Rotor Brake Caution Light

The rotor brake caution light, marked ROTOR BRAKE, is located on the caution advisory panel (figure FO-4), on the pilot's side of the instrument panel. The light is provided as an aid in the prevention of rotor engagement while the rotor brake is engaged. Whenever the rotor brake hydraulic pressure is 10 ± 1 psi or above, and electrical power is supplied to the dc primary bus, the caution light will go on. When the rotor brake pressure drops below 10 ± 1 psi, the light will go out. Normally with rotor brake off, pressure should be zero; however, after the rotor brake is released and pressure at 10 ± 1 psi or above is trapped in the system, the caution light will remain on. If the pressure reaches 20 psi, the pucks will begin to drag.

TRANSMISSION SYSTEM

The transmission system (figure 1-9) consists of three gear boxes that transmit power to the main and tail rotors. The main gear box reduces engine rpm and interconnects the two engines to the rotor head. A freewheeling unit, located at each engine input to the main gear box, permits the rotor head to autorotate without power turbine drag in event of engine (or engines) failure, or when power turbine rpm decreases below that of the rotor rpm. Engine torque is transmitted through the main gear box to the main rotor drive shaft to drive the main rotor, and aft through a tail rotor drive shaft to the intermediate gear box at the base of the pylon. From the intermediate gear box, a pylon drive shaft extends upward to the tail gear box to drive the tail rotor. Each of the three gear boxes has a chip detector.

MAIN GEAR BOX

The main gear box, mounted above the cargo compartment aft of the engines, is a four-stage reduction gear system which reduces engine rpm at a ratio of approximately 93.4 to 1 for driving the rotor head. The main gear box contains a spur, helical bevel gear, and single planetary gear stage. Shafting extends from the main gear box lower housing to the intermediate gear box and then to the tail gear box to drive the tail rotor. The main gear box accessory section, located at the rear of the main gear box lower housing, drives the primary, utility, and auxiliary hydraulic pumps, the main gear box oil pumps, the high pressure torquemeter oil pump, and the two generators. The auxiliary power unit drives the accessory section of the main gear box, on the ground, prior to starting the engines. When operating the APU at 100% speed, the APU drive shaft will drive the accessory section until the rotor reaches 100% Nr. The APU has a clutch, which contains a freewheeling unit, that enables shutdown of the APU when the rotor head is engaged. There is a through shaft, driven by the No. 1 engine, in the main gear box. Under normal conditions with the rotor turning, the gear box accessories are driven by the tail takeoff drive unit, which is provided with a freewheeling unit. In the event this freewheeling unit fails, the through shaft will drive the accessories.

INTERMEDIATE GEAR BOX

The intermediate gear box, located at the base of the tail rotor pylon, contains a bevel gear direct-drive system to change the direction of the shafting that transmits engine torque to the tail gear box. The intermediate gear box is splash-lubricated. Screened air

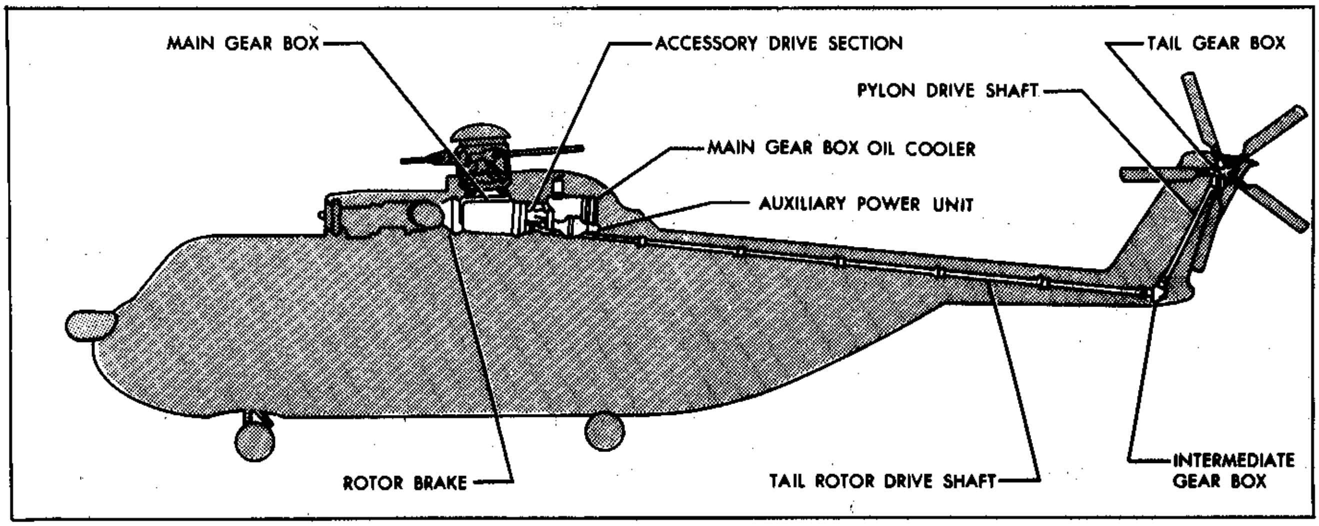


Figure 1-9. Transmission System

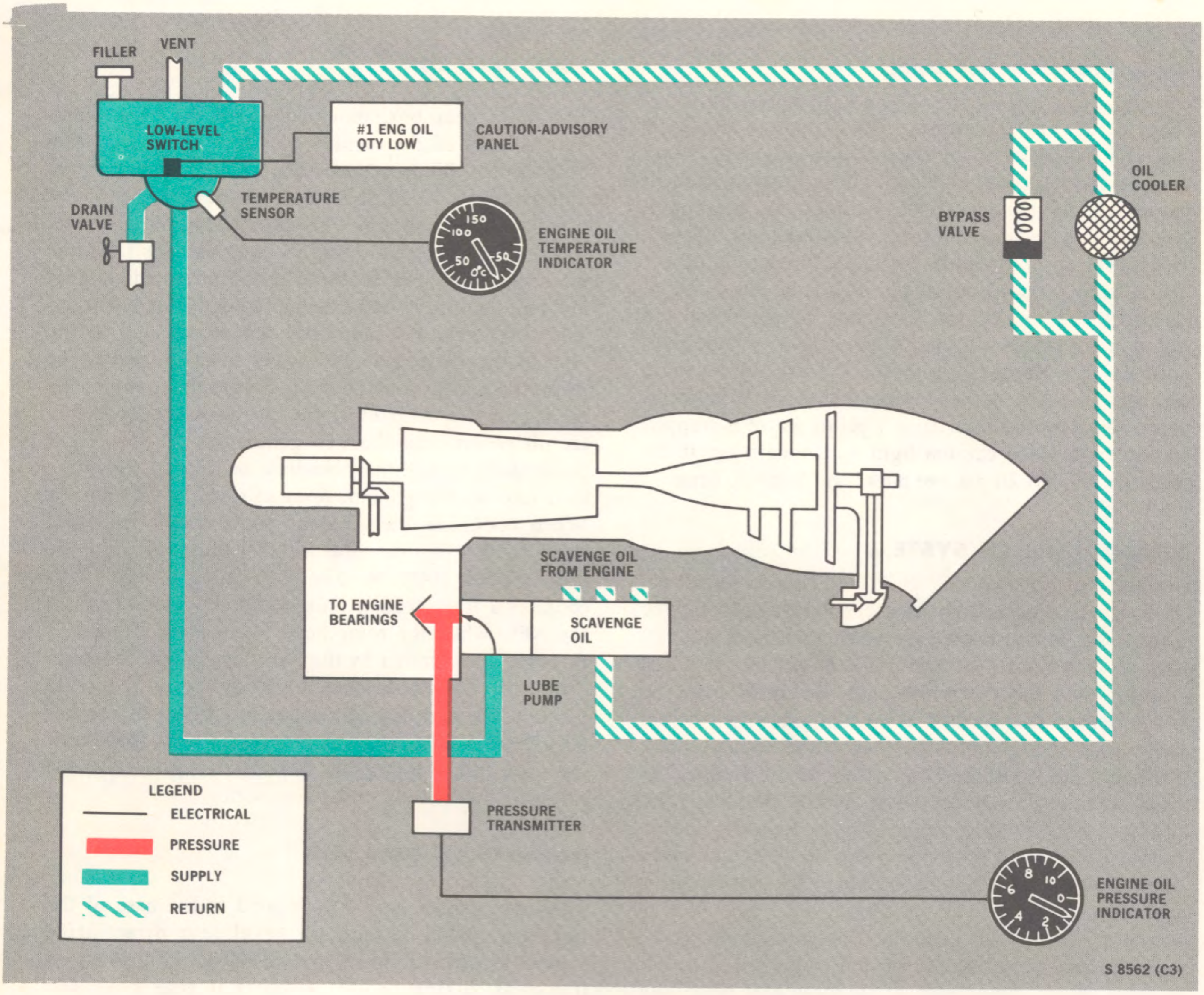


Figure 1-10. Engine Oil System

outlets in the pylon fairing permit the gear box to be cooled by the rotor downwash.

TAIL GEAR BOX

The tail gear box, located at the upper end of the tail rotor pylon, contains a bevel gear reduction-drive system to transmit engine torque to the tail rotor. The tail gear box also contains part of the pitch change linkage which extends through the hollow output shaft to the tail rotor hub. The tail gear box is splash-lubricated.

TRANSMISSION CHIP DETECTOR LIGHTS

Three amber transmission chip detector caution lights, marked XMSN CHIP MAIN, XMSN CHIP INTMED and XMSN CHIP TAIL, are located on the caution-advisory panel on the instrument panel (figure FO-4). The lights provide a visual indication of metallic chips detected in the main, intermediate, or tail gear boxes. The system operates on current from the dc primary bus and is protected by circuit breakers, marked MAIN, INTMED, and TAIL under the general heading CHIP DET, located on the overhead DC circuit breaker panel.

OIL SUPPLY SYSTEMS

The oil supply systems consist of the engine and transmission oil systems.

ENGINE OIL SYSTEM

Each engine has an independent oil tank and dry sump full scavenge oil system (figure 1-10). Oil is gravity-fed from the tank to the engine-driven oil pump, mounted on the forward right-hand side of the engine. The engine-driven pump distributes oil, under pressure, through a filter to accessory gears and engine bearings. The oil serves both lubricating and cooling purposes and the system is completely automatic. The scavenge side of the pump returns the oil through an oil cooler to the oil tank. The oil cooler is an oil-to-fuel heat exchanger with an associated oil bypass system. The oil flow through the cooler depends on oil temperature. At low oil temperature, most of the oil bypasses the cooler. Higher oil temperatures close the bypass valve and cause all of the oil to flow through the cooler. Each engine oil system has a useful capacity of 2.5 U.S. gallons of oil in a 3.0 U.S. gallon tank (0.5 gallon expansion space). The circular tanks are located around the forward section of each engine.

Engine Oil Low Level Caution System

The engine oil low level caution system consists of a separate indicating system for each engine. Each system is separately powered through a 5 ampere circuit breaker by the dc primary bus, and each has a respective caution panel light capsule, marked #1 ENG OIL QTY LOW and #2 ENG OIL QTY LOW. A float switch is installed in each engine oil tank. Should the oil level in a tank fall below that desired for safe engine operation (1.5 gallons), the float switch contacts close, causing the light to illuminate. Power for the engine oil low level caution system is supplied by the dc primary bus through circuit breakers, marked ENG OIL LOW LEVEL, located on the dc circuit breaker panel.

TRANSMISSION OIL SYSTEMS

Main Gear Box Oil System

Primary and secondary oil pumps (figure 1-11) are provided for lubrication. The torque system oil pump is mounted piggy-back to the primary pump on the lower portion of the rear cover of the main gear box, and uses main transmission oil for the torque sensing system. The secondary oil pump is mounted between the utility hydraulic pump and the rear cover mounting pad. Oil is pumped from the gear box sump to the oil cooler located behind the main gear box. Cooling air is forced through the cooler by a blower driven by belts from the tail drive shaft. The air is then exhausted through a screened transmission accessories cooling air outlet at the rear of the fairing. After passing through the oil cooler, the oil returns to the main gear box where it is sprayed onto the gears and bearings through jets built into the gear box castings. An oil filler, reached from the left side of the main gear box fairings, is on the left side of the gear box. A window in the gear box below the oil filler is for checking the oil level. Oil capacity is 15 gallons; normal servicing is 11 gallons.

Main Gear Box Oil Pressure Indicator and Caution Light The main gear box oil pressure indicator (38, figure FO-4) is located on the instrument panel. The indicator, graduated in pounds per square inch, is actuated by a pressure transmitter connected to the gear box oil inlet port. The main gear box oil pressure indicator operates on 26 volts ac from the No. 1 generator, through the autotransformer, and is protected by a circuit breaker, marked XMSN under the general headings OIL PRESS and 26V AC, located on the copilot's overhead circuit breaker panel. The main gear box oil low pressure caution light, marked XMSN OIL PRESS, is located on the caution-advisory panel (figure FO-4), and is actuated by a pressure switch located in the forward part of the main gear box. The amber caution light operates on dc from the primary bus and is protected by a circuit breaker, marked XMSN OIL PRESS, located on the overhead circuit breaker panel. The light will illuminate when the main gear box oil system pressure drops below approximately 4 psi. The different locations of the pressure sensors for the gage and caution light will warn the pilot of an oil blockage within the gear box which may not be indicated on the pressure gage.

Augmented Main Gear Box Oil System The augmented main gear box oil system will permit the helicopter to continue operating for approximately 45 minutes if there is a failure in the main lubrication system. In this case, the auxiliary sump, at the base of the main gear box, provides an oil supply of about 1.5 gallons to lubricate the input sleeve bearings in the high speed section of the main gear box. The torquemeter pump uses oil from the auxiliary sump to lubricate these bearings and to supply oil to the torque sensing system. An additional chip detector is installed in the auxiliary sump and lights the XMSN CHIP MAIN caution light.

Main Gear Box Oil Temperature Indicator and Caution Light The main gear box oil temperature indicator (27, figure FO-4), marked XMSN TEMP and located on the instrument panel, is graduated in degrees Celsius. The indicator is connected by dc from the primary bus to an oil temperature bulb adjacent to the main gear box oil outlet port, and is protected by a circuit breaker, marked XMSN TEMP, located on the overhead circuit breaker panel. The main gear box oil

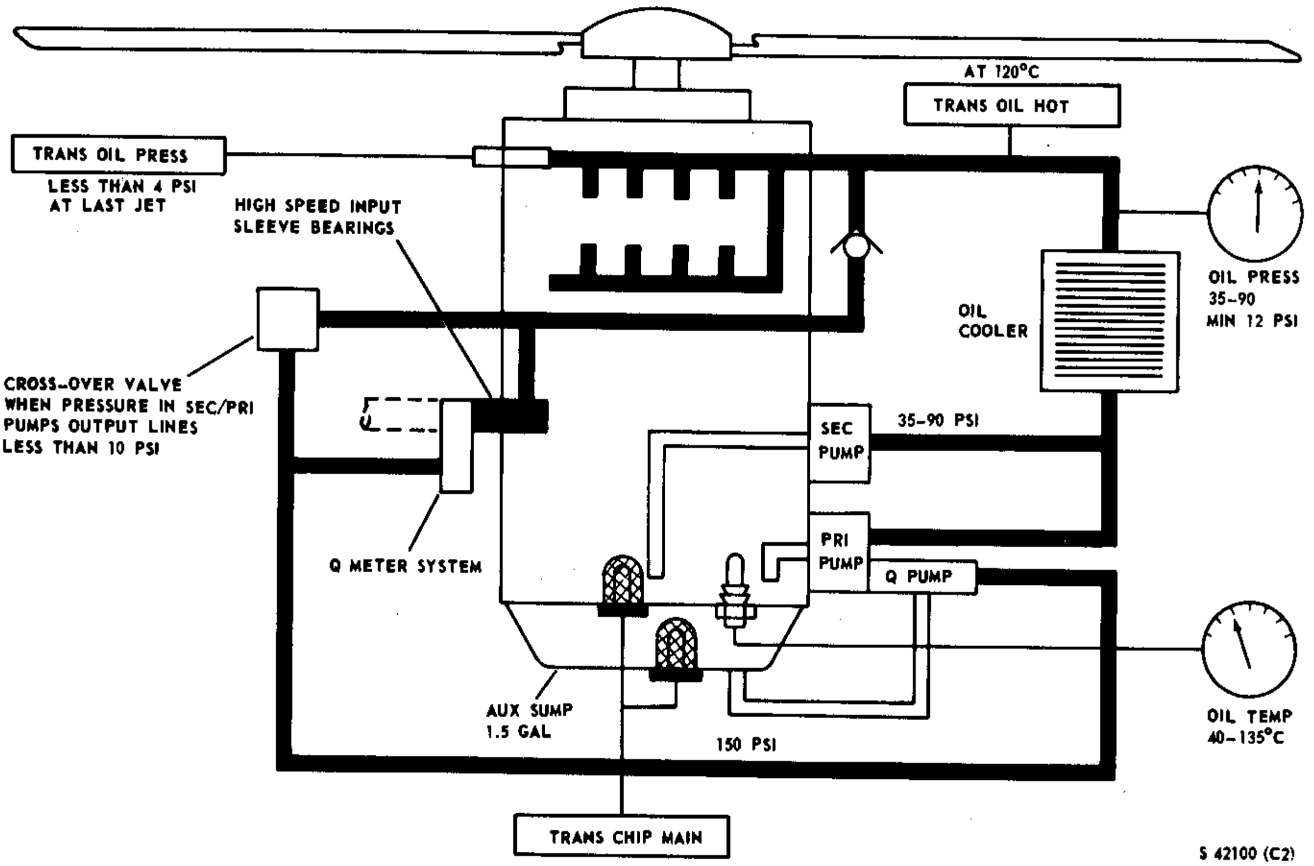


Figure 1-11. Main Transmission with Auxiliary Sump

temperature caution light is located on the caution-advisory panel and actuated by a temperature sensor located at the outlet of the oil cooler. The caution light, marked XMSN OIL HOT, operates on dc from the primary bus and is protected by a circuit breaker, marked XMSN OIL TEMP, located on the overhead circuit breaker panel (figure FO-8). The transmission oil temperature caution light will come on when the transmission oil temperature reaches 120°C. The different locations of the temperature sensors for the gage and light allow the pilot to monitor the gear box operation by means of the gage and the oil cooler operation by means of the caution light. Thus, if a malfunction occurs in the oil cooler, (blockage, fan belt failure, etc.) the caution light will illuminate before the gear box oil temperature rises to a dangerous level.

