2A-32-10: General Description

The Gulfstream IV landing gear has dual wheels on both the main and steerable nose gear. The main gear retract laterally into wheel wells on either side of the fuselage keel at the wing root area. The auto-centered nose gear retracts forward into a wheel well beneath the cockpit. Integrated gear doors close upon completion of gear extension and retraction providing an aerodynamically conformal fuselage surface.

Landing gear operation normally uses 3000 psi pressure from the Combined hydraulic system. Extension and retraction selections are made with the cockpit landing gear handle that is mechanically linked to the landing gear selector valve. The selector valve directs hydraulic pressure in a defined sequence to open the gear doors, extend or retract the gear and close the doors when the gear have reached the selected position. If Combined hydraulic system pressure is not available, the Utility system pressure may be used for landing gear operation, provided adequate fluid remains in the Combined system. In the event that Combined system fluid is lost, a single-use emergency gear extension system operated with a 3000 psi nitrogen gas bottle will extend the landing gear, but leave the gear doors open. The landing gear operating system is shown in Figure 1.

Landing gear position is indicated on the gear control panel by a green light for each gear and a red light in the landing gear control handle. Illumination of the red landing gear handle light indicates a disagreement between the selected and actual position of one or more of the landing gear. The individual gear green lights come on when the corresponding landing gear is down and locked.

In addition to the red landing gear position disagreement light in the gear handle, a warning horn/klaxon will sound if the landing gear is not down and locked and the aircraft configuration, altitude and/or power setting approximates the landing configuration.

A proximity, or nutcracker switch is installed on each gear. As the weight of the aircraft compresses the shock struts of the landing gear, the nutcracker switches close completing circuits for relays to aircraft systems that only operate on the ground. Conversely, when airborne, the nutcraker switches open, providing an in-the-air signal to systems that operate only in flight.

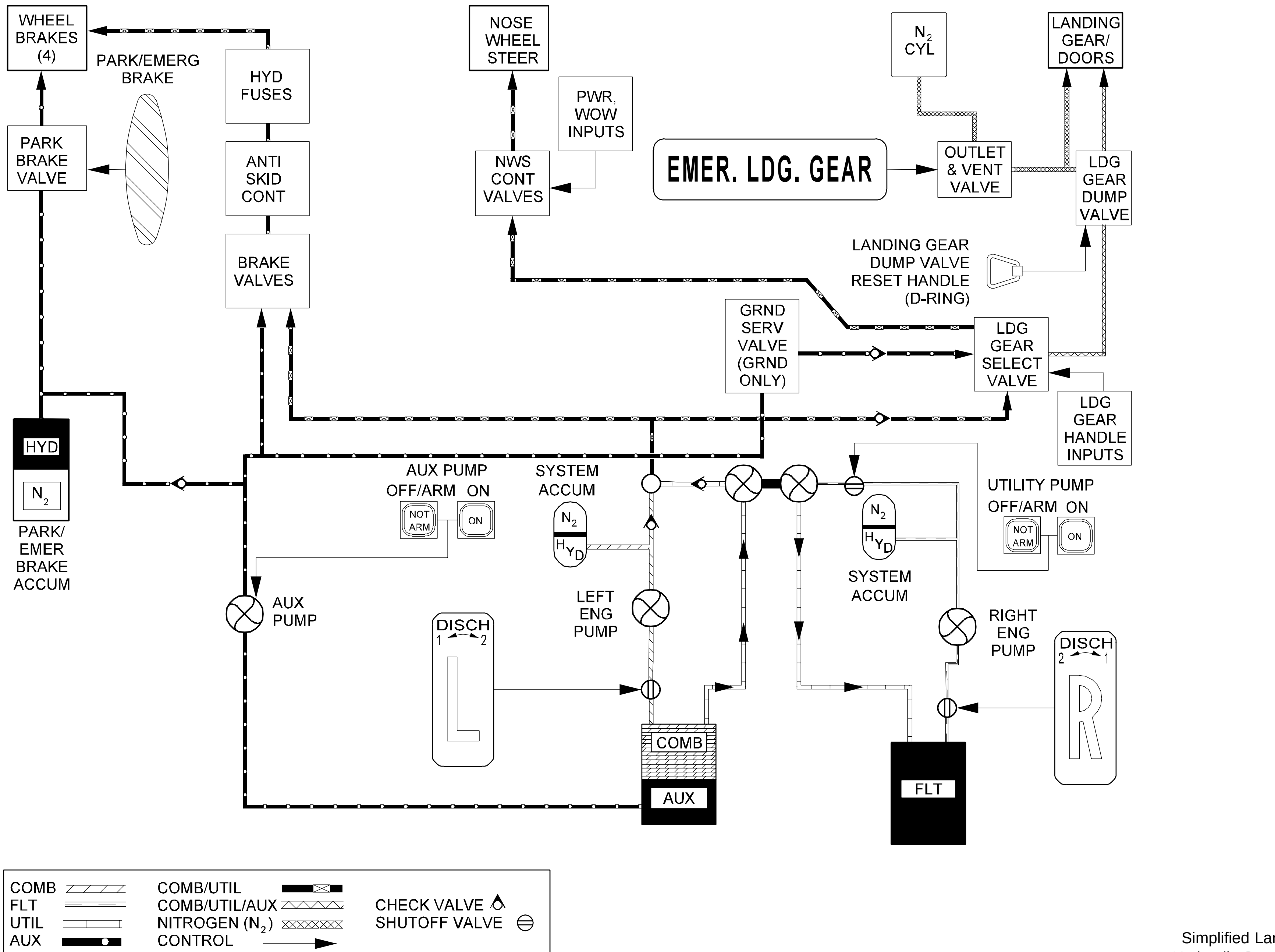
Main landing gear brakes are air-cooled multiple carbon-fiber discs with anti-skid protection. Overheat protection for the tires/wheels is provided by fusible plugs in the wheels that melt, releasing tire pressure, if high temperature thresholds are exceeded. Brake temperatures are monitored (on aircraft S/Ns 1000 to 1155 with ASC 167 and S/N 1156 and subsequent) and indicated on cockpit displays.

The Combined hydraulic system provides 3000 psi for normal brake operation. Hydraulic fuses are incorporated into the brake hydraulic lines and close to prevent fluid loss if a hydraulic line is damaged or cracked. If Combined/Utility system pressure is not available, but brake system hydraulic lines are intact, the Auxiliary hydraulic system can be used for brake operation. If no hydraulic pressure is available from aircraft systems, accumulator pressure from the Parking/Emergency brake system will provide approximately five to six brake applications without anti-skid protection.

Control of the brake system is either through a hydro-mechanical analog brake system (HMAB) for aircraft S/Ns 1000 - 1213 with ASC 307 and S/Ns 1214 and subsequent, or a brake-by-wire system for aircraft S/Ns 1000 - 1213 without ASC 307. In the analog system, cockpit brake pedal application is mechanically linked to the system brake metering valves that apply hydraulic pressure proportional to pedal depression. In the brake-by-wire system, cockpit brake pedals incorporate linear variable differential transducers (LVDTs) that supply pedal deflection signals to a brake electronic control unit (ECU) that opens the left and right brake control valves proportional to LVDT commands. Brake feel is supplied by bungees that apply proportional resistance to pedal deflection. Anti-skid circuitry is integrated into the brake-by-wire system. Analog braking uses a separate anti-skid control box.

The Landing Gear System is composed of the following components.

- 2A-32-20: Landing Gear and Doors
- 2A-32-30: Extension and Retraction
- 2A-32-40: Wheels and Brakes
- 2A-32-50: Nose Wheel Steering



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Simplified Landing Gear Hydraulic System Diagram
Figure 1

2A-32-20: Landing Gear and Doors System

1. General Description

The major components of each main landing gear assembly are: a landing gear door, a shock strut, a sidebrace actuator/downlock mechanism and an uplock mechanism. Each of the main landing gear assemblies has dual wheels with integrated brakes mounted on an articulated trailing arm attached to the landing gear structural post. The pneumatic/hydraulic shock strut is attached between the wheel end of the articulated trailing arm and the top of the structural post. In flight with the landing gear down, the shock strut extends and the articulated trailing arm pivots downward, positioning the wheels for ground contact and compression of the shock strut. The major components of the main landing gear are shown in Figure 2 and Figure 3.

A side brace hydraulic actuator/downlock extends and retracts the main gear. In the extended position, hydraulic pressure positions mechanical downlocks that prevent gear retraction. The downlocks include mechanical springs loads that maintain the locked position if hydraulic pressure is lost. In gear retraction, hydraulic pressure compresses the springs, displaces the downlocks and allows gear retraction inward into the wheel well in the wing root next to the fuselage keel. An uplock roller mounted on the dual wheel axle assembly engages an uplock hook in the wheel well and rotates to latch the main gear in the uplocked position. The main landing gear door, hinged at the inboard side of the wheel well, closes after the gear retracts and mates with the fairing door panel mounted on the outside of the main gear assembly to seal the wheel well.

The nose gear assembly includes steerable dual wheels mounted on a shock strut attached to a trunnion and a hinged two-piece truss brace. A drag brace between the truss brace and the shock strut provides additional structural support and limits the extension of the nose gear shock strut. The truss brace and trunnion pivot axes allow the hydraulic actuator cylinder to retract the nose gear assembly forward and up and extend the gear aft and down. See the illustration in Figure 4.

When the nose gear is fully extended, the forward and aft sections of the truss brace move over center to lock the nose gear down. A downlock hydraulic cylinder extends an actuating arm holding the truss brace in the over center position. Springs are also incorporated in the over center downlock to maintain the over center lock in the event of hydraulic failure. When the nose gear retracts, the downlock actuator arm moves aft, allowing the forward and aft portions of the truss brace to fold at the hinge point. As the nose gear reaches the fully retracted position, a roller attached to the dual wheel axle assembly pushes aside a spring latch and engages an uplock hook, with the spring latch then falling into place securing the roller into the uplock hook. A hydraulically actuated pushrod then rotates the uplock hook upward to provide positive nose gear latching.

Two symmetrical nose wheel doors, hinged on the outside of the forward nose wheel well and mechanically linked together, operate in conjunction with the nose gear. Timing valves integrated into the extension/retraction cycle operate the hydraulic door actuating cylinder in the correct sequence. When the nose gear is retracted and seated in the uplock hook, the doors close to cover the forward wheel well, with the fairing panel attached to the rear of the nose gear drag brace enclosing the aft portion of the wheel well.

Indication of landing gear position is provided to the flight crew through lights on the landing gear control panel, an aural warning horn/klaxon, and by messages on the Crew Alerting System (CAS) in the event of malfunction.

2. Description of Subsystems, Units and Components

A. Landing Gear Doors

The landing gear doors are shaped to conform to the surrounding fuselage areas, providing an aerodynamically smooth covering of the wheel well areas when closed. The main landing gear wheel well is enclosed by a single door in conjunction with a fairing panel attached to the outside of the gear strut. The nose gear wheel well is enclosed by two symmetrical curved doors and a fairing panel attached to the aft side of the nose gear. When the landing gear is fully extended, the gear doors close to reduce drag and air noise. All gear door movement is sequenced by mechanical linkages operating hydraulic actuators to coordinate actuation with landing gear extension and retraction.

Procedures exist to open and close the doors on the ground in order to accomplish maintenance tasks. Hydraulic power from the Auxiliary system and operation of a ground service valve, located adjacent to the nose wheel well, allow movement of the gear doors by pressurizing the landing gear on the ground. The ground service valve is shown in Figure 5.

B. Shock Struts

Standard oleo-pneumatic (gaseous nitrogen and hydraulic fluid) shock struts are incorporated into each gear assembly. The shock struts consist of a movable stainless steel piston within a cylinder containing hydraulic fluid pressurized with gaseous nitrogen. O-ring seals maintain strut pressure and allow the movement of the steel piston. The struts absorb the shock of landing, and provide damping during taxi, takeoff and landing rollout. An air/oil filler valve is provided at the top of each strut for servicing. During normal operations, approximately three to five inches of inner cylinder chrome is exposed at the bottom of each main landing gear strut, depending on aircraft gross weight and outside air temperature. A placard attached to upper portion of the landing gear strut indicates the correct extension for ambient conditions.

C. Extension / Downlock Mechanisms

(1) Main Landing Gear Sidebrace Actuator / Downlock Mechanism

The sidebrace hydraulic piston and actuator arm extends the main landing gear outboard and down from the wheel well and locks the landing gear in the down position using internal locking keys located in the lower end of actuator cylinder. The keys slide into annular slots when the sidebrace actuator reaches full extension. A lock ram positioned by down hydraulic pressure and a spring maintain the keys in the slots preventing any further actuator movement. Spring pressure is necessary to maintain the keys in the locked position in the event of hydraulic pressure failure. The movement of the keys into the slots also provides the input for the downlock microswitch that provides the cockpit indication that the gear is down and locked. When the gear is selected up, hydraulic pressure is ported to the other side of the lock ram, displacing the keys from the slots and overriding spring pressure, allowing the actuator to move in the retract direction.

The sidebrace actuators have provisions for inserting ground lock pins to maintain the main landing gear in the extended position during ground operations. See the illustration in Figure 6.

(2) **Nose Landing Gear Extend / Retract Actuator**

Unlike the main landing gear sidebrace actuator that pushes the main gear into the extended position, the nose gear actuator pulls the nose gear aft and down to the extended position. One end of the actuator is attached to the aft structural wall of the nose wheel well. The other end is attached to the hinged nose gear trunnion that pivots aft into the extended position.

(3) **Nose Landing Gear Downlock**

Nose gear downlock protection is provided by the over center position of the hinged two part truss brace with integrated downlock actuator and spring. The forward and aft sections of the truss brace unfold as the nose gear extends. As the truss brace unfolds, it forces open the “C” shaped suitcase springs. When the nose gear is fully extended, a hydraulic downlock actuator and linkage pulls the truss brace slightly (approximately 0.050 inch) over center at the hinge point. The truss brace hinge is also held in the over center position by the energy of the “C” shaped suitcase springs that close slightly in the over center position. The springs are necessary to maintain the nose gear in the downlocked position in the event of hydraulic pressure failure. Once locked into the over center position, hydraulic pressure on the retract side of the downlock actuator is required to push the truss brace past the over center position and overcome spring pressure.

The truss brace hinge has a provision for inserting a down lock pin to secure the nose gear in the extended position. See the illustration in Figure 6.

(4) **Downlock Switches**

Microswitches installed on the main and nose landing gear provide downlock electrical signals for the landing gear position indicators on the cockpit control panel. On the main landing gear, when the internal locking keys move into the annular groves of the actuator in the fully extended position, a plunger with an integrated cam moves, depressing the contact of the downlock microswitch.

The downlock microswitch on the nose gear is located on the truss brace. As the truss brace unfolds to the over center position the contact on the nose gear microswitch is depressed sending a downlocked signal to the landing gear control panel.

D. Uplock Mechanisms

(1) **Main and Nose Landing Gear Uplock Mechanisms**

The uplock mechanisms of the main and nose landing gear operate in an identical manner, although the components are shaped somewhat differently due to the direction of movement of the nose and main landing gear in the extension/retraction cycle. Operation of the main gear uplocks is described as typical of all landing gear installations. The landing gear uplocks are shown in Figure 2 and Figure 4.

When the main landing gear is fully retracted into the wheel well, the gear is locked into position by a hook and latch that engage an uplock roller mounted on the front of the dual wheel axle housing. The uplock hook and latch mechanism is mounted on the interior forward side of the wheel well. The uplock hook is mechanically positioned by the uplock roller on the landing gear and the uplock latch is positioned by dual springs. (The uplock hook also has dual springs that will maintain the hook in the locked position in event of hydraulic failure.) The integrated latch assembly mechanically blocks movement of the uplock hook until the latch is moved aside by the gear uplock roller as the gear reaches the retracted position. When the latch is moved by the gear uplock roller from the position blocking movement of the uplock hook, the uplock roller enters the concave portion of the uplock hook as the hook pivots into a cradling position. Spring tension on the latch drops the latch into position behind the uplock roller, securing the uplock roller in the uplock hook. The action of the spring held latch counteracts any tendency for the uplock roller to rebound out of the uplock hook and retains the uplock roller in the grasp of the uplock hook after system hydraulic pressure is released.

CAUTION

WHEN THE LANDING GEAR IS EXTENDED, THE GEAR UPLOCK HOOKS ARE OPEN AND HYDRAULIC PRESSURE IS REMOVED FROM THE UPLOCK ACTUATORS. IT IS POSSIBLE TO MANUALLY PUSH UP THE UPLOCK LATCH AND ROTATE THE UPLOCK HOOKS TO THE LOCKED POSITION. SUBSEQUENT RETRACTION OF THE GEAR WILL RESULT IN THE LANDING GEAR DOORS CLOSING PREMATURELY, AND THE LANDING GEAR WILL IMPACT THE DOORS, FAILING TO RETRACT. ALL UPLOCKS SHOULD BE VERIFIED OPEN DURING PREFLIGHT WALK AROUND INSPECTION.

3. Limitations

A. Maximum Tire Speeds

Maximum tire speeds for installed configuration are shown in the following table:

Aircraft Serial Number	Configuration	Nose Tire	Main Tire	Overall Limit
1000-1213	Without ASC 190 / 266	182 kts / 210 mph	182 kts / 210 mph	182 kts / 210 mph
1000-1213	With ASC 266	182 kts / 210 mph	195 kts / 225 mph	182 kts / 210 mph
1000-1213	With ASC 190	195 kts / 225 mph	195 kts / 225 mph	195 kts / 225 mph
1214 and subsequent	Production	195 kts / 225 mph	195 kts / 225 mph	195 kts / 225 mph

B. Nose Strut Servicing

Recommended nose strut servicing is shown in the following table.

