SECTION I

DESCRIPTION

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THE HELICOPTER

The U.S. Air Force Series UH-1N utility helicopter is manufactured by the Bell Helicopter Company. It is capable of operating from prepared or unprepared takeoff or landing areas, under visual (VFR) or instrument conditions (IFR), day or night (figure 1-1.) The helicopter is certified for personnel paradrop static line and HALO, aerial flare drop M8A1 flares, rappelling, infiltration/exfiltration rope ladder, and aerial spray rig Model 3090-3 operations.

Cockpit entrance is afforded through the hinged doors located in the forward cabin next to the pilot's positions (figure 1-2). Entrance to the cargo-passenger area is accomplished through two sliding doors, one on each side of the aft cabin area. The cargo-passenger area provides loading space for equipment transportation (figure 1-3). The sliding doors permit items too long to be carried internally to be loaded straight through the cabin. Seats may be installed for thirteen passengers. For medical evacuation and ambulance service, litter racks may be installed within the cabin, providing a litter capacity of six patients and one medical attendant.

The propulsion system consisting of the engine and drive system is located aft of the cabin and mounted above the fuselage on a platform. The engine and drive systems are enclosed by cowling that may be swung open or removed for access.

The fuselage consists of two main sections: the forward section, and the aft or tail boom section. The forward fuselage section consists primarily of longitudinal beams with transverse bulkheads and metal covering. The beams provide the supporting structure for the cabin sections, landing gear, fuel tanks, transmission, engine, and tail boom. The lift beam is the attaching point for the external cargo suspension unit. The aft (tail boom) section is a semimonocoque structure with metal covering, and attaches to the forward fuselage section with four bolts.

The tail boom structure supports the tail rotor vertical fin and synchronized elevator. The landing gear system is of the skid type, attached to the fuselage at four points. Ground handling wheels are provided for moving the helicopter on the ground.

HELICOPTER DIMENSIONS

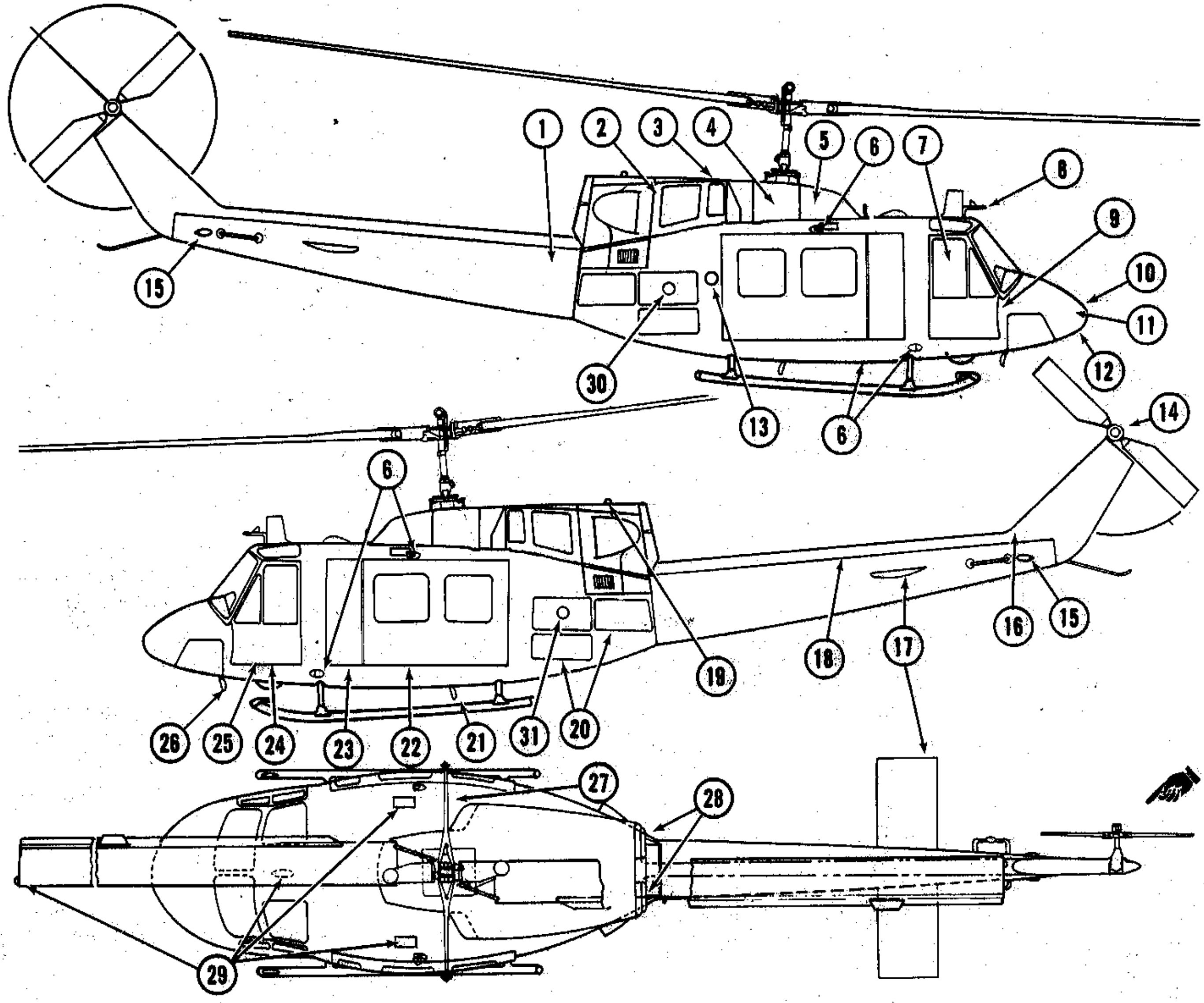
PRINCIPAL DIMENSIONS (APPROXIMATE) (Figure 1-4)

Length

Overall (main rotor fore and aft and tail rotor horizontal)	57 ft. 3.3 in.
Overall (main rotor fore and aft and tail rotor vertical) to end of tail skid	55 ft. 1.3 in.
Nose of cabin to tail skid	42 ft. 4.7 in.
Nose of cabin to tail rotor horizontal	45 ft. 11.2 in.

Width

Skid gear	9 ft. 1.0 in.
Elevator	9 ft. 4.5 in.



1. Tail Boom
2. Engine Compartment
3. Engine Oil Reservoirs
4. Transmission
5. Hydraulic Oil Reservoir
6. Navigation Light (7)
7. Pilot's Position
8. Pitot Static Tube
9. Pilot's Door
10. Electrical and Electronic Compartment
11. Battery
12. External Power Receptacle
13. Fuel Tank Filler
14. Tail Rotor (90°) Gear Box
15. Aft Navigation Light (2)
16. Tail Rotor Intermediate (42°) Gear Box
17. Synchronized Elevator
18. Tail Rotor Drive Shaft
19. Anti-Collision Light
20. Electrical and Electronic Compartment
21. Landing Light
22. Cargo-Passenger Compartment Door
23. Hinged Panel
24. Copilot's Position
25. Copilot's Door
26. Searchlight
27. Stabilizer Bar
28. Tailpipes
29. Formation Lights (When Installed)
30. Aft Auxiliary Fuel Tank Filler (RH)
31. Aft Auxiliary Fuel Tank Filler (LH)

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Figure 1-2. General arrangement