Intermediate and Tail Gear Box Oil Systems

Both the intermediate and tail gear boxes are splash-lubricated from individual sump systems. Internal spiral channels ensure oil lubrication to all bearings. An oil filler plug, drain plug, and oil level sight gage are located in each gear box casting. When the oil in the intermediate gear box is at FULL on the oil level sight gage, it contains 0.2 gallons. When the oil in the tail gear box is at FULL on the oil level sight gage, it contains 0.4 gallons.

FUEL SUPPLY SYSTEM

The helicopter is equipped with two independent pressure-type fuel systems (figure FO-5) joined together by a crossfeed system to ensure maximum fuel utilization, and two internal auxiliary systems. The two main systems are divided into a forward and aft system and are augmented by the auxiliary systems. The main systems consist of fuel tanks with individual bladder-type cells, collector cans, and ejector units in which two submerged boost pumps are located. The auxiliary systems consist of a forward and aft tank with a bladder type cell and ejector units. Transfer of fuel from auxiliary to main tanks is controlled by the pilot. The ejector system and boost pump arrangement provides for a minimum of unusable fuel. The crossfeed system provides diversified main fuel system operation. The crossfeed system is electrically controlled by a crossfeed valve switch and allows fuel from both main systems to be directed to one engine during single engine operation, or fuel from one system to supply both engines should the need arise. Under normal conditions, the forward main system supplies fuel to the No. 1 engine and the aft main system supplies fuel to the No. 2 engine. All tanks may be filled by either a pressure refueling system or by gravity through conventional filler necks on each tank. Auxiliary tanks may also be refueled through inflight gravity refueling connections. Aircraft modified by AR&SC Engineering Specification 13470.14 do not have inflight gravity refueling connections. For fuel specification and grade, see figure 1-12.

FUEL TANKS

There are two main fuel tanks and two auxiliary fuel tanks located beneath the cabin floor in the following sequence from nose to tail; forward auxiliary tank, with a capacity of 184 gallons; forward main tank, with a capacity of 348 gallons; aft auxiliary tank, with a capacity of 245 gallons; and aft main tank with a capacity of 345 gallons (figure 1-13). The forward main tank system, consisting of two cells, contains one collector can, five check valves, two pressure switches, two pump failure caution light circuits, one ac primary bus powered boost pump, one monitor bus powered boost pump, one flapper valve, one pressure refueling and defueling valve, one fuel ejector unit, one gravity filler, two sump drain valves, one dual high level sensor, and two float-operated shutoff valves. The aft main tank system, consisting of two cells, contains the same items as the forward main tank except that it has one instead of two float-operated shutoff valves. The fuel boost pumps within each main tank provide fuel to the associated fuel ejectors and the appropriate engine. The boost pumps are located in collector cans. The fuel level in the collector cans is maintained by ejector and flapper valve action. This fuel ejector unit and boost pump arrangement provides integral fuel transfer within each main tank at all operating attitudes, and provides for a maximum of usable fuel. The single cell forward auxiliary fuel tank system contains one check valve, one gravity filler, one fuel ejector, one sump drain valve, one pressure refueling and defueling valve, one dual high level sensor, one gravity refueling connection in the cabin, and one motor-operated gate valve. The single cell aft auxiliary fuel tank system contains the same items as the forward auxiliary tank system except that it has two check valves, two motor-operated gate valves, and two fuel ejectors. All fuel tank vent lines are routed through the cabin to minimize vent icing, then to both sides of the helicopter, where the tanks are vented to the atmosphere.

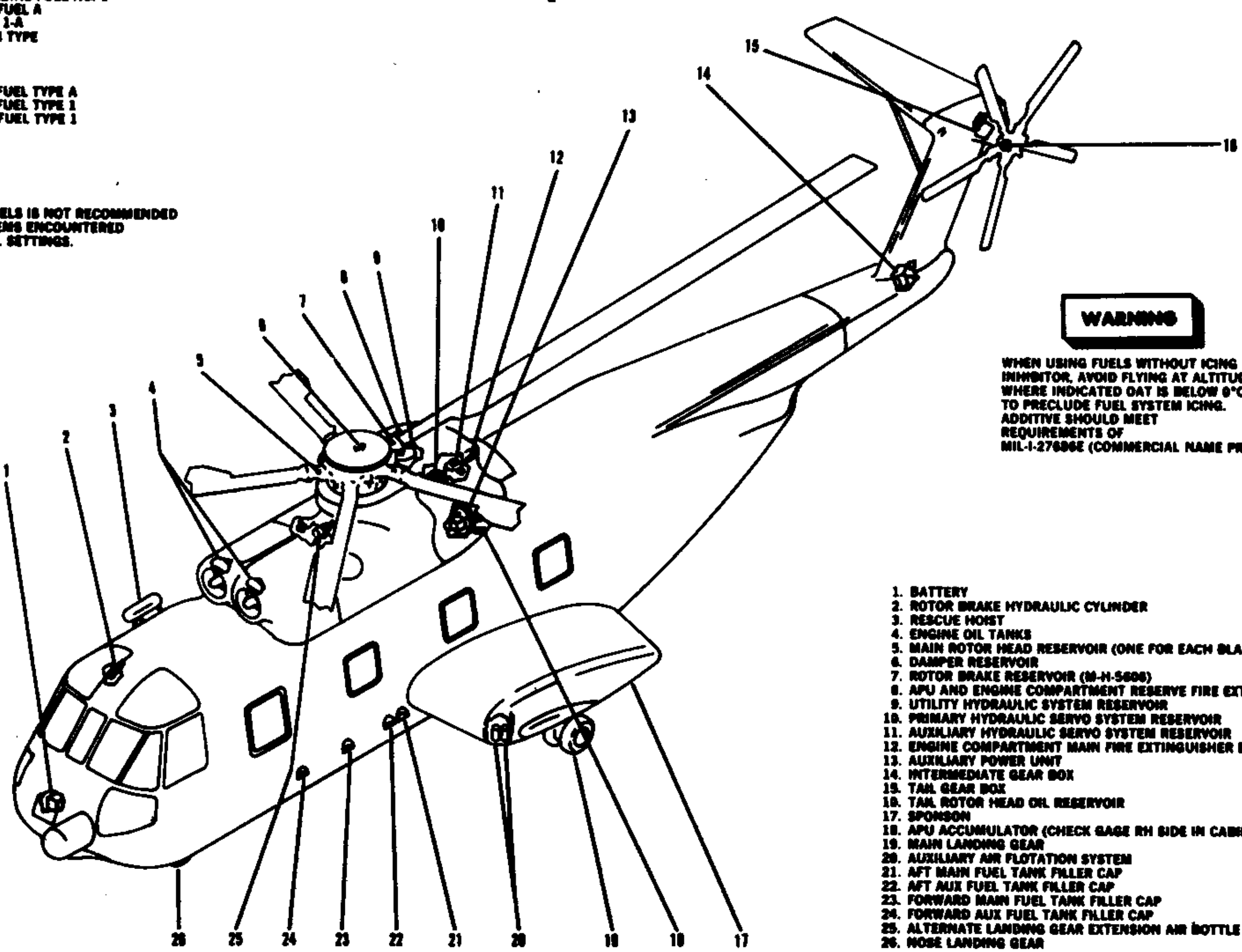
FUEL BOOST PUMPS

Two fuel boost pumps are located in each main fuel tank. Under normal conditions, fuel is pumped by the boost pumps from the collector can in each main tank through the main fuel line to the engine, and also through an ejector unit in the main tank, which maintains a constant quantity of fuel in the collector can. Ejector units in the auxiliary tanks are used to transfer fuel to the main tanks. When transferring fuel, the boost pumps in the forward main tank provide fuel to the forward auxiliary and aft auxiliary ejectors through motor-operated gate valves. The aft main boost pumps furnish fuel to the aft auxiliary ejectors through a motor-operated gate valve. The pumps in each main tank are designated No. 1 and No. 2 and each pump has a separate power source. The No. 1 boost pump in the forward tank is powered by the No. 1 ac primary bus and the No. 1 boost pump in the aft tank is powered by the No. 2 ac primary bus. The No. 2 boost pump in the forward tank is powered by the No. 1 ac monitor bus. The No. 2 boost pump in the aft tank is powered by the No. 2 ac monitor bus. Control of the pumps is accomplished by switches on the fuel management panel which operate on 28 volts dc from the primary bus. The No. 1 boost pumps are protected by circuit breakers, marked NO. 1 FUEL PUMP FWD TANK and NO. 1 FUEL PUMP AFT TANK, on the pilot's and copilot's overhead circuit breaker panels. The No. 2 boost pumps are protected by circuit breakers, marked NO. 2 FUEL PUMP FWD TANK and NO. 2 FUEL PUMP AFT TANK, on the pilot's and copilot's overhead circuit breaker panels. The engines may be efficiently operated with one, both, or no boost pumps. However, it is necessary to operate the helicopter with at least one boost pump operating in each main tank during the following conditions:

FUEL-JP-4, JP-5 OR OTHER APPROVED ALTERNATE FUELS
ALTERNATE APPROVED FUELS
ALTERNATE APPROVED FUELS CONFORM TO ASM TYPE A-1, JET A-1
THESE FUELS LIMIT STARTING TO -25°F, NATO F-34.
ARCOJET-1
AMERICAN TYPE A-1
CALTEX JET A-1
440 UNIVERSAL TURBINE FUEL NO. 1
GULFLITE TURBINE FUEL A
EXXON TURBO FUEL 1-A
KEROSENE-AVIATION TYPE
PUREJET TYPE A-1
AEROSHELL 650
AVTUR 50
AVIATION TURBINE FUEL TYPE A
AVIATION TURBINE FUEL TYPE 1
CHEVRON TURBINE FUEL TYPE 1
ATF-1A
407 AVJET K-50

NOTE: MIXING OF FUELS IS NOT RECOMMENDED BECAUSE OF PROBLEMS ENCOUNTERED WITH FUEL CONTROL SETTINGS.

OIL, ENGINE APU AND ALL GEAR BOXES
MIL-L-23699 NATO SYMBOL 0-156
HYDRAULIC FLUID-PRIMARY, AUXILIARY AND UTILITY SYSTEMS
MIL-H-5606 NATO SYMBOL H-515
FIRE EXTINGUISHER AGENT
ENGINE AND APU-CF_3 Br
PORTABLE - CO_2



WARNING

WHEN USING FUELS WITHOUT ICING INHIBITOR, AVOID FLYING AT ALTITUDES WHERE INDICATED OAT IS BELOW 0°C TO PRECLUDE FUEL SYSTEM ICING. ADDITIVE SHOULD MEET REQUIREMENTS OF MIL-I-27686E (COMMERCIAL NAME PRIST)

1. BATTERY
2. ROTOR BRAKE HYDRAULIC CYLINDER
3. RESCUE HOIST
4. ENGINE OIL TANKS
5. MAIN ROTOR HEAD RESERVOIR (ONE FOR EACH BLADE)
6. DAMPER RESERVOIR
7. ROTOR BRAKE RESERVOIR (M-H-5606)
8. APU AND ENGINE COMPARTMENT RESERVE FIRE EXTINGUISHER BOTTLES
9. UTILITY HYDRAULIC SYSTEM RESERVOIR
10. PRIMARY HYDRAULIC SERVO SYSTEM RESERVOIR
11. AUXILIARY HYDRAULIC SERVO SYSTEM RESERVOIR
12. ENGINE COMPARTMENT MAIN FIRE EXTINGUISHER BOTTLE
13. AUXILIARY POWER UNIT
14. INTERMEDIATE GEAR BOX
15. TAIL GEAR BOX
16. TAIL ROTOR HEAD OIL RESERVOIR
17. SPONSON
18. APU ACCUMULATOR (CHECK GAGE RH SIDE IN CABIN)
19. MAIN LANDING GEAR
20. AUXILIARY AIR FLOTATION SYSTEM
21. AFT MAIN FUEL TANK FILLER CAP
22. AFT AUX FUEL TANK FILLER CAP
23. FORWARD MAIN FUEL TANK FILLER CAP
24. FORWARD AUX FUEL TANK FILLER CAP
25. ALTERNATE LANDING GEAR EXTENSION AIR BOTTLE
26. NOSE LANDING GEAR

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Figure 1-12. Servicing Diagram

1. Above 6000 feet pressure altitude.

2. Dumping fuel. (See section III.)

3. Above 43° Celsius FAT.

4. Transferring fuel.

5. Below 600 pounds of fuel per tank.

6. Whenever a fuel filter bypass light is illuminated.

7. Whenever a fuel low pressure caution light is illuminated.

8. One pump per tank during takeoff and landing.

NOTE

Continuous use of one boost pump per engine is mandatory to prevent inadvertent operation without boost pumps when the exceptions are encountered. Actual or simulated generator failure will cause loss of both number two boost pumps.

FUEL QUANTITY DATA – JP4

FUEL TANKS	USABLE US GALLONS	USABLE POUNDS	UNUSABLE US GALLONS	UNUSABLE POUNDS	FULLY SERVICED US GALLONS	FULLY SERVICED POUNDS
FWD MAIN	344.6	2239.9	3.48	22.62	348	2262.0
AFT MAIN	342.4	2225.6	2.61	16.96	345	2242.5
FWD AUXILIARY	183.2	1190.8	0.8	5.20	184	1196.0
AFT AUXILIARY	244.2	1587.3	0.8	5.20	245	1592.5
TOTAL	1114.4	7243.6	7.69	49.98	1122	7293.0

FUEL QUANTITY DATA – JP5

FUEL TANKS	USABLE US GALLONS	USABLE POUNDS	UNUSABLE US GALLONS	UNUSABLE POUNDS	FULLY SERVICED US GALLONS	FULLY SERVICED POUNDS
FWD MAIN	344.6	2343.28	3.48	23.66	348	2366.4
AFT MAIN	342.4	2328.32	2.61	17.74	345	2346.0
FWD AUXILIARY	183.2	1245.76	0.8	5.44	184	1251.2
AFT AUXILIARY	244.2	1660.56	0.8	5.44	245	1666.0
TOTAL	1114.4	7577.92	7.69	52.29	1122	7629.6

1. Usable fuel determined at 0 degree nose-down attitude.
2. Fuel density of 6.5 lb/gal (JP-4), 6.8 lb/gal (JP-5) at 60°F (15.6°C)
3. Gravity refueling method used.