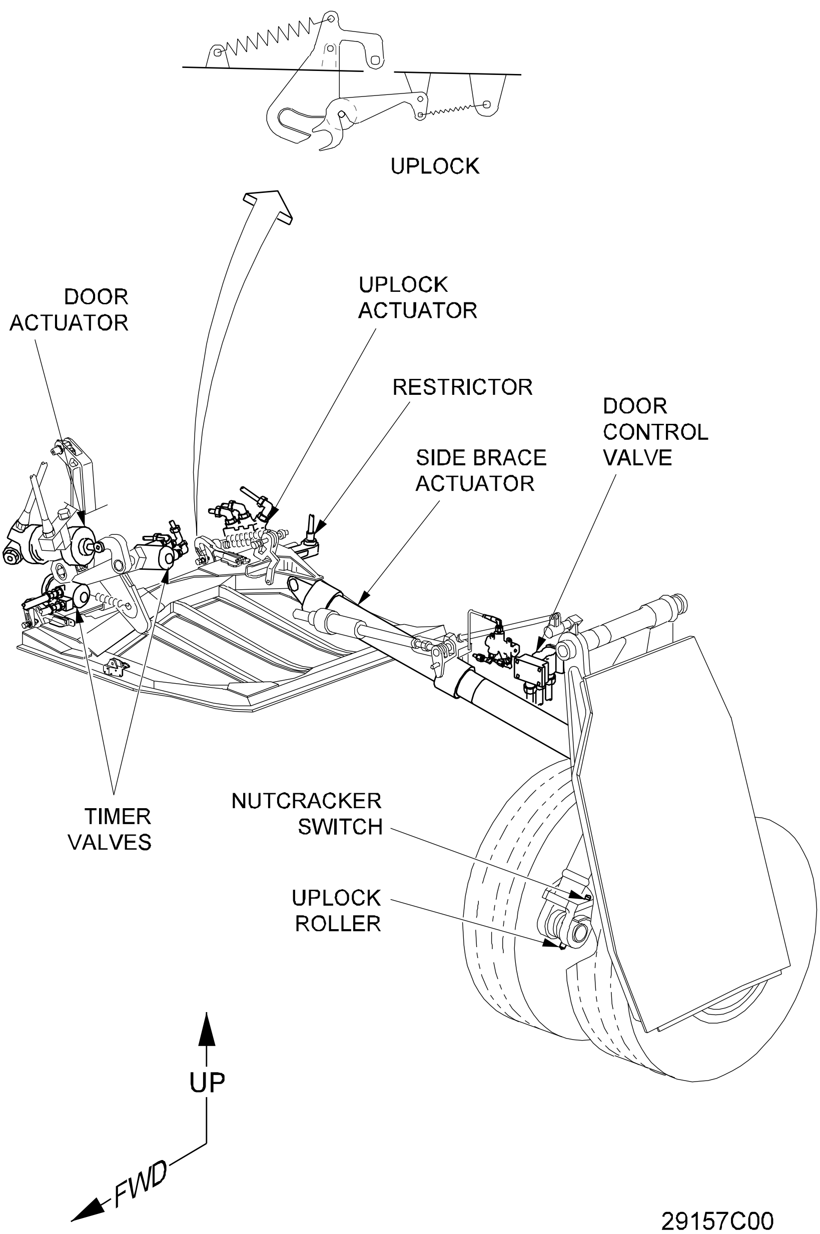
Service Temp:	Cold Day Service (-40° F / -40° C)	Std Day Service (70° F / 21° C)	Hot Day Service (110° F / 43° C)
Piston Extension (inches)	**Strut Pressure (psig)**		
13.04	188	224	246
11.24	220	264	289
9.46	260	312	341
8.34	300	365	399
7.34	350	419	457
6.54	400	475	518
5.84	450	537	586
5.30	500	596	651
4.94	550	644	703
4.54	600	706	770
4.24	650	761	830
3.94	700	825	900
3.64	750	901	982
3.44	800	959	1045
3.24	850	1025	1117
3.04	900	1100	1198
2.94	950	1142	1244
2.84	1000	1187	1293
2.74	1050	1236	1346
2.64	1100	1289	1403
2.44	1150	1408	1533
2.34	1200	1477	1607

NOTE:

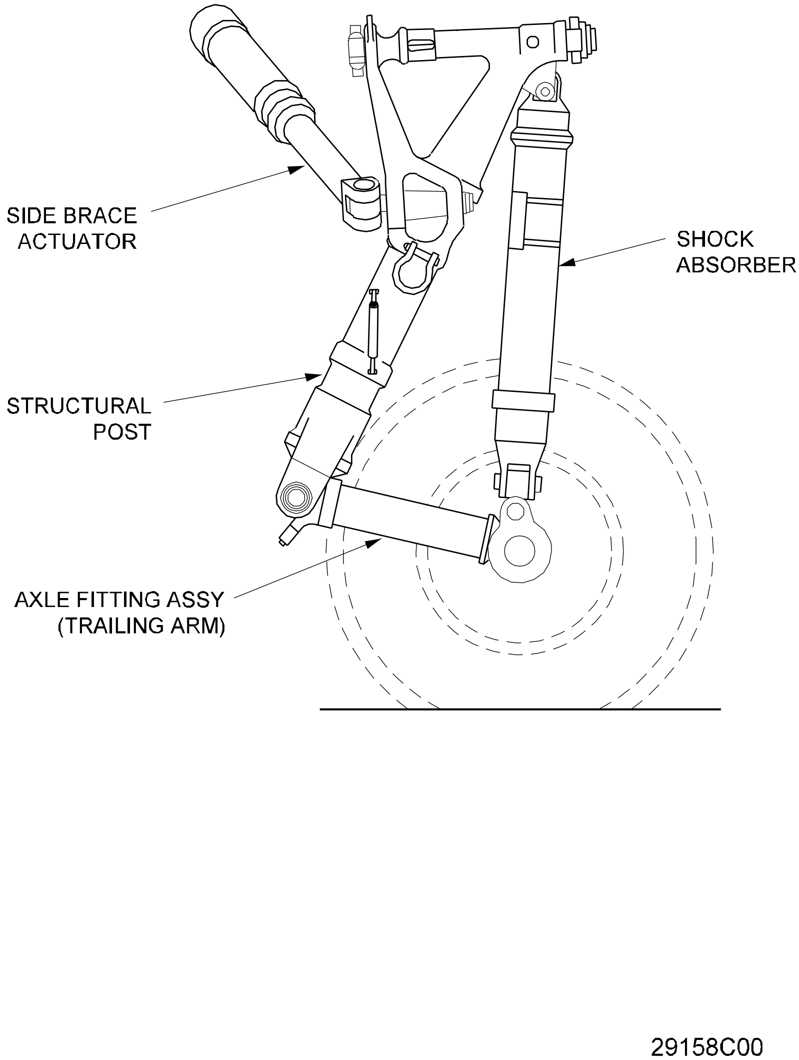
(1) Piston extension is measured from bottom surface of gland nut to black circumferential line on piston / axle.

(2) This table is used only for checking and adjusting strut service pressure and should not be used for full strut servicing.

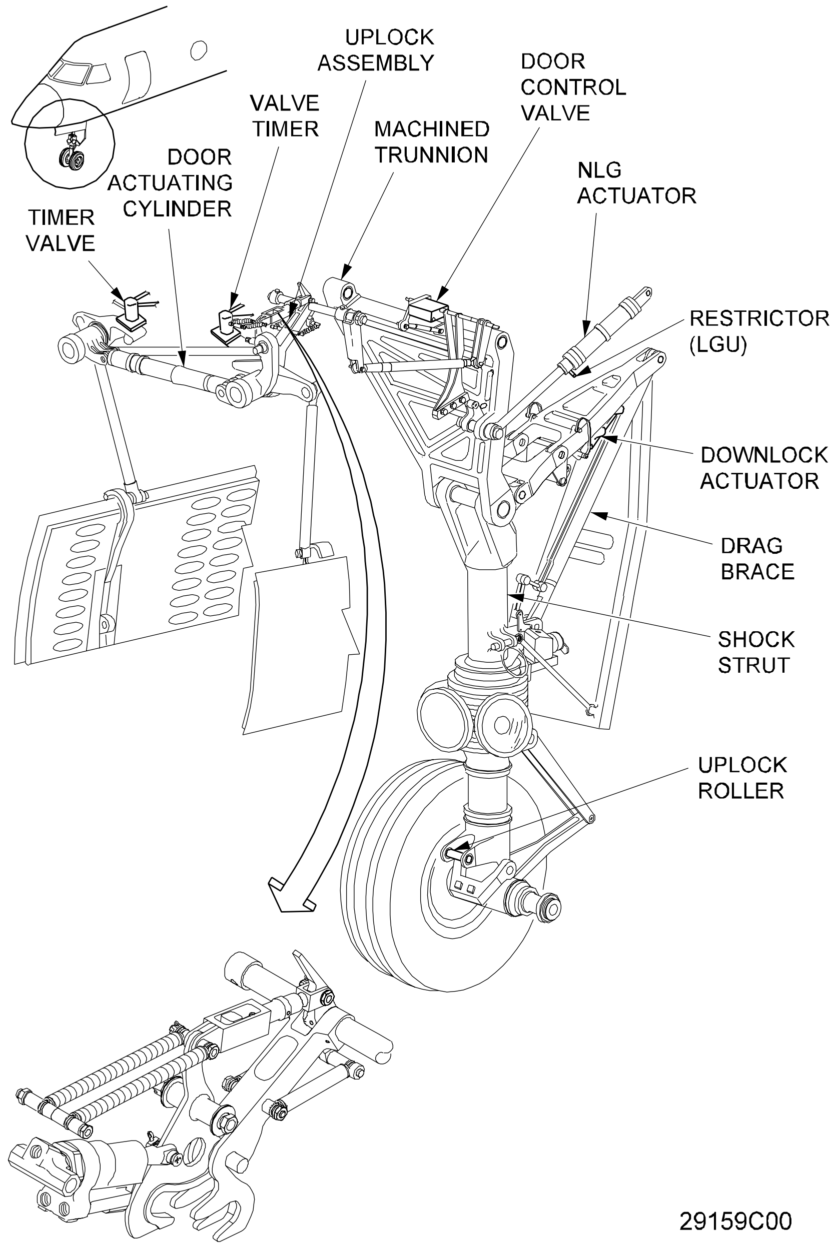
(3) Interpolate between temperatures for proper service pressure.



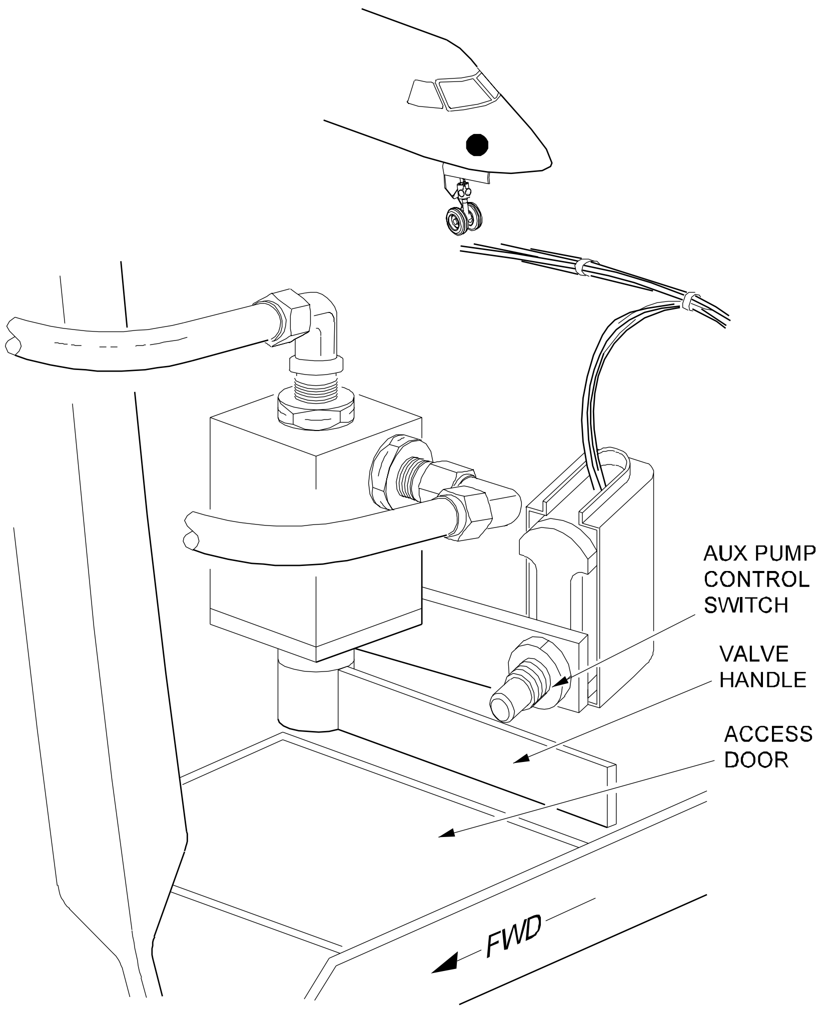
Main Landing Gear Components
Figure 2



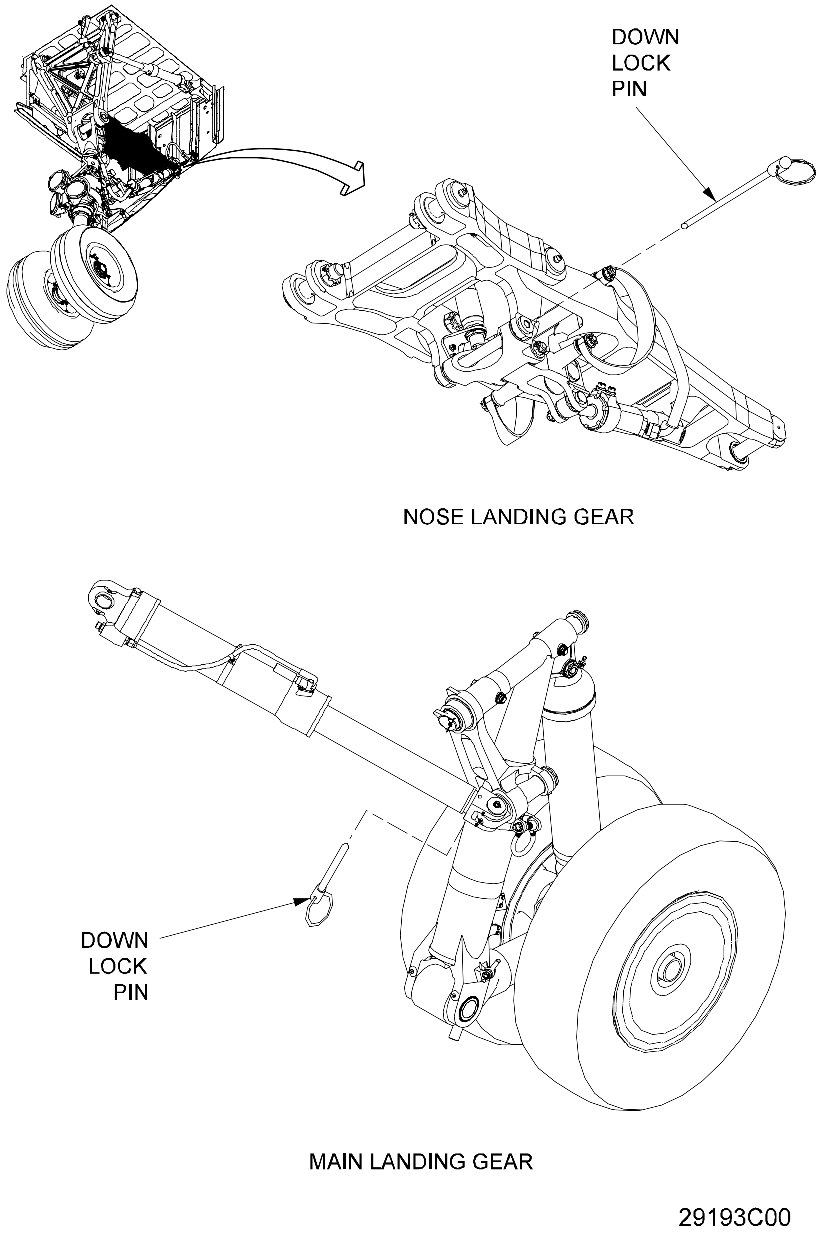
Main Landing Gear Structure
Figure 3



Nose Landing Gear Components
Figure 4



Ground Service Valve
Figure 5



Landing Gear Safety Pins
Figure 6

2A-32-30: Extension and Retraction System

1. **General Description**

Landing gear extension and retraction is controlled with the landing gear lever on the copilot side of the center instrument panel. The landing gear lever is connected to a mechanical linkage behind the instrument panel that extends to the landing gear selector valve mounted in the unpressurized nose section. The linkage is shown in Figure 7. Moving the landing gear handle mechanically opens and closes ports in the selector valve, routing Combined system hydraulic pressure to actuators for the main and nose gear and doors. If Combined system pressure is not available, but hydraulic lines and fluid quantity are intact, Utility system pressure will operate the landing gear using normal control linkages. If no hydraulic pressure is available, but system lines remain intact, the landing gear may be lowered by pressurizing the hydraulic lines with nitrogen gas stored in an emergency bottle located in the nose wheel well. The emergency extension system may only be used once and cannot retract the landing gear.

A nutcracker (squat) switch is installed on each landing gear. The nutcracker switches are closed when the aircraft is on the ground and the pneumatic-oleo struts are compressed with the aircraft weight. For this reason the switches are sometimes referred to as weight-on-wheels switches. The switches provide sensing information to aircraft systems / sub-systems that operate only in the air or only on the ground.