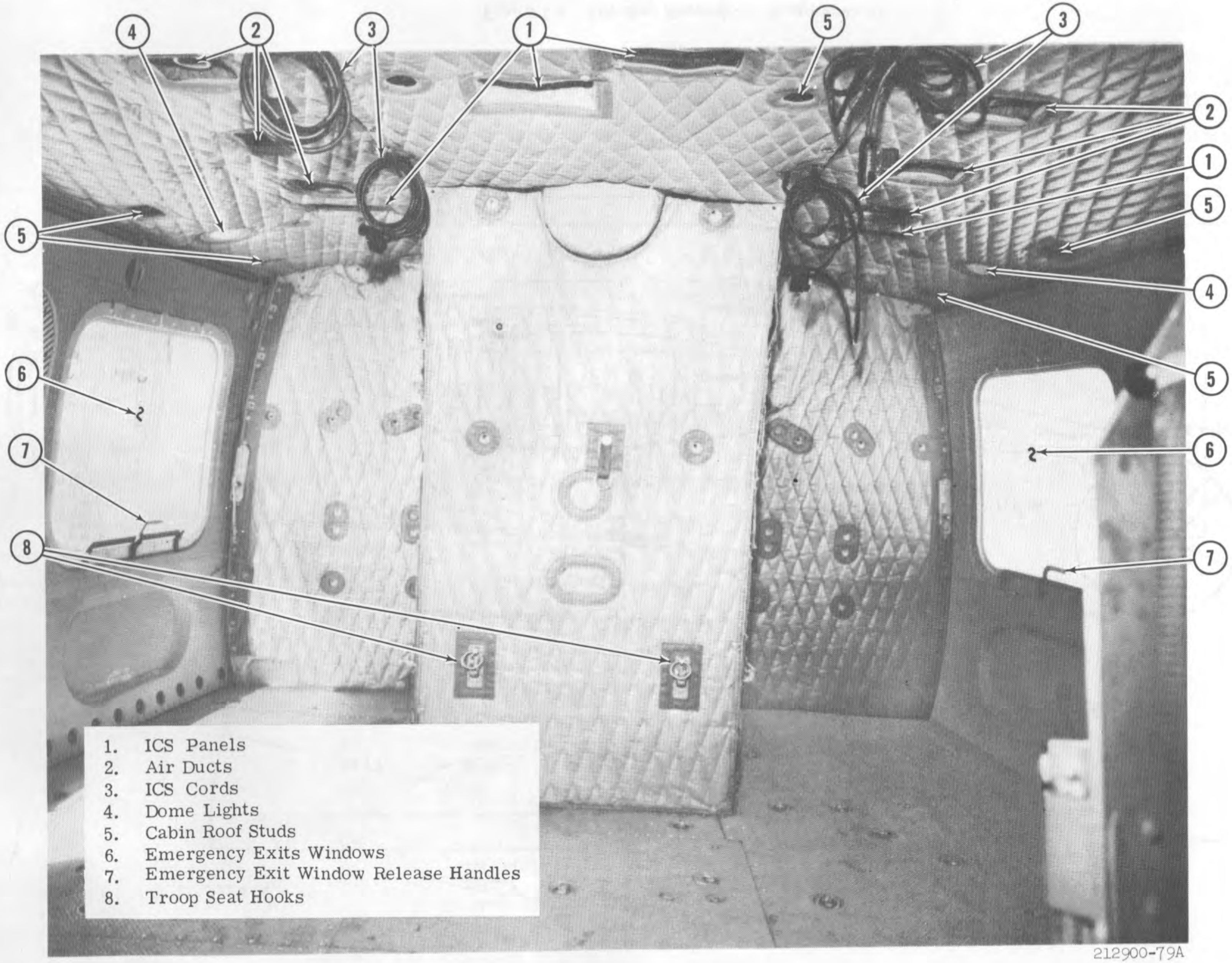

Figure 1-3. Compartment diagram

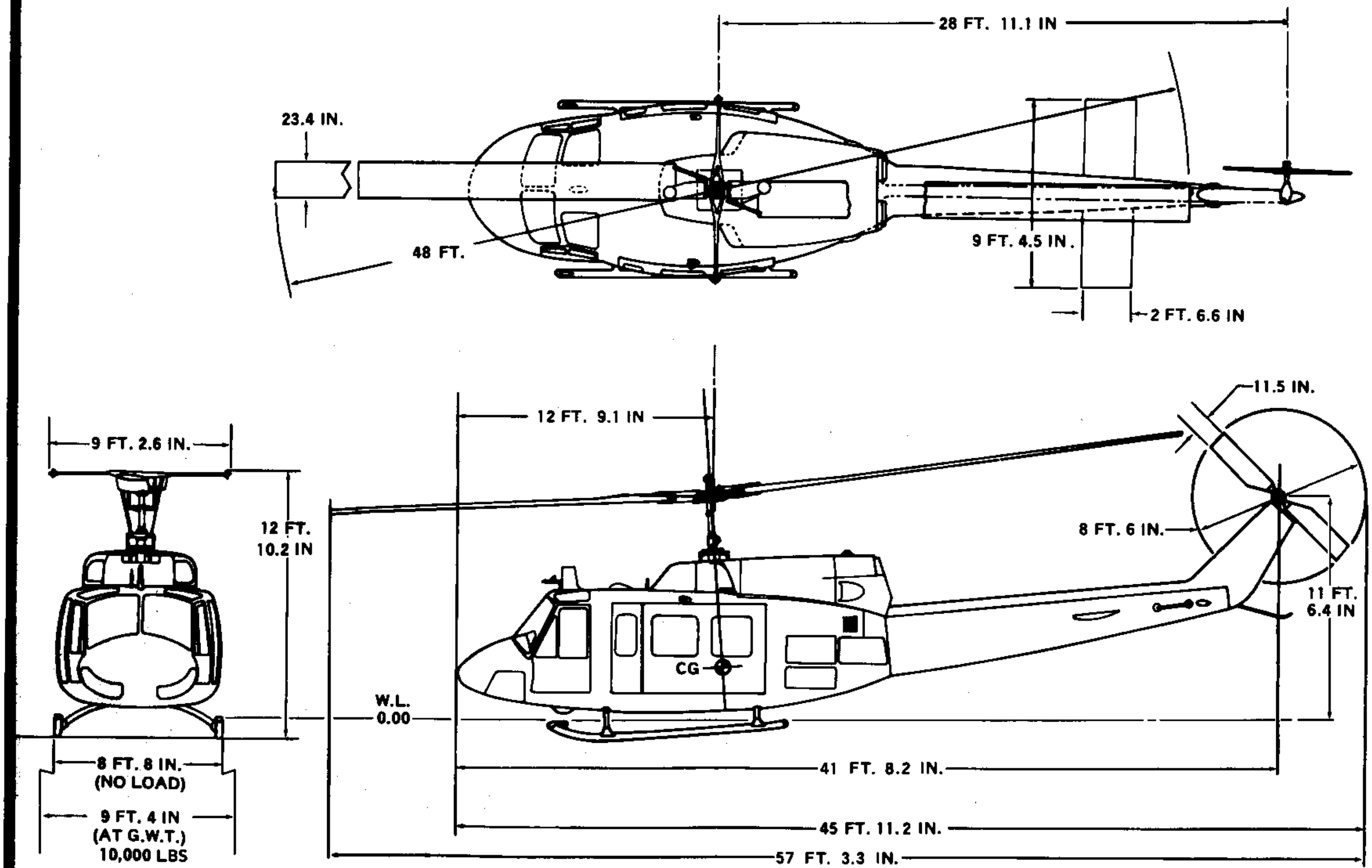

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Figure 1-4. Principal dimensions (approximate)

Height Above Ground (Static Position)

Tip of main rotor forward blade to ground	7 ft. 0.2 in.
Stabilizer Bar to Ground	12 ft. 10.2 in.
Tip of main rotor forward blade with droop to ground	5 ft. 11.16 in.
Tip of main rotor forward blade, tied down position	16 ft. 10.5 in.
Tip of tail rotor blade, vertical position	14 ft. 4.7 in.
Top of cabin	7 ft. 8 in.
Minimum height with blades in horizontal position	13 ft. 1.3 in.
Maximum height from whip antenna	13 ft. 4.6 in.

Cabin floor to ground

Front of hinged panel	2 ft. 8.7 in.
Front of cargo door	2 ft. 7.5 in.
Rear of Cargo door	2 ft. 1.5 in.

Diameters

Main Rotor	48 ft. 0.0 in.
Tail Rotor	8 ft. 6.0 in.
Stabilizer Bar	9 ft. 2.6 in.

Cargo Door with Hinged Panel Open

Width	7 ft. 0.6 in.
Height	4 ft. 0.1 in.

NOTE

Ground handling wheels in the lowered position for towing will add approximately five inches to overall height.

DESIGN GROSS WEIGHT

The design gross weight of the aircraft is 6600 pounds. For detailed information, refer to Section V.

POWER PLANT

The helicopter is powered by a T400-CP-400 turboshaft engine (1800 SHP rating). The engine (figure 1-5) consists of two independent power sections driving into a combining gearbox. The combining gearbox contains an overrunning clutch as well as a torquemeter for each power section.

NOTE

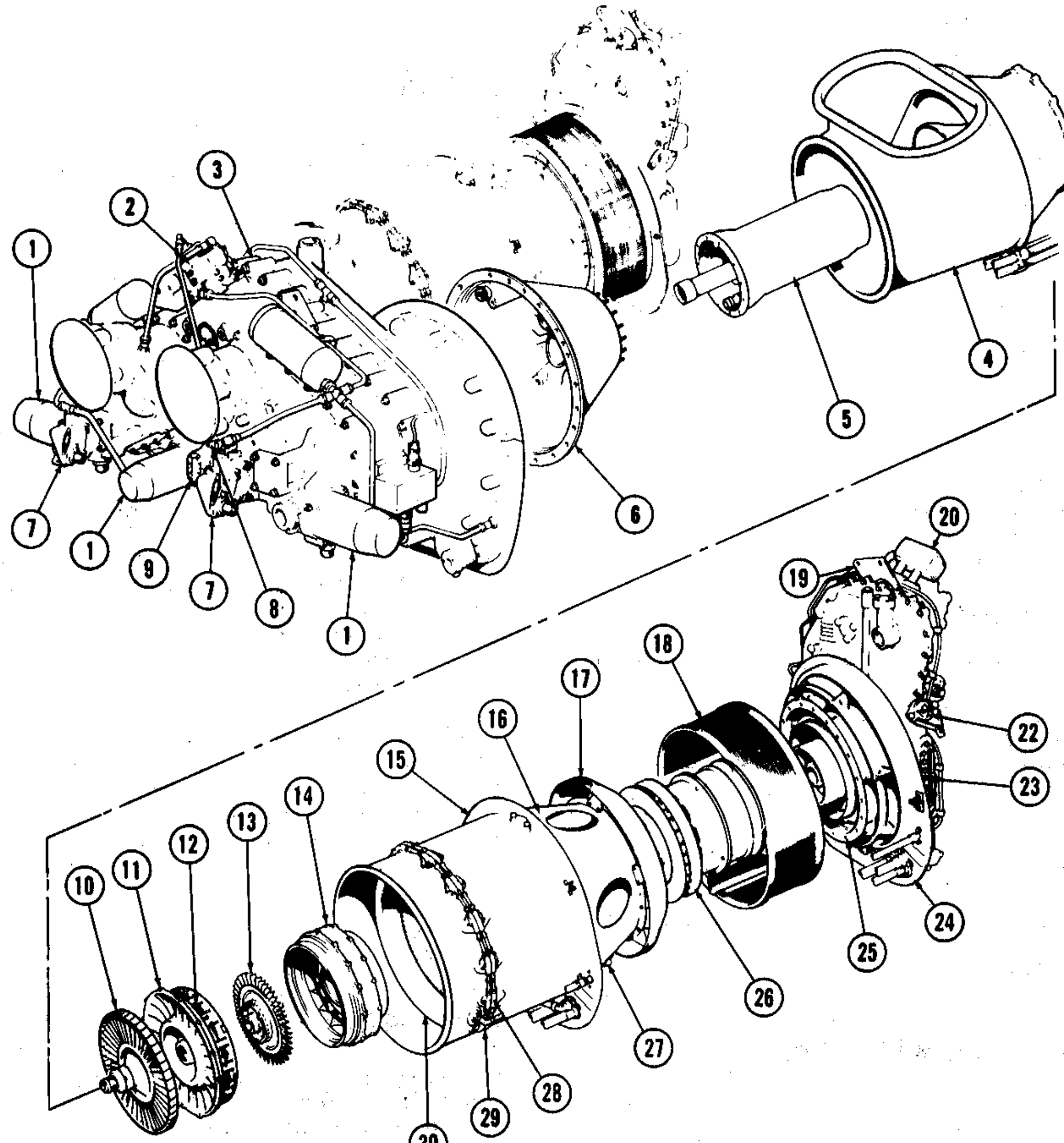
To enable the use of standard terminology in this manual, the individual power sections will be referred to as engine 1 (left) and engine 2 (right).