Figure 1-13. Fuel Quantity Data

Fuel Boost Pump Switches

Four boost pump switches are located on the fuel management panel (figure FO-4). The two boost pump switches for the pumps in the forward tank, located on the left side of the panel, are marked PUMP 1 and 2, and those for the aft tank on the right side of the panel are marked PUMP 1 and 2. Each switch has marked positions ON and OFF. All boost pump switches are connected to the dc primary bus through circuit breakers, marked FWD TANK 1 and 2, and AFT TANK 1 and 2, under the general headings FUEL SYSTEM and PUMP CONT, located on the overhead dc circuit breaker panel. When the switches are placed in the ON position, dc power from the primary bus closes relays in the circuit between the appropriate ac bus and the respective boost pump. The OFF position de-energizes the relays, cutting off ac power to the respective boost pump.

Fuel Boost Pump Failure Caution Lights

Each of the four boost pumps are provided with a pressure switch connected to the pressure feed line from each boost pump. The fuel pressure switches actuate the boost pump failure caution lights, marked FAIL, located on the fuel management panel (figure FO-4), when the boost pump pressure falls below a safe operating pressure. Each boost pump is provided with an individual boost pump failure caution light located above the respective boost pump switch. The boost pump failure caution lights should illuminate, and then go off, when the boost pumps are first turned on. The pressure switches close if the boost pump pressure decreases to, or is below, approximately 16 psi and energizes the respective boost pump failure caution light circuit, which lights the boost pump failure caution light. The boost pump pressure switches and failure caution lights operate from the dc primary bus and are protected by circuit breakers, marked FWD TANK, 1 and 2, AFT TANK, 1 and 2, under the general headings INDICATOR LIGHTS and FUEL PUMPS, located on the cockpit center overhead dc circuit breaker panel.

FUEL SHUTOFF VALVE SWITCHES

The two fuel shutoff valve switches are located on the fuel management panel (figure FO-4). The switch, marked NO. 1 ENG with marked positions ON and OFF, controls flow of fuel to the No. 1 engine. The switch, marked NO. 2 ENG with marked positions ON and OFF, controls flow of fuel to the NO. 2 engine. The switches control the fuel shutoff valves located overhead in the cabin. Placing either switch in the OFF position, shuts off the flow of fuel to the appropriate engine. The fuel shutoff valves and switches operate on power from the dc primary bus and are protected by circuit breakers, marked EMER SHUT-OFF, 1-ENG-2 under the general headings FUEL SYSTEM and VALVES, located on the overhead dc circuit breaker panel. The fuel shutoff valves are also actuated to the OFF position when the appropriate fire emergency shutoff selector handle is pulled.

FUEL CROSSFEED VALVE SWITCH

The fuel crossfeed valve switch, marked CROSSFEED, is located on the fuel management panel (figure FO-4). The switch has marked positions OPEN and CLOSE and controls the fuel crossfeed valve. With the switch in the CLOSE position, the forward main fuel tank supplies fuel to the No. 1 engine and the aft main fuel tank supplies fuel to the No. 2 engine. When the switch is placed in the OPEN position, the crossfeed valve opens and allows fuel under pressure to be supplied from both main fuel tanks to either or both engines. The crossfeed system does not transfer fuel between tanks. The fuel crossfeed valve operates on direct current from the dc primary bus, and is protected by a circuit breaker, marked X FEED under general headings FUEL SYSTEM and VALVES, located on the overhead dc circuit breaker panel. (See section VII for crossfeed procedures).

LOW FUEL PRESSURE CAUTION LIGHTS

Two low fuel pressure caution lights, marked LOW PR, are superimposed on a fuel flow schematic contained on the fuel management panel. The caution lights will illuminate whenever fuel pressure drops below a safe operating pressure at the pressure switch, located between the fuel shutoff valve and the engine. It should be noted that this is incorrectly depicted on the fuel management panel. The caution lights receive electrical power from the dc primary bus through circuit breakers, under the general heading INDICATOR LTS and marked FUEL PRESS, 1 and 2, located on the overhead DC circuit breaker panel.

FUEL QUANTITY GAGES AND TEST SWITCHES

Four fuel quantity gages, located on the fuel management panel, indicate the fuel quantity in each tank in pounds. Fuel quantity data is shown in figure 1-12. The fuel quantity indicating system may be tested by pressing the fuel quantity gage test switch, marked FUEL GAGE TEST, on the fuel management panel. Pressing the button-type switch for approximately 10 seconds will induce a current reversal which causes the pointers to indicate slightly below zero. Upon release of the test switch, pointers should return to the previous reading.

FUEL LOW LEVEL CAUTION LIGHTS

The fuel low level caution lights, marked FWD FUEL LOW and AFT FUEL LOW, are located on the caution-advisory panel (figure FO-4). The fuel low level caution light will illuminate when 210 to 280 pounds of fuel remain in the respective fuel tank, in a 5° nose-down attitude, or when 170 to 200 pounds of fuel remain in the respective fuel tank, when in a level attitude. The caution lights operate on current from the dc primary bus, through circuit breakers, marked LOW LEVEL, FWD and AFT under the general headings INDICATOR LIGHTS and FUEL, located on the cockpit center overhead dc circuit breaker panel

FUEL FILTER BYPASS CAUTION LIGHTS

The fuel filter bypass caution lights, marked FWD FUEL BYPASS and AFT FUEL BYPASS, are located on the caution-advisory panel (see figure FO-4). The fuel filter bypass caution light will illuminate whenever the respective system filter screen has become partially clogged and fuel bypass is imminent. The caution lights are tested by the master TEST button on the caution panel and are powered by the 28 volt dc primary bus through circuit breakers, marked FWD and AFT under the headings INDICATOR LIGHTS and FUEL BY-PASS, located on the overhead dc circuit breaker panel.

FUEL TRANSFER

Complete control by the pilots of fuel transfer is achieved by use of three dc motor-operated gate valves located in the fuel ejector motive flow lines. Fuel transfer from auxiliary to main tanks is accomplished by fuel ejectors, with motive flow obtained from the boost pumps. Fuel in the forward auxiliary tank can be transferred only to the forward main tank. Overfilling of the forward main tank is prevented by use of a float-operated shutoff valve. Fuel from the aft auxiliary tank can be transferred to either the forward or aft main tank. Overfilling of either main tank is prevented by use of float-operated shutoff valves. With either main tank level above 1600 pounds, any attempt to transfer fuel will result in a transfer to the auxiliary tank in use due to the inability of the fuel to pass the float-operated gate valve in the respective main tank.

CAUTION

Do not transfer fuel when main tank level is above 1600 pounds, as fuel may be forced overboard through the auxiliary tank vents if the auxiliary tank fuel level is high.

Fuel Transfer Switches

Two fuel transfer switches are located on the fuel management panel. The switch, marked AFT AUX, OFF, controls the transfer of fuel from the aft auxiliary tank to the aft main tank. The switch, marked FWD AUX, OFF, AFT AUX, controls the transfer of fuel from the forward auxiliary tank and the aft auxiliary tank to the forward main tank. The switches are powered by 28 volts dc from the primary bus and are protected by circuit breakers, marked TRANSFER, 1, 2, 3, located on the overhead circuit breaker panel.

NOTE

The position of the fuel transfer lines may cause fuel to feed from the auxiliary to the main fuel tanks without the use of transfer system. This condition occurs when the fuel level in the auxiliary tank is above the fuel transfer lines and at the same time the fuel level in the main tanks is lower.

FUEL DUMP SYSTEM

Fuel may be dumped at a limited rate from either main tank, or from both main tanks simultaneously, through a single overboard drain located in the trailing edge of the right sponson. Fuel flow is provided by the fuel boost pumps. Two manually-operated dump valves, one for each main tank, are located overhead in the forward area of the cabin. The dump system is activated by holding or locking over center one or both of these valves in the open position. Rate of dump from one tank is approximately 80 pounds per minute and from both tanks approximately 160 pounds per minute.

PRESSURE REFUELING SYSTEM

The pressure refueling system consists of one external pressure fueling adapter, four dual high level sensors, and four pressure refueling and defueling valves, one in each tank. The location of the pressure fueling adapter directly beneath the cabin door permits refueling while the helicopter is in a hover. Each dual high level sensor contains one primary and one secondary valve, each of which can be solenoid-operated during precheck or float-operated by fuel level. The shutoff capability of the system is tested by positioning the high level shutoff switch, located above the fueling adapter, to PRI TEST and SEC TEST independently. This energizes the corresponding solenoids of all four dual high level sensors. The solenoids raise the floats electrically, which in turn close the pressure refueling/defueling valves, stopping flow into all tanks. When the switches are returned to the OFF position, the floats operate in conjunction with the fuel level of each tank. When each tank reaches full capacity, its dual high level sensor closes its pressure refueling and defueling valve and stops fuel flow into that tank. Electrical power for the system TEST is supplied by the 28 volt dc primary bus through the HI-LEVEL SHUT-OFF TEST circuit breaker on the overhead circuit breaker panel. A pressure regulator and surge control valve are installed in the pressure refueling lines to prevent damage to the fuel system by faulty or improperly operated pressure refueling equipment.

NOTE

Make sure fueling pressure is not over 50 psi.

PRESSURE DEFUELING SYSTEM

Limited pressure defueling is accomplished by attaching the pressure fueling nozzle to the pressure refueling adapter and applying suction. Fuel will then be drawn from all tanks simultaneously until one of the four tanks is empty. Defueling will then cease from the remaining tanks due to air ingestion from the empty tank. The amount of fuel remaining in the various tanks depends entirely on the initial quantities of fuel in each tank.

ELECTRICAL POWER SUPPLY SYSTEM

The electrical power supply system consists of an alternating current power supply system (figure FO-6), and a direct current power supply system (figure FO-7). Two permanent magnet generators (PMG) supply 115/200-volt, three-phase, 400 Hz ac power to the electrical system. Three transformers provide 26 volt single-phase ac power. Both generators also supply 28 volt dc power to the electrical system control circuits. Two converters provide 28 volt dc control and operating power. One battery supplies 24 volt dc power.

ALTERNATING CURRENT POWER SUPPLY SYSTEM

Alternating current power is supplied by two generators designated as No. 1 and No. 2. Associated system components are designated in a similar manner. System operation is automatic, and control switches on the overhead switch panel and monitoring caution-advisory light capsules on the caution-advisory panel are provided. Normally, associated primary and monitored bus loads are assumed by each generator. Primary bus loads are those that are essential for night instrument flight and monitor bus loads are not essential for this type of flight. If either generator fails, its primary bus load is automatically transferred to the remaining generator. With a failed generator, all monitor bus loads are automatically dropped. An ac external power receptacle permits use of an ac external power unit for ground power application.

Supervisory Panels

The supervisory panels, designated No. 1 and No. 2, provide control for all ac relays and one dc relay in the electrical system. When the No. 1 generator is developing normal ac power and the generator switch is placed ON, dc power from the same generator will be used by the No. 1 supervisory panel to close the No. 1 generator contactor relay. Closing the No. 1 generator contactor relay permits the No. 1 generator to power the No. 1 ac primary bus and to deliver 28 volts dc to the ac monitor bus relay. In addition, it opens the No. 1 generator caution light circuit causing the light to go out. The No. 2 supervisory panel operates the same way to power the No. 2 ac primary bus and to turn out the No. 2 caution light. Dc power from the No. 2 supervisory panel also closes the ac monitor bus control relay, which permits 28 volts dc from the No. 1 supervisory panel to close the dc monitor bus relay and the No. 1 and No. 2 monitor bus contactor relays. Therefore, 28-volt dc power is required from both the No. 1 and No. 2 supervisory panels to energize the ac and dc monitor buses. If either generator fails to produce 28 volts dc, the primary dc bus supplies back up dc voltage to each supervisory panel through circuit breakers. The circuit breakers, located on the overhead circuit breaker panel, are marked 1 and 2 under the general heading PMG BACK-UP. The supervisory panels provide protection for the electrical system. Ac power delivered to the panel from its associated generator is monitored by the panel for open phase, over voltage, and under voltage at all times. The panel monitors for underfrequency when the helicopter is on the ground with its main landing gear struts compressed activating the scissor switches. If any of the monitored conditions are not normal, the generator contactor relay will open, taking the associated generator off the line, de-energizing all monitor buses, and illuminating the associated generator caution light. In event of a generator failure, primary ac bus loads will be switched automatically to the remaining generator.

NOTE

> Should a low voltage condition occur in either generator, there will be a six-second delay before the load is shifted to the other generator.

Generators

Two 115/200-volt, three-phase, 400 Hz, PMG, ac generators are mounted on and are driven by the accessory section of the main gear box. Generator output varies with temperature and altitude (approximately 20 KVA at sea level to 25 KVA at 15,000 feet altitude). Generator ac voltage is delivered to the associated supervisory panel and generator contactor relay. The permanent magnet sections of the generators also develop dc power to be used in the control circuits. This dc power is delivered to the associated supervisory panel. The auxiliary power unit (APU) drives the generators through the main gear box accessory section when the rotor rpm is below 100% N_r. When the rotor speed reaches 100% N_r, the accessory section is driven through the main gear box.

Generator Switches The generator switches are located on the overhead switch panel (figure FO-3) under the general heading 1 GEN 2 and have marked positions ON-OFF, RESET-TEST. Placing the switch in the ON position puts the respective generator on the line by closing the generator contactor relay. The OFF-RESET position turns the generator off and resets the cycle. The TEST position is only used for maintenance.

Generator Caution Lights Two generator caution lights, marked 1 GEN and 2 GEN respectively, are located on the caution-advisory panel. These lights will illuminate whenever the associated generator is taken off the line by the opening of the generator contactor relay, which causes the caution light circuit to be completed. The generator caution lights are powered by the dc primary bus and are protected by circuit breakers, marked No. 1 and No. 2 under the general headings GENERATOR and INDICATOR LTS, on the overhead circuit breaker panel.

Autotransformers

Four autotransformers in the ac system convert 115 volt power from the primary ac buses to 26 volts. Three of the autotransformers are powered by the No. I primary bus and are protected by circuit breakers on the copilot's overhead circuit breaker panel, marked 26 V XMFR and RADIO XMFR 1 and 2, 26V ϕB under the general heading No. 1 AC PRI. The autotransformer protected by the circuit breaker marked 26V XMFR, supplies 26 volts ac to the copilot's course indicator azimuth card, copilot's ID-250 RMI card, primary hydraulic pressure gage, transmission oil pressure gage, and the No. 1 engine torque sensor and oil pressure gages. The autotransformers protected by the circuit breakers marked RADIO XMFR 1 and 2, 26 V ϕB, supply 26 volts ac to the pilot's course indicator azimuth card, pilot's ID-250 RMI card, both pilot's ID-250 needles, TACAN (AN/ARN-52(V) azimuth and DME, doppler radar (AN/APN-175(V)-1), heading information to the navigation computer (AN/AYN-1) and VOR inputs to the AN/AYN-2 computer. Two radio autotransformers are installed to provide a redundant source of 26 volts ac power for the navigation instruments. The output of each autotransformer is applied to the contacts of the RADIO XFMR switch on the overhead switch panel. The switch selects either the No. 1 or the No. 2 autotransformer output to energize the transformer fail relay and the NAV ϕB and PILOT ϕB circuit breakers. Should the relay become de-energized, 28 volts dc from the RADIO XMFR CAUT circuit breaker is applied to the caution/advisory panel, illuminating the RADIO XFMR caution capsule to indicate loss of 26 volts ac from the selected autotransformer. The fourth autotransformer is powered by the No. 2 primary bus and is protected by a circuit breaker in the pilot's overhead circuit breaker panel, marked 26V XMFR under the general heading NO. 2 AC PRI. This autotransformer supplies 26 volts ac to operate the auxiliary and utility hydraulic pressure gages and the No. 2 engine oil pressure and torque sensor gages.

AC Utility Power Receptacles

Two 115 volt ac 400 Hz utility receptacles are provided. The receptacles receive power from the No. 1 ac monitor bus through two circuit breakers on the copilot's overhead circuit breaker panel, marked CABIN and XMSN under the general headings UT RECP and NO. 1 AC MON.

ALTERNATING CURRENT CIRCUIT BREAKERS

Alternating current circuit breakers are located on the pilot's and copilot's overhead circuit breaker panels (figure FO-8).