To facilitate maintenance on the landing gear or on components located in the wheel wells, the Auxiliary hydraulic system may be used to operate the landing gear system and/or the landing gear doors. A ground service valve, located inside a hinged panel near the nose wheel well, is positioned and held with safety locking pin to port Auxiliary system pressure to operate the doors only. If the aircraft is raised on jacks, the Auxiliary system may also be used to raise and lower the landing gear.

2. **Description of Subsystems, Units and Components**

   A. **Landing Gear Control Panel**

   The landing gear control panel, located on the copilot side of the center instrument panel, contains the gear lever, position and warning indicators and safety features. See Figure 8. Landing gear position and warning lights on the panel are powered by the Emergency DC bus and may be dimmed and tested with switches on the cockpit side consoles. The landing gear warning horn/klaxon is powered by the Essential DC bus.

   (1) Landing Gear Lever:

   The landing gear lever is selected into either the UP or DOWN detents on the control panel. To move the lever from one selection to the other, the lever must first be moved to the left to clear the detent. Lever movement is mechanically linked to the landing gear hydraulic selector valve in the nose compartment behind the radome. The selector valve routes Combined or Utility system pressure to the extend or retract sides of actuators for the landing gear and doors. Lever position has no effect on operation of the emergency landing gear extension system, but the position of the lever is integral to a correct landing gear position indication.

   The transparent wheel shaped handle of the landing gear lever has two clear bulbs for night illumination and a red bulb for annunciation of disagreement between gear lever position and the selected position of the landing gear (when gear selected down) or landing gear doors not closed (when gear selected up). With the gear lever up, the red light illuminates until all landing gear are up and the gear doors closed. With the gear lever down, the red light illuminates until all landing gear are down and locked. The warning light circuits are configured such that a ground is supplied to the red light, powering the light, whenever there is a difference between the gear lever selected position and proximity switches on the landing gear downlocks (DOWN) or proximity switches on the gear doors (UP). Agreement between selected and actual position removes the ground from the circuit.

(2) Ground Safety Lock and LOCK RELEASE Button:

On the ground with weight on the aircraft wheels (nutcracker switches on both main landing gear compressed), the landing gear lever is blocked from movement out of the DOWN detent by a solenoid actuated pin. The pin prevents inadvertent retraction of the landing gear due to unintentional movement of the landing gear lever. If the blocking solenoid or nutcracker switches malfunction during takeoff in the transition to the weight off wheels mode, preventing movement of the landing gear lever to the UP position, pressing the LOCK RELEASE button will manually move the blocking pin clear of the landing gear lever.

(3) Down and Locked Lights:

Three green light modules, one for each landing gear, are installed above the landing gear lever. Each light is an independent circuit, illuminating only when the associated gear downlock proximity switch contact is closed. The green lights do not annunciate landing gear door uplock engagement, and are also separate from the red warning light in the landing gear lever handle. Thus, if a landing gear does not fully extend and engage the downlock switch, a red light will illuminate in the gear lever handle and the green light associated with the malfunctioning landing gear will not illuminate, identifying the malfunctioning gear. However, if a landing gear does not fully retract (downlock not engaged, but gear door uplock not engaged) the red light in the landing gear lever handle will illuminate, but none of the green lights will illuminate.

(4) Gear Warning Horn and HORN SILENCE Switchlight:

If the landing gear are not down and locked during flight conditions associated with landing, a warning horn (aircraft with SPZ-8000 are equipped with a horn, aircraft with SPZ-8400 are equipped with a klaxon) will sound, alerting the crew that the landing gear is not in the correct position. In some conditions, the gear warning horn may be silenced by pressing the HORN SILENCE switchlight on the landing gear control panel to the left of the landing gear lever, or by pressing either of the HORN SILENCE buttons on the outside of the power levers. In other conditions, the warning horn can be silenced only by configuring the aircraft correctly for landing. If the HORN SILENCE switchlight or buttons have been pressed, the HORN SILENCE switchlight will illuminate.

The following conditions will sound the landing gear warning horn:

- The aircraft at or below 1,200 feet radio altitude and a power lever retarded to 70% or less and all landing gear not down and locked. The warning horn may be silenced in this condition. If the power lever(s) is subsequently advanced to more than 70%, the HORN SILENCE switchlight will extinguish and the horn warning circuit will rearm.
- The flaps selected to more than 22° and all landing gear not down and locked regardless of altitude. The warning horn cannot be silenced and will sound until the aircraft configuration is corrected.

B. Normal Retraction and Extension System

(1) Retraction Sequencing

(a) Landing gear lever is placed in UP position

(b) Red light in landing gear lever handle is illuminated

(c) Landing gear selector valve is moved to RETRACT position

(d) Hydraulic pressure is routed to OPEN side of landing gear door actuators

(e) Landing gear doors open

(f) The timer valves route hydraulic pressure to the RETRACT side of the main landing gear sidebrace actuators, unlocking the internal downlock keys, and to the nose gear actuator and release the nose gear downlock actuator

(g) The nose gear downlock actuator unlocks

(h) Three green DOWN AND LOCKED lights on landing gear control panel are extinguished

(i) Landing gear retracts and locks into uplock hooks

(j) Hydraulic pressure is routed to CLOSE side of landing gear door actuators

(k) Landing gear doors close

(l) Red light in landing gear lever handle is extinguished

(2) Extension Sequencing

(a) Landing gear lever is placed in DOWN position

(b) Red light in landing gear lever handle is illuminated

(c) Landing gear selector valve is moved to EXTEND position

(d) Hydraulic pressure is routed to OPEN side of landing gear door actuators

(e) Landing gear doors open

(f) Hydraulic pressure is routed to UNLOCK side of landing gear uplock actuators

(g) Uplock hooks unlock

(h) Hydraulic pressure is routed to EXTEND side of main landing gear sidebrace actuators and nose landing gear extend / retract actuator

(i) Landing gear extends down and downlocks engage

(j) Three green DOWN AND LOCKED lights on landing gear control panel are illuminated

(k) Red light in landing gear lever handle is extinguished

(l) Hydraulic pressure is routed to CLOSE side of landing gear door actuators

(m) Landing gear doors close

C. Emergency Extension System

The landing gear emergency extension system will operate only if the hydraulic lines for gear extension are intact. The system substitutes gas pressure for hydraulic pressure to move the actuators of the landing gear components. Emergency gear extension takes approximately six seconds to extend and lock the landing gear. Emergency extension nitrogen pressure will not close the landing gear doors after the landing gear are down and locked, the gear doors remain open. Once activated, emergency system pressure remains in the three actuators of each landing gear (uplock release, door open and gear extend) until the pressure is vented overboard when the emergency extension T-handle is stowed. Controls for the emergency extension system are shown in Figure 9.

(1) Units and Components

(a) Nitrogen Bottle

Pressurized nitrogen used for emergency landing gear extension is stored in a bottle located on the left side of the nose wheel well. A fully charged bottle contains 150 cubic inches of nitrogen pressurized to 3,000 psi at 70° F. The bottle has a pressure relief valve set at 3,750 psi to prevent damage to aircraft components should gas pressure exceed the structural limits of the bottle. The location of the emergency extension bottle is shown in Figure 10.

(b) Gear Emergency Extension Cable and T-Handle

The emergency extension T-handle is located on the forward end of the copilot side console and is labeled EMER LDG GEAR. The T-handle is connected by a flexible cable to the pressurized nitrogen bottle. Pulling the T-handle to full extension opens the bottle outlet valve, releasing compressed nitrogen into the landing gear hydraulic lines. If the emergency T-handle is subsequently returned to the fully stowed position, the nitrogen bottle vent port opens, and nitrogen pressure is discharged through an overboard vent.

(c) Dump Valves

Compressed nitrogen enters the hydraulic gear extension lines through dump valves integrated into the normal hydraulic system pressure lines. Nitrogen pressure activates the dump valves to close off the normal hydraulic system pressure lines and return hydraulic fluid present in the gear extend, gear retract and gear door lines and actuators to the system reservoir. This action prevents any hydraulic lock in the landing gear actuators. Nitrogen then pressurizes the unlock port of the three gear uplock actuators, the open port of the three door actuators and the extend port of the three landing gear actuators.

(d) Dump Valve Reset Handle

A reset handle (D-ring), shown in Figure 9, is located on the inside of the copilot side console and labeled EMERGENCY LANDING GEAR DUMP VALVE RESET. The D-ring is mechanically connected to the hydraulic system dump valves. Pulling the D-ring returns the dump valves to the normal hydraulic system setting. If emergency extension of the landing gear with the pressurized nitrogen bottle has been satisfactorily completed, the landing gear would normally be left extended until a landing could be made. If however, the landing gear must be retracted and hydraulic fluid and Combined or Utility system pressure is available, resetting the dump valves will allow retraction of the landing gear.

(2) Landing Gear Emergency Extension Sequence

(a) Landing gear lever is placed in DOWN position

(b) Red light in landing gear lever handle is illuminated

(c) EMER LDG GEAR T-handle is pulled to full limit of travel

(d) Compressed nitrogen is released to landing gear dump valves

(e) Dump valves isolate landing gear extend lines from remainder of hydraulic system and open returns for hydraulic fluid in actuators to hydraulic reservoirs

(f) Pressurized nitrogen is routed to OPEN side of landing gear door actuators, UNLOCK side of landing gear uplock actuators, EXTEND side of main landing gear sidebrace actuators and nose landing gear extend / retract actuator

(g) Landing gear doors open

(h) Uplock actuators unlock

(i) Landing gear extends down and locks

(j) Three green DOWN AND LOCKED lights on landing gear control panel illuminate

(k) Red light in landing gear lever handle extinguishes

(l) Landing gear doors remain open

D. Nutcracker System

The nutcracker (squat) switch system provides AIR or GROUND sensing to aircraft systems and components. A nutcracker switch is installed on each landing gear. The nutcracker switch contacts are depressed when the landing gear oleo-pneumatic struts are compressed by the weight of the aircraft on the ground. When the aircraft is in flight, the struts extend, releasing pressure on the nutcracker switches and opening the switch contacts. The nutcracker switch system is connected to seventeen relays. Circuits may be opened or closed corresponding to AIR or GROUND states of operation by wiring systems and components through the nutcracker switch relays.

The left and right main landing gear nutcracker switch relays are powered by the Essential DC bus. The nose gear nutcracker switch relay is powered by the Emergency DC bus. The main gear nutcracker switches are incorporated into a test circuit that verifies switch integrity. (The nose landing gear nutcracker switch is not tested.) The test switch is located on the center pedestal aft of the throttle quadrant. Each nutcracker switch has a dedicated circuit breaker. If a nutcracker switch fails in the ground (closed) position, pulling the respective circuit breaker will change the switch input to relays to the air position.

Nutcracker switch inputs from the main landing gear control the landing gear lever safety lock solenoid and the nose landing gear nutcracker switch weight-on-wheels signal is necessary for nosewheel steering. For a list of nutcracker system relay inputs to aircraft systems and components, see the following table.

Ground Mode (Weight on Wheels)	**Flight Mode (Weight off Wheels)**
Thrust reverser operation Engine ground idle Ground spoiler operation APU control Air flow control Cockpit clocks FMCS EDS Autopilot Cockpit voice recorder ILS Engine starting Speedbrake / Flap alarm Cabin pressure control EICAS DAU / FWC #1 EICAS DAU / FWC #2 G-meter Stall barrier	Thrust reverser REVERSE ALERT light Engine flight idle Angle of attack indication Gear lever downlock solenoid pin retract Pitot heat VHF nav Brake-by-wire Flight data recorder

NOTE:

Failure of the nutcracker system will result in inoperative and/or abnormal operation of aircraft systems, depending upon whether the nutcracker system fails in the ground or flight mode. See the following table for nutcracker failure effects.

In flight with Nutcracker System failed in ground mode	**On the ground with Nutcracker System failed in flight mode (Aircraft with ASC 166A)**	**On the ground with Nutcracker System failed in flight mode (Aircraft without ASC 166A)**
Landing gear lever cannot be positioned to up unless the solenoid lock release button is pressed	Red REV UNLOCK light may illuminate when thrust reversers are used. Thrust reversers will stow at low ground speed.	Thrust reversers inoperative

In flight with Nutcracker System failed in ground mode	On the ground with Nutcracker System failed in flight mode (Aircraft with ASC 166A)	On the ground with Nutcracker System failed in flight mode (Aircraft without ASC 166A)
Auto-pressurization will maintain 0.25 in differential, and cabin altitude will climb with aircraft. Manual pressurization is available.	Red ACFT CONFIG anunciation with speed brake deployment	Red ACFT CONFIG anunciation with speed brake deployment
Thrust reversers will operate in flight if power levers are retarded to idle.	Ground spoilers will deploy with wheel spin-up, but will stow at low ground speed.	Ground spoilers will deploy with wheel spin-up, but will stow at low ground speed.
If Ground Spoiler switch is armed, ground spoilers will deploy in flight if power levers are retarded to idle.	Anti-skid may not operate. All braking may be inoperative at low ground speed unless anti-skid is selected off.	Anti-skid may not operate. All braking may be inoperative at low ground speed unless anti-skid is selected off.
Stick shaker and stick pusher are inoperative	Engine idle set to 67% HP at low ground speed	Engine idle set to 67% HP at low ground speed
	APU bleed air inoperative	APU bleed air inoperative
	Pressurization outflow valve will not automatically open, valve must be manually opened.	Pressurization outflow valve will not automatically open, valve must be manually opened.
	Stick shaker and stick pusher remain armed.	Stick shaker and stick pusher remain armed.
	Landing gear lever handle not protected by lock release solenoid and gear may be retracted.	Landing gear lever handle not protected by lock release solenoid and gear may be retracted.
	Engines may not be restarted after shutdown	Engines may not be restarted after shutdown

3. Controls and Indications

A. Circuit Breakers (CBs)

Circuit Breaker Name	CB Panel	Location	Power Source
LDG GEAR POS IND	P	E-5 (1)	Emergency DC Bus
LDG WARN HORN	P	E-4	Essential DC Bus
L NUTCRACKER	CPO	A-13	Essential DC Bus
NUTCRACKER	CPO	B-13	Essential DC Bus
R NUTCRACKER	CPO	C-13	Essential DC Bus
NUTCRACKER BAT PWR	CPO	D-13	Battery Bus (2)

NOTE(S):

(1) S/N 1000-1279; B-5 on S/N 1280 and subsequent.

(2) Aircraft with ASC 242.

B. Warning (Red) Messages and Annunciations

CAS Message	SWLP Indication	Cause or Meaning
ACFT CONFIGURATION	ACFT CONFIG	Landing Gear not down and locked with radar altitude ≤ 1200 ft and power lever < 70%, or Landing Gear not down and locked with Flaps > 22° at any altitude

NOTE:

See 2A-27-60, Flaps System for a description of the Configuration Warning System

C. Other Warning Annunciations

Annunciation	Cause or Meaning
Warning horn (SPZ-8000 equipped aircraft) or Klaxon™ hi-lo, hi-lo (SPZ-8400 equipped aircraft) sounds	Landing gear unsafe

D. Nutcracker Switch Test

The Nutcracker Switch Test button on the center pedestal is used to verify that the main landing gear nutcracker switches are in the air (weight-off-wheels) mode during flight (the landing gear retracted). Pushing the button should result in a green indication for the L (left) and R (right) strut nutcracker switches on the main landing gear, indicating the air mode. See the illustration in Figure 11. The switch should not be activated on the ground, since this would place the nutcracker switches in the air mode, interrupting the circuits to aircraft systems that are dependent upon ground (weight-on-wheels) mode for proper operation (this action would retract the solenoid pin from the landing gear lever, allowing the handle to be placed in the up position).

E. Brake Nutcracker Override Switch (Brake-by-Wire System Only)

During flight with the nutcracker switches in the air (weight-off-wheels) mode, hydraulic pressure is removed from the main landing gear brakes. Brake operation and correct brake pressure application may be tested in flight by depressing the Brake Nutcracker Override Switch on the center pedestal, shown in Figure 18. With the switch depressed, brake pressure may be applied with the landing gear retracted. With the EICAS selected to the hydraulics system page (SPZ-8000) or the brakes system page (SPZ-8400) depressing either set of brake pedals will result in the applied brake pressure indicated on the brake system page, allowing verification of brake operation prior to landing.

4. Limitations

A. Landing Gear Extended Speed (V_{LE} / M_{LE})

Do not exceed 250 KCAS / 0.70 MT with landing gear extended (gear doors open or closed).

B. Landing Gear Operation Speeds (V_{LO} / M_{LO})

(1) Normal Operation:

Do not lower or raise landing gear at speeds in excess of 225 KCAS

/ 0.70 MT.