Each engine has a three stage axial, single stage centrifugal compressor driven by a single stage turbine. Another single stage turbine counter-rotating with the first, drives into the combining gearbox. Fuel is sprayed in the annular combustion chamber by fourteen individually removable fuel nozzles mounted around the gas generator case. A high tension ignition unit and two spark igniter plugs are used to start combustion.

A hydro-pneumatic fuel control schedules fuel flow to provide the power required to maintain the desired output shaft speed. Automatic load sharing, as well as engine torque limiting is provided. In addition, a manually operated fuel system is provided within the fuel control system.

COMBINING GEARBOX

The combining gearbox located on the aft portion of the engines (figure 1-5) has two identical reduction geartrains which transmit torque from each engine to a common output shaft. Each geartrain has three stages and an overrunning clutch incorporated with the second stage shaft allowing torque to be transmitted in one direction only. A gearbox output section is composed of the common output-shaft and both second stage shafts. The lubrication of the gearbox output section is independent of the engines and self-contained within the combining gearbox. The first stage and accessory drives obtain lubrication from their respective engines. The two torquemeter oil pressures, one from each reduction geartrain, are compared in a Torque Control Unit mounted on the combining gearbox. A power turbine governor and a tachometer-generator are fitted for each engine on separate mounts on the combining gearbox and are driven by their respective engines.



1. Oil Filter
2. Torque Control
3. Reduction Gearbox
4. Exhaust Case Assembly
5. Power Turbine Shaft Housing
6. Power Turbine Shaft Housing Support
7. Nf Governor
8. Oil Filler
9. Oil Level Indicator
10. Power Turbine Disk Assembly
11. Power Turbine Guide Vane
12. Tt5 Probe
13. Compressor Turbine Disk Assembly
14. Compressor Turbine Guide Vane Assembly
15. Fire Seal Mount Ring
16. Gas Generator Case
17. Inlet Screen Support
18. Inlet Screen
19. Lifting Bracket
20. Oil-To-Fuel Heater
21. Deleted
22. Oil Filler
23. Oil Level Indicator
24. Fire Seal Mount Ring
25. Compressor Inlet Case
26. Compressor Assembly
27. Compressor Bleed Valve
28. Fuel Manifold
29. Spark Igniter Plug
30. Combustion Chamber Liner

Figure 1-5. Major sections of engine 212060-58

ENGINE AIR MANAGEMENT SYSTEM.

The air management system is a series of inter-connected ducts and doors to provide the engine with air that is free of ice, dust, and other foreign particles. During normal operating conditions, air approaching the engine inlet is partially bypassed to an ejector (figure 1-6). Air flow to the ejector is provided by a two position control door located downstream of the engine inlet duct. With the control door in the normally open position, the inertial action of the air passing into the inlet duct and through the ejector door will remove approximately 93 percent of foreign particles over 100 microns and approximately 80 percent of particles over 20 microns from engine inlet air. When the helicopter is in hover, the engine exhaust system acts as a power source to promote air flow through the ejector.

Particle Separator Switch

The particle separator switches (figure 1-8) are three-position switches labeled ON, AUTO, and OFF. In the AUTO position the air management system electrically actuated doors will be open unless caution panel segment PART SEP OFF is illuminated. If the low engine rpm warning light illuminates due to Ng dropping below 52 ±2%, with the particle separator in the AUTO position, the door will close directing all induction air into the engine. The ON position on the switch opens the air management door by retraction of electrically operated actuator and caution light will extinguish. The OFF position extends the actuator, closes the door to route all induction air into the engine and the caution panel segment will illuminate. Actuation of FIRE PULL HANDLE will close the particle separator door, regardless of switch position. Power is supplied through the 28V DC essential bus and circuit protection by breakers. labeled, ENG 1 PART SEP, and ENG 2 PART SEP.

NOTE

*

The AUTO position should be used at all times when visible moisture is evident at temperature below 5°C.

Particle Separator Caution Light

The particle separator caution lights are located on the instrument caution panel (figure 1-20) and labeled PART SEP OFF. The closing of the door (figure 1-6) in the air management system will cause the light to illuminate. Power is supplied to the caution light by the 28V DC essential bus and protected by circuit breaker CAUTION LTS.

ENGINE FUEL SYSTEM

The engine fuel system consists of separate identical engine fuel control systems, one for each engine. Fuel from the tank boost pump passes through the fuel filter oil-to-fuel heat exchanger, and engine fuel pump to the fuel control units. Metered fuel from the fuel control units is piped to the flow divider thence into the fuel manifolds (figure 1-7).