DIRECT CURRENT POWER SUPPLY SYSTEM

Direct current power is supplied by two 28 volt, 200 ampere converters, designated as No. 1 and No. 2, which are powered by respective No. 1 and No. 2 ac primary buses. The dc system operation is automatic and control switches and converter caution lights are provided. Normally, primary and monitor bus loads are assumed by both converters. Primary bus loads are those loads essential for flight under night instrument conditions, while monitor bus loads are those not essential for this type of flight. If one converter fails, the associated reverse current cutout relay will remove the failed converter from the dc primary bus. The remaining converter will assume the dc primary bus loads, and the dc monitored bus loads will be dropped. The battery can provide power to the dc primary bus when no other source is available. The dc external power receptacle and associated circuitry permit use of an external power unit for ground power application.

Converters

Two 200 ampere, 28-volt dc converters are located in the electronics compartment. The converters require an ac input from the generators or from an ac external power source. The two converters are designated as No. 1 and No. 2, and the associated components are designated in a similar manner. Both converters normally supply power to the dc primary bus. The dc primary bus supplies power to the dc monitor bus. The No. 1 converter receives three-phase power from the No. 1 ac primary bus, and the No. 2 converter receives three-phase power from the No. 2 ac primary bus. The ac input is stepped down, rectified, and filtered within each converter, and the dc output is fed to the associated reverse current cutout relay. During normal operation, dc power from the reverse current cutout relay is fed to the dc primary bus. If either converter or either ac generator fails, the monitor bus is automatically dropped from the line and an appropriate caution light will illuminate. The reverse current cutout relay prevents current flow from the dc primary bus to a failed converter. The dc monitor bus will be dropped from the line if the ac monitor bus relay is open. The dc monitor bus relay must be closed for the monitor bus to receive power. Power to close this relay comes from the dc primary bus and is routed through the No. 2 and No. 1 reverse current cutout relays. If either converter, either reverse current cutout relay, or either ac generator is inoperative (or if either converter switch is OFF), the dc monitor bus will be dropped from the line. The No. 1 converter is protected by a circuit breaker on the copilot's overhead circuit breaker panel, marked No. 1 CONVERTER under the general heading No. 1 AC PRI. The No. 2 converter is protected by a circuit breaker on the pilot's overhead circuit breaker panel, marked No. 2 CONVERTER under the general No. 2 AC PRI.

Converter Switches The converter switches, located on the overhead switch panel under the general heading 1 CONVERTER 2, have marked positions ON and OFF.

Converter Caution Lights Two converter caution lights marked #1 CONV and #2 CONV are located on the caution-advisory panel. Failure of a converter, or reverse current cutout relay (or turning a converter switch OFF), will illuminate the associated caution light.

Battery

A 24 volt, 22 ampere hour nickel cadmium battery, located in the nose section forward of the pilot's compartment, is accessible from outside the helicopter. Battery power is used for limited ground operations, when no external power is available, and as an emergency source of power to the dc primary bus. The battery bus is continually energized and provides power for the anchor lights. The battery bus is connected to the dc primary bus when the battery switch is ON.

Battery Overtemperature Warning System When a battery temperature of 135°(±5°)F is detected by the battery overtemperature sensing unit, a BAT OVTEMP light on the caution panel will go on. At this time, the battery must be removed from its charging source by placing the battery switch OFF. The BAT OVTEMP caution light will go out when the battery cools to about 95°F or when all power is secured to the helicopter and the battery switch is turned off. The caution light is powered by the dc primary bus.

Battery Switch The battery switch, located on the overhead switch panel, is labeled BATTERY, with positions marked ON and OFF.

DC Utility Power Receptacle

Three 28 volt dc utility receptacles are installed. The receptacles receive power from the dc monitor bus through three circuit breakers on the pilot's overhead circuit breaker panel, marked COCKPIT, CABIN, and XMSN under the general headings UT RECEPTACLE, MON and DC.

Direct Current Circuit Breakers

Direct current circuit breakers are located on all three overhead circuit breaker panels and on the battery bus circuit breaker panel.

EXTERNAL POWER

External Power Switch

The external power switch is located on the overhead switch panel, under the heading EXT PWR, with positions marked ON and OFF. The external power switch is protected by a circuit breaker on the overhead circuit breaker panel, marked EXT PWR under the general heading DC PRI BUS.

External Power Advisory Light The external power advisory light, located on the caution-advisory panel and marked EXT POWER, will illuminate when the external power switch is ON and external power is being supplied to the aircraft.

AC External Power An ac external power receptacle, located on the left side of the fuselage aft of the sponson, is used to connect 115/200 volt, three-phase, 400 Hz power to the helicopter. A phase sequence relay is incorporated to sense proper external power phase rotation. When the phase rotation is correct, the EXT PWR switch is ON, and the BATT switch is ON, three-phase power will be admitted to the aircraft and the ac systems will function normally. The battery switch must be ON to power the dc primary bus, and the external power switch must be ON to permit the dc primary bus to supply control power to the phase sequence relay. With the converter switches ON, the dc monitor bus control relay and the dc monitor bus relay will close and the dc power system will function normally. If external power is to be used for an extended period of time, the battery switch should be placed OFF to avoid possible overcharging.

DC External Power A dc external power receptacle, located on the right side of the fuselage below the pilot's window, is used to connect 28 volt dc power to the helicopter. With the external power switch ON, 28 volt dc power from an external power source is delivered through the dc power receptacle to illuminate the external power advisory light, close the external power relay, energize the dc primary bus, and close the dc monitor bus relay; providing power to the dc monitor bus.

CAUTION

When securing either ac or dc external power, secure the EXT PWR switch before removing the power cables to prevent possible arcing and damage to the external power receptacles.

UTILITY HYDRAULIC SUPPLY SYSTEM

The utility hydraulic supply system (figure 1-14) provides hydraulic pressure for all hydraulic equipment not included in the flight control servo hydraulic systems. The utility hydraulic system reservoir (9, figure 1-12), located aft of the main gear box, has a capacity of 3.05 gallons of hydraulic fluid. The utility hydraulic pump is located on the accessory drive section of the main gear box and provides 3000 psi hydraulic pressure. An oil cooler is provided in the hydraulic line to maintain utility hydraulic oil temperatures within limits. The blower in the oil cooler operates on power from the No. 1 ac primary bus, and the control relay is actuated by power from the dc primary bus. The blower operates continuously when these buses are energized. The utility hydraulic system operates the main landing gear, nose landing gear, APU start system, ramp actuating system, and the rescue hoist.

UTILITY HYDRAULIC PRESSURE INDICATOR

The utility hydraulic pressure indicator (28, figure FO-4), located on the instrument panel, operates on 26 volt ac power from an autotransformer. The gage, marked UTI, indicates pressure in the utility hydraulic system in psi. The indicator receives electrical power from the No. 2 ac primary bus through a circuit breaker, located on the pilot's AC circuit breaker panel, under the heading HYD PRESS IND and marked UT.

FLIGHT CONTROL SYSTEM

The flight control system is divided into three systems: The main rotor flight control system, the tail rotor flight control system, and the flight control hydraulic supply systems. When the automatic flight control

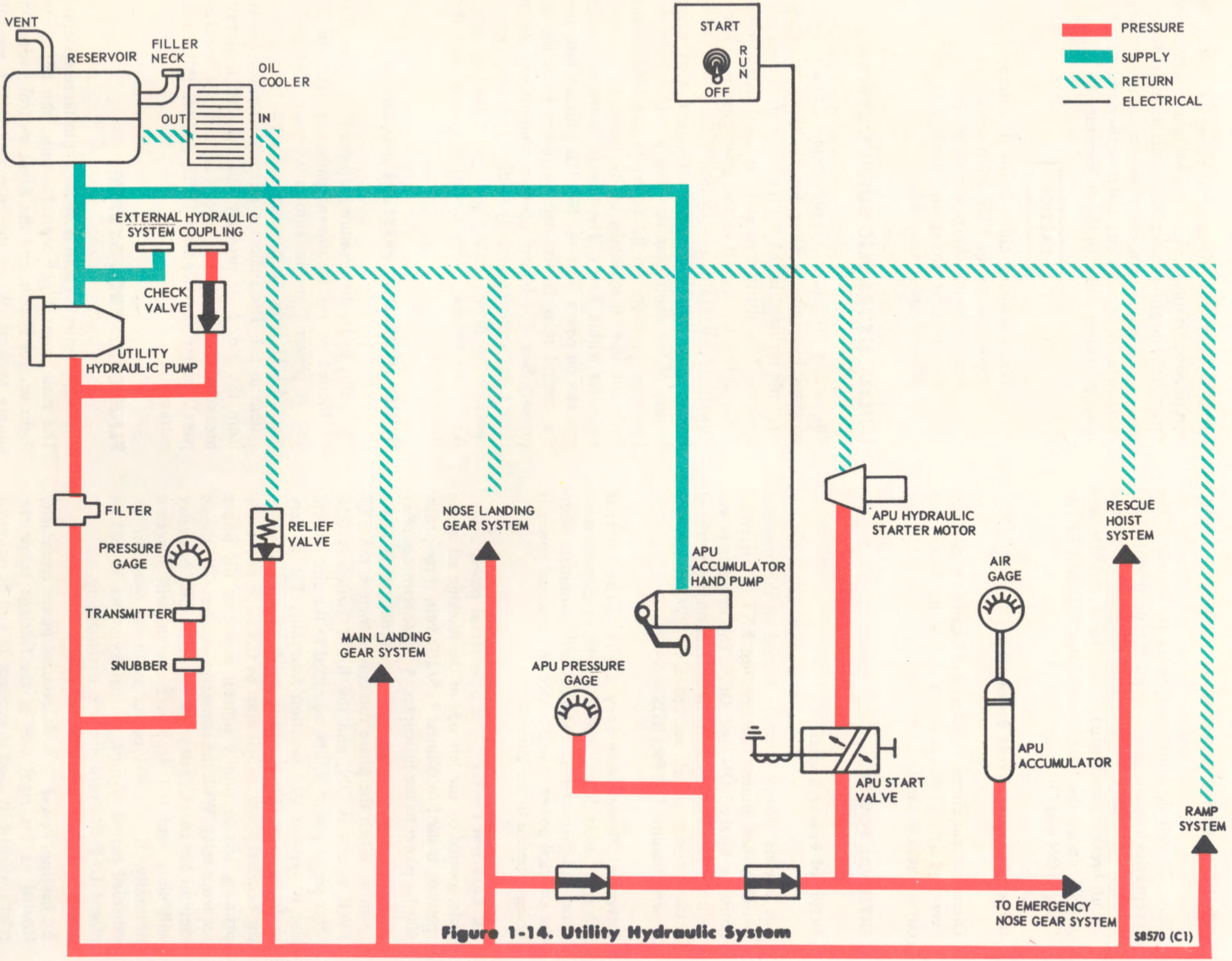


Figure 1-14. Utility Hydraulic System

system and coupler system are engaged, they provide corrections of limited authority to the flight control system, causing the helicopter to respond in a stable manner to the maneuver called for by the position of the cyclic stick or referenced ground speed from AFCS control panel drift pots. This equipment also functions to provide a constant altitude. The description and operation of the automatic flight control system and coupler system are included in the paragraphs, **AUTOMATIC FLIGHT CONTROL SYSTEM (AFCS) AND COUPLER SYSTEM** in this section. A cyclic stick trim system is installed.

MAIN ROTOR FLIGHT CONTROL SYSTEM

The main rotor flight control system provides both vertical, lateral, and longitudinal control. Control motions from the collective pitch lever for vertical control, and from the cyclic stick for lateral and longitudinal control, are combined in a mixing unit, located in the AFCS control compartment aft of the pilot's seat, and are transmitted to the main rotor assembly by mechanical linkage. Control action is assisted by two hydraulically-operated flight control servo systems. The main rotor flight controls terminate at the stationary swashplate of the main rotor head. Control action is transmitted through the rotating swashplate and linkage on the main rotor hub to the blades.

Collective To Yaw Coupling

A collective to yaw coupling, located in the mixing unit, provides automatic tail rotor blade angle changes to compensate for changes in collective pitch settings. In flight, the coupling is irreversible in that collective pitch motion will act to displace the tail rotor but tail rotor pedal motion will not affect main rotor collective pitch blade angle.

Collective To Cyclic Coupling

A bias in the collective to cyclic coupling is incorporated in the mixing unit to apply a slight left roll correction when the collective pitch is increased.

Collective Pitch Levers

Two collective pitch levers are located in the pilot's compartment. Both levers operate simultaneously to change the collective pitch of the main rotor blades. A friction lock on the pilot's collective pitch lever can be rotated to apply friction to prevent the collective pitch levers from creeping while in flight.

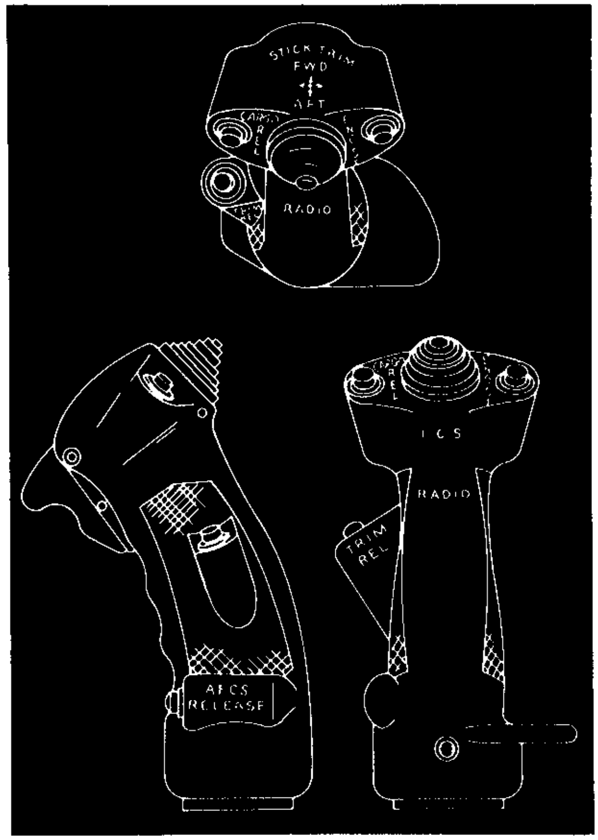


Figure 1-15. Cyclic Stick Grip

Cyclic Sticks

The cyclic sticks provide lateral and longitudinal control of the helicopter. Moving the cyclic stick in any direction tilts the tip path plane of the main rotor blades in that direction and moves the helicopter in the same direction. The stick grip (figure 1-15) contains pushbutton and thumb-operated switches for controlling various equipment installed in the helicopter.

Cyclic Stick Trim System

The cyclic stick trim system holds the stick in a selected trim position. Two actuators are hydraulically powered by the auxiliary servo system and energized electrically from the dc primary bus. One actuator positions the cyclic stick laterally, and the other positions the cyclic stick fore and aft. The actuators are operated by a four-way cyclic trim switch mounted on both the pilot's and copilot's cyclic stick grips. To trim the cyclic stick, the cyclic trim switch is pushed in the direction of desired cyclic stick movement, causing the actuators to move the stick until the cyclic trim switch is released. The cyclic stick may be manually displaced from the trimmed position against a resistance force caused by spring compression, which increases progressively with stick displacement. The spring action provides cyclic stick feel and amounts to approximately 1.5 pounds initial force, plus 0.5 pound for each one inch of cyclic stick displacement. When the pressure on the cyclic stick is released, spring action returns the stick to the original trim position. The cyclic stick trim system will operate as long as there is both dc power to the primary bus and auxiliary hydraulic pressure to the actuators.

Cyclic Stick Trim Master Switch A cyclic stick trim master switch, marked STICK TRIM MASTER, ON and OFF, is located on the overhead switch panel (see figure FO-3). When the switch is ON, the cyclic stick is held in position. When the switch is OFF, the force gradient system is inoperative and the cyclic trim system will not maintain the position of the stick.

Cyclic (Beeper) Trim Switches The cyclic trim switches, located on the pilot's and copilot's cyclic stick grips (figure 1-15), have marked positions FWD, AFT, L, and R. The four-way thumb switch is spring-loaded to the center (off) position. When the switch is placed in any of the four positions, hydraulic pressure will drive the cyclic stick in the same direction. Release of the switch stops stick motion. The action of the cyclic stick trim system will then function about this location of the cyclic stick. The cyclic trim switches receive electrical power from the dc primary bus through a circuit breaker, marked BEEPER-TRIM, located on the overhead dc circuit breaker panel.

Cyclic Trim Release Button The spring-loaded, pushbutton switches are located on the pilot's and copilot's cyclic stick grips (figure 1-15) and marked TRIM REL. Cyclic trim position can be changed by depressing the cyclic trim release button, manually moving the stick to the new position, and then releasing the cyclic trim release button. The cyclic trim system will then hold the new selected position of the cyclic stick. The cyclic trim release button controls dc primary bus power to the trim actuators.