(2) Alternate Operation:

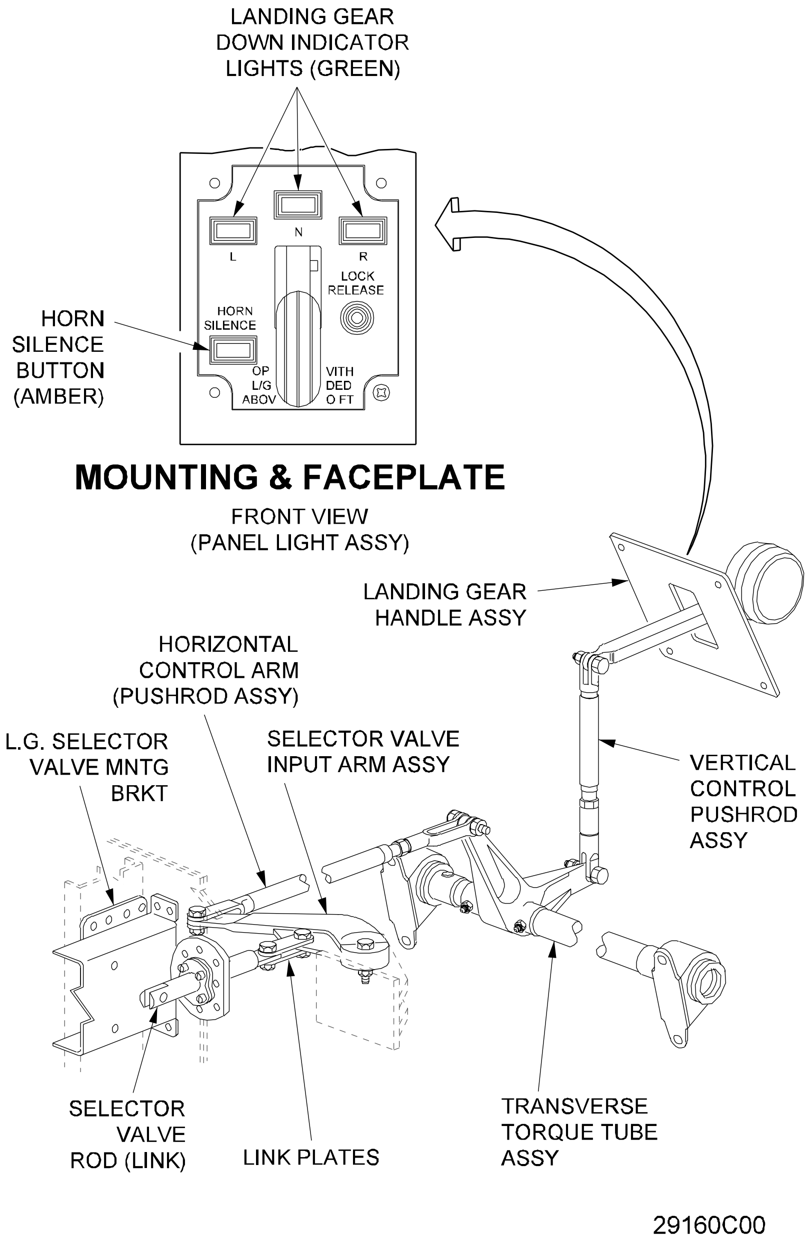
Do not lower landing gear using alternate system at speeds in excess of 175 KCAS.

C. Landing Gear Extension / Operation Altitudes

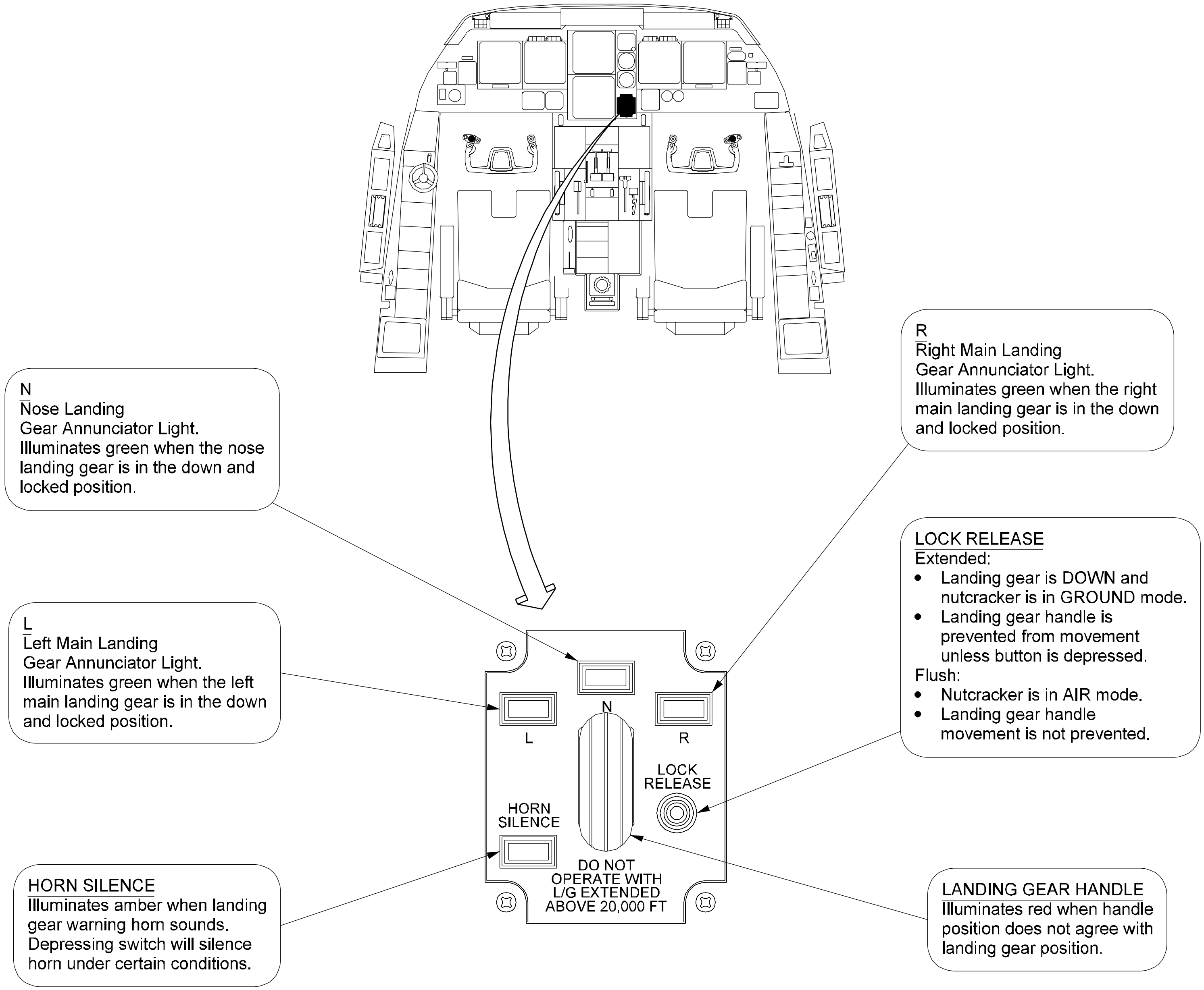
Maximum operating altitude for extending landing gear or flying with landing gear extended is 20,000 ft. MSL.

D. Speed Brakes Extension With Landing Gear Extended

During flight, speed brakes are not to be extended with flaps set at 39°, or with the landing gear extended.

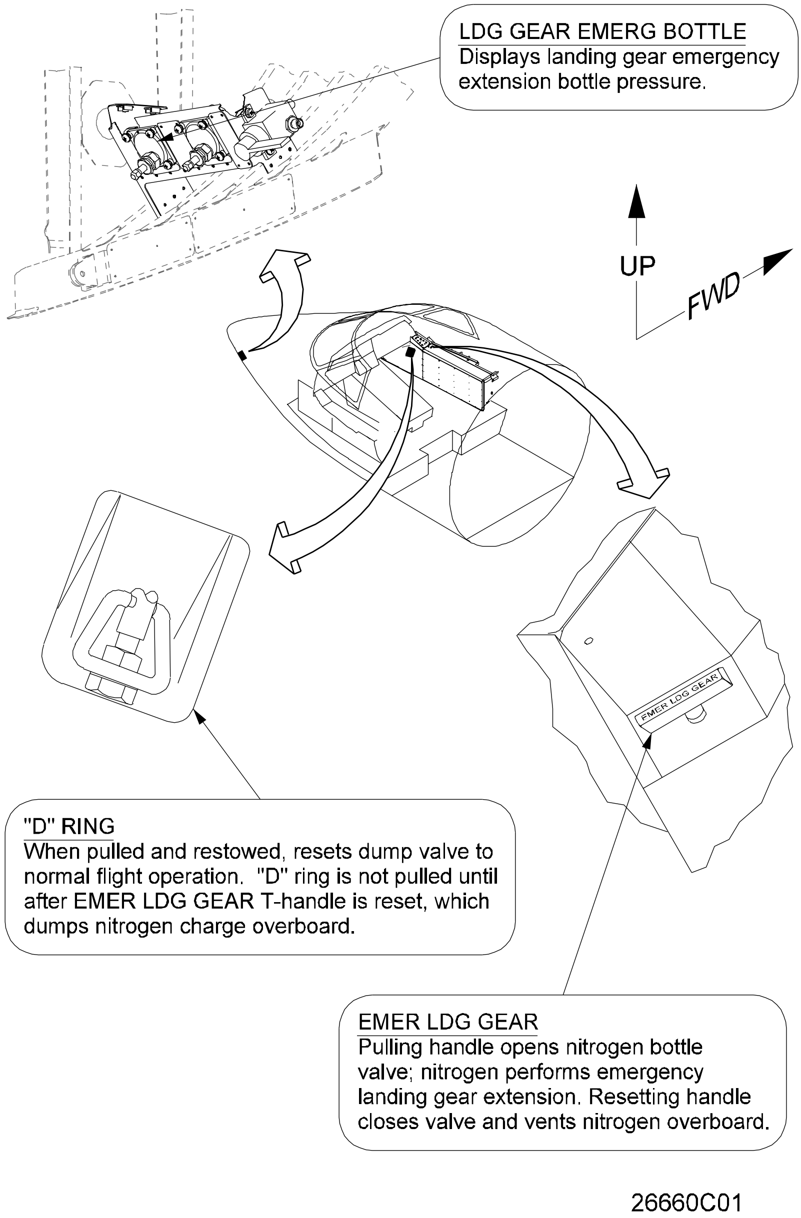


Landing Gear Lever Linkage
Figure 7

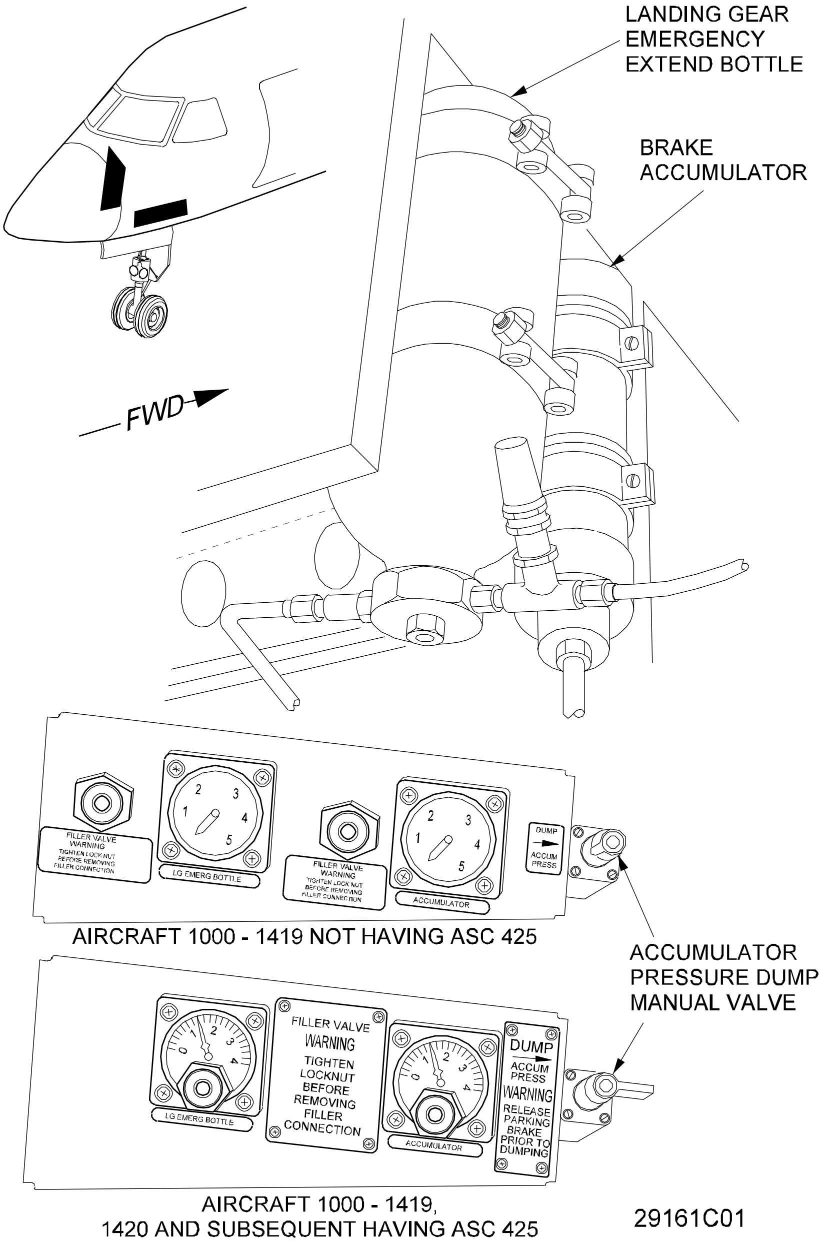


26659C00

Landing Gear Control
Panel
Figure 8



Emergency Gear Extension Controls
Figure 9



Emergency Extension Bottle
Figure 10

NUTCRKR SW TEST pushbutton

Momentary action pushbutton, L and R halves of pushbutton will illuminate with amber background when pressed if Left and Right MLG nutcracker switches are in AIR (weight-off-wheels) mode. NOTE: Pressing pushbutton while on ground (weight-on-wheels) will place MLG nutcracker switches in AIR mode. Operation of some aircraft systems will be altered:

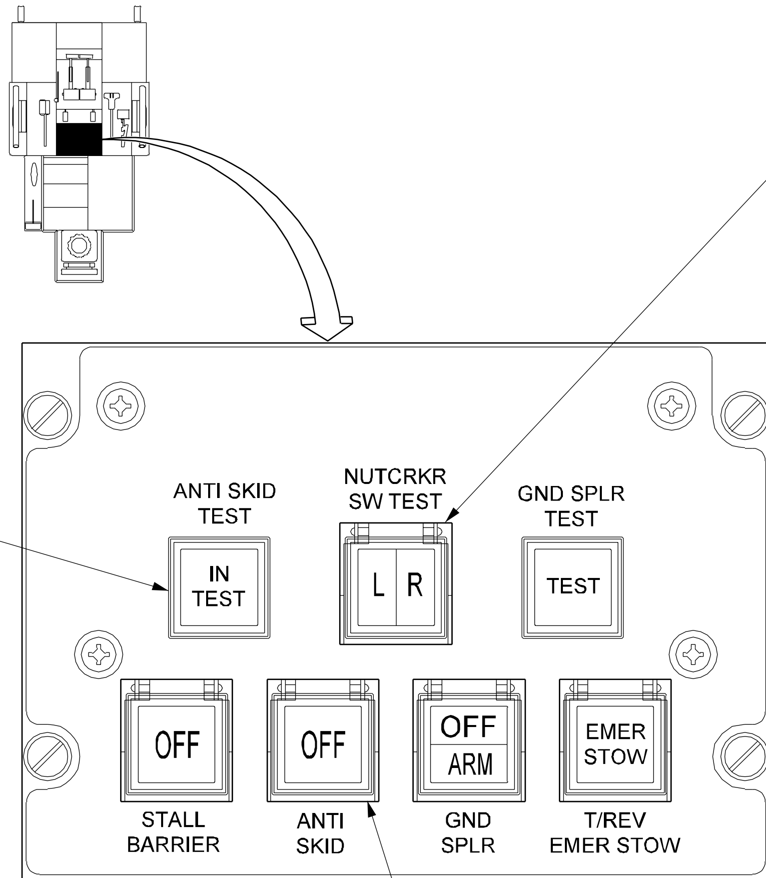
- Nose wheel steering inoperative
- Thrust reversers inoperative
- Ground spoilers inoperative
- Landing gear lever solenoid will retract and landing gear handle may be moved to retract position

ANTI SKID TEST pushbutton

Pushing momentary action ANTI SKID TEST pushbutton initiates BIT test of anti-skid system. BIT test should only be initiated when aircraft speed is less than 12 knots. During the BIT test the following indications are displayed on EICAS:

- Green IN TEST light in pushbutton illuminated
- ANTI SKID FAIL (amber)
- BRAKE FAIL (amber)
- BRAKE MAINT REQ'D (blue)
- SYSTEM TEST light (amber) on front of Skid Control Box is illuminated

If system fault is detected, system will remain in BIT test mode, and all lights / annunciations will remain on. Faults are isolated using FAULT ISOLATION switch on front of Skid Control Box.



ANTI SKID pushbutton

Alternate action pushbutton selects Anti-skid system OFF (amber) or ON

29199C00

Nutcracker Switch Test
Pushbutton
Figure 11

2A-32-40: Wheels and Brakes System

1. General Description

The GIV aircraft is equipped with one of two types of main landing gear wheels/ brakes and tires, and one of two types of braking systems, depending upon serial number (S/N) and/or Aircraft Service Change (ASC) retrofitted into the aircraft.

Aircraft S/N 1000 through S/N 1213 (with the exception of S/N 1183) are equipped with main landing gear wheels and brakes manufactured by Aircraft Braking Systems (ABS) and tires manufactured by Goodyear, speed rated at 182 knots. Beginning with S/N 1214, aircraft were equipped with main landing gear wheels / brakes manufactured by Dunlop, and Goodyear tires. The later model GIV wheels/brakes and tires are speed rated at 195 knots, and thus support increased speeds associated with higher gross weight takeoffs. Any aircraft not equipped with Dunlop wheels/brakes may optionally have them retrofitted with the installation of ASC 266.