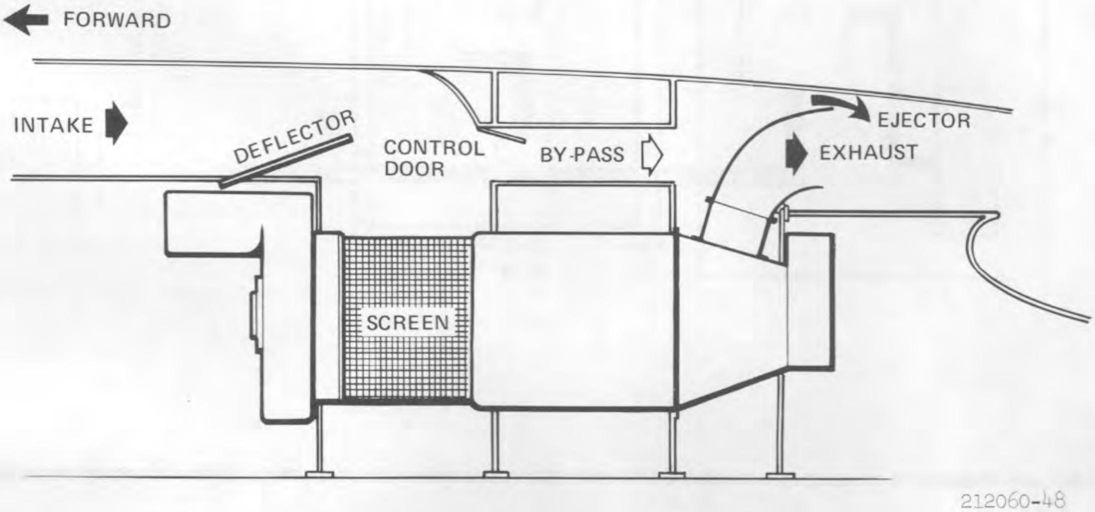


Figure 1-6. Engine air management system

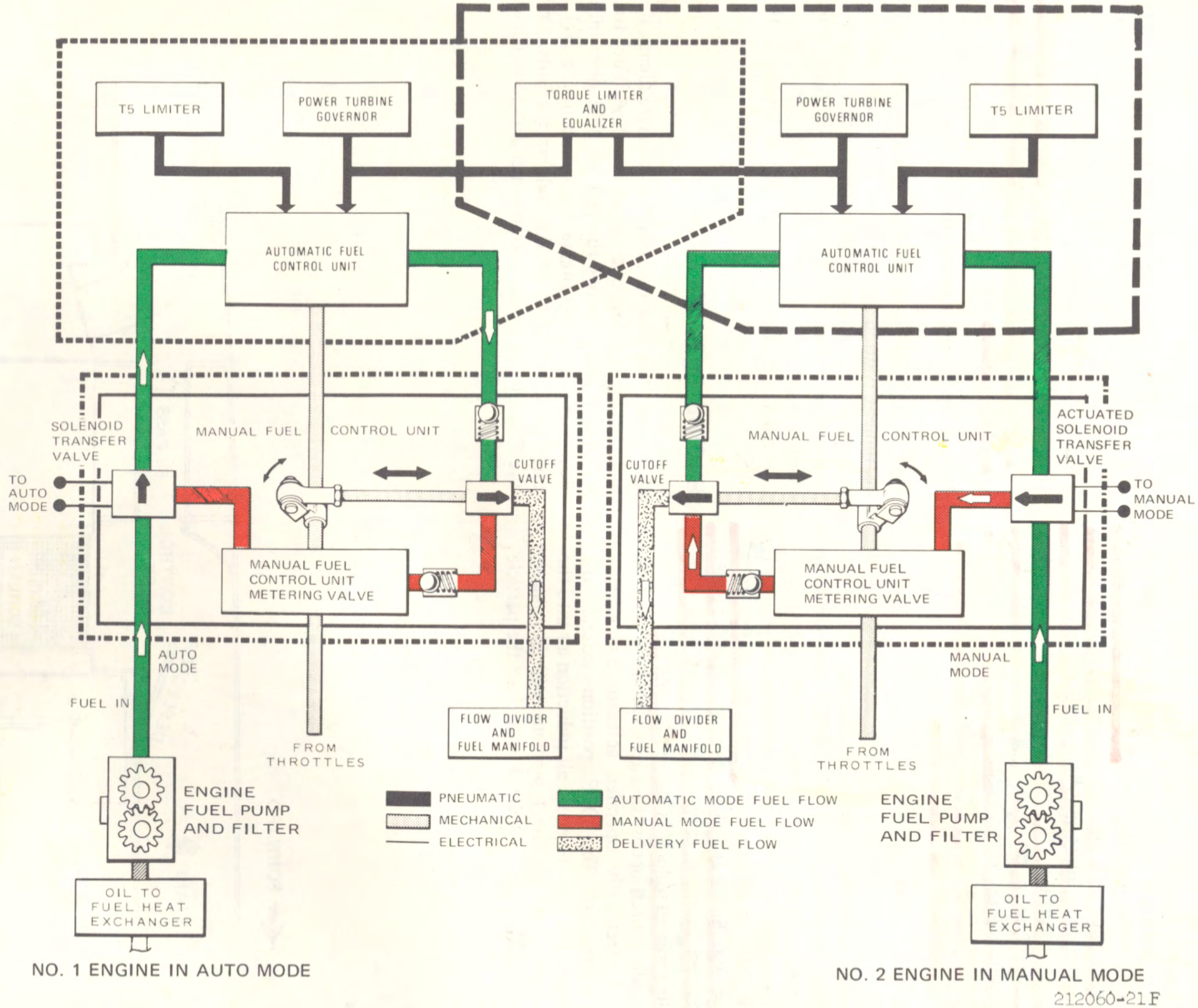


Figure 1-7. Engine fuel system

OIL-TO-FUEL HEAT EXCHANGER

The oil-to-fuel heat exchangers are located on the front face of each accessory gearbox. Accumulation of ice in each engine fuel system is prevented by the use of engine lubricating oil to heat cold low pressure fuel to a maximum temperature of 80° ± 10° F. Above 90°F a wax filled bellows actuated valve cuts off the supply of oil to the heat exchanger. The fuel heater maintains fuel temperature at inlet to the fuel pump to at least 4.5° C (40° F) at a minimum oil temperature of 71° C (160° F) with a maximum fuel flow of 540 lb/hr per engine at -53.9° C (-65° F).

ENGINE DRIVEN FUEL PUMP

A positive displacement gear-type pump is located on the front face of each accessory gearbox, between the fuel control unit and gearbox cover of each engine. These pumps deliver pressure fuel to the fuel control units. A coupling transmits drive from the accessory gearbox to the pump gearshaft. A filter is incorporated in each engine driven fuel pump. An output coupling on the pump gearshaft transmits drive to the automatic fuel control for each engine (figure 1-7).

ENGINE FUEL CONTROL UNITS

Each engine has an automatic and a manual fuel control unit. Major fuel control elements consists of a metering valve for each fuel control unit, and a pressure regulating valve. The pressure regulating valve maintains a constant pressure at the metering valve inlet by by-passing excess fuel back to the engine fuel pump inlet. In the manual position (figure 1-8) the metering valve is manually positioned; in the automatic position, the metering valve is positioned in response to various internal operating signals.

Automatic Fuel Control Units

Under normal operating conditions (AUTO position, figure 1-8) fuel passes from the engine driven fuel pump through a governor solenoid and transfer valve within the manual fuel control unit, then to the automatic fuel control unit. Metered fuel flows from the automatic fuel control unit through a cutoff valve within the manual fuel control unit and then to the flow divider and fuel manifold. Each automatic fuel control system prevents Ng overspeed and Nf overspeed, maintains Ng idle speed, and balances and limits torque output from each engine.

Manual Fuel Control Unit

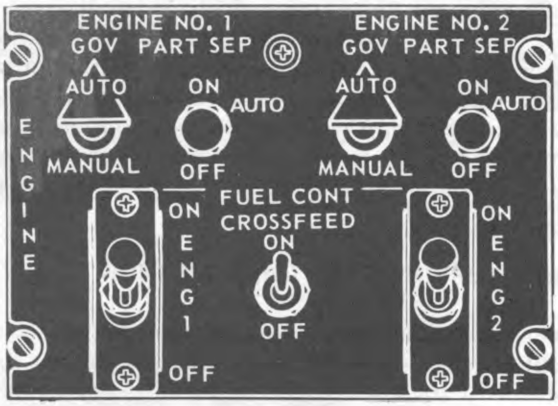
The fuel control system provides a manual control, which may be selected to operate the engines in the event of a malfunction of the automatic system. The power control is a fully hydro-mechanical unit which derives its operation solely from a direct geared connection to the Ng section of the engine. The fuel control requires no electrical or other outside source of power except for manual switchover. However, if electrical power is lost while operating in the manual mode, the transfer valve will return to the de-energized position placing engine operation on the automatic fuel control.

POWER TURBINE SPEED (Nf) GOVERNOR

The engine fuel and power control system permits the pilot to obtain maximum performance from the engines with a minimum of attention. Under normal flight conditions the Nf speed is controlled by the power turbine speed governor.

TORQUE CONTROL UNIT

A single torque control unit, mounted on the reduction gearbox and common to automatic fuel control systems of both engines, receives torquemeter oil pressure signals proportional to the torque outputs of the two engines. Engine total torque is limited by summing the torquemeter pressures and comparing with a spring reference, if an overtorque condition is imminent, pneumatic signals are sent to both AFCU's to reduce fuel flow and thus reduce torques. Limiter setting range is 102 to 105% of transmission torque. Equalization of engine output torques is achieved by comparing the two torquemeter pressures and then sending an "increase fuel flow" signal to the AFCU of the relatively low-torque engine.



212070-12

Figure 1-8. Engine panel

FUEL CONTROL LINE (P3) HEATER

The fuel control sense line heaters prevent accumulation of ice in the engine governing system by maintaining P3 air temperature of at least 40° F regulated by a thermostat. The fuel control sense line heaters are energized by the essential DC bus through the FUEL CONT HTR circuit breaker (figure 1-18). ENG 1 FUEL CONT switch (figure 1-8) controls the operation of engine No. 1 fuel control sense line heater, and ENG 2 FUEL CONT switch (figure 1-8) controls the operation of engine No. 2 fuel control sense line heater. The heaters are automatic and no cockpit indications are provided.

FUEL FLOW DIVIDER (FUEL MANIFOLD AND NOZZLES)

Metered fuel is fed to a dual manifold system consisting of two separate assemblies of seven simplex fuel nozzles interconnected by a flow divider assembly. Each manifold is similar, and the flow divider maintains a pressure differential between the primary and secondary manifold. The primary manifold is the only manifold used during initial starting sequence.

ENGINE CONTROLS

THROTTLES

The throttles are actuated by a dual-twist grip on the collective pitch control lever (figure 1-10) labeled ENGINE 1 and ENGINE 2. A friction control for each engine is located at the top on ENGINE 1 throttle and bottom on ENGINE 2 throttle. Friction may be induced by rotating the friction control ring to the left, to increase friction, and to the right to decrease friction. RPM selected within the governing range of 97% to 100% will be automatically maintained on both engines, within the limits of the engines' automatic fuel control and droop compensator systems.

ENGINE IDLE RELEASE (STOP) SWITCH

The engine idle stop release switches, located on the pilot's collective switch box (figure 1-10) and the instructor pilot engine control panel (figure 1-9), are three position type, labeled ENG 1 (left) and ENG 2 (right), and are spring loaded to the off (center) position. Power is furnished to the switch through the 28V DC essential bus (figure 1-17) and protection is provided by the IDLE STOP circuit breaker (figure 1-18). There is a solenoid and plunger for each throttle that is mounted to act as a stop in the power control linkage. When the idle stop release switch is actuated, the corresponding solenoid is energized, which retracts the plunger, allowing full throttle movement. A time delay circuit holds the plunger in the retracted position for approximately five seconds after the idle stop release switch is released. During engine start the plunger prevents throttle movement beyond the ground idle position, permitting engine shutdown in the event of a hot start without activating the idle stop release switch. The idle stop release switch must be actuated to advance the throttle beyond the ground idle position. After the throttle is advanced to the normal operating range, the plunger will not allow throttle reduction below flight idle, effectively preventing inadvertant engine shutdown.

NOTE

Throttle pressure against the idle stop in either an advance or retard direction, when the switch is actuated, will prevent plunger release.

BEEP (RPM INCREASE – DECREASE) SWITCH

A beep switch is mounted in the switch box located on each collective control lever (figure 1-10). Each switch is a three-position, momentary type and is held forward to INCR, aft to DECR the power turbine (Nf) speed. Regulated power turbine speed may be adjusted in flight through the operating range of 97 to 101.5% ± 0.5% rpm. For single engine operation, operating range can be expected to drop approximately 2%. Power is supplied through the 28V DC essential bus and protected by circuit breaker GOV CONT (figure 1-18).