TAIL ROTOR FLIGHT CONTROL SYSTEM

The functions of the tail rotor flight control system are to compensate for main rotor torque and to provide a means for changing the heading of the helicopter. The torque developed by the main rotor blades turning counterclockwise tends to rotate the fuselage in a clockwise direction. Any change in power setting will vary the amount of main rotor torque. To compensate for torque variations, the pitch and resulting thrust of the tail rotor blades can be increased or decreased. Turns are accomplished by increasing tail rotor thrust, which overcompensates for main rotor torque and changes the heading of the fuselage to the left, or by decreasing the tail rotor thrust, which undercompensates for the main rotor torque and changes the heading of the fuselage to the right. Tail rotor control pedal movements are transmitted to the tail rotor assembly by mechanical linkage and cables. Control action is assisted by the auxiliary servo system only. A hydraulic damping device incorporated in the auxiliary servo resists abrupt movements of the pedals, which could cause sudden yaw acceleration and possible damage to the helicopter. The pedal damper is inoperative when the auxiliary servo system is inoperative or shut off. Yaw compensation is accomplished by mechanical linkage in the mixing unit, which automatically changes tail rotor blade angles for changes in collective pitch without moving the pedals. A tail rotor negative force gradient system is installed to relieve the pilot of tail rotor forces created by aerodynamic loads when the auxiliary servo system is inoperative. Because of this, when the system is checked on the ground with tail rotor stationary and the auxiliary servo off, a negative spring centering effect is created. The normal tendency of the pedals is then to go to either extreme. Under these conditions, considerable force is required to push the pedals from the extreme position; however, the force will decrease as the neutral pedal position is approached.

Tail Rotor Pedals

The tail rotor pedals change the pitch and thrust of the tail rotor and consequently the heading of the helicopter. Electrical switches, mounted on the pedals, null the reference heading retention feature of the automatic flight control system when feet are placed on the pedals. Toe brake pedals for the main landing gear wheel brakes are mounted on both the pilot's and copilot's pedals.

Tail Rotor Pedal Adjustment Knobs Pedal adjustment knobs are located on each side of the fuselage, (16 and 31, figure 1-23). The adjustment knobs are connected to mechanical linkages that provide for fore-and-aft adjustment of the pedals. The knobs are rotated to the right, as indicated by the arrow marked FWD, for forward adjustment and to the left, as indicated by the arrow marked AFT, for aft adjustments. Adjustments should be made with feet away from the pedals to avoid damage to the adjustment cables, striker plates or microswitches.

FLIGHT CONTROL HYDRAULIC SYSTEMS

A primary and an auxiliary flight control servo hydraulic system (figure 1-16) provides power boost to operate the controls. The servos also prevent feedback of rotor system vibratory loads to the controls. Each servo system operates from an independent hydraulic system and utilizes similar servo hydraulic units to vary the main and tail rotor blade pitch through the mechanical linkage of the flight control system. The servo unit output is connected to the flight control linkage to provide power boost. The continuity of the direct control linkage is maintained from the controls in the pilot's compartment through the auxiliary and the primary servos to the main rotor blades, except for a slight amount of endplay at each servo unit to permit the pilot valves to move before the direct control linkage. Normally, both servo systems are in operation.

Primary Flight Control Servo System

The primary flight control servo system consists of three hydraulic servo units which connect the flight control linkage to the stationary swashplate of the main rotor assembly. All three servo units respond simultaneously and move in the same direction in response to movements of the collective. Two of the servo units (lateral servo units) respond simultaneously, but move in opposite directions in response to lateral movements of the cyclic stick. One of the servo units (fore-and-aft servo unit) responds to the fore-and-aft movements of the cyclic stick. Since all three movements can occur simultaneously through the action of the mixing unit, the position of any primary servo unit is the result of the combined input of the cyclic stick and collective. This results in a primary servo system in which any one servo may have an effect on both collective pitch and cyclic (lateral or fore-and-aft) pitch. The servos provide power boost only to the main rotor flight control system. The primary servo hydraulic pump is driven by the accessory section of the main gear box. The primary hydraulic system reservoir, mounted aft of the main gear box, has a capacity of approximately 0.45 gallon of hydraulic oil. A light on the caution panel, marked PRI-HYD PRESS, illuminates when primary servo pressure falls below 1,000 psi.

WARNING

Helicopters are limited to no PRIMARY SERVO OFF operation except for functional check flights. This limitation is necessary to minimize the vibratory loads to the mixer unit.

Auxiliary Flight Control Servo System

The auxiliary flight control servos, consisting of a bank of four hydraulic servo units, are located between the pilot's controls and the flight control system mixing unit. Each control input acts independently on the corresponding auxiliary servo. The collective positions the collective servo. The tail rotor control pedals position the directional servo. The cyclic stick positions either, or both, the fore-and-aft servo and the lateral servo. This results in an auxiliary servo system in which only one servo has an effect on collective pitch, one on fore-and-aft cyclic pitch, one on lateral cyclic pitch and one on tail rotor pitch. They provide power boost to both the main and tail rotor flight control systems and the means of introducing AFCS corrective signals into the flight control systems. The auxiliary servo hydraulic pump is driven by the main gear box accessory section. The auxiliary hydraulic system reservoir, located aft of the primary hydraulic system reservoir, has a capacity of approximately 0.45 gallon of hydraulic oil. A light on the caution panel, marked AUX-HYD PRESS, illuminates when auxiliary servo hydraulic pressure falls below 1,000 psi.

Flight Control Servo Switches

The primary and the auxiliary flight control servo systems are controlled by the servo switches, marked SERVO, located on the pilot's and copilot's collective pitch lever grip (figure 1-5). The marked switch positions are PRI OFF and AUX OFF. Both servo systems are normally in operation with both switches in the unmarked center (ON) position. To turn off the primary servos, either of the switches is placed in the forward (PRI OFF) position. To turn off the auxiliary servos, the switch is placed in the aft (AUX OFF) position. The systems are interconnected electrically in such a way that regardless of switch position, it is impossible to turn either system off unless there is 1000 psi in the remaining system. The servo shutoff valves operate on current from the dc primary bus. Protection is provided by circuit breakers on the overhead dc circuit breaker panel, marked SERVO CUT-OFF, PRI and AUX.

Servo Hydraulic Pressure Indicators

The primary and auxiliary servo hydraulic pressure indicators (30 and 29, figure FO-4), located on the instrument panel, operate on 26 volts ac from the autotransformers powered by the ac primary buses.

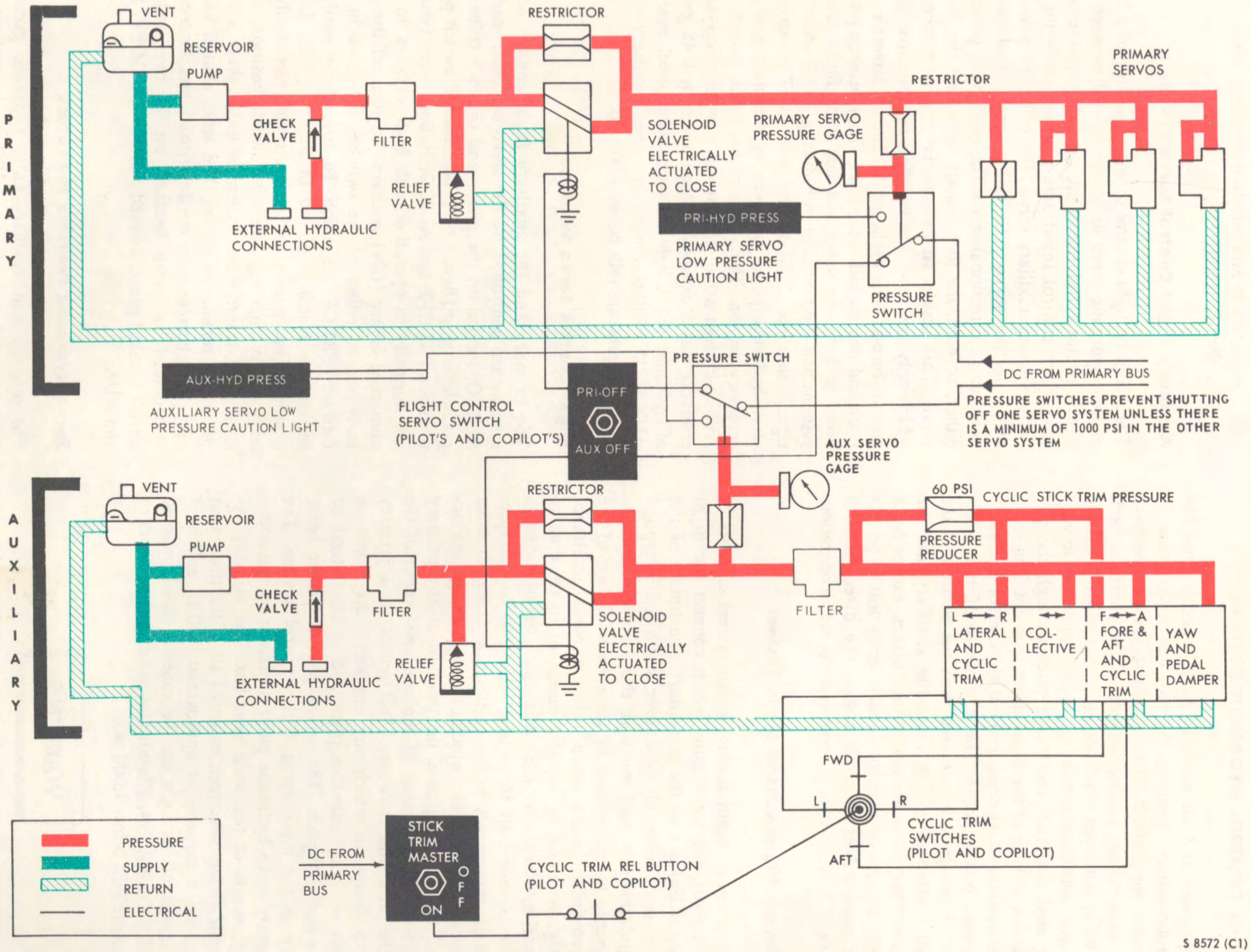


Figure 1-16. Flight Control Servo Hydraulic Systems

The primary hydraulic pressure indicator is protected by a circuit breaker on the copilot's overhead circuit breaker panel, marked HYD PRESS IND PRI. The auxiliary hydraulic pressure indicator is protected by a circuit breaker on the pilot's overhead circuit breaker panel, marked HYD PRESS IND AUX.

Primary and Auxiliary Servo Pressure Caution Lights

The primary and auxiliary servo pressure caution lights, marked PRI HYD PRESS or AUX HYD PRESS, are located on the caution panel (see figure FO-4). When servo pressure in either system drops below 1000 psi, the associated light will illuminate.

AUTOMATIC FLIGHT CONTROL (AFCS) AND COUPLER SYSTEMS

The automatic flight control system (AFCS) provides added stability in pitch, roll, and yaw, plus attitude, heading, and altitude hold. The coupler system is used in conjunction with the AFCS for hovering operations when accurate ground speed and altitude control is desired.

AUTOMATIC FLIGHT CONTROL SYSTEM (AFCS)

The AFCS used in this helicopter differs from the autopilot used in fixed wing aircraft in that it may be engaged at all times, has less control authority than the primary flight control system, and may be easily overridden through normal use of the flight controls. The pilot has direct control of the system at all times and can engage or disengage the entire AFCS or any channel, as desired, by means of switches on the AFCS control panel, channel monitor panel, cyclic sticks, and collective pitch levers. The AFCS indicators provide the pilot and copilot with visual indication of all AFCS signals. The AFCS has two modes of operation: (1) Attitude and directional stabilization, and (2) barometric altitude hold. Attitude and directional stabilization is controlled through the pitch, roll, and yaw channels, and barometric altitude hold is controlled through the collective channel. The AFCS is capable of maintaining the barometric altitude of the helicopter within ± 25 feet or 5% of the altitude, whichever is greater, during straight unaccelerated flight, or when hovering out of ground effect by utilizing barometric altitude reference. In the pitch and roll channels, the fuselage attitude is held constant by comparing the actual attitude signal received from the vertical gyro with the reference attitude signal received from the cyclic stick position sensor. Automatic pitch and roll attitude stability correction occurs any time the helicopter is displaced from the reference attitude. Pitch and roll gyro information source is selected on the channel monitor panel. In the yaw channel the heading is held constant by reference signals developed within the ASN-50 compass system. See OS-27 These reference signals pass through the YAW TRIM control on the AFCS control panel prior to reaching the AFCS. When the AFCS is engaged and controlling the heading, the YAW TRIM knob may be used to make minor heading changes. When the pilot establishes a new reference heading using the pedals, the yaw synchronizer becomes active and the yaw channel goes into the synchronizing mode. During the synchronizing mode, the yaw rate gyro develops a signal proportional to the manual heading displacement rate of the helicopter. This signal initiates an open-loop spring condition that produces a proportional feedback force at the pedals. As the pilot presses either pedal, he feels the proportional feedback force opposing the pedal pressure applied. The feedback force remains until the pilot has established the new reference heading. Heading stability correction occurs any time the helicopter is displaced from the desired reference heading. In the collective channel, the altitude of the helicopter is held constant by signals developed from the altitude controller, which senses changes in barometric pressure from the engage point. Automatic barometric altitude stability correction occurs any time the helicopter is displaced up or down from the reference altitude. AFCS utilizes power from the No. 1 ac primary bus and the dc primary bus. A thermal time delay relay is incorporated to allow approximately 120 seconds for the vertical gyros to reach a stabilized state before dc power is applied to the system. The AFCS ENG button may then be depressed to engage the pitch, roll, and yaw channels. The collective channel is engaged by depressing the BAR ALT ENG button. With BAR ALT engaged, collective friction should be removed so as not to inhibit collective movement. The AFCS must be engaged in order to engage the BAR ALT. Ac power to the AFCS is protected by circuit breakers, marked ϕA and ϕB, located on the copilot's overhead circuit breaker panel, under the general heading of AFCS. DC power to the AFCS is protected by a circuit breaker, marked AFCS, on the overhead circuit breaker panel.

COUPLER SYSTEM

Radar controlled ground speed and drift rate signals are fed from the AN/APN-175(V)-1 doppler through the cyclic coupler, allowing the pilot to set ground speed and drift rate. Radar signals from the AN/APN-171 radar altimeter are fed through the collective coupler as reference for the selected altitude between zero and 200 feet. The AFCS must be in operation before the coupler can be engaged. The doppler and radar altimeter must be on for normal operation. Should the doppler be inoperative a stable coupled hover can still be attained on the pitch and roll accelerometers. The coupler controls are on the AFCS control panel. The coupler output can be monitored on the AFCS indicator (in the AFCS mode) by placing the meter selector switch on the channel monitor panel in the CPLR position. The coupler is engaged by placing either or both of the coupler control switches located on the AFCS control panel in the ON position, and depressing the coupler engage button marked CPLR. These switches, one for the cyclic coupler and one for the collective coupler, are marked CYC CPLR and ALT CPLR. The barometric altitude controller, and the BAR ALT light will automatically be on when the collective coupler is engaged, and cannot be disengaged until after the collective coupler is disengaged. Upon engagement, the coupler will maintain the selected altitude, ground speed and drift relative to the surface. The altitude can be between zero and 200 feet by use of the altitude set knob. The speed and drift can be controlled up to approximately 10 knots in each direction by use of the speed and drift knobs on the AFCS control panel and by the crewman's hover trim control, which can change speed and drift up to approximately 8 to 9 knots in each direction from the setting of the speed and drift knobs on the AFCS control panel. The crewman's roll and pitch bias pots can also control the speed and drift up to three knots in each direction from that set on the AFCS control panel. The limited authority of the AFCS is supplemented by the beeper trim system to reposition the cyclic stick when the coupler output reaches half the total authority of the servo valve. The collective is repositioned through operation of the collective open loop spring. Both channels can be disengaged simultaneously by depressing the CPLR REL button on either collective stick.