Aircraft S/N 1000 through S/N 1213 (with the exception of S/N 1183) are equipped with an electronic braking system that transmits signals proportional to cockpit brake pedal movement to a Brake Electronic Control Unit that then applies hydraulic pressure to the brakes in response to the variable electric signal. This type of braking system is commonly referred to as Brake-by-Wire. Beginning with S/N 1214, aircraft are equipped with a Hydro-Mechanical Analog Braking system (HMAB). In the HMAB system, the cockpit brake pedals are mechanically connected to valves that apply hydraulic pressure to the brakes in response to brake pedal displacement. Aircraft produced prior to S/N 1214 may have the HMAB system retrofitted with the installation of ASC 307. Both braking systems incorporate full anti-skid protection. Each of the primary braking systems is supplemented with an independent emergency/parking brake system that uses stored accumulator pressure for brake application.

2. Nose Landing Gear Wheels And Tires

The nose landing gear has dual wheels. Each wheel is 22 x 6.6 inches and made of two forged aluminum halves. The two wheel halves are mated together by eight bolts, with the inner joint fitted with an O-ring seal, forming an airtight structure for mounting tubeless tires. The tires are 21 x 7.25 -10 rated at 182 knots for aircraft S/N 1000-1213 (except S/N 1183) without ASC 190 installed. For aircraft 1214 and subsequent, and S/N 1000-1213 with ASC 190, the nose wheel tire speed rating is 195 knots. For more information concerning nose wheel tire and wheel selection, see GIV Illustrated Parts Catalog section 32.

3. Main Landing Gear Wheels and Tires

Each main landing gear has dual wheels. The wheels are forged aluminum with a removable flange to facilitate tire servicing. The flange is mated to the wheel with an O-ring that provides and airtight seal for the tubeless tires. Each wheel is mounted to the landing gear axle with two tapered roller bearings. A brake assembly is integrated into the wheel, fitting into the space between the bearing housing and inside wheel rim.

S/Ns 1000 through 1213 (except 1183) have wheels and brakes manufactured by ABS, and Goodyear 34 x 9.25-16, 18 ply rating, type VII tires, speed rated at 182 knots. Aircraft in this production sequence may be retrofitted with ASC 190 that increases tire speed rating to 195 knots, and / or with ASC 266 that installs Dunlop wheels, brakes and tires that provide increased braking capability. On aircraft S/N 1214 and subsequent, Dunlop wheels, brakes and tires are production installed, supporting higher aircraft gross weights and consequent higher tire speeds. The

Dunlop wheels have a greater radius between the bearing housing and the inside of the wheel hub to accommodate the larger braking package. This requires a lower profile tire, measuring 34 x 9.25-18, in order for the tire and wheel combination to fit into the wheel wells.

ABS wheels have three fusible plugs that melt at 430°F, releasing tire pressure if the wheel overheats. Dunlop wheels have four fusible plugs that melt at 390°F and a wheel safety plug that will deflate the tire if pressure reaches 375 - 650 psi.

4. **Brake Assemblies**

Brakes manufactured by ABS have four rotating discs (rotors) three stationary discs (stators) an end plate and a pressure plate. See Figure 12 for an illustration of an ABS brake. All of these elements are composed of carbon-metallic alloy, and are referred to as a whole as the disc stack or heat pack. The stators are attached to the torque tube that in turn is bolted to the aircraft gear assembly and holds the wheel bearings. The rotors are attached to the wheel, with notches in the rotors fitting keys on the interior of the wheel hub. The wheel with attached rotors turns on the wheel bearing, with the rotors spinning between the brake stators. Brake bolts attached at the brake housing and fastened to the outside of the end plate maintain sufficient clearance between the rotors and the stators to permit the wheels to turn freely. Five hydraulic actuating pistons are built in the brake housing. When the brakes are applied, hydraulic pressure is applied to the pistons that then move outward from the housing squeezing the pressure plate, and reducing the clearance between the rotors and stators. The surfaces of the rotors and stators are pressed against each other, producing the friction that slows the spinning wheel.

Dunlop brakes have three rotors, two inner stators of double thickness, and a stator of single thickness at the pressure plate and the end plate. The components of a Dunlop brake are shown in Figure 13. Operation of the Dunlop brakes is similar to that of the ABS brakes, but because of the greater energy absorbing mass afforded by the increased diameter of the wheel and resulting larger surface area of the rotors and stators, a higher breaking efficiency is attained.

Both brake types have provisions for brake temperature monitoring, anti-skid protection, and application of hydraulic pressure at a reduced level (400 ± 50 psi) to stop wheel spin as the landing gear are retracted into the wheel well.

2A-32-41: Hydro-Mechanical Analog Braking (HMAB) System

1. **General Description**

The Hydro-Mechanical Analog Braking (HMAB) system is installed on aircraft S/N 1214 and subsequent, and is available for retrofit on aircraft S/N 1000 through 1213 with the installation of ASC 307. The HMAB system applies up to 3000 psi hydraulic pressure from the Combined system to the main landing gear brakes in response to cockpit brake pedal commands. The left and right pilot and copilot brake pedals are mechanically linked to the left and right brake metering valves in the nose wheel well. The metering valves open in response to brake pedal deflection, porting increased pressure with greater brake pedal application. From the brake metering valves, pressurized hydraulic fluid is routed to the augmenter valve in the main wheel well. The augmenter valve increases hydraulic fluid flow rates and intensifies hydraulic pressure to compensate for any system response loss caused by the distance between the brake metering valves in the nose wheel well and the main landing gear. From the augmenter valve, hydraulic pressure then passes through the anti-skid control valve modules, hydraulic line fuses, pressure transducers and brake shuttle valves and each brake piston assembly.

Hydraulic fuses detect a broken or damaged hydraulic line indicated by increased fluid flow and close to prevent system fluid loss. The anti-skid control valve modules regulate hydraulic pressure to the brake assemblies to maintain wheel rotation during braking. Pressure transducers electronically transmit readings of hydraulic pressure applied to the brakes to the Data Acquisition Units (DAUs). The DAUs formulate this information for display on the cockpit Engine Instrument/ Crew Alerting System (EICAS). See Figure 14 for a schematic of the HMAB system.

The left and right brake metering valves are plumbed to both the Combined and Auxiliary hydraulic systems. If Combined system pressure drops below 1500 psi, internal stops in the metering valves shift, allowing Auxiliary system pressure to operate the brakes with a slight increase in pedal effort required. The Auxiliary hydraulic pump will pressurize automatically (if armed) when the loss of Combined system pressure requires increased brake pedal travel and subsequent engagement of the pedal limit switches. The limit switches in combination with the Combined system low pressure switch complete the circuit to power the Auxiliary pump. When the Auxiliary system is in use, the augmenter valve is bypassed, and pressure is routed through alternate anti-skid modules to the emergency brake shuttles and hydraulic fuses, and then to the brakes.

If Combined or Auxiliary hydraulic pressure are not available, a parking / emergency brake system with stored accumulator pressure may be used to stop the aircraft. The brake accumulator is in the nose wheel well, and has a nitrogen gas pre-charge of 1200 psi. The accumulator is shown in Figure 10. When normal hydraulic pressure is initially applied to the aircraft, the accumulator is fully pressurized with hydraulic fluid at 3000 psi. Accumulator pressure may be read at the accumulator in the nose wheel well, or on the indicator on the copilot lower instrument panel. The accumulator, when fully charged contains sufficient pressurized hydraulic fluid for approximately 5 to 6 brake applications. The PARK / EMER BRAKE handle on the cockpit center console is connect by cable to the modulating valve on the right side of the nose wheel well. Brake modulation can be accomplished by pulling out the handle between ¼ and ½ inch. Handle travel beyond approximately ½ inch results in the application of full accumulator pressure. No anti-skid is available with emergency braking. To apply accumulator pressure for parking the aircraft, the PARK / EMER BRAKE handle is pulled up and then rotated clockwise. To release the PARK / EMER BRAKE, pull the handle up and rotate counterclockwise.

An anti-rotation valve is incorporated in the braking system to stop wheel spin upon landing gear retraction. When the landing gear is down and locked, the anti-rotation valve is closed, but as the gear retracts, pressure from the main landing gear timer valve is routed to the anti-rotation valve where it is reduced to 400±50 psi, and applied to the wheel brakes through the parking / emergency brake lines.

2A-32-42: Hydro-Mechanical Analog Braking (HMAB) Anti-skid Protection

1. General Description

NOTE:

The following system description applies to aircraft S/N 1214 and subsequent, S/N 1000 - 1213 with ASC 307 installed, and S/N 1183.

Hydro-mechanical Analog Braking (HMAB) anti-skid protection includes a primary system and an alternate system. The primary system provides anti-skid protection during normal operations using the Combined hydraulic system. The alternate system provides anti-skid protection during a hydraulic malfunction of the Combined system, and uses the Auxiliary hydraulic system for braking.

Anti-skid protection for the main landing gear brakes operates by regulating hydraulic pressure applied to the brakes to maintain wheel rotation. HMAB anti-skid is effective at taxi speeds above ten (10) kts. (Below 10 knots, the main wheel brakes may be locked by brake pedal commands in order to turn the aircraft within a minimum radius.) Hydraulic pressure is metered by the anti-skid control valve modules in response to commands from the skid control box or Electronic Control Unit (ECU). The ECU, located in the left avionics equipment rack aft of the cockpit, monitors wheel speeds using data from the Wheel Speed Sensors (WSSs). Each main landing gear wheel is equipped with a WSS mounted on the wheel axle that is spun by the rotation of the wheel. The WSSs act as generators, producing an alternating current (AC) signal at a frequency proportional to wheel revolution. The ECU compares the frequency of the AC signals produced by the rotation of the wheels to detect a skidding wheel, indicated when one wheel is not rotating as fast as others. If the ECU detects a rotational speed difference on one of the main landing gear wheels during braking, it signals the anti-skid control valve module to reduce hydraulic pressure to the brake on the slower wheel and to the corresponding wheel on the opposite strut. The ECU controls the hydraulic pressure to wheel brakes in pairs - inboard wheels and outboard wheels. By regulating the same hydraulic pressure to symmetrically paired wheel brakes, aircraft directional control is easier to maintain.

The ECU has three circuit boards for monitoring wheel speeds and controlling brake application. Two boards are used during primary anti-skid system operation: one board for the inboard wheels on each strut, and one board for the outboard wheels on each strut. The third circuit board is used during alternate anti-skid operation and controls the application of Auxiliary system hydraulic pressure to both the inboard and outboard wheel brakes on each strut. Each circuit board is powered separately for redundancy. The board for the outboard wheels is powered by the Emergency Battery 2B Bus through circuit breaker OUTBD ANTI SKID, located at position B-2 on the Copilot Overhead (CPO). The inboard wheel circuit board is powered by the Emergency Battery 1A Bus through circuit breaker INBD ANTI SKID located at position B-1 on the CPO. The circuit board for the alternate anti-skid system is powered by Emergency Battery 2B Bus through the WHEEL SPEED circuit breaker at location C-10 on the CPO.

The ECU circuit boards communicate with three anti-skid control valve modules. There are two modules controlled by the primary anti-skid system circuit boards, one for each main landing gear strut. The third module is controlled by the alternate anti-skid board, and is linked to all four main landing gear wheels. The alternate module and corresponding circuit board are always electrically powered, even if the primary system components are operating the anti-skid system.

During primary system operation, the circuit board for the outboard wheels compares and controls outboard wheel speed though the WSSs and the left and right strut anti-skid control valve modules. The inboard wheels of each strut are monitored and controlled in the same way. During alternate system operation with Auxiliary hydraulic system pressure, a single circuit board and anti-skid control valve module controls all four wheel brakes. The alternate anti-skid system is not directly linked to the WSSs, it relies upon the inboard and outboard ECU circuit boards for wheel speed monitoring and comparison. Unlike the primary system that controls symmetrically paired wheel brakes, the alternate system controls each strut as a unit. If, for instance, a decrease in wheel speed is detected in the left inboard wheel, the alternate system reduces hydraulic pressure application to both the inboard and outboard wheel brakes of the left strut.

Additional protective features are provided by the anti-skid system. To avoid flat-spotted and/or blown tires, touchdown protection prevents hydraulic pressure from reaching the wheel brakes until the nutcracker (squat) switch is in the ground mode (weight-on-wheels) or the wheels have spun up to twenty-two (22) knots. When the wheels are lowered prior to landing, the nutcracker switch is in the flight (weight-off-wheels) mode and the wheels are not rotating. In this condition, with the anti-skid system on, hydraulic pressure to the brakes is blocked, since the lack of wheel rotation signals a locked wheel. On touchdown, compression of the nutcracker switch to the ground (weight-on-wheels) mode starts a seven second delay during which hydraulic pressure remains blocked to the wheel brakes unless the wheels spin up and attain a speed equivalent of 22 knots. The time delay provides protection for the wheels/brakes in the event of a bounce upon landing that would compress the nutcracker switch and allow application the brakes prior to the subsequent touchdown. Normal hydraulic pressure is available at wheel speeds above 22 knots regardless of nutcracker switch position. Should a nutcracker switch fail in the flight, or weight-off-wheels mode, normal braking with full anti-skid protection is available until the aircraft slows to below 22 knots. Below wheel speeds of 22 knots, the logic of the touchdown protection circuitry reverts to nutcracker switch position. The brakes on the wheels of the strut with the normally operating nutcracker switch will operate with full anti-skid (since the seven second delay has expired), but the brakes on the wheels of the strut with the inoperable nutcracker switch will be without hydraulic pressure, since the nutcracker switch did not transition to the ground (weight-on-wheels) mode to start the timing of the seven second delay. This hazardous condition of full anti-skid braking present on one strut and no braking available on the other strut is avoided by selecting anti-skid off prior to slowing the aircraft below 22 knots.

Similar protection acts as a backup to normal anti-skid protection to avoid locked wheels on the ground during roll out. If a rotational speed difference of approximately thirty-three percent (33%), (as measured between inboard wheels or between outboard wheels) is detected on one wheel, hydraulic pressure is completely removed from the slow wheel brake. This feature is useful in circumstances of severe hydroplaning or icing conditions where wheel rotation is minimal or nonexistent. Removing all hydraulic brake pressure to the locked or non-rotating wheel prevents damage / blow outs when the aircraft exits conditions of very low friction. This feature differs from touchdown wheel protection in that brake pressure is removed from a single wheel, rather than both symmetrical wheels.

In addition to the wheel speeds supplied by the WSSs to the anti-skid system, wheel rotation data is supplied to the automatic spoiler system to operate ground spoilers if there is a failure in the nutcracker switch system. When automatic ground spoilers are armed, they normally deploy when the nutcracker switch on each main landing gear is in the ground, or weight-on-wheels mode. If a nutcracker switch fails, wheel speed can initiate spoiler deployment. If at least one WSS on each strut supplies a signal equivalent to a speed of approximately forty-eight (48) knots, the ECU will supply a signal to the ground spoiler system to deploy. However, as the aircraft slows, the strength of the signal supplied by the WSSs will degrade, and when wheel rotation has slowed to approximately thirty-nine (39) to twenty-three (23) knots, the ground spoilers will retract.

2A-32-43: Electronic Braking System (Brake-by-Wire)

1. General Description

The Electronic or Brake-by-Wire Braking System uses electronic signals in place of mechanical linkages to transmit cockpit brake pedal commands to the main wheel brakes. All wiring is dual channel, providing redundancy for system components. DC power to channel 1 is provided by the forward emergency battery through circuit breaker BCS CHL #1, located on the CPO at position A-2 , and channel 2 is powered from the DC Essential Bus through circuit breaker BCS CHL #2, located on the CPO at position B-2. Anti-skid protection for braking is fully integrated into the system using information from wheel speed sensors (WSSs). The wheel speed sensor circuits are powered by the Left Main DC Bus through the WHEEL SPEED circuit breaker, located on the CPO at position C-10.

Pilot and copilot left and right brake pedals are equipped with linear variable differential transducers (LVDTs). As a brake pedal is depressed, an electrical signal proportionate to pedal displacement is generated. Brake pedal feel is provided by bungees incorporated into the pedal structure. The electrical signals from the LVDTs are sent to the Electronic Control Unit (ECU). If both pilots depress their respective brake pedals, only the signal corresponding to the greatest pressure is sent (the electric signals are not additive).

The ECU is the primary interface between system components. Circuit boards in the ECU control Combined hydraulic system braking and anti-skid functions, Auxiliary hydraulic system braking and pulsed anti-skid braking. One circuit board is dedicated to the Built-in Test Equipment (BITE). The ECU receives signals from the LVDTs, WSSs, and commands from cockpit switch panel selections (Anti-skid ON / OFF, Brake Test, and Brake Nutcracker Override). In response to received data, the ECU is programmed to apply braking in response to LVDT commands, reduce brake application if a wheel skid is detected, or provide a fault detected signal to the DAUs for annunciation on EICAS displays. Other ECU functions include wheel speed data for control of automatic ground spoiler deployment, prevention of a locked wheel on landing, and wheel spindown on landing gear retraction.

The ECU controls the main wheel brakes with inputs to the Hydraulic Brake Control Module (HBCM). The HBCM is connected to pressure lines from the Combined hydraulic system and the Auxiliary hydraulic system. A simplified diagram of the hydraulic components of the Brake-by-Wire system is shown in Figure 15. During normal operations, Combined system pressure displaces a shuttle valve in the HBCM and is routed to right and left brake control valves that are electronically operated by the ECU. The ECU signals the valves to open proportional to LVDT inputs. Combined system pressure is then directed through brake lines incorporating hydraulic fuses and pressure transducers to the shuttle valves at the wheel brakes. Indication of hydraulic pressure applied to the brakes is available on the cockpit EICAS displays using data transmitted from the pressure transducers to the DAUs. If there is a failure of the Combined hydraulic system, the Auxiliary hydraulic system will provide wheel braking. A pressure switch in the HBCM monitors Combined hydraulic system pressure. If Combined system pressure falls, the HBCM will automatically switch to the Auxiliary system when all of the following conditions are met: (with AUX Pump armed) Combined system pressure drops below 1500 psi, either main landing gear nutcracker switch is in the ground mode, and a brake pedal (LVDT) is depressed to a least 10% of full travel. As Auxiliary system pressure increases, it displaces the HBCM shuttle valve that closes off Combined system pressure input and routes Auxiliary system pressure to the brake control valves. All normal brake system functions are available with Auxiliary hydraulic power.