GOVERNOR SWITCH

The fuel control solenoid valves, through the governor switch, provide a method of bypassing the automatic control governors. Each solenoid is protected by a separate GOV MAN CONT circuit breaker. When ENG No. 1 GOV and ENG No. 2 GOV switches are to the AUTO position, the solenoids are de-energized, the twist-grip throttle controls are manually set full on and the speed of each engine is controlled by the governor. When the switches are set to MANUAL, the solenoids are energized, the governors are bypassed, and engine control is by the manual twist-grip position. Placing either switch to MANUAL also provides a 28 volt signal to the master caution panel and the legend GOV MAN is illuminated.

AUTOMATIC FUEL FLOW

Fuel from the helicopter fuel tanks is supplied by the tank-mounted fuel boost pumps (figure 1-15) to the engine fuel pumps, and then is delivered to the solenoid actuated valve. With the governor switch in the AUTO position (figure 1-8) fuel flows through the solenoid valve to the automatic fuel control unit.

CAUTION

When operating on automatic fuel flow it is possible to exceed maximum ITT.

MANUAL FUEL FLOW

With the governor switch in the MANUAL position, fuel is metered to the engine without automatic overspeed or over temperature, protective features.

CAUTION

When operating on manual it is possible to overspeed the Ng turbine, Nf turbine and to exceed maximum ITT.

Manual Fuel Flow - Caution Light

A caution light for each engine is provided on the caution panel (figure 1-20) labeled GOV MANUAL. Illumination of these worded segments indicate the appropriate switch (figure 1-8) is in the manual position and the fuel control solenoid valve is closed.

NOTE

It is possible to have a GOV MANUAL light and still be in the automatic mode if a malfunction exists in the fuel control valve, solenoid or wiring.

INSTRUCTOR PILOT ENGINE CONTROL PANEL

The instructor pilot engine control panel provides the instructor pilot with the capability to start, stop, or deenergize the pilot's starter switch. The panel is located on the left-hand side of pedestal (figure 1-12). The panel (figure 1-9) contains an idle stop switch, starter-ignition switch, and an override switch. The panel operation is the same as the operation of the switches used in the start sequence on the pilot's collective with the exception of the override switch.

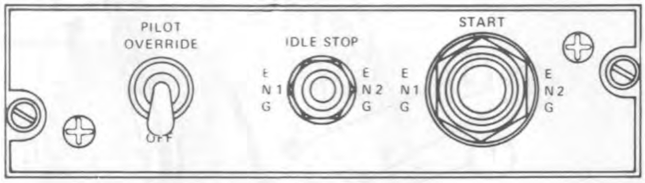


Figure 1-9. Instructor pilot engine control panel

Override Switch

With the override switch on, the instructor pilot denies power to the pilot's starter switch. In the OFF position, power is available for engine start from both the pilot's start switch and the instructor pilot's panel. Regardless of override switch position both idle stop release switches are energized and when actuated will retract engine idle release plunger. Power is furnished through the 28V DC essential bus (figure 1-17) by tieing into existing wiring for the pilot's starter switch. The override switch is furnished circuit protection by the ENG 1 START and IGN and ENG 2 START and IGN circuit breakers (figure 1-18).

NOTE

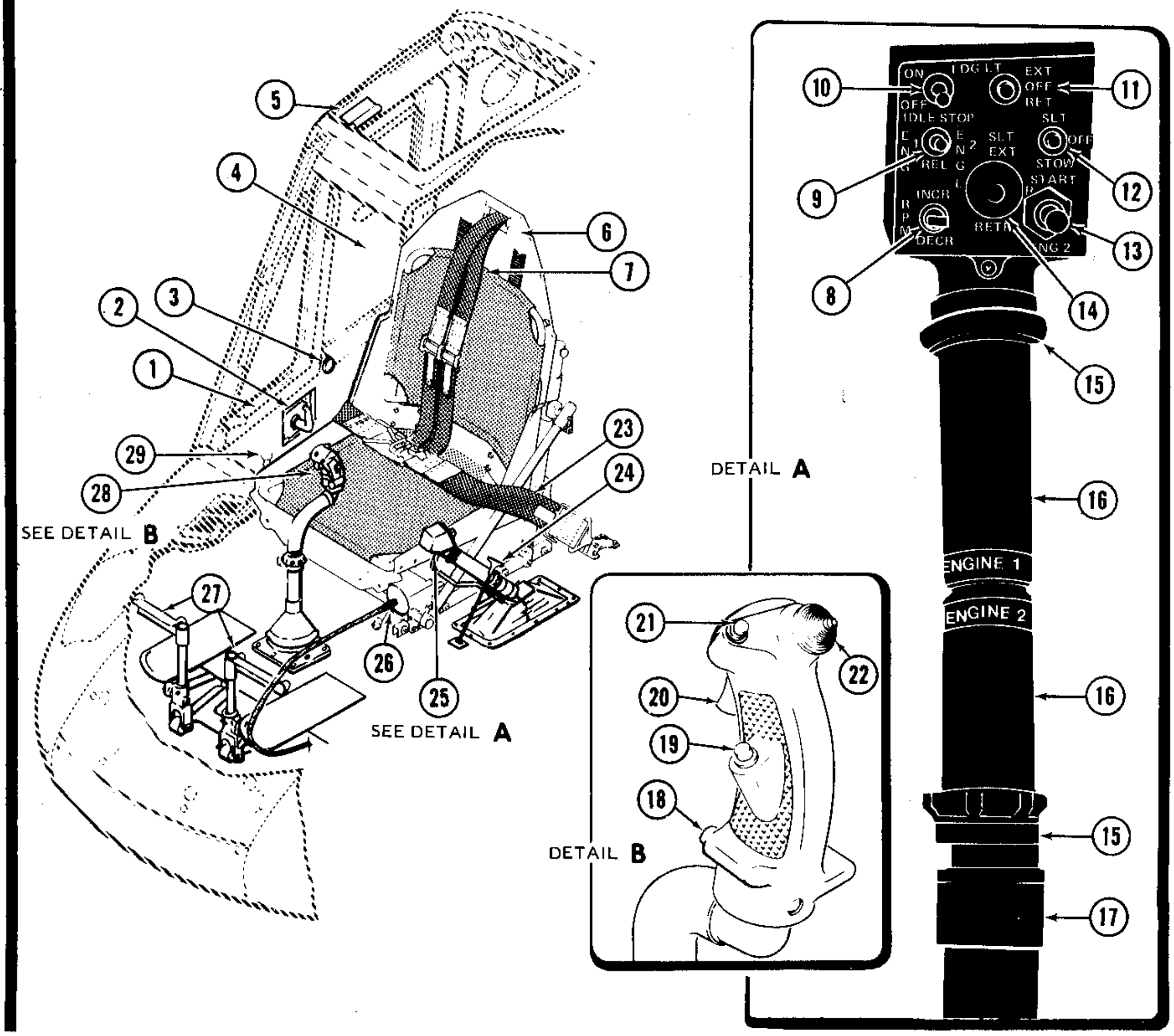
With the override switch in the off position it is possible to engage both starters simultaneously. This should not be attempted as an extreme voltage drain will occur and possible destruction of the aircraft battery could occur.

Idle Stop Switch

The idle stop switch is identical to the switch located on the pilot's collective and operation is the same. Power is furnished through the 28V DC essential bus by tieing into the pilot's idle stop wiring. Circuit protection is furnished by the same IDLE STOP circuit breaker.