Automatic Flight Control System Control Panel

The AFCS and coupler controls are located on the panel, marked AFCS CONTROL PANEL, mounted on the center console between the pilot and copilot. Five pushbutton switches, marked AFCS BAR ALT, CPLR, BAR OFF and HOVER TRIM ENG are located on the AFCS control panel. All except BAR OFF illuminate when engaged. When the AFCS button is pressed, the pitch, roll, and yaw channels become operative. Pressing the BAR ALT button engages the barometric altitude controller. Pressing the CPLR button engages the cyclic and collective coupler, if the coupler control switches are both on. In addition, the barometric controller will be engaged if the collective coupler switch is ON. Pressing the button marked HOVER TRIM ENG energizes the hover trim panel, illuminating the button and a red engage light on the top of the hover trim stick. The fifth pushbutton, marked BAR OFF, is used for permanent disengagement of the barometric altitude controller, provided the collective coupler is not engaged. The barometric altitude controller can be released momentarily by pressing and holding the button marked BAR REL on the pilot's or copilot's collective strick grips. Hover trim is disengaged by disengaging the coupler or by cycling the cyclic coupler control switch to OFF then ON. Two coupler control switches are provided for individual operation of the cyclic or collective channels of the coupler when it is engaged. The five remaining controls are knobs used for trimming various systems. The knobs are designed with characteristic shapes to enable identification by touch. Specific knobs and their identifying shapes and markings are as follows: DRIFT, bar shape; CG TRIM, clover leaf shape; SPEED, indented circle shape; YAW trim, triangular shape; ALTITUDE, cross shape. The DRIFT knob allows the pilot to control the lateral drift of the helicopter with an approximate ± 10 knot knob ground speed authority when the cyclic coupler is engaged. The CG TRIM knob is used to trim the pitch channel of AFCS for actual cg location. The SPEED knob allows the pilot to control the forward and aft ground speed with an approximate ± 10 knot authority when the cyclic coupler is engaged. The YAW knob enables the pilot to make small heading corrections in forward flight with the yaw channel of AFCS engaged. In a hover, the knob may be used for larger heading changes. One rotation of the knob turns the helicopter approximately 72°. The ALTITUDE knob allows the pilot to accurately select hovering altitudes and make altitude changes between zero and 200 feet. The knob has a graduated scale from zero to 200 feet with increments of 5 feet.

CHANNEL MONITOR PANEL

The channel monitor panel located on the starboard side of the cockpit underneath the pilot's window, provides switches that enable individual disengagement of the channels of the AFCS, switches for minor testing, a switch to select gyro input to the AFCS, and a switch to select the inputs to the hover indicators. The channel disengage switches, located in the forward row, enable the pilot to disengage individual channels as desired. The four hardover switches enable technicians to introduce hardover signals to the individual AFCS channels. The channel monitor test switch, located on the overhead switch panel, must be in the TEST position to power the hardover switches. The switches are protected by red safety covers that should be in the down position for flight. The gyro selector switch enables the pilot to select either the port or starboard gyro for pitch and roll inputs to the AFCS. The port gyro position selects the AN/ASN-50 gyro and is normally selected for flight. See 85-A1 The starboard gyro position selects the 1080Y gyro. The meter selector switch enables either AFCS or coupler inputs to be monitored on the hover indicators when they are in the A mode.

NOTE

> The position of the gyro selector switch has no effect on the inputs to the pilot's or copilot's AYN-2 flight director system or the yaw channel of AFCS.

INSTRUMENTS

Electrical instruments that operate on either ac, dc, or both, are protected by appropriately marked circuit breakers on the three overhead circuit breaker panels in the cockpits.

MAGNETIC COMPASS

A standby compass, located on the right side frame of the center windshield above the instrument panel, is marked TO BE USED AS A STBY COMPASS ONLY. A light switch is located at the lower right of the compass. A standby compass correction card is located at the upper right of the compass.

FREE-AIR TEMPERATURE GAGE

The free-air temperature indicating system consists of a gage, marked FREE AIR, and a temperature sensing bulb. The temperature sensing bulb, extending through the center windshield above the instrument panel, is a direct reading instrument calibrated in degrees Celsius.

CLOCKS

Three 8-day, 12-hour clocks are installed in this helicopter; two on the instrument panel, and one at the navigator's position.

PITOT-STATIC SYSTEM

There are two pitot-static systems. The pitot portion of the pilot's and copilot's systems are independent of each other, but the static portion of each system is common. Each pitot-static pressure system consists of a heated pitot-static tube, altimeter, airspeed and vertical velocity indicator. The pitot and static lines both originate at the pitot-static tubes. The opening at the head of the tubes furnishes ram air pressure, and ports near the center of the tubes furnish static pressure. The static system vents the airspeed, altimeter, and vertical velocity indicators to atmospheric pressure. The pitot-static tube on the right side of the cockpit canopy furnishes ram air pressure to the pilot's airspeed indicator and static pressure to the common static system. The pitot tube on the right side of the cockpit canopy also furnishes pitot and static pressures to the true airspeed transducer. The pitot-static tube on the left side of the cockpit canopy furnishes ram air pressure to the copilot's airspeed indicator and static pressure to the common static system. Capped tees in the lines in the electronics compartment and in the cabin permit draining moisture from the lines. The AFCS barometric altitude control sensing unit is connected into the common static system. Resistance-type heaters in the pitot-static tubes, controlled by the PITOT HEAT switch on the overhead control panel, prevent formation of ice at the openings. Power for the pitot-static tube heaters is supplied by the dc primary bus system, through the circuit breakers marked, ICE PROTECTION, PITOT HEAT 1 and 2, on the overhead circuit breaker panel.

VERTICAL VELOCITY INDICATORS

Two vertical velocity indicators (5 and 51, figure FO-4), located on the instrument panel, indicate the rate of ascent or descent up to 3000 fpm. The vertical velocity indicators utilize the same static source as the altimeter.

ALTIMETER-ENCODER AAU-21/A OR AAU-32/A

One altimeter-encoder (figure 1-17) is installed in the pilot's instrument panel. The altimeter-encoder combines a conventional barometric type altimeter, possessing a counter-drum-pointer display, with an altitude reporting encoder in one self-contained unit. The 10,000- and 1,000-foot counters and the 100-foot drum provide a direct digital output and readout of altitude in units of 100 feet, from -1000 to 38,000 feet. The digital output is referenced to 29.92 in Hg and is not affected by change of barometric setting. The pointer repeats the indications of the 100-foot drum, and serves both as a vernier for the drum and as a quick indication of the rate and sense of altitude changes. Two methods may be used to read indicated altitude on the counter-drum-pointer altimeter: (1) read the counter-drum window, without reference to the pointer, as a direct digital readout in thousands and hundreds of feet; (2) read the thousands of feet on the two counter indicators, without referring to the drum, and then add the 100-foot pointer indication. The self-contained servo driven encoder provides altitude encoded in 100-foot units for automatic transmission when the AIMS/IFF transponder is interrogated on mode C. In case of power loss to the encoder servo system, a CODE OFF flag appears automatically in a window in the upper left portion of the display, indicating that altitude information is no longer being transmitted to the ground. In this condition, the instrument continues to function as a normal barometric altimeter. The barometer pressure is entered by use of a barometric set knob in the lower left front of the instrument case. The altimeter setting appears on counters in the window at the lower right of the display and has a range of settings from 28.1 to 31.0 in Hg. On those helicopters with an altimeter-encoder AAU-21A installed, an internal vibrator operators continuously whenever DC power is turned on. The vibrator minimizes internal mechanical friction, enabling the instrument to provide a smoother display during changing altitude conditions. Should vibrator failure occur, the altimeter will continue to function pneumatically, but a less-smooth movement of the instrument display will be evident with changes in altitude. The altimeter-encoder AAU-32/A does not require an internal vibrator due to its solid-state construction.

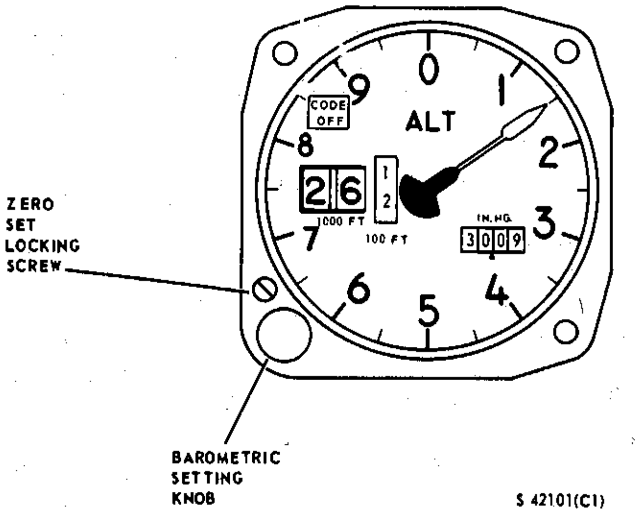


Figure 1-17. Altimeter-Encoder AAU-21/A or AAU-32/A

WARNING

If the internal vibrators of the altimeter-encoder (AAU-21/A) or altimeter (AAU-24/A) are inoperative due to either internal failure or DC power failure, the 100-foot pointers may momentarily hang up when passing through 0 (12 o'clock position). If the vibrators have failed, hangup of the 100-pointers can be lessened by tapping the case of the altimeters. Pilots should be especially watchful for this failure when the minimum approach altitude lies within the 800-1000-foot part of the scale (1800-2000 feet, etc.).

NOTE

When each 1,000-foot unit is nearly completed, the counter(s) abruptly index to the next digit. The counter-drum-pointer altimeter mechanism may also cause a noticeable pause of hesitation of the pointer due to the additional intermittent friction and inertial loads applied to the mechanism to turn over the 1000-foot counter. This effect may be more pronounced at 10,000-foot intervals where both counters are turned over simultaneously. This momentary pause is followed by a noticeable acceleration as the altimeter mechanism overcomes the counter wheel load and rolls the dial over to the next digit. The pause occurs during the "9" to "1" portion of the scale. The pause-and-accelerate behavior is normally more pronounced at high altitudes and high rates of ascent and descent. During normal rates of descent or ascent and at low altitudes, the effect will be minimal.

ALTIMETER AAU-24/A

One altimeter (figure 1-18) is installed in the copilot's flight instrument panel. The instrument is similar to the altimeter-encoder, except that it does not have an altitude-encoder nor the CODE OFF display mechanism. The indicated altitude on the altimeter is from -1000 to 38,000 feet. The altitude display, altimeter

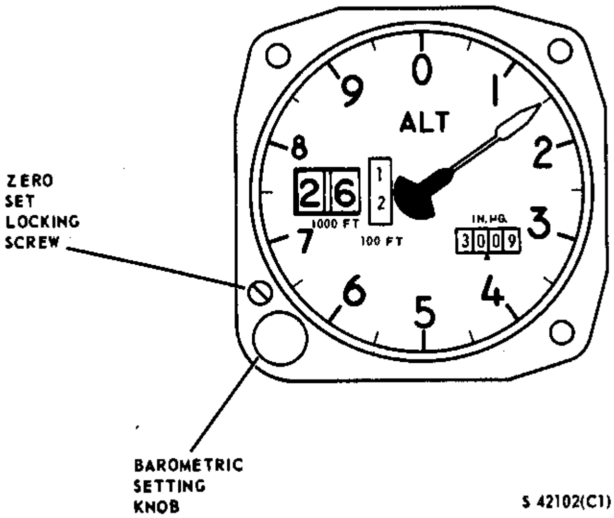


Figure 1-18. Altimeter AAU-24/A

setting, and vibrator considerations described for the altimeter-encoder also apply to the copilot's altimeter.

ATTITUDE AND HEADING REFERENCE SYSTEM (AN/ASN-50)

SEE OS-21

The AN/ASN-50 provides continuous attitude and heading information representing the displacement of the helicopter about the pitch, roll, and yaw axes. The system supplies attitude inputs to the doppler, radar set, AFCS, and the copilot's attitude indicator. Heading inputs are supplied to the VHF/NAV, TACAN, RMI's, course indicators, AFCS and the navigation computer. Power for the system is supplied from the No. 1 ac primary bus, the dc primary bus, and the two autotransformers that operate from the No. 1 ac primary bus. The system is protected by circuit breakers on the copilot's overhead circuit breaker panel. One circuit breaker, under the general heading NO. 1 AC PRI, is marked GYRO COMPASS, and the other under the general heading DC PRI, is marked GYRO COMPASS.

SEE OS-21

AN/ASN-50 OFF/GYRO FAST ERECT PANEL

The AN/ASN-50 OFF and GYRO FAST ERECT switches (figure 1-19), marked VERT GYRO and FAST ERECT, are on the instrument panel at the lower left corner of the fuel management panel. The VERT GYRO switch, with marked positions ON/OFF, enables the pilot to control power to the AN/ASN-50 displacement gyro after the power passes the AC GYRO COMPASS circuit breaker in the copilot's overhead circuit breaker panel. The OFF position of the VERT GYRO switch secures power to the displacement gyro but does not interfere with power to the compass control system. The FAST ERECT switch, with marked positions NORMAL/FAST ERECT, is spring-loaded to the NORMAL position. Holding the switch in the FAST ERECT position enables the pilot to select a fast erect cycle for the AN/ASN-50 displacement gyro. When FAST ERECT is selected, normal system safeguards are bypassed and an increased erection voltage is applied to the displacement gyro. If the AN/ASN-50 gyro is in a partially erect state and is coasting down, the FAST ERECT switch may be used to maintain the gyro in the fast erect mode until it is completely erected. When the switch is in the FAST ERECT position or if the gyro is not completely erected, the GYRO warning flag on the copilot's attitude indicator will remain in view. The switch should be held to FAST ERECT for 30 seconds, then released to check for flag drop indicating that the gyro has erected. If the GYRO flag still shows, activate the FAST ERECT switch for 30-second intervals until the gyro flag disappears. The FAST ERECT switch allows the pilot to regain the AN/ASN-50 displacement gyro and should only be used during the coastdown period.

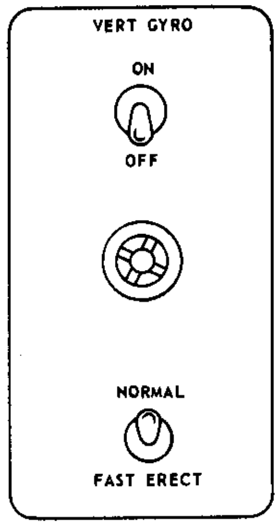


Figure 1-19. AN/ASN-50 OFF and GYRO FAST ERECT Switches

NOTE

The FAST ERECT switch will be used for operational necessity only.

NOTE

The AN/ASN-50 gyro can be damaged during ground operations if the following precautions are not observed: the helicopter should not be moved for a period of 5 to 20 minutes after ac power is secured because of the danger of damaging the ANASN-50 during coastdown. If during this period, ac power is to be restored to the helicopter, the VERT GYRO switch on the instrument panel must be off.

SEE OS-21

Attitude Reference System (1080Y) The 1080Y Vertical Gyro provides pitch and roll reference signals for operation of the pilot's attitude indicator and, if selected, will provide pitch and roll reference signals used in the AFCS. This selection is made by placing the vertical gyro selector switch, on the Channel Monitor Panel, to STBD. The system is powered by the No. 2 AC Primary Bus and is protected by a circuit breaker marked GYRO, on the pilot's overhead circuit breaker panel under the general heading NO. 2 AC PRI.

Compass Control Panel (ASN/ASN-50)

The control panel, marked COMP, on the pilot's console provides the pilot with system operating controls. The mode switch has three marked positions, SLAVE, FREE and COMP. SLAVE is the normal operating mode between 70° N and 70° S, unless a distorted magnetic field is present. When the SLAVE mode is selected, fast synchronization occurs; the SLAVE mode synchronizes directional gyro output to the remote compass heading. FREE is an alternate mode which does not use remote compass inputs and only directional gyro outputs are used. The COMPASS mode is used in the event the directional gyro fails and, in this case only, the remote compass will provide outputs. The PUSH TO SYNC button is momentarily depressed to accelerate fast synchronization when operating in the SLAVE mode. The SYNC IND meter displays the degree of synchronization when operating in the SLAVE mode. The hemisphere switch and its associated display window, at the upper right of the control panel, compensates for apparent gyro drift due to the earth's rotation by setting the switch to N or S when operating in the FREE mode. The LATITUDE DEGREES dial and its associated display window, below the hemisphere switch, compensates further for apparent gyro drift when the local latitude is dialed in, used when operating in the FREE mode. The PUSH TO TURN switch with marked positions L and R is used to set the compass card to the helicopters heading when operating in the FREE mode. When either the PUSH TO SYNC button or the PUSH TO TURN switch is used, the yaw channel of the AFCS is designed to be automatically nulled. However, when using the PUSH TO SYNC button or PUSH TO TURN switch, yaw kicks may be experienced through the AFCS yaw channel.