If Combined or Auxiliary hydraulic pressure is not available, a parking/emergency brake system with stored accumulator pressure may be used to stop the aircraft. The brake accumulator is in the nose wheel well, and has a nitrogen gas pre-charge of 1200 psi. The accumulator is shown in Figure 10. When normal hydraulic pressure is initially applied to the aircraft, the accumulator is fully pressurized with hydraulic fluid at 3000 psi. Accumulator pressure may be read at the accumulator in the nose wheel well, or on the indicator on the copilot lower instrument panel. The accumulator, when fully charged contains sufficient pressurized hydraulic fluid for approximately 5 to 6 brake applications. The PARK / EMER BRAKE handle on the cockpit center console is connect by cable to the modulating valve at the accumulator. Brake modulation can be accomplished by pulling out the handle between ¼ and ½ inch. Handle travel beyond approximately ½ inch results in the application of full accumulator pressure. No anti-skid is available with emergency braking. To apply accumulator pressure for parking the aircraft, the PARK / EMER BRAKE handle is pulled up and then rotated clockwise. To release the PARK/EMER BRAKE, pull the handle up and rotate counterclockwise.

The Brake-by-Wire system provides automatic spindown of the wheels during landing gear retraction. When the landing gear lever is selected to the up position and at least one nutcracker (squat) switch is in the air (weight off wheels) mode, the ECU signals an initial partial brake application to slow wheel spin, followed by a full brake application to stop wheel spin. The initial brake application is for two (2) seconds, the subsequent application is for eight (8) seconds, with the hydraulic pressure applied ranging from 400 psi to 1500 psi.

2A-32-44: Electronic (Brake-By-Wire) Anti-skid Protection

1. **General Description**

NOTE:

This section pertains to aircraft S/N 1000 - 1213, excluding 1183, without ASC 307.

To prevent wheels from skidding during braking, the Electronic or Brake-by-Wire System provides anti-skid protection. Anti-skid is operable between ten (10) and one hundred seventy (170) knots. (Below 10 knots, the main landing gear wheels may be locked to accomplish tight-radius turns.) Anti-skid is accomplished by maintaining wheel rotation during braking. Each wheel assembly is equipped with a Wheel Speed Sensor (WSS). The speed sensors are mounted each wheel hub and act as generators driven by wheel rotation. As a wheel turns, the WSS produces alternating current (AC) voltage at a frequency proportional to wheel speed. The variable frequency voltage of each wheel is monitored by the Brake Electronic Control Unit (ECU). Circuit boards in the ECU are programmed to compare the frequencies, and therefore the speeds, of adjacent wheels on each strut (the speed of the outboard wheel on the left strut is compared with the speed of the inboard wheel of the left strut). If the ECU detects a difference in rotational speed between the two wheels, the circuit boards in the ECU determine that the slow wheel is in a skid, and the ECU signals the brake control valve on the Hydraulic Brake Control Module (HBCM) to reduce hydraulic pressure to both wheel brakes on the strut with the skidding wheel. When the rotational speed of the slow wheel recovers to match that of the faster wheel, hydraulic pressure is restored to both wheel brakes. Anti-skid protection is also available using Auxiliary system pressure if the Combined system fails.

If the anti-skid system fails or is disabled, a type of anti-skid braking with degraded performance is available. This type of braking is termed open-loop pulsed anti-skid. This mode does not compare wheel rotational speeds. Brake pressure is proportional to brake pedal LVDT displacement, but is applied only three (3) to five (5) times per second, resulting in pressure fluctuations from zero to commanded pressure levels. Because full brake pressure is not constantly applied, wheel skids are less likely. The abrupt fluctuations between zero and full commanded brake pressure can be felt in the brake pedals, thus this mode of braking is popularly called “bang-bang” braking.

The anti-skid Wheel Speed Sensors in combination with the nutcracker (squat) switches are also used to provide data for other braking functions and protective features. With the anti-skid system selected on, the nutcracker switches block brake application to the wheels when the aircraft is in flight (weight-off-wheels). When the landing gear is lowered in preparation for landing, brake pressure should not be available until the aircraft is on the ground and both the nutcracker switches compressed (weight-on-wheels). An additional locked wheel circuit in the ECU protects against a failure of the nutcracker switch system that would allow application of the wheel brakes prior to landing, causing flat-spotted or blown tires and/or wheel damage. The circuit incorporates the following logic. When the landing gear is initially lowered, the wheels do not rotate. The WSSs signal the lack of rotation to the ECU that equates this condition to locked wheels and blocks hydraulic pressure to the brakes. The locked wheel signal is preserved in the ECU until the aircraft wheels spin up on runway contact and reach a speed equivalent of thirty-five (35) knots as sensed by the WSSs, or until one of the nutcracker switches compresses and enters the ground (weight-on-wheels) mode. A time delay in the circuit will also maintain the locked wheel signal for seven seconds after the nutcracker system transitions from flight to ground mode in order to provide protection in the event of an aircraft bounce on landing. If a nutcracker system malfunction prevents a switch from achieving the ground (weight-on-wheels) mode, the locked wheel circuit preserves brake system operation with anti-skid protection until the aircraft slows to thirty-five (35) knots or below, at which time anti-skid protection is lost, but full pulsed braking is available if the anti-skid selected off.

In conditions of severe hydroplanning, a separate locked wheel circuit is effective at aircraft wheel speeds above twenty-six knots. This circuit releases hydraulic pressure to the brake on a wheel that is rotating at a speed thirty percent (30%) less than the speed of the other paired wheel. This protection decreases the risk of loss of directional control when the aircraft exits the hydroplanning condition, and would otherwise encounter a sudden full brake application.

Speed signals from the WSSs are used to provide a ground spoiler activation signal in the event of a malfunction in the nutcracker switches. Automatic ground spoiler activation is provided on landing when the system is armed, flaps are extended to a least 22° and both nutcracker switches are in the ground (weight-on-wheels) mode. If a nutcracker switch fails to transition to the ground mode on landing , then speed signals from the WSSs can prompt activation. When at least one wheel on each strut spins up to approximately fifty-six (56) knots, the ground spoilers are activated. The spoiler activation signal provided by the WSSs is preserved until the wheels slow to approximately thirty-nine (39) to twenty-two (22) knots, at which time the electrical signal generated by the WSSs is too weak to maintain automatic ground spoiler deployment and the spoilers will stow.

For aircraft S/N 1000 - 1143 with ASC 166/166A incorporated, and S/N 1144 and subsequent, a circuit similar to the ground spoiler activation circuit is installed for activation of the thrust reversers. If WSSs on at least one wheel of each strut indicate a rotation speed of fifty-seven (57) knots, thrust reversers may be deployed even if the nutcracker system has not transferred to the ground (weight-on-wheels) mode.

2A-32-45: Brake Temperature Monitoring System (BTMS)

1. **General Description**

The temperature of each main landing gear brake is monitored by a probe incorporated into one of the brake assembly bolts on each wheel. The probes are resistance type devices (RTDs) that vary resistance to electrical current with temperature at a predetermined rate. Brake temperatures sensed by the probes are transmitted for view on cockpit displays. Electrical power for the temperature probes and monitoring circuits is provided by 28V DC through the BTMS circuit breaker at position D-10 on the CPO.

Temperature readings and overheat warning displays are presented in two different formats. For aircraft with SPZ 8000 DAFCS, a Brake Temperature Monitoring System (BTMS) panel, is installed in the cockpit center console beginning with aircraft S/N 1156, and available for installation in prior aircraft S/Ns as ASC 167. For aircraft with SPZ 8400 DAFCS, temperature readings and overheat warnings are displayed on the EICAS.

2. **Brake Temperature Monitoring System (SPZ-8000 AIRCRAFT)**

   A. **Description**

A Brake Temperature Monitor System (BTMS) panel is installed in aircraft S/Ns 1156 through 1252 (except S/N 1236) and those aircraft S/Ns 1000-1155 retrofitted with ASC 137. The BTMS panel, shown in Figure 16, is located on the cockpit center console (location optional) and contains a temperature gage with readout selector, an overheat light, and a test switch. The temperature readout selector is normally positioned to ALL. In this position, the highest temperature of the four main landing gear brakes is displayed on the temperature gage. The selector is rotated to monitor the temperature of each individual brake. If temperature limits for the type of brake installed are exceeded, the OVHT light on the BTMS panel illuminates. If the OVHT light on the BTMS panel is illuminated, rotating the temperature display selector through the brake positions will determine which brake(s) is overheating. On aircraft S/Ns 1156-1167, a temperature limit exceedance also prompts the illumination of the master caution lights, and an illumination of the BRAKE TEMP light installed above the pilot primary flight display. An amber BRAKE OVHT message is also annunciated on the EICAS for aircraft S/Ns 1168 -1252.

NOTE:

> Temperature limits for ABS brakes are 400° ±10°C - the OVHT light will remain on until the brake(s) has cooled below 370°C. For Dunlop brakes the temperature limit is 625° ±25° C - the OVHT light will remain on until the brake(s) has cooled to below 595°C.

The TEST push-button on the BTMS verifies that system circuits are operating properly. With the temperature selector switch in the ALL position, the test function checks all four wheel circuits and the over temperature warning circuit. A proper test is indicated by a temperature gage reading of 425°C to 475°C (for both ABS and Dunlop brakes) and the illumination of the OVHT light. If either of these conditions is not valid, selecting each wheel brake with the readout selector and pressing the TEST push-button will identify the faulty circuit. A temperature reading of 0°C indicates a short in the selected sensor circuit.

3. Brake Temperature Monitoring System (SPZ-8400 EQUIPPED AIRCRAFT)

A. Description

The Brake Temperature Monitor System (BTMS) on aircraft with SPZ 8400 DAFCS installed is mechanically and electrically the same as that installed on prior S/N aircraft. Only the display format differs. Temperature data from the brake temperature probes is supplied to the Data Acquisition Units (DAUs) that communicate the data for presentation on the CAS display. The Brake page is selected to the CAS with Line Select Key (LSK) entries on the Display Controller.

On the CAS, brake temperatures are formatted as bar graphs rising vertically with temperature. A center bar displays temperature scale increments in C° X 100, with the temperature of the inner and outer brakes on the left and right struts aligned on either side of the scale. Above the bar graph display is a presentation of applied brake pressure for left and right brakes in increments of psi X 100. Whenever the landing gear is extended, a readout of the peak temperature recorded during the last brake application in °C and identification of the brake with the peak temperature is displayed beneath the bar graphs. The existing record of prior brake application peak temperature is erased whenever the landing gear is lowered for landing, and when the aircraft accelerates faster than 60 knots on take off roll, initiating a new cycle for temperature recording.

If a brake temperature exceeds 625°C ±25°C, the bar graph display for that brake will be displayed in amber, and a BRAKE OVHT caution message is displayed on the CAS. The maximum temperature and time (in GMT) of the exceedance is recorded and is available for display by selecting the exceedance page on the CAS. The bar graph display will return to the normal white color display and the BRAKE OVHT caution message will clear when the brake cools to 595°C or below.

If an individual brake sensor or monitor circuit malfunctions, the corresponding bar graph is replaced with a red X. If a DAU fails, both brake temperature graphs on the strut monitored by the DAU are replaced by X's and the peak temperature readout is replaced with amber dashes.

For additional information on the Brake System Temperature displays, refer to the Honeywell SPZ-8000 or SPZ-8400 Digital Automatic Flight Control System Pilot Manual for the Gulfstream IV.

4. Controls and Indications:

A. Circuit Breakers (CBs)

The wheels and brakes are protected by the following circuit breakers:

Circuit Breaker Name	CB Panel	Location	Power Source
WHEEL SPEED	CPO	C-10	Left Main 28V DC Bus **(1)** Emergency 28V DC Bus 2B **(2)**
BTMS **(3)**	CPO	D-10	Right Main 28V DC Bus
APPLIED BRAKE PRESSURE	CPO	D-6	Essential 28V DC Bus
BCS CHL #1 **(1)**	CPO	A-2	Fwd Emergency Battery 28V DC Bus
BCS CHL #2 **(1)**	CPO	B-2	Essential 28V DC Bus
INBD ANTI SKID **(2)**	CPO	A-2	Emergency 28V DC Bus 1A
OUTBD ANTI SKID **(2)**	CPO	B-2	Emergency 28V DC Bus 2B
WHEEL BRK ACCUM PRESS	CPO	D-3	Essential 28V DC Bus

NOTE(S):

(1) Electronic Brake-by-Wire System

(2) Hydro-Mechanical Analog Braking System

(3) Aircraft with SPZ-8000

B. Caution (Amber) CAS Messages:

Caution CAS messages associated with wheels and brakes are:

CAS Message	Cause or Meaning
ANTI-SKID FAIL (Brake-by-Wire	Normal anti-skid control circuit or other failure: system has switched to open loop pulsed anti-skid control (bang-bang braking) **OR:** ANTI-SKID switch selected to OFF
ANTI-SKID FAIL (Hydro-Mechanical Analog Braking)	Miscompare in 28V DC power supplies or in nutcracker switches **OR:** ANTI-SKID switch selected to OFF or system circuit problems **OR:** Auxiliary hydraulic system fails to energize or pressure fails to rise within three (3) seconds after brake pedals are depressed
BRAKE FAIL (Brake-by-Wire)	Failure of brake system. BCS cannot provide braking

CAS Message	Cause or Meaning
BRAKE FAIL (Hydro-Mechanical Analog Braking)	Loss of Combined and Auxiliary hydraulic system pressure. No braking is available
BRAKE OVHT	One or more brake assemblies has exceeded temperature limit: • 400°C for S/N 1156-1213 and aircraft with ASC 167 • 625°C for S/N 1214 and subsequent, and S/N 1000-1213 with ASC 346
BRAKE PEDAL (Brake-by-Wire)	Malfunctioning brake pedal

C. **Advisory (Blue) CAS Messages**

Advisory CAS messages associated with the brake system are:

CAS Message	Cause or Meaning
BRAKE MAINT REQ'D (Brake-by-Wire)	(For S/N 1000-1213 with ASC 190, 266, or 296) If message is displayed after landing gear retraction and extinguishes approximately five (5) seconds later, the auto spin-down feature has failed
BRAKE MAINT REQ'D (Hydro-Mechanical Analog Braking)	(For S/N 1214 and subsequent) If message is displayed during takeoff, landing ground roll, or during flight with the landing gear extended, a wheel speed sensor (WSS) miscompare lasting more than five (5) seconds has been detected

5. **Built-in Test (BIT) - HMAB System**

The Anti-skid Test pushbutton, shown in Figure 11, is used to initiate a system BIT test from the cockpit. Pressing the pushbutton performs the same function as pressing the System Test button on the face of the Skid Control Box (ECU) in the avionics equipment rack, illustrated in Figure 17. For a valid test, the following indications should appear.

- The Anti-skid Test switch illuminates IN TEST
- Amber ANTISKID FAIL and BRAKE FAIL messages appear on EICAS
- Blue BRAKE MAINT REQ'D message appears on EICAS
- System Test light illuminates on the Skid Control Box (ECU)

6. **Built-in Test Equipment (BITE) – Brake-by Wire System**

Pressing the BRAKE TEST pushbutton, shown in Figure 18 will initiate a brake system test. During the test, the following indications are displayed.

- Amber ANTISKID FAIL, BRAKE FAIL, and BRAKE PEDAL FAIL messages appear on EICAS
- Blue BRAKE MAINT REQ'D message appears on the EICAS
- The pushbutton illuminates IN TEST
- The BITE indicator illuminates on the ECU

If one (or more) of the indications fails to appear during the BITE test, a fault is indicated. Faults may be isolated by rotating the selector on the front of the ECU. See the illustration in Figure 19

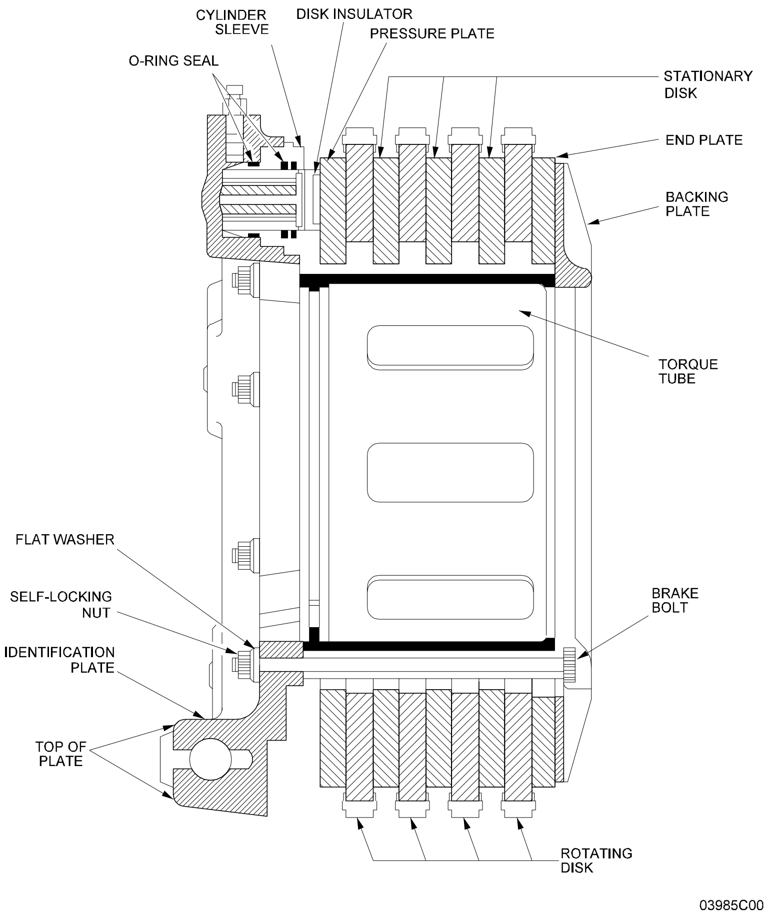
NOTE:

BIT or BITE tests should be performed at taxi speeds less than 10 knots. For the duration of the BIT or BITE tests, normal anti-skid operation is interrupted.