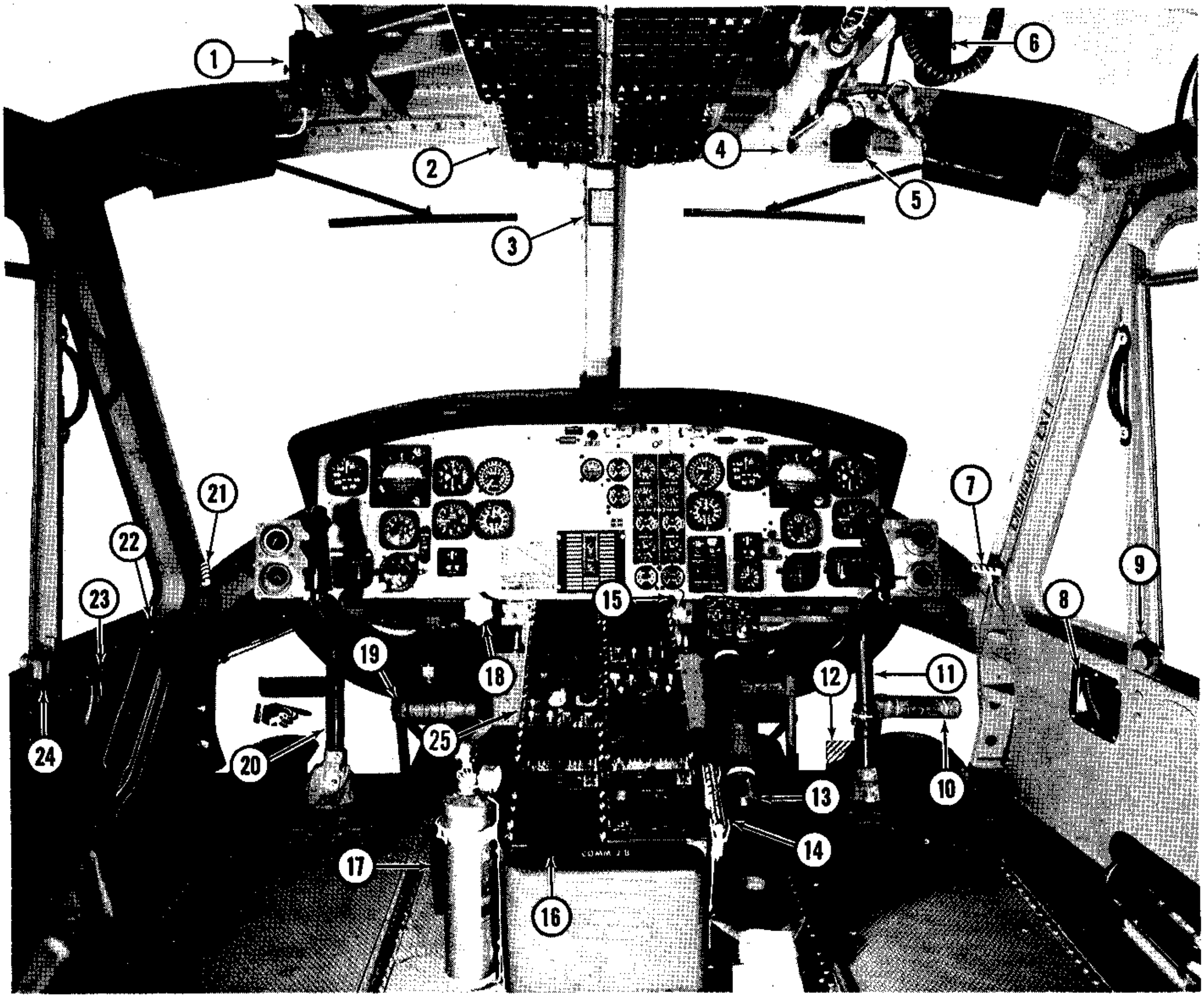
Starter-Ignition Switch

The starter-ignition switch is supplied power by tieing into existing wiring from the pilot's starter-ignition switch which is supplied power by the 28V DC essential bus. Circuit protection is provided by the same ENG START & IGN circuit breakers as the pilot's starter switch.



1. Pilot's Door
2. Door Handle
3. Window Friction Knob
4. Sliding Window Panel
5. Handhold
6. Pilot's Seat
7. Shoulder Harness
8. Rpm Increase-Decrease Beep Switch
9. Engine Idle Stop Release
10. Landing Light On-Off Switch
11. Landing Light Extend-Retract Switch
12. Searchlight On-Off Stow Switch
13. Engine Start Switch
14. Searchlight Extend-Retract-Left-Right Switch
15. Friction Adjustment - Throttle
16. Throttles
17. Friction Adjustment - Collective Pitch Control
18. Cargo Hook Release
19. Armament
20. Communications Switch - ICS and Radio
21. Force Trim
22. Rescue Hoist
23. Safety Belt
24. Collective Pitch Down Lock
25. Collective Pitch Control Lever
26. Adjuster - Directional Control Pedals
27. Directional Control Pedals
28. Cyclic Control Stick
29. Shoulder Harness Lock-Unlock Control

Figure 1-10. Pilot's compartment — typical (Sheet 1 of 2)



1. Copilot's Cockpit Light
2. Circuit Breaker and Overhead Console Panel
3. Compass Card
4. Rotor Brake
5. Standby Compass
6. Pilot's Cockpit Light
7. Pilot's Door Jettison Handle
8. Pilot's Door Handle
9. Pilot's Window Adjuster
10. Pilot's Directional Control Pedals
11. Pilot's Cyclic Stick
12. Manual Cargo Release Pedal
13. Pilot's Collective Control Lever
14. External Stores Manual Release Handle
15. Heater/Defrost Control Lever
16. Pedestal Panel
17. Data Case
18. Copilot's Directional Control Pedal Adjuster
19. Copilot's Directional Control Pedals
20. Copilot's Cyclic Stick
21. Copilot's Door Jettison Handle
22. Copilot's Collective Control Handle
23. Copilot's Door Handle
24. Copilot's Window Adjuster
25. Copilots Checklist Holder

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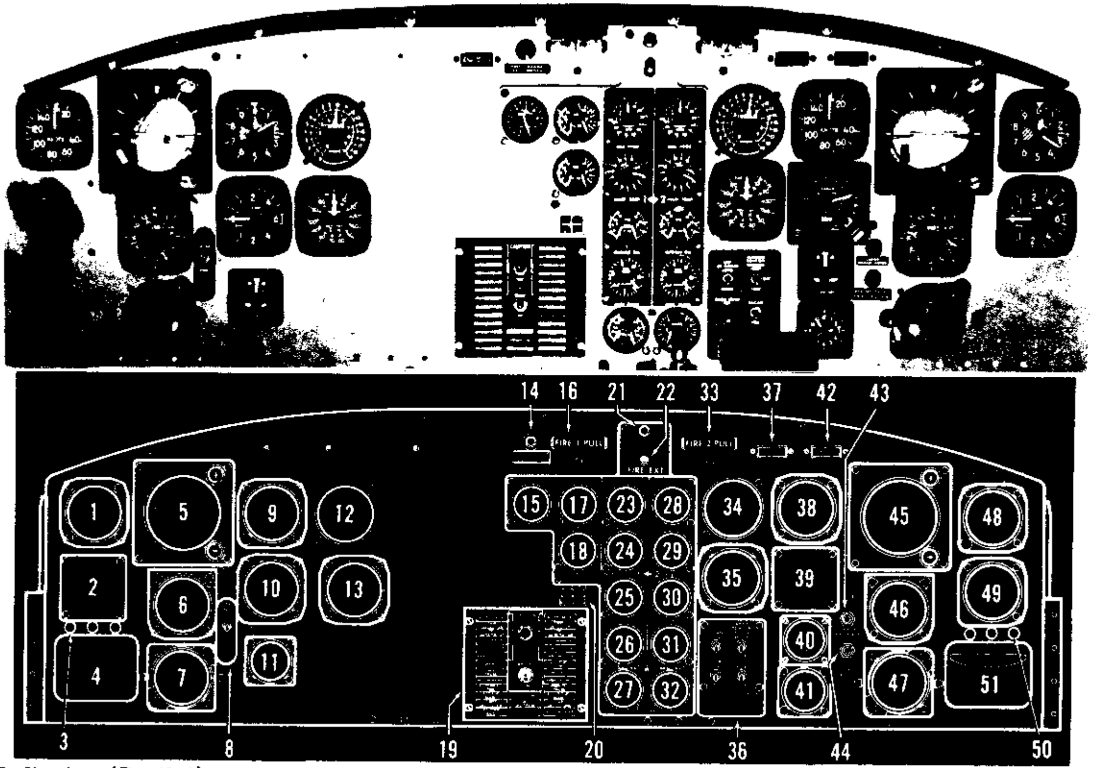
Figure 1-10. Pilot's compartment – typical (Sheet 2 of 2)



1. Airspeed Indicator (Copilot)
2. Radar Altimeter (When Installed)
3. Marker Beacon Indicator Lights *
4. Ash Tray
5. Attitude Indicator
6. Bearing Distance Heading Indicator (BDHI)
7. ID-387 Course Indicator
8. TACAN-VOR Selector Switch
9. Altimeter
10. Vertical Velocity Indicator
11. Turn and Slip
12. Dual Torque Indicator
13. Tripple Tachometer
14. Fuel Gauge Test Switch
15. Fuel Quantity Indicator
16. Fire Pull Handle (Eng. 1)
17. Transmission Temperature and Pressure Indicator
18. Combining Gearbox Temperature and Pressure Indicator
19. Caution Panel
20. Chip Detector Panel
21. Press To Test Switch (Fire Handle Lights)
22. Fire Extinguisher Selector Switch
23. Gas Producer Tachometer (Eng. 1)
24. Inlet Temp. Indicator (ITT, Eng. 1)
25. Engine Oil Temperature and Pressure (Eng. 1)
26. Fuel Flow Indicator (Eng. 1)
27. Voltmeter
28. Gas Producer Tachometer (Eng. 2)
29. Inlet Temp. Indicator (ITT, Eng. 2)
30. Engine Oil Temperature and Pressure (Eng. 2)
31. Fuel Flow Indicator (Eng. 2)
32. Ammeter
33. Fire Pull Handle (Eng. 2)
34. Dual Torque Indicator
35. Triple Tachometer
36. NAV Equipment Selector Panel
37. Master Caution Light
38. Airspeed Indicator (Pilot)
39. Radar Altimeter (When Installed)
40. Turn and Slip
41. Clock
42. RPM Warning Light
43. Cargo Release Armed Light
44. Rescue Hoist (20 foot) Caution Light
45. Attitude Indicator
46. Bearing Distance Heading Indicator (BDHI)
47. ID-387 Course Indicator
48. Altimeter
49. Vertical Velocity Indicator
50. Marker Beacon Indicator Lights *
51. Ash Tray