CAUTION

Landing sites with abnormally strong magnetic fields may affect helicopter compass synchronization. Extended time at these areas can cause the remote compass transmitter to slew off heading. The use of the free mode during operations in the vicinity of these magnetic fields will provide proper heading information after initial departure from the site.

NOTE

If the COMP position is used, disengage the yaw channel of the AFCS.

SCE 05-21

CAUTION-ADVISORY PANEL

The caution-advisory panel (39, figure FO-4) is located in the right center of the instrument panel. The caution section of the panel provides visual indication of certain failures or unsafe conditions through illumination of amber lights. The advisory section provides visual indication of certain non-critical conditions through illumination of green lights. Each light has its own operating circuit and will remain illuminated as long as the condition which caused the light to illuminate prevails. The caution and advisory lights are powered by the dc primary bus through a circuit breaker, marked PWR, on the overhead circuit breaker panel under the heading CAUTION PANEL. A switch, marked DIM and BRT, located on the caution panel, enables selection of a dim or bright brilliance of the caution and advisory lights. The switch cannot be utilized until the rheostat, marked PILOT, FLT INST, located on the overhead switch panel, has been turned on.

MASTER CAUTION LIGHT

Two master caution lights, marked MASTER CAUTION, located on the instrument panel hood, one in front of each pilot, illuminate whenever a caution light on the caution-advisory panel illuminates. The purpose of these lights is to alert the pilots and direct attention to the caution-advisory panel. The lights are press-to-reset type. After the specific condition or malfunction has been noticed on the caution panel, the master lights should be reset to provide a similar indication if a second condition or malfunction occurs while the first is still present.

CAUTION-ADVISORY LIGHTS TEST SWITCH

The caution-advisory lights test switch, marked TEST, located on the caution panel, provides a means of sumultaneously checking all lights by a single push-button switch. The switch receives power from the 28 volt dc primary bus through a circuit breaker, under the heading CAUTION PANEL and marked TEST, located on the overhead circuit breaker panel.

LANDING GEAR SYSTEM

The tricycle landing gear system consists of two retractable main landing gear assemblies, a partially retractable nose gear assembly and an actuating system. The landing gear hydraulic system (figure 1-20) operates on 3000 psi hydraulic pressure from the utility hydraulic system. The necessary electrical power is provided from the dc primary bus, through circuit breakers, under the general heading LAND GEAR and marked EMER DN, NOSE, MAIN and WARN, located on the overhead circuit breaker panel. The main landing gear system is equipped with a one-shot pneumatic, alternate extension system, and the APU accumulator is utilized as an alternate extension power source for the nose gear assembly. The nose gear assembly retraction system can be used on the ground to kneel the helicopter to facilitate cargo handling. The landing gear control panel is located on the instrument panel.

MAIN LANDING GEAR

The two main landing gear assemblies are located below the sponsons and retract forward and upward into the sponsons. Each main landing gear is equipped with dual wheels and hydraulic brakes, a retracting cylinder, a pneudraulic strut, attaching drag links and supports, up and down lock mechanisms, and emergency release mechanisms.

NOSE LANDING GEAR

The single nose landing gear, mounted vertically at the centerline of the helicopter, is free to rotate 360 degrees about the strut centerline. All shock stroke, kneeling, jacking and retraction motion is vertical. The nose gear assembly is equipped with dual wheels, a retracting cylinder, a pneudraulic strut and shimmy damper, and attaching drag links and supports. The entire pneudraulic strut acts as a piston, which is lowered or raised for retraacting, jacking, and kneeling. The nose gear may be retracted (kneeled) so as to alter the ground clearance (figure 1-21) of the tail section to facilitate cargo handling. The nose gear assembly is hydraulically locked in the extended, retracted, or kneeled positions. A centering cam centers the nose gear assembly when the helicopter is airborne. A nosewheel lock is installed to improve ground stability of the helicopter on uneven terrain.

Nosewheel Lock Handle

The lock handle, marked PARK LOCK, is on the pilot's side of the center console (figure 1-23). The nosewheel is locked by pulling the lock handle aft and up, and unlocked by pulling aft and pushing down.

LANDING GEAR ACTUATING SYSTEM

The landing gear actuating system operates on 3000 psi hydraulic pressure, supplied by the utility hydraulic system to raise or lower the main and nose landing gear assemblies. Each main landing gear is equipped with downlock release limit switches which prevent inadvertent retracting of the landing gear when the weight of the helicopter compresses the oleo struts. When airborne, the struts extend and close the contacts of the downlock release limit switches. The landing gear control panel (figure 1-22) is located on the instrument panel. Placing the landing gear control handle in the UP position retracts the landing gear. As the landing gear retracts, limit switches are actuated that cause the landing gear warning light in the control handle to show an unsafe condition, the landing gear position lights to go out, and a circuit to be completed that assures electrical power to lower the gear. When the landing gear is fully retracted, limit switches are actuated that cause the landing gear warning light in the handle to go out. The main gears are held up by mechanical uplocks. The retraction cycle of the nose gear system remains energized to maintain the nose gear in the retracted position. The landing gear is extended by placing the landing gear lever in the DN position. This completes the electrical circuit to the solenoid valve that directs fluid to the uplock cylinders of the main landing gears, unlocks them from the up position, simultaneously directs fluid to the actuator, energizes the landing gear control handle warning light, and causes the landing gear to extend. Mechanical spring-loaded locks are engaged to lock the main gear in the down position. As the main landing gear extension phase is initiated, the retraction port of the actuator is vented to relieve pressure that had been holding the nose gear in the retracted position, and

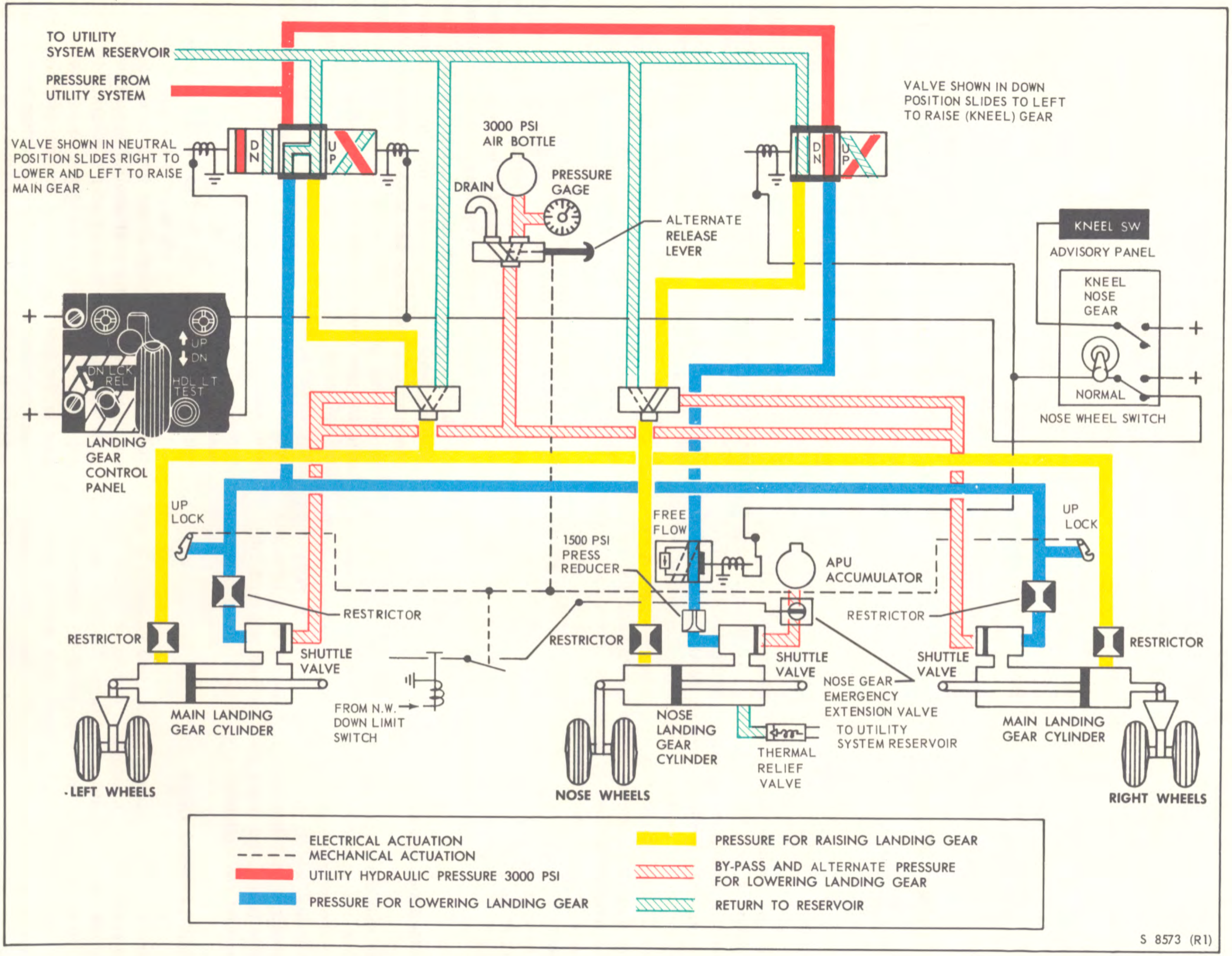


Figure 1-20. Landing Gear Hydraulic System

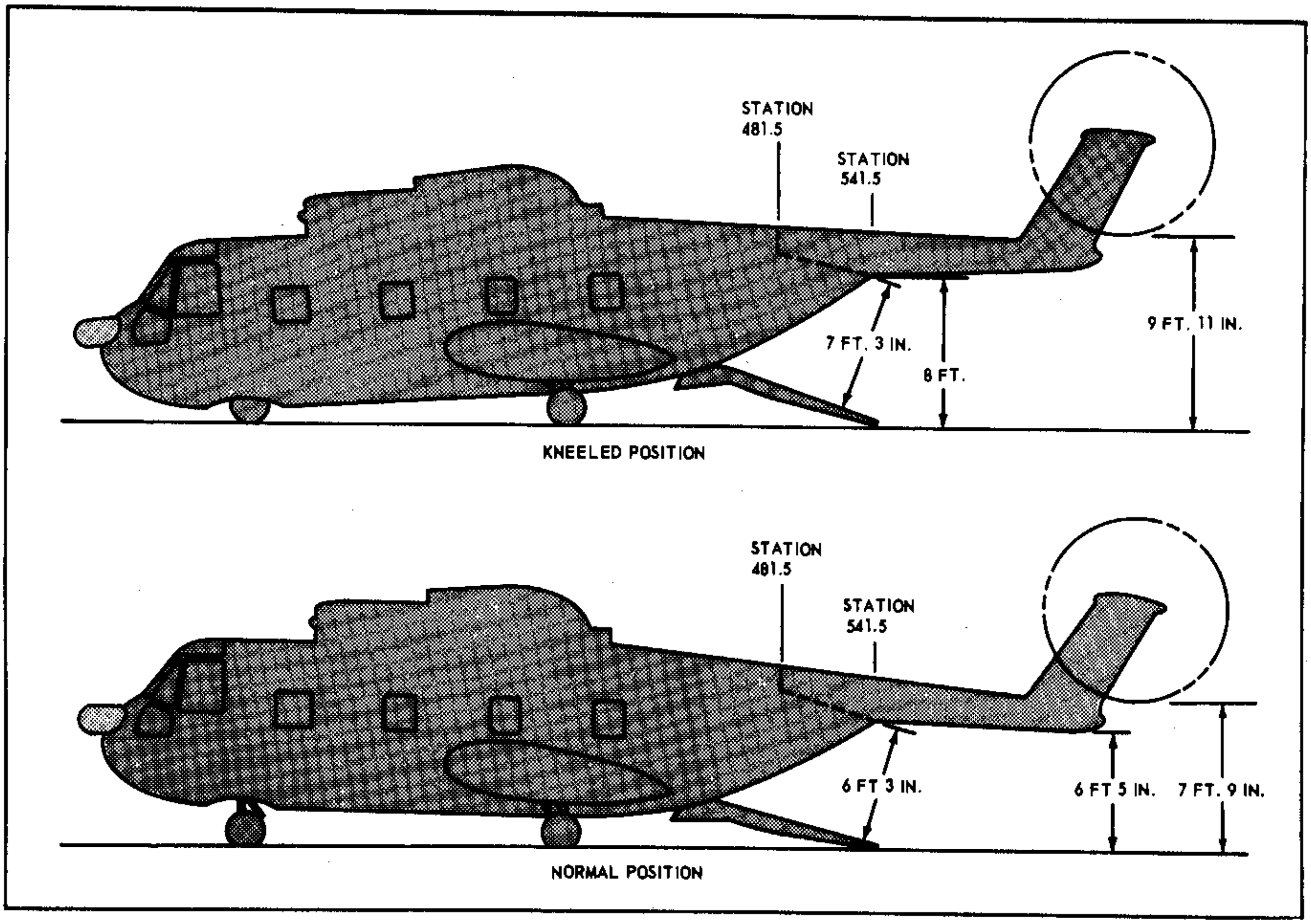


Figure 1-21. Ground Clearances (Normal and Kneeled)

hydraulic pressure is directed to the extension port of the actuator to lower the nose gear. Hydraulic pressure is retained in the actuating cylinder to lock the nose gear down. When all gear are fully extended, limit switches are actuated that energize the landing gear position lights and extinguish the control handle warning lights.

Landing Gear Control Handle Downlock Release

A manually-operated downlock release, located on the landing gear control panel, marked DN LCK REL, provides a mechanical override of the landing gear control handle downlock solenoid in the event of an interruption of electrical power to the solenoid. Should the downlock solenoid become inoperative, the downlock release can be actuated to mechanically release the landing gear control handle from the DN position.

Nose Gear Switch and Caution Light

Kneeling is accomplished by placing the switch, marked NOSE GEAR, NORMAL, KNEEL, located on the overhead switch panel (figure FO-3), in the KNEEL position. Placing the switch in the NORMAL position extends (jacks) the nose gear to the normal down position. An advisory light, marked KNEEL SW, located on the caution-advisory panel, will illuminate when the kneel switch is in the KNEEL position. The green nose gear position light will be out and the red warning light in the landing gear handle will be on when the nose gear is kneeled.

Landing Gear Position Lights

The landing gear position lights, located on the landing gear control panel under the general heading GEAR DOWN and marked LEFT, RIGHT, and NOSE, are three press-to-test green lights. The lights operate on power from the dc primary bus through a circuit breaker, under the general heading and marked LAND GEAR WARN, located on the overhead dc circuit breaker panel. Each light will illuminate when the associated landing gear is down and locked.

Landing Gear Warning Light and Test Button

The landing gear warning light, located in the handle of the landing gear control lever, will illuminate whenever any or all landing gears are neither fully up and locked nor fully down and locked. The warning light operates on power from the dc primary bus, through a circuit breaker under the general heading and marked LAND GEAR MAIN, located on the overhead circuit breaker panel. A landing gear warning light test button, marked HDL LT TEST, is located on the landing gear control panel. The test button is pressed to test the warning light.

Landing Gear Alternate Extension System

An alternate gear handle, located on the left side of the center console, is used to lower the landing gear in the event of an electrical or hydraulic failure. The handle mechanically unlocks the main landing gear uplocks, positions a directional valve, and discharges a 3000 psi air bottle. The compressed air charge actuates valves that vent the return side of the actuators to the reservoir, and then pressurizes the actuators to lower and lock the main landing gear. Simultaneously, fluid from the APU accumulator is directed through an electrically actuated valve to the top side of the nose gear actuating cylinder to lower the nose gear. After an actuation, the valves must be manually reset before the main landing gear air cylinder can be recharged and the landing gear retracted. In the event the air charge in the cylinder has been depleted, when the alternate landing gear handle is actuated, main landing gear hydraulic pressure is vented back to the utility reservoir, the uplocks are disengaged, and the main landing gear will lower by its own weight. The mechanical downlocks may not engage.

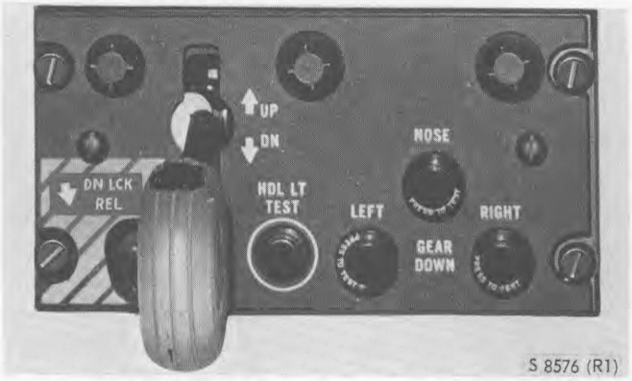


Figure 1-22. Landing Gear Control Panel

Landing Gear Pins

When the landing gear pins are inserted into the main landing gear drag link assembly, they provide an additional mechanical downlock.

