shall illuminate in less than 1.5 seconds after the hydraulic UNLOCK light illuminates. An erroneous sequencing or a delay in the illumination of the thrust reverser lights indicates a failure in the thrust reverser system. Either or both conditions requires a maintenance check.

**WARNING**

**DO NOT ATTEMPT TO FLY THE AIRPLANE IF THE THRUST REVERSER PREFLIGHT CHECK IS UNSUCCESSFUL.**

As previously mentioned, either of the landing gear squat switches must be activated to complete the electrical circuit necessary to initiate deployment of the thrust reversers.

The thrust reverser lever(s) should not be placed in the idle reverse detent position in flight since a single failure of either squat switch could permit deployment of the thrust reverser(s). If the thrust reverser lever is placed in the idle reverse detent position while airborne, the airplane MASTER WARNING light will flash along with illumination of the ARM and HYD PRESS annunciator lights. A MASTER WARNING light, when thrust reversers are moved to deploy on the ground, means that neither landing gear squat switch has activated. To ensure actuation of the squat switches and to eliminate any delay in the deployment of the thrust reversers, it is recommended that the speed brakes be extended immediately following touchdown.

After deployment, power may be increased by moving the thrust reverser throttle levers aft for maximum reverse thrust. Thrust reverser throttle stops are set to give approximately 75% of takeoff thrust. These stops will allow the pilot to keep his/her attention on the landing rollout instead of diverting attention to the reverse power settings.

For increased aerodynamic drag on landing roll, it is suggested that the thrust reversers remain in the deployed idle reverse power position after reverse thrust power has been terminated at 60 KIAS unless loose pavement, dirt or gravel is present on the runway. Idle reverse thrust is capable of causing ingestion of small grit at very low ground speed.

To stow the thrust reversers, move the reverse thrust lever through the idle reverse detent to the stow position. This actuates a switch in the pedestal which moves the thrust reverser control valve to the stow position. Hydraulic pressure is directed by the valve to the two actuators in the reverser which move the thrust reverser doors to the stowed position. Initial movement of the linkage toward the stowed position deactivates the deploy switch extinguishing the DEPLOY light. As each actuator moves to the fully stowed and locked position, they deactivate a thrust reverser unlocked switch. When both switches in a reverser have been deactivated, the UNLOCK light is extinguished, the airplane hydraulic system is depressurized and the affected thrust reverser isolation valve closes. This puts the ARM light out as the pressure in the line downstream of the isolation valve drops.

# THRUST REVERSER STOW SWITCHES/LIGHTS

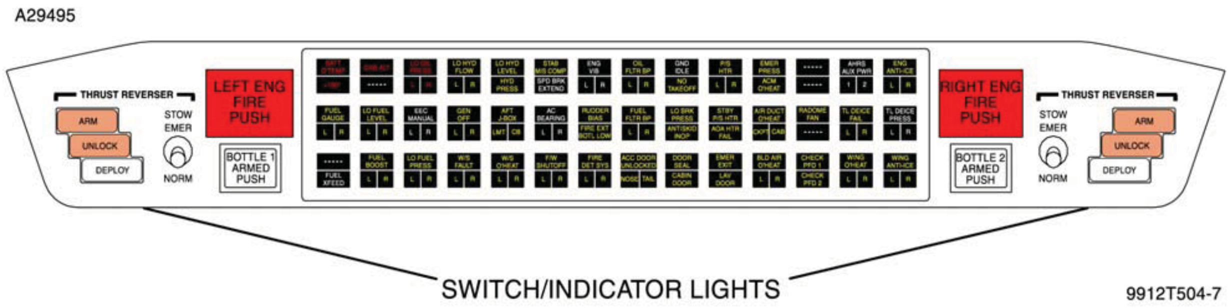


Figure 2-4

The thrust reversers are not to be used during touch and go landings. A full stop landing must be made once reverse thrust has been selected. Less distance is required to stop, even on a slick runway, once the reversers have been deployed, than is required to restow the reversers and takeoff.

Landings with a crosswind component of 24 knots at 10 meters above runway were demonstrated. Adequate control of the airplane was maintained during and after thrust reverser deployment. Single-engine reversing has been demonstrated during normal landings and is easily controllable.