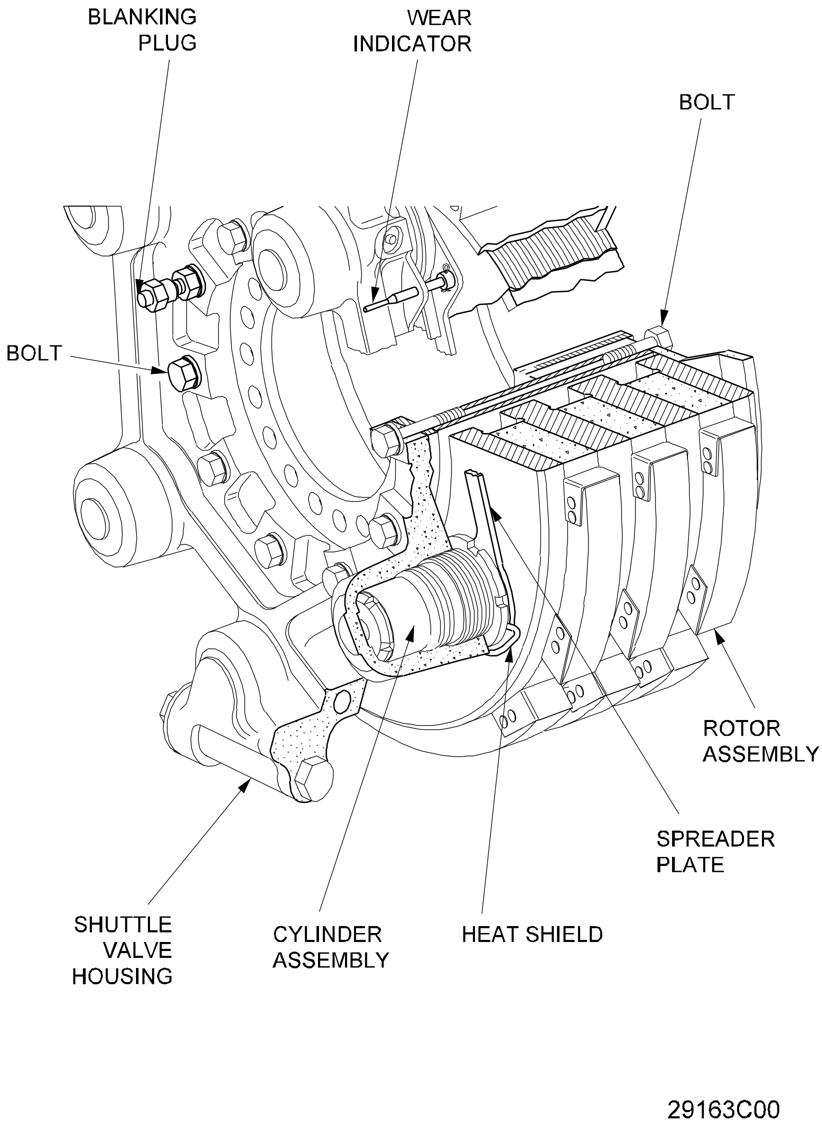
7. **Limitations:**

Takeoff is prohibited with BRAKE FAIL or BRAKE PEDAL message displayed.

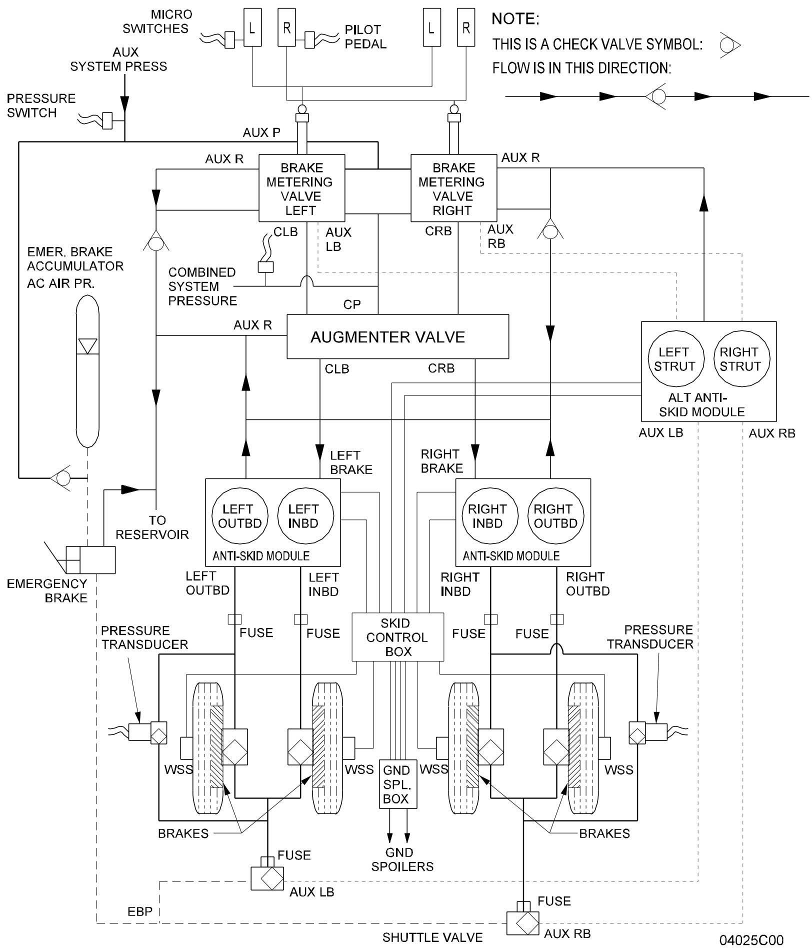
Takeoff is permitted with Anti-Skid system inoperative, provided Ground Spoilers are operative, 20° flaps are used, and the Cowl and Wing Anti-Icing systems are not used.



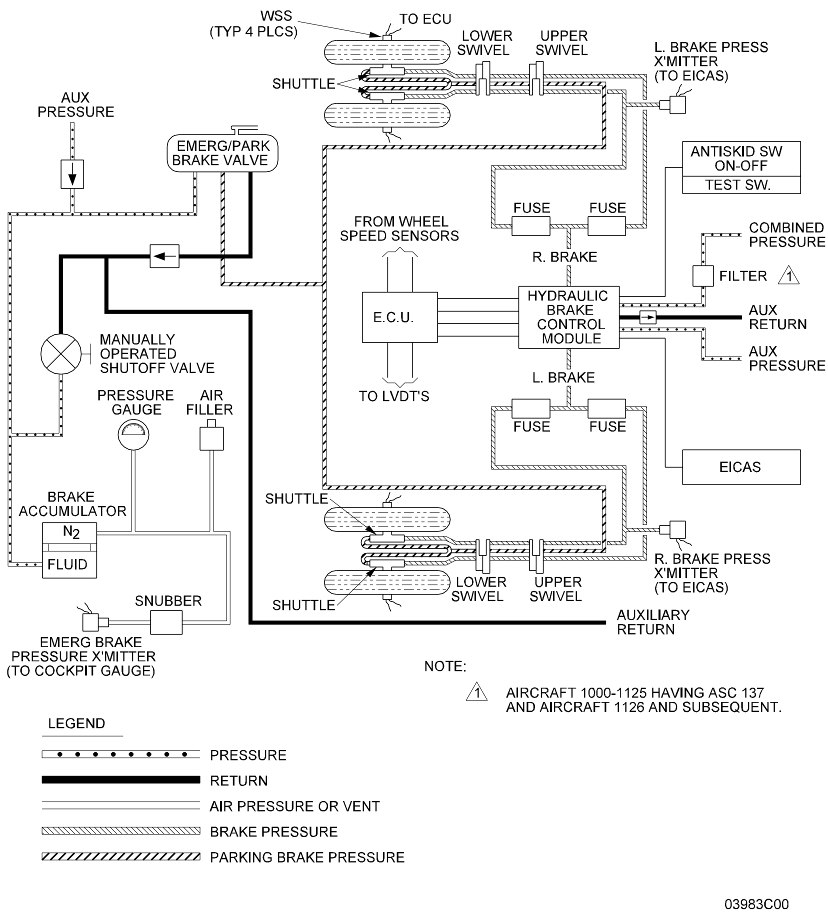
ABS Brake Components
Figure 12



Dunlop Brake Components
Figure 13



HMAB Hydraulic Schematic
Figure 14



Brake-by-Wire Hydraulic Schematic
Figure 15

2A-32-50: Nose Wheel Steering System

1. General Description:

The aircraft has a steerable nose wheel operated by an electronic and hydraulic steer-by-wire system using inputs from a control wheel mounted on the pilot side console or inputs from rudder pedal position. Hydraulic pressure from the Combined powers nose wheel steering.The nose wheel may be turned 78° (± 2°) left or right of center with the cockpit control wheel, or 7° (±1°) using only rudder pedal input. Control wheel and rudder pedal inputs are additive, but total nose wheel displacement from center is limited to 80° (±2°) by mechanical stops. Nose wheel steering must be selected ON with the guarded system switch next to the control wheel. The system switch and other system components and controls are

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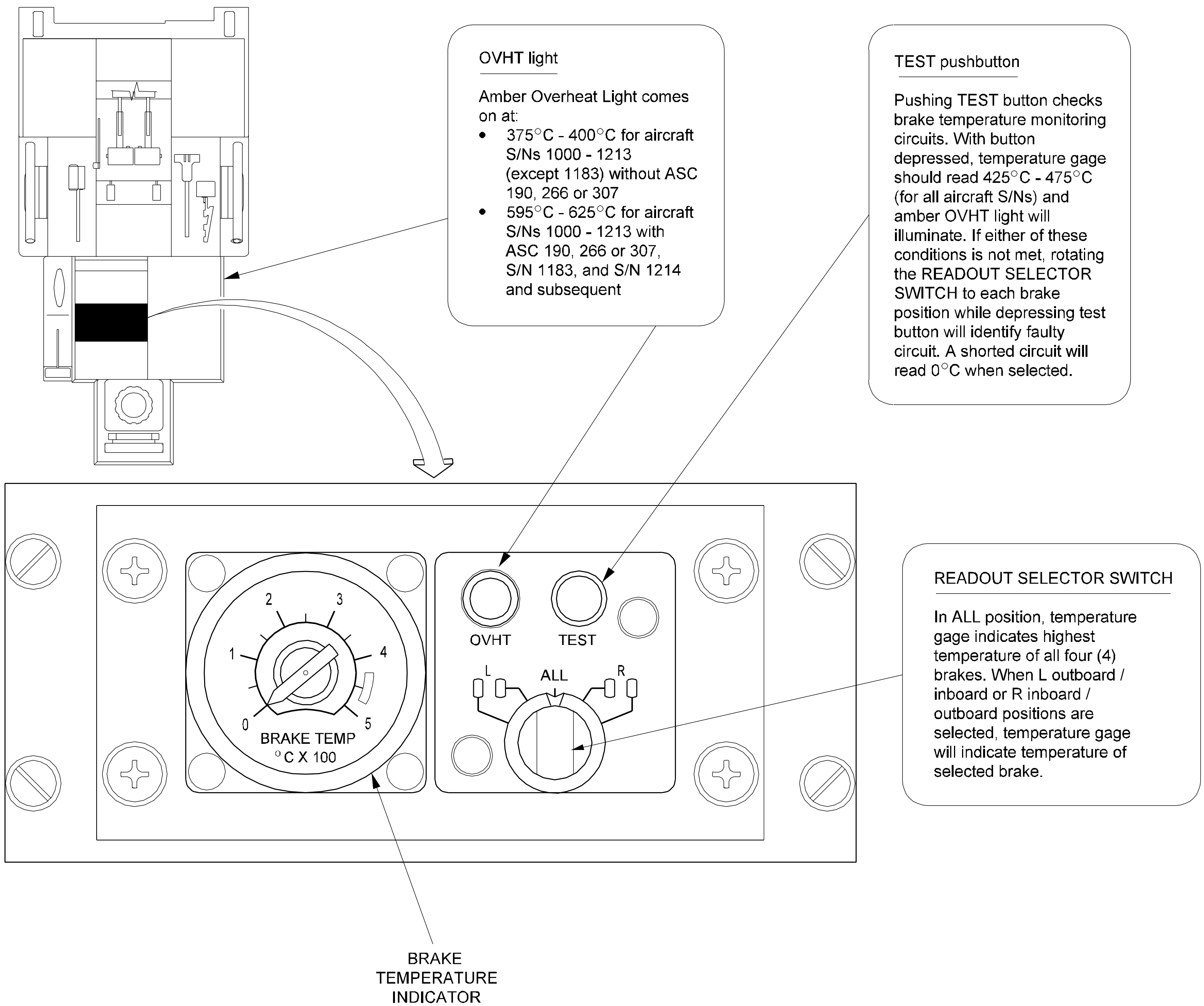


29164C00

Brake Temperature
Monitoring System Panel
(SPZ-8000 Aircraft)
Figure 16

RESET button

Button resets magnetic monitor circuits.

GR/FL indicator

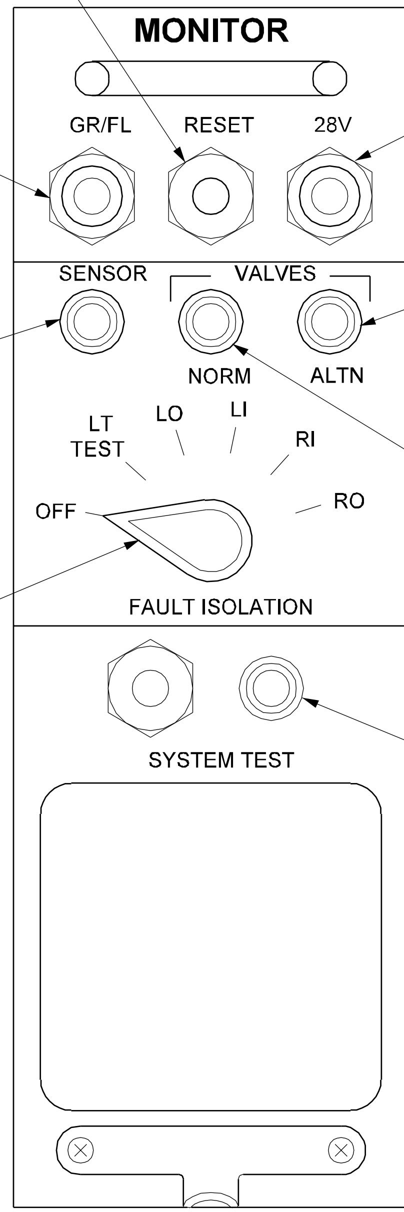
Flag indicator magnetic monitor will indicate disagreement between Left and Right Main Landing Gear nutcracker switches that lasts for more than ten (10) seconds.

SENSOR light

Amber light illuminates if fault is detected in a Wheel Speed Sensor (WSS) circuit.

FAULT ISOLATION SWITCH

Six position switch:
OFF - Fault Isolation off
LT TEST - Tests lights and circuits
LO - Left Outboard brake
LI - Left Inboard brake
RI - Right Inboard brake
RO - Right Outboard brake
If fault is detected during BIT test, rotation of FAULT ISOLATION switch to individual brake positions will detect faulty system element (e.g. with switch set to LO brake position, a fault in Left Outboard WSS circuit will illuminate SENSOR light).



28 V indicator

Flag indicator magnetic monitor will indicate loss of 28 volt DC power to INBD ANTI SKID or OUTBD ANTI SKID circuit breakers that lasts more than ten (10) seconds.

ALTN VALVES light

Amber light illuminates when fault in Alternate (Auxiliary hydraulic system) Anti-skid valves is detected.

NORM VALVES light

Amber light illuminates when fault in Normal (Combined hydraulic system) Anti-skid valves is detected.

SYSTEM TEST light

Amber light illuminates during a BIT test initiated by pressing SYSTEM TEST button on Skid Control Box or by pressing ANTI SKID TEST pushbutton on cockpit center console.