BRAKE SYSTEM

The main landing gear wheels are each equipped with hydraulic brakes. The self-contained brake system is operated by toe pedals located on the pilot's and copilot's tail rotor pedals. A parking brake system is also provided. The parking brake handle, marked PARKING BRAKE, is located on the right side of the center console. The parking brakes are applied by depressing the brake pedals, manually pulling the parking brake handle to the PARK position, and then releasing the brake pedals. Depressing the brake pedals will release the parking brakes, allowing the parking brake handle to return to the OFF position.

NOTE

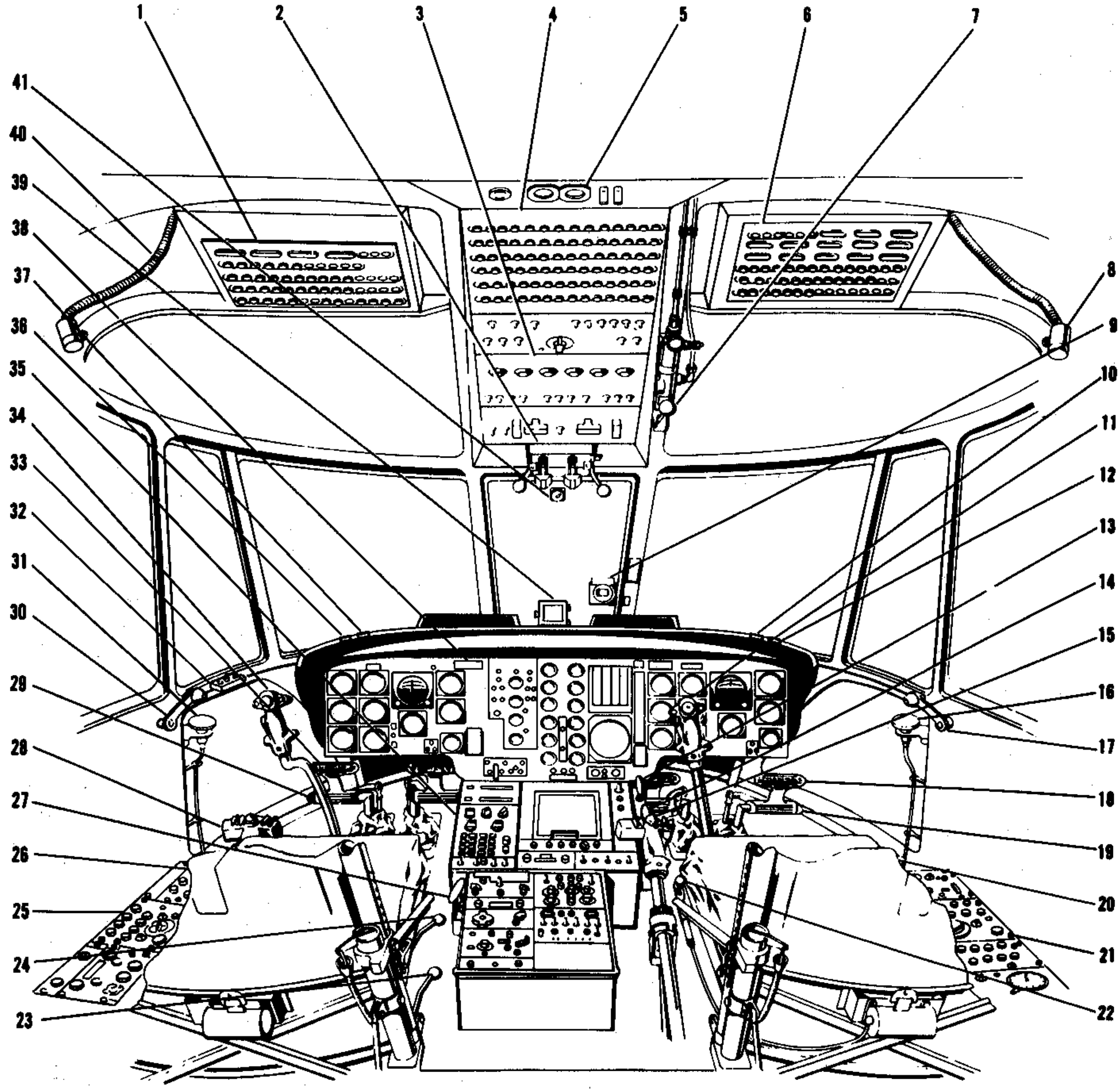
> The parking brake handle can be raised and locked in the extended position, thereby illuminating the advisory light, without pressurizing the wheel brake system.

FIRE DETECTION SYSTEMS

Three fire detection systems, one for each engine compartment and one for the APU compartment ~~and one for the APU compartment~~, provide warning in event of fire. A single switch is used to test the three systems.

ENGINE AND APU COMPARTMENT FIRE DETECTION SYSTEMS

Four heat-sensitive fire detector elements, located in each engine compartment, are wired into a closed series loop and connected to control units. The elements are mounted on the inside of the engine compartment doors and on both sides of the center firewall. The fire detector elements terminate at the aft firewall. The control units, located overhead in the cargo compartment, continuously monitor the output of the detector elements. In event of fire in an engine compartment, the detector element senses excessive heat and the control unit will close the circuit to fire warning lights in the cockpit. Power for the No. 1 engine fire detection system is supplied by the No. 1 ac primary bus through a circuit breaker, marked (FIRE DET), on the copilot's overhead circuit breaker panel.



1. COPILOT'S CIRCUIT BREAKER PANEL
2. ENGINE CONTROL QUADRANT
3. OVERHEAD SWITCH PANEL
4. OVERHEAD DC CIRCUIT BREAKER PANEL
5. PILOT'S COMPARTMENT DOME LIGHT
6. PILOT'S CIRCUIT BREAKER PANEL
7. ROTOR BRAKE LEVER
8. PILOT'S MAP LIGHT
9. STANDBY COMPASS
10. FIRE WARNING LIGHT
11. PILOT'S CYCLIC STICK
12. MASTER CAUTION LIGHT
13. PARKING BRAKE HANDLE
14. NOSE WHEEL LOCK HANDLE
15. PILOT'S COLLECTIVE PITCH LEVER
16. PILOT'S TAIL ROTOR PEDAL ADJUSTMENT KNOB
17. PILOT'S WINDOW EMERGENCY RELEASE HANDLE
18. PILOT'S TOE BRAKES
19. PILOT'S TAIL ROTOR PEDALS
20. PILOT'S SEAT
21. PILOT'S CONSOLE
22. PILOT'S SHOULDER HARNESS LOCK LEVER
23. COPILOT'S SEAT HEIGHT ADJUSTMENT LEVER
24. COPILOT'S SEAT FORWARD AND AFT ADJUSTMENT LEVER
25. COPILOT'S CONSOLE
26. COPILOT'S SEAT
27. ALTERNATE LANDING GEAR HANDLE
28. COPILOT'S COLLECTIVE PITCH LEVER
29. COPILOT'S TAIL ROTOR PEDALS
30. COPILOT'S WINDOW EMERGENCY RELEASE HANDLE
31. COPILOT'S TAIL ROTOR PEDAL ADJUSTMENT KNOB
32. LDG LTS AND REMOTE ICS CONTROL PANEL
33. COPILOT'S TOE BRAKES
34. COPILOT'S CYCLIC STICK
35. COCKPIT CONSOLE
36. FIRE WARNING LIGHT
37. MASTER CAUTION LIGHT
38. INSTRUMENT PANEL
39. SCROLL CHECK LIST
40. COPILOT MAP LIGHT
41. FREE-AIR TEMPERATURE GAGE

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Figure 1-23. Cockpit Arrangement

Power for the No. 2 engine fire detection system is supplied by the No. 2 ac primary bus through a circuit breaker, marked FIRE DET, on the pilot's overhead circuit breaker panel. The APU fire detection system uses five probe-type detector switches mounted at strategic locations in the APU compartment. When any one of these switches sense excessive heat, it closes a circuit to the APU fire warning light. Current is supplied from the dc primary bus through a circuit breaker, marked APU, FIRE DET, on the overhead circuit breaker panel.

Fire Warning Lights and Test Switch

When fire is detected in an engine compartment, two red master fire warning lights and red lights installed in the ends of the T-shaped fire emergency shutoff selector handle for that engine will illuminate. The master fire warning lights, marked FIRE, are mounted in the instrument panel hood. One light is (10, figure 1-23) located in front of the pilot and the other (36, figure 1-23) in front of the copilot. Fire detected in the APU compartment causes illumination of a light, marked FIRE, on the APU control panel, and on helicopters modified by T.O. 1H-3(H)F-563 a capsule marked APU FIRE on the caution panel. A fire warning test switch (44, figure FO-4), marked FIRE WARN TEST, is on the right side of the instrument panel. The switch is spring-loaded to the center (off) position and has two test positions marked NO. 1 ENG and NO. 2 ENG and APU. Testing the engine fire detection system is done by holding the test switch in either test position. The two master fire warning lights and the respective fire emergency shutoff selector handle lights should go on. The test switch should return to center when released and all fire warning lights should go off. Testing of the APU fire detection system is done by holding the test switch at NO. 2 ENG and APU. The fire warning light on the APU control panel and on helicopters modified by T.O. 1H-3(H)F-563 the APU FIRE capsule on the caution panel will go on. Current for the engine fire warning test system is supplied by the ac and dc primary buses through appropriately marked circuit breakers. Current for the APU fire warning test system is supplied by the dc primary bus through a circuit breaker marked APU, FIRE DET.

FIRE EXTINGUISHING SYSTEMS

There are two similar Bromotrifluoremethane CF_3BR fire extinguishing systems. The engine compartments system has two CF_3BR containers with associated circuitry and plumbing. The APU compartment system has only one container.

WARNING

CF_3BR is highly volatile and is not easily detected by odor. It is not toxic and is considered to be about the same as other freons and carbon dioxide, causing danger primarily by reduction of oxygen. Do not allow liquid to contact the skin as it may cause frostbite or low temperature burns because of its low boiling point.

ENGINE COMPARTMENT FIRE EXTINGUISHING SYSTEMS

The CF_3BR containers, located aft of the main gear box, are charged with 2.5 pounds of CF_3BR plus a nitrogen charge to propel the agent through tubing to the engine compartments. The tubes for each engine compartment are mounted on the center firewall with discharge nozzles directed at the combustion and power turbine sections of the engine. Each container is equipped with a pressure gage and a thermal discharge valve which will allow discharge of the containers through a common line when the temperature of the containers reaches 96° to 104°C (208°F to 220°F). Each container is equipped with two frangible discs and two explosive cartridges and slugs. The tubing from the disc of each container is tied together to form a single line to each engine. Choice of engine compartment and explosive cartridges is made by pulling out the appropriate engine fire emergency selector handles. Firing of the explosive cartridge and discharge of the extinguishing agent is accomplished by use of the engine fire extinguisher switch. Current for the engine compartment fire extinguishing systems is supplied by the dc primary bus through the circuit breaker, marked ENGINE FIRE EXT, located on the overhead circuit breaker panel.

Engine Fire Emergency Shutoff Selector Handles

Two T-shaped handles, located on the overhead switch panel (figure FO-3) are marked FIRE EMER SHUT-OFF SELECTOR, NO. 1 ENGINE and NO. 2 ENGINE. When either handle is pulled down, power from the dc primary bus actuates the fuel shutoff valve, which closes the fuel for that engine, selects that engine compartment to receive the fire extinguishing agent, and energizes the circuit to the fire extinguisher switch. A fire warning light is located in each end of each handle.

Engine Fire Extinguisher Switch

An engine fire extinguisher switch, marked FIRE EXT, located on the overhead switch panel (figure FO-4) in the pilot's compartment, has marked positions RESERVE, OFF, and MAIN. The lock lever type switch is operative only after one of the fire emergency shutoff selector handles has been pulled. When the engine fire extinguisher switch is held in the MAIN position, after a fire emergency shutoff selector handle has been pulled, the contents of the main fire extinguisher container are discharged into the corresponding engine compartment. When the engine fire extinguisher switch is held in the RESERVE position, after a fire emergency shutoff selector handle has been pulled, the contents of the reserve fire extinguisher container is discharged into the last selected engine compartment.

Thermal Discharge Indicator

Two red thermal discharge indicators are located aft of the third window on the left side of the fuselage. The indicator marked ENG is connected through a common line to both engine fire extinguisher containers. The indicator marked APU is connected to the APU engine fire extinguisher container. When the temperature reaches 96°C to 104°C (208°F to 220°F) in any of the containers, the valve in that container will open to relieve pressure. The fire extinguisher agent will then be forced through the lines, eject the seal, and be discharged overboard.

APU FIRE EXTINGUISHER SYSTEM

The fire extinguisher system for APU unit consists of a charged container of CF_3BR, located adjacent to the APU compartment, with lines, nozzles, and controls similar to the engine compartment extinguisher system. The container holds 2.5 pounds of CF_3BR.

APU Fire Detector and Extinguisher Control Panel

The APU unit fire detector and extinguisher control panel is located on the APU control panel on the pilot's console (figure FO-9). The APU fire extinguisher system is energized by first placing the toggle switch, marked FUEL SHUT-OFF and NORM, in the FUEL SHUT-OFF position, then placing the toggle switch, marked FIRE EXTING and OFF, in the FIRE EXTING position. Power for this system is provided by the dc primary bus, through a circuit breaker marked APU, EXT.

EMERGENCY ENTRANCES AND EXITS

For emergency entrances or exits, see foldout FO-10 and accompanying descriptions in Section III. A complete description of the cargo door and ramp is found in Section IV.

CREW SEATS

PILOT'S AND COPILOT'S SEATS

The pilot's and copilot's seats are track-mounted in the cockpit. The pilot's seat is on the right. The track-mounted seats are designed to accommodate a thin back type parachute, RK-2 pararaft kit and SP-1 seat pan. Both seats have a 5-inch range of height adjustment, a 3.5-inch forward and aft adjustment, and are equipped with cushions which are interchangeable with the pararaft and parachute.

Seat Height Adjustment Lever bb

The seat height adjustment levers (figure 1-23) are the rear levers at the right of the pilot's and copilot's seats. The spring-loaded levers are pulled up to release the height adjustment lockpins.

Seat Forward and Aft Adjustment Lever

The seat forward and aft adjustment levers (figure 1-23) are the front levers on the right side of the pilot's and copilot's seats. The spring-loaded levers are pulled up to release the forward and aft seat adjustment lockpins.

Shoulder Harness Lock Lever

A two-position shoulder harness inertial reel lock lever (figure 1-23) is located at the left side of each seat. When the lever is in the unlocked (aft) position, the shoulder harness cable will extend to allow the occupant to lean forward; however, the inertial reel will automatically lock if an impact force between two and three G's in any direction is encountered. When this occurs, the inertial reel will remain locked until the lever is cycled. When the lever is placed in the locked (forward) position, the shoulder harness cable is locked so that the occupant is prevented from leaning forward. The locked position is used to provide an added safety precaution over that of the automatic lock on the inertia reel.

CREWMAN'S SEATS

A swivel type crewman's seat is located adjacent to each of the large search windows. Each is equipped with a seat belt and shoulder harness identical to the cockpit seats and is of high stress design. The left seat is track mounted for lateral adjustment. The right seat is mounted on the aft frame of the personnel door and has a pivot adjustment forward and aft.

JUMP SEAT

A folding jump seat (figure FO-2), located in the cockpit entrance, is folded against the entrance wall to facilitate passage. This seat is equipped with a safety belt, but is not designed to withstand as much stress as the other crew seats.

AUXILIARY EQUIPMENT

The following major systems and items are covered in Section IV:

Heating System

Anti-icing Systems

Communication and Associated Electrical Equipment

Lighting Equipment

Auxiliary Power Unit

Cargo Compartment Equipment

Rescue Hoist

External Cargo Sling

Troop Carrying Equipment

Miscellaneous Equipment

Windshield Wiper System

Bilge Pump

Load Adjuster

Air Deliverable Anti-pollution Transfer System (ADAPTS)

Flotation System, Auxiliary

Navigation Radios and Sensors