## EMERGENCY STOW OPERATION

An emergency stow switch for each thrust reverser is located on the cockpit glare shield and will provide the same stow sequence (using the alternate 28 volt thrust reverser power source) as the thrust reverser throttle levers, in the event of a failure of the pedestal-mounted deploy and stow switch, or of the respective 28 volt direct current (VDC) bus.

Each emergency stow switch receives its electrical power through the opposite thrust reverser circuit breaker. The emergency stow function can be checked on the ground by deploying the reversers normally and then actuating each emergency stow switch. The DEPLOY and UNLOCK lights shall extinguish. The ARM and HYD PRESS lights remain illuminated. Return the thrust reverser lever to stowed position, then turn each emergency stow switch off. All lights shall be extinguished.

**WARNING**

**DO NOT ATTEMPT TO FLY THE AIRPLANE IF THE THRUST REVERSER PREFLIGHT CHECK IS UNSUCCESSFUL.**

# FIRE PROTECTION

Engine fire detection consists of a closed-loop sensing system and detector control unit which illuminates the respective red ENGINE FIRE warning light on the cockpit glare shield if a fire or overheat condition is present. The warning light, under a transparent, spring-loaded guard, also serves as a firewall shutoff switch.

Lifting the guard and depressing the warning light simultaneously closes the respective firewall fuel and hydraulic valves, de-energizes the starter/generator and arms the two freon extinguishing bottles. Firewall shutoff and extinguisher arming are indicated by illumination of the respective LO FUEL PRESS, HYD PRESS, F/W SHUTOFF and GEN OFF annunciator panel lights and both white BOTTLE ARMED lights.

Once armed, either bottle may be discharged to the selected engine by pushing the BOTTLE ARMED light. The light will go out as the light is pushed. System plumbing is such that both bottles can be directed to the same engine if necessary.

Each fire extinguisher bottle is equipped with a thermally compensated pressure switch. If the pressure in either bottle decreases below the minimum level demonstrated for adequate fire extinguishing, an amber FIRE EXT BOTL LOW annunciator on the annunciator panel will illuminate.

Function of the lights and continuity of the sensor and detector control units is checked by placing the rotary TEST selector in the FIRE WARN position and observing illumination of both red lights. Depressing either fire light will then illuminate both BOTTLE ARMED lights.

All test, detection and extinguishing features are electrically powered from the main Direct Current (DC) buses requiring either external power, the battery switch in BATT, or a generator on the line for operation.

## FIRE DETECTION INDICATING LIGHTS

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Figure 2-5

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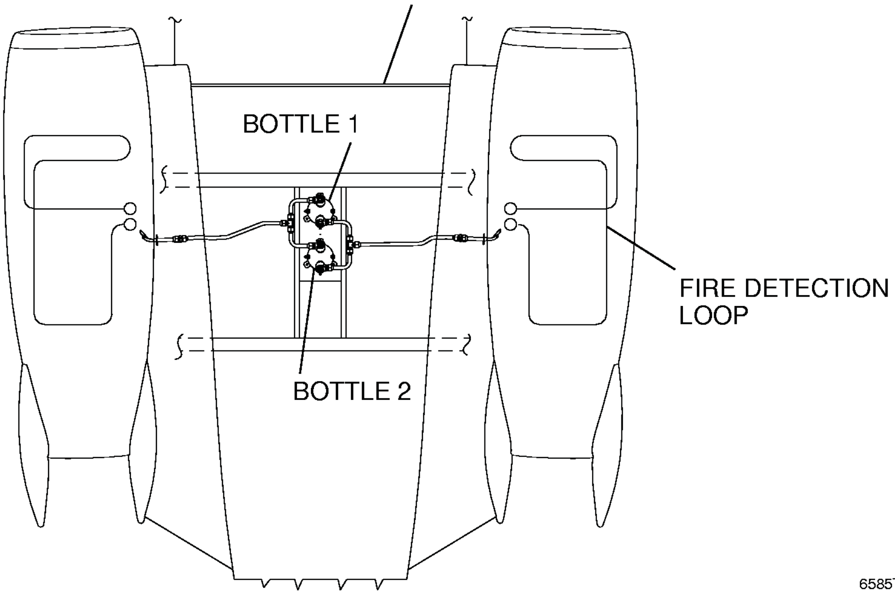


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