* A/C Modified by T.O. 1H-1(U)N-591

Figure 1-11. Instrument panel typical

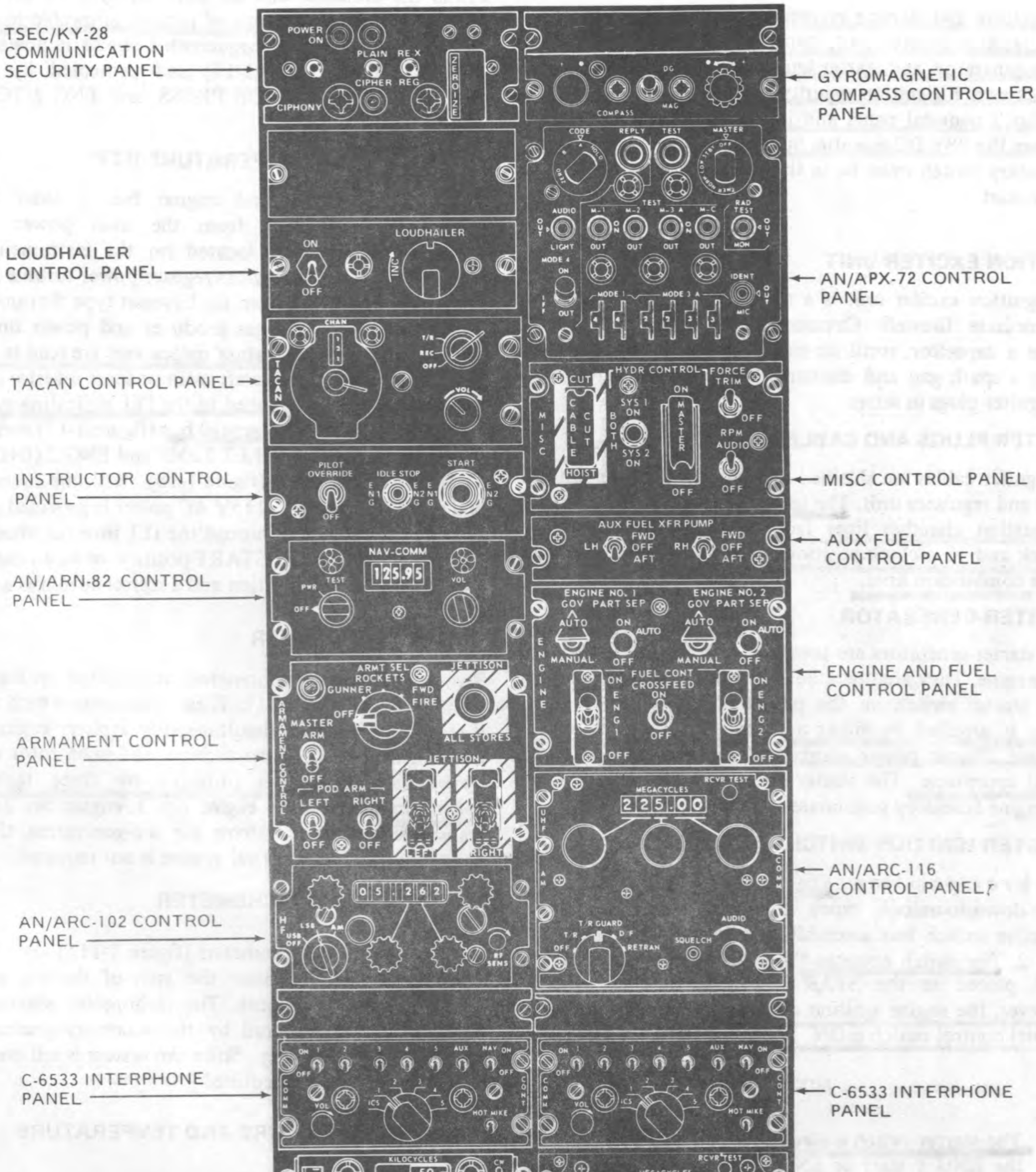


Figure 1-12. Pedestal console – typical

ENGINE IGNITION AND STARTING SYSTEM

The ignition and starting system for each engine consists of an ignition exciter unit, two shielded igniter plugs, starter-generator, and starter ignition switch. A rotary keylock switch (ignition security device) is installed on the pilot's pedestal panel and is electrically connected between the 28V DC essential bus and the starter switch. The rotary switch must be in the On position to initiate engine start.

IGNITION EXCITER UNIT

The ignition exciter unit is a sealed unit mounted to the intermediate firewall. Circuits in the unit progressively charge a capacitor, until its stored energy is sufficient to ionize a spark gap and discharge the capacitor across the two igniter plugs in series.

IGNITER PLUGS AND CABLES

The igniter cables are shielded and grounded to the igniter plugs and regulator unit. The igniter plugs protrude into the combustion chamber liner from bosses positioned at 4 o'clock and 11 o'clock positions near the rear, domed end of the combustion liner.

STARTER-GENERATOR

Two starter-generators are provided, one for the starting of each engine. They are operated independently of each other by a starter switch on the pilot's collective switch box. Power is supplied by either a 24-volt battery or from an external 28-volt power source plugged into the external power receptacle. The starter generators are mounted on the engine accessory gear-boxes.

STARTER IGNITION SWITCH

A three-position starter ignition switch (push-down-to-unlock type) is mounted on the pilot's collective switch box assembly and is marked ENG 1 and ENG 2. The switch actuates the starter and ignition circuit when placed in the START position for that engine. However, the engine ignition circuit is not energized unless the fuel control switch is ON.

NOTE

The starter switch is electromagnetically held in the ENG 1 start or ENG 2 start position and must be manually returned to center position.

ENGINE INSTRUMENTS AND INDICATORS

DUAL TORQUEMETER

The pilot's and copilot's dual torquemeter located on the instrument panel (figure 1-11) provide continuous readings of engine output shaft torque in percentage by means of two torque pressure transmitters, one for each engine. Each torque indicator simultaneously displays the torque output of both engines on the inner dial, and the torque to the transmission (combined torque of both engines) on the outer dial. The torque output of engine No. 1 is indicated by pointer No. 1, and the torque output of engine No. 2 is indicated by pointer No. 2. These torques are totalized within the indicator and the sum displayed by the cursor. All indications are in terms of percent allowable torque to the transmission. The torquemeter circuit is powered by 26V AC BUS (figure 1-17) and protected by circuit breakers ENG 1 TORQUE PRESS, and ENG 2 TORQUE PRESS (figure 1-18).

INTER TURBINE TEMPERATURE (ITT)

The engine No. 1 and engine No. 2 inter turbine temperature is read from the inlet power turbine temperature indicators located on the instrument panel (figure 1-11). The indicators register power turbine inlet air temperature received from the bayonet type thermocouples mounted between the gas producer and power turbine in the engine. The temperature indications are read in degrees centigrade. With the aircraft main or spare inverter in operation, power is supplied to the ITT indicating system through the 115V AC essential bus (figure 1-17) and protected by the ENG 1 INLET TEMP and ENG 2 (INLET TEMP) circuit breakers (figure 1-18). With the aircraft main and spare inverters off, 115V AC power is provided to the ITT indicating system through the ITT inverter when the battery switch is in the START position or when the battery switch is in the ON position and a starter switch is actuated.

TRIPLE TACHOMETER

Two triple pointer tachometers are located on the instrument panel (figure 1-11). These tachometers each contain three pointers which simultaneously register engine No. 1 and engine No. 2 power turbine, and main rotor rpm, in percentage. Power is provided by three tachometer generators mounted on engine No. 1, engine No. 2 and the transmission. These systems are self-generating, therefore connection to the electrical system is not required.

GAS PRODUCER TACHOMETER

The gas producer tachometers (figure 1-11) located on the instrument panel, register the rpm of the gas producer turbines (Ng), in percent. The tachometer generators are located on and powered by the accessory gearboxes for each engine respectively. Since the system is self contained, no electrical power is required.

ENGINE OIL PRESSURE AND TEMPERATURE

The engine No. 1 and No.2 oil temperature and pressure indicators (figure 1-11) are dual-type indicators, each registering temperature in centigrade and pressure in psi. The indications registered on the gage must be multiplied by ten to obtain actual value. The temperature portion receives indications from an electrical resistance-type bulb located on the lower part of the accessory gearbox. The pressure portion receives indications from an engine oil pressure transmitter (figure 1-14) located on the lower part of the accessory gearbox adjacent to the oil temperature bulb. The oil temperature portion is electrically powered by the 28V DC essential bus (figure 1-17) and protected by circuit breakers ENG 1 OIL TEMP and ENG 2 OIL TEMP (figure 1-18). The pressure portion is 26V AC powered (figure 1-17) from the 26V AC bus through the ENG 1 OIL PRESS and ENG 2 OIL PRESS circuit breakers (figure 1-18) when the aircraft main or spare inverter is operating. With the aircraft main and spare inverters off, 26V AC power is provided to the engine oil pressure indicating system from the ITT inverter when the battery switch is in the START position or when the battery switch is in the ON position and a starter switch is actuated.

ENGINE OIL LOW PRESSURE – CAUTION

Engine OIL PRESSURE caution lights for each engine are located on the instrument panel (figure 1-11). The lights are connected to a pressure switch on each engine (figure 1-14) which makes contact upon a pressure drop below safe limits and illuminates the appropriate caution light at approximately 30 psi. The power is supplied through the 28V DC essential bus (figure 1-17) and protected by circuit breakers CAUTION LTS (figure 1-18).

GEARBOX OIL TEMPERATURE AND PRESSURE

The gearbox oil temperature and pressure indicator is a dual-type indicator (figure 1-11), registering oil temperature in centigrade and oil pressure in psi in the combining gearbox. The indications registered on the gage must be multiplied by ten to obtain actual value. The temperature portion receives indications from an electrical resistance-type bulb and the pressure portion receives its signal from the combining gearbox oil pressure transmitter (figure 1-14) located on top of the combining gearbox. The temperature portion is powered by the 28V DC essential bus (figure 1-17) and protected by circuit breaker C BOX OIL TEMP (figure 1-18). The pressure portion is powered by the 26V AC bus and protected by circuit breaker C BOX OIL PRESS. If low oil pressure has occured in the gearbox a light on the caution panel (figure 1-20) labeled C BOX OIL PRESS will illuminate at approximately 31 psi. A chip (magnetic particle) in the oil will illuminate a light on the caution panel labeled CHIP DETECTORS and in turn will illuminate a light on the chip detector panel (figure 1-21), labeled C BOX.