29167C00

Skid Control Box (ECU)
for HMAB Anti-skid
Figure 17

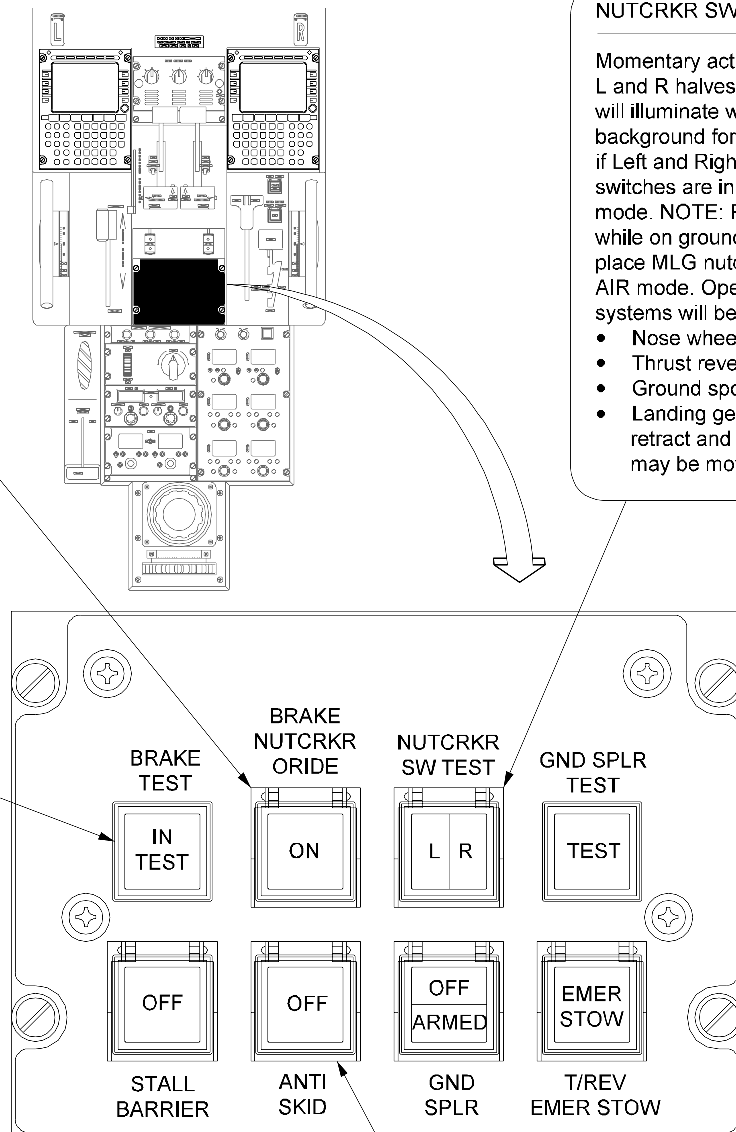
BRAKE NUTCRACKER ORIDE pushbutton

Momentary action amber ON switch, when pressed in air (weight-off-wheels) overrides air signal of MLG nutcracker switches, sending ground signal (weight-on-wheels) to brake system ECU, allowing brake pressure application with brake pedals, monitored on EICAS.

NUTCRKR SW TEST pushbutton

Momentary action pushbutton, L and R halves of pushbutton will illuminate with amber background for when pressed if Left and Right MLG nutcracker switches are in AIR (weight-off-wheels) mode. NOTE: Pressing pushbutton while on ground (weight-on-wheels) will place MLG nutcracker switches in AIR mode. Operation of some aircraft systems will be altered:

- Nose wheel steering inoperative
- Thrust reversers inoperative
- Ground spoilers inoperative
- Landing gear lever solenoid will retract and landing gear handle may be moved to retract position



BRAKE TEST pushbutton

NOTE: Test should only be initiated if aircraft speed is less than 12 knots. Momentary pushbutton initiates a BITE test in ECU (BITE test may also be initiated with toggle switch on front of ECU box). When pushbutton is depressed, IN TEST green light in pushbutton will illuminate for approximately three (3) seconds during which BRAKE test takes place. The following annunciations will be displayed on CAS:

- BRAKE MAINT REQ'D (blue)
- ANTISKID FAIL (amber)
- BRAKE FAIL (amber)
- BRAKE PEDAL FAIL (amber)

System fault is indicated if BRAKE TEST light and / or CAS annunciations are not present.

ANTI SKID pushbutton

Alternate action pushbutton selects Anti-skid system OFF (amber) or ON

29168C00

Brake Test Pushbutton for
Brake-by-Wire System
Figure 18

FAULT light

ON (amber) when fault detected in anti-skid system during BITE test

LEFT light

ON (amber) when fault detected (FAULT light ON) in left hand component of anti-skid system and FAULT ISOLATION SELECTOR SWITCH rotated to faulty component type

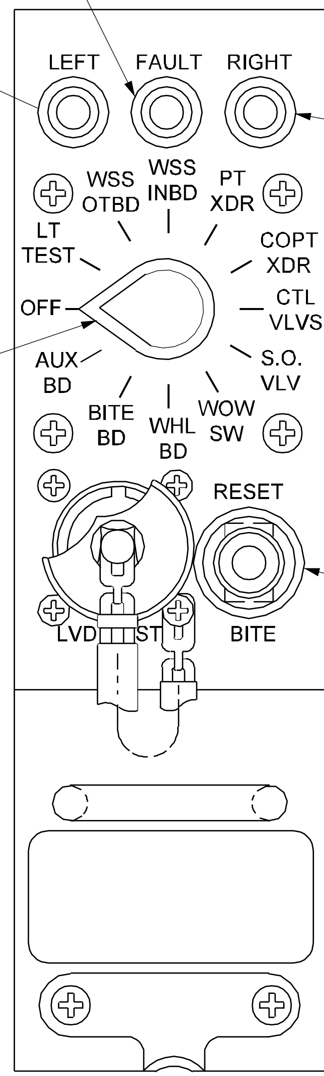
RIGHT light

ON (amber) when fault detected (FAULT light ON) in right hand component of anti-skid system and FAULT ISOLATION SELECTOR SWITCH rotated to faulty component type

FAULT ISOLATION SELECTOR SWITCH

Selector rotates through twelve (12) positions to diagnose source of FAULT light:

- OFF - selector off
- LT TEST - tests lights on ECU
- WSS OTBD - selects outboard wheels speed sensors
- WSS INBD - selects inboard wheels speed sensors
- PT XDR - selects pilot brake pedal LVDTs
- COPT XDR - selects copilot brake pedal LVDTs
- CTL VLVS - selects HBCM brake control valve
- S.O. VLV - selects HBCM shutoff valve
- WOW SW - selects main landing gear nutcracker switches
- WHL BD - selects Wheel circuit board
- BITE BD - selects BITE circuit board
- AUX BD - selects Auxiliary circuit board



BITE / RESET toggle switch

Initiates a BITE test of anti-skid system / resets BITE system and extinguishes amber lights (LEFT / FAULT / RIGHT) if illuminated during BITE test

29195C00

Brake-by-Wire BITE / Fault Indications
Figure 19

shown in Figure 20. With the nose wheel steering activated, moving the control wheel turns a dual potentiometer located beneath the control wheel inside the side console. Rudder pedal movement turns a similar potentiometer located under the copilot floorboard. Electrical signals proportional to control wheel / rudder displacement are routed from the potentiometers an Electronic Control Module (ECM) installed in the left hand avionics rack. The ECM sums the control wheel and rudder pedal inputs to obtain the desired steering command. The steering command is compared to the present position of the nose wheel, and an electric signal is transmitted to the Electro-Hydraulic Servo Valve (EHSV) in the nose wheel well, porting hydraulic pressure to the steering actuator that turns the nose wheel in the desired direction by moving the torque linkage connecting the actuator and the nose wheel. The steering control wheel is returned to the neutral, or straight ahead position by centering springs if it is released. The rudder pedals are centered by the bungees that provide artificial feel for rudder inputs.

To ensure that the nose wheel remains centered prior to nose wheel retraction into the nose wheel well, internal centering cams are incorporated in the shock strut, and the steering system is restricted to ground operation only by several safeguards. The landing gear lever handle must be in the down position to complete the electrical circuits for steering. Hydraulic pressure for steering is provided through the nose landing gear extend lines. The hydraulic pressure must pass through two (2) shut-off valves (SOVs) before reaching the EHSV and steering actuator. See the schematic in Figure 21. The first, SOV, #1, is powered by 28V DC through circuit breaker STEER BY WIRE #1 (position C-12) on the COP. For the SOV to open, the guarded PWR STEER switch next to the control wheel must be on, and the nose gear down lock switch must be engaged. SOV #2 is powered by 28V DC through circuit breaker STEER BY WIRE #2 (position D-12) on the COP, and requires that the nose gear nutcracker (squat) switch be compressed in the ground (weight-on-wheels) position for the valve to admit hydraulic pressure for the steering actuator. When the nose gear nutcracker switch is compressed, the ECM initiates a one second delay before responding to steering commands from the rudder pedals or the control wheel, maintaining the nose wheel in the centered or straight ahead direction. The delay avoids abrupt steering commands caused by large rudder pedal inputs during crosswind landings.

The nose wheel steering linkage may be disconnected for aircraft towing by removing the connecting pin on the torque link between the actuator and the nose wheel. A tow bar is then connected to the nose wheel axle.

Several Aircraft Service Changes (ASCs) have modified the nose wheel steering system from the original configuration.

- ASC 176: modifies aircraft S/Ns 1000-1242 (modification incorporated into aircraft S/N 1243 and subsequent) to include additional circuits that monitor operation of the system and prompt a STEER BY WIRE FAIL message on the CAS if any of the following occur: nose wheel nutcracker switch stuck in the ground (weight-on-wheels) position, causing the No. 2 SOV to remain in the open position, loss of electrical power to the system, an improperly installed ECU or ECU connectors, or the No. 2 SOV fails to open after touchdown and nose wheel nutcracker switch compression.
- ASC 302A: available for installation on all aircraft, installs a separate switch that shuts off rudder pedal nose wheel steering and restricts steering to the hand control wheel only. The two (2) position switch is labelled NORMAL and HANDWHEEL ONLY. When HANDWHEEL ONLY is selected, a blue

light adjacent to the switch is illuminated. For aircraft S/Ns 1000-1252 without SPZ 8400, a light installation, mounted above the pilot Primary Flight Display (PFD) will illuminate with the legend RUD STRG OFF when nose wheel steering is selected to HANDWHEEL ONLY. For aircraft S/Ns 1000-1252 with SPZ 8400 installed, and aircraft S/N 1253 and subsequent an annunciation of RUDDER STRG OFF will appear on the CAS when the steering switch is selected to HANDWEEL ONLY.

- ASC 302A AM2: installs an additional light above the copilot PFD, identical to the one installed on the pilot side for aircraft S/Ns 1000-1252 without SPZ 8400. The additional light legend is also RUD STRG OFF, and the light will illuminate in conjunction with the light on the pilot side when nose wheel steering is selected to HANDWHEEL ONLY.
- ASC 323: installs a geared hand control wheel for nose wheel steering in aircraft S/Ns 1000-1230. The geared control wheel is production installed in aircraft S/N 1231 and subsequent. The two to one ratio wheel provides a less sensitive input for nose wheel steering, allowing smoother turns, by increasing the control wheel rotation required for steering commands.

2. Controls and Indications:

A. Circuit Breakers (CBs)

Nosewheel steering is protected by the following circuit breakers:

Circuit Breaker Name	CB Panel	Location	Power Source
STEER BY WIRE #1	CPO	C-12	Essential 28V DC Bus
STEER BY WIRE #2	CPO	D-12	Right Main 28V DC Bus

B. Caution (Amber) CAS Messages

Caution CAS messages associated with nosewheel steering are:

CAS Message	Cause or Meaning
STEER BY WIRE FAIL	Both steering channels have failed: steering not available. System is in shimmy damping mode. With ASC 176 incorporated, a switch position / system status miscompare is detected

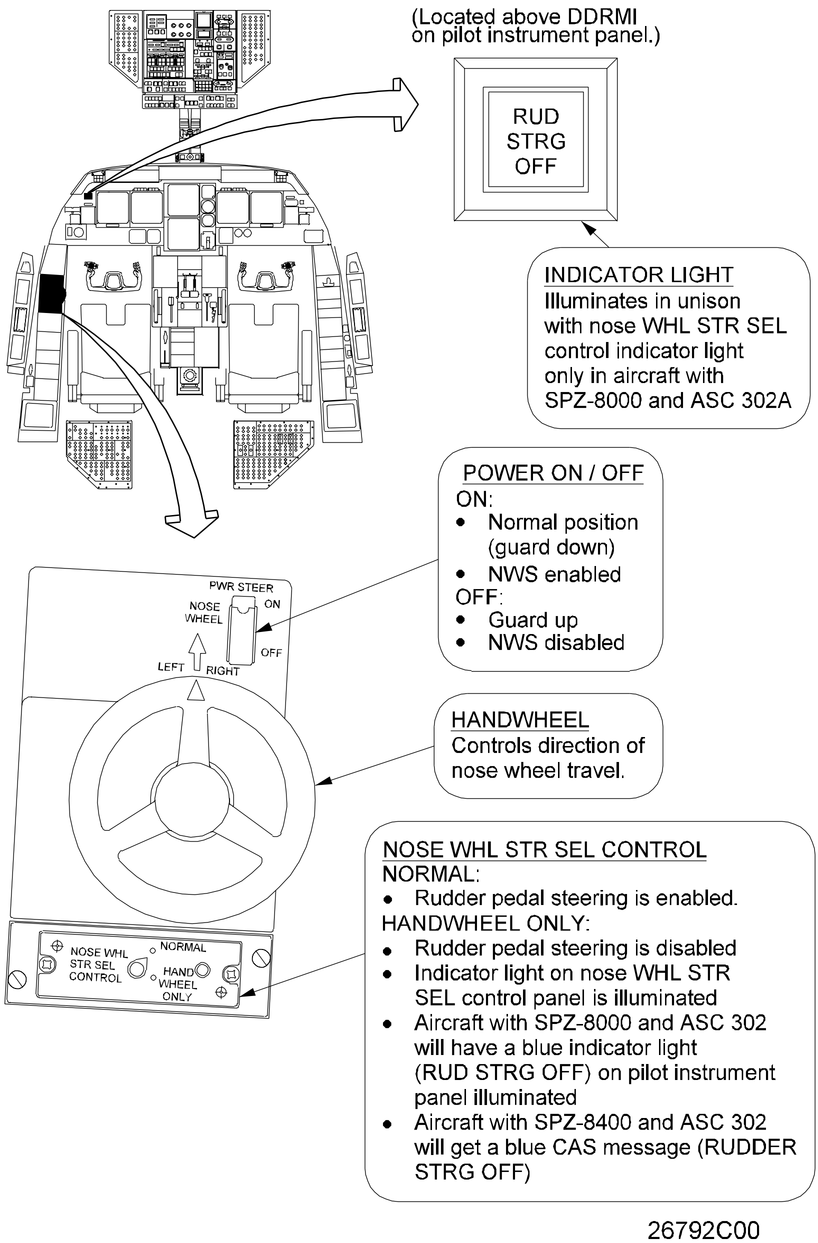
C. Advisory (Blue) CAS Messages

Advisory CAS messages associated with the nosewheel steering system are:

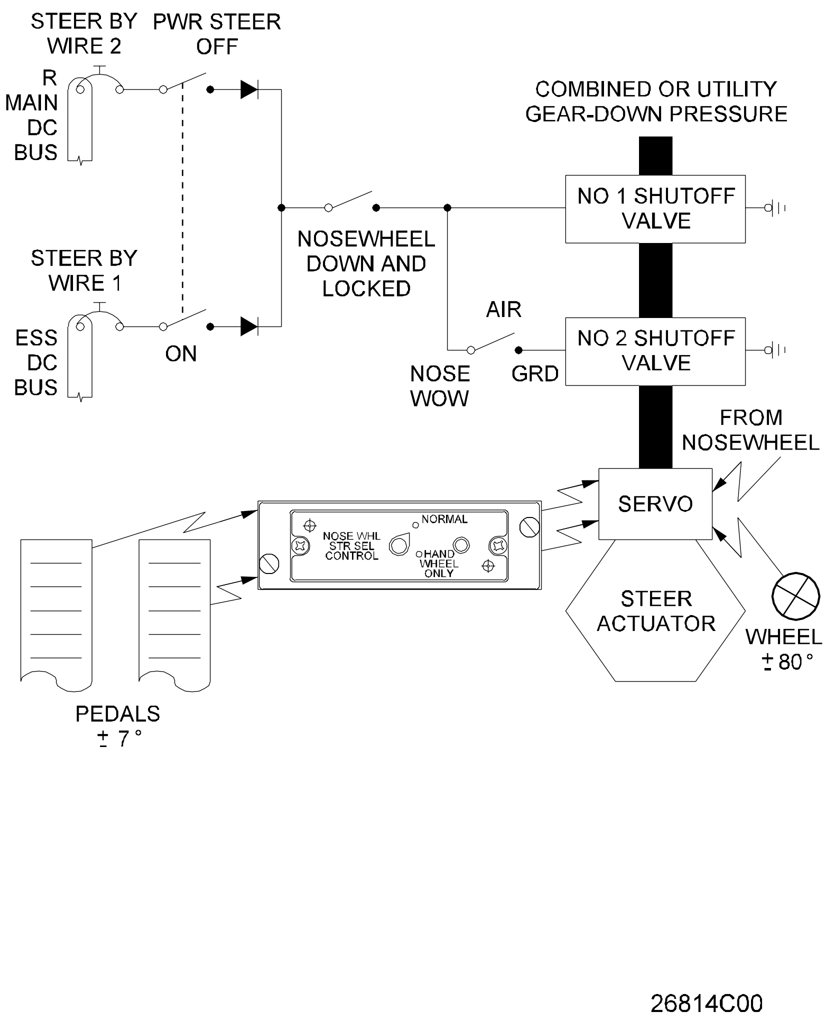
CAS Message	Cause or Meaning
RUDDER STRG OFF	(With ASC 302A) HAND WHEEL ONLY mode is selected on nose wheel steering control panel

D. RUD STRG OFF Indicator (S/N with SPZ-8000)

If the NOSE WHL STR SEL CONTROL on the pilot side console is selected to HAND WHEEL ONLY, a blue RUD STRG OFF indicator on the pilot instrument panel (ASC 302A) and on the copilot instrument panel (ASC 302A AM2) will illuminate.



Nose Wheel Steering Controls and Components
Figure 20



Nose Wheel Steering Hydraulic Schematic
Figure 21