COMBINING GEARBOX CLOGGED FILTER-CAUTION

A combining gearbox clogged filter caution light is located on the caution panel (figure 1-20), labeled C BOX OIL FILTER. When differential pressure exceeds approximately 30 psi the caution light illuminates. The power is supplied through the 28V DC essential bus (figure 1-17) and protected by circuit breaker CAUTION LTS (figure 1-18).

FUEL FLOW

Engine No. 1 and engine No. 2 fuel flow indicators register rate of fuel flow in lb/hr. The transmitters are located in the fuel lines between the fuel control and fuel manifold on each engine. Power is supplied through the 115V AC essential bus (figure 1-17) and protected by circuit breakers ENG 1 FUEL FLOW and ENG 2 FUEL FLOW (figure 1-18).

ENGINE OIL SUPPLY SYSTEM

The engine oil supply system (figure 1-14) consists of three independent oil systems: one for each engine and a third for the combining gearbox output section. The operation of the oil systems is completely automatic and self-regulating; therefore, no operator control is necessary other than monitoring of the instrument indications. The engine oil system lubricates and cools bearings, mesh gears, and is also used in the torquemeter system to indicate engine output torque. The combining gearbox oil system has a lubrication and cooling function. Each system has its own integral tank, oil level sight glass, filler opening, filter, and drain plugs. The sight glass and filler opening is located on the accessory gearbox of each engine. The oil level sight glass, filler opening, and drain plug of the combining gearbox (output section) oil system is located on the combining gearbox. Each engine oil system has two drain plugs; one on the accessory gearbox, the other on the combining gearbox. Oil cooler blowers for all three systems are located on the combining gearbox. (figure 1-14).

OIL COOLERS

Engine and combining gearbox output section oil is cooled by three radiators with forced airflow. Engine oil is delivered to the oil coolers by a pressure pump located in the engine oil tank. Combining gearbox oil is cooled by oil drawn through an inlet screen from the oil tank and delivered to the combining gearbox oil cooler. The oil coolers are three individual coolers located aft of the combining gearbox. One cooler is used for each engine, and the third for the output section of the combining gearbox. This cooler has two sections, one the combining gearbox, the other for the main transmission. Air is supplied to the oil coolers by two geardriven blowers mounted on the aft end of the combining gearbox. Blower operation will cease with engine shutdown. Either blower will provide sufficient cooling air to maintain oil temperatures during single engine operation.

ENGINE OIL TANKS

The engine oil tank is an integral part of the accessory gearbox assembly, its rear face being formed by the compressor inlet case front face, and its forward face being formed by the accessory gearbox housing casting lower half. The tank has a total capacity of 6.4 quarts, of which 3 quarts is usable oil. An expansion space of 2 quarts is provided with an oil filler neck on either side, one of which will be blanked off in use. A filler cap and strainer element are fitted in the tank filler neck. A drain plug in the bottom of the tank facilitates oil drainage. The combining gearbox oil tank is an integral part of the reduction gearbox assembly and has a total capacity of 5 quarts of which 1.0 quart is usable oil. A filler plug and oil screen are incorporated in the oil filler assembly. A drain plug in the bottom of the tank facilitates oil drainage. The oil grade and specification are shown in the servicing diagram figure 1-22.

ROTOR SYSTEM

The rotor system consists of the main rotor, tail rotor and drive system, rotor tachometer, rotor brake, and rpm warning system.

MAIN ROTOR

The main rotor is 48 feet in diameter, and is a two-bladed, semirigid flapping type, employing preconing and underslinging. The main rotor blade is a thin tip blade tapering from a 12% airfoil at the 80% radius to a 6% airfoil at the tip. The blades are bonded (metal to metal) and interchangeable. The assembly consists of two aluminum bonded blades, with corrosion and scuff resistant stainless steel edges, blade grips, yoke, mast stabilizer bar, and rotating controls. Each blade is connected to a common yoke by a bladegrip and pitch change bearings, with tension straps to carry centrifugal forces.

The rotor assembly is attached to the mast with a bearing mounted trunnion, allowing the rotor to flap. The trunnion is secured to the mast by splines and a nut-cap fitting, which incorporates provisions for cable attachment used in hoisting the helicopter. Blade pitch change is accomplished by movement of the collective pitch control lever and a series of controls terminating at the blade grip horn. Power for the rotor system is transmitted by the two-stage planetary transmission.

ROTOR BRAKE

A rotor brake is mounted on the left side of the transmission and provides positive means of stopping and holding the rotors in a stationary position. The rotor brake hydraulic system is a separate and independent system. A hand operated lever mounted adjacent to the overhead console in the cockpit is actuated by the pilot pulling the lever to the down position. This action applies hydraulic pressure from a hydraulic cylinder and compresses the brake linings against the disc, stopping the rotors through the mechanical couplings. The rotor brake caution light (figure 1-20) is actuated by the hydraulic pressure switch (located on the cylinder) at 10 psi. When pressure drops to 5 psi, light will extinguish. Power is supplied by the 28V DC essential bus (figure 1-17). A relief valve actuates preventing excessive rate of rotor deceleration when rotor brake handle is placed in the lock (on) position.

RPM WARNING SYSTEM

The rpm limit warning system includes the RPM Limit Warning Control, the RPM Warning Light on the instrument panel (figure 1-11), an audio rpm switch (figure 1-12), and the RPM WARN circuit breaker (figure 1-18). The RPM Limit Warning Control, receives information relative to the rpm of the main rotor and the rpm of each engine through tachometer sensor circuits. The RPM Limit Warning Control interprets these tachometer signals and causes the RPM warning light to illuminate whenever a low or high rpm condition exists with the main rotor or a low rpm condition with engine No. 1 or engine No. 2. It also provides an audio control signal (28V DC) which causes an audio tone to be present in pilot's and co-pilot's headsets whenever a low rpm condition exists with the main rotor. The audio tone may be shut off by means of RPM AUDIO switch (figure 1-12). The switch will automatically reset to the ON position when rotor rpm rises above the lower limit. A Low Engine RPM signal (through the RPM Limit Warning Control) will also cause the particle separator on the low engine to be by-passed and the bleed air solenoids on both engines to be closed. The particle separator system will only be activated if the PART SEP switch is in the AUTO position.

Light warning	– Rotor rpm drops below 92% ± 2 or exceeds 103% ± 2. Either engine Ng speed drops below 52% ± 2.
Audio warning	– Rotor rpm drops below 92% ± 2.

ROTOR SYSTEM TACHOMETER

This tachometer is part of the triple tachometer and is located in the instrument panel (figure 1-11). The rotor rpm is indicated on inner scale and the pointer is marked with an R. The indicator is powered by a tachometer generator mounted on and driven by the transmission. The indicator and tachometer generator operate independently of the helicopter electrical system. The tachometer system is a synchronous motor type; as rpm changes, the frequency and magnitude output of the generator varies. The variable output frequency and magnitude from the generator operates the motor in the indicator at the same speed, thus providing a direct reading of the rotor rpm in percent.

TAIL ROTOR

The tail rotor is a two-bladed tractor type located on the right hand side of the tailboom fin and turns counterclockwise (as viewed facing the tail rotor). It is a semi-rigid delta-hinged type, employing preconing and under slinging. Each blade is connected to a common yoke by means of a grip and pitch change bearing. The blade and yoke assembly is mounted on the tail rotor shaft by means of a delta-hinge trunnion to minimize rotor flapping.

TAIL ROTOR DRIVE SYSTEM

Two gearboxes are employed in the tail rotor drive system. A 42-degree gearbox is located at the base of the tail fin mounted on the boom and changes direction of the tail rotor drive shaft 42 degrees. At the top of the tail boom vertical fin is 90 degree gearbox which changes direction of drive 90 degrees and provides a 2.6:1 speed reduction for the tail rotor. Each gearbox has an electric indicating, magnetic drain plug. These magnetic chip detectors illuminate caution lights (figures 1-20 and 1-21) when metal is detected in the gearbox. The tail rotor drive shaft is composed of shaft sections supported along the tail boom by bearings and hanger assemblies. The shafts are coupled to one another and the gearboxes by flexible couplings.

TRANSMISSION SYSTEM

TRANSMISSION

The transmission is located forward of the engine and coupled to the engine by a main drive shaft into the combining gearbox. The transmission is limited to 1290 shp. The transmission drives the main rotor mast through a series of gears and has accessory drives for the hydraulic pumps, rotor brake disc, tachometer-generator, and tail rotor drive shaft. A magnetic chip detector is installed in the transmission which activates the caution warning system (figure 1-20) and illuminates a light on the chip detector panel (figure 1-21), if magnetic particles occur in the oil.

FREEWHEELING UNIT

The freewheeling unit is located in the input quill coupling of the transmission and disengages to allow main rotor and gear train to turn freely when the engines are below rotor driving speed. The unit provides a positive disconnect from the engines in case of power failure.

TRANSMISSION OIL SYSTEM

Transmission lubrication is accomplished by a self contained pressure oil system, with the oil pump immersed in the wet sump located at the lower end of the transmission unit. Quick disconnect couplings are used on the tubing and electrical connectors on the cable assemblies. Oil level sight gauges on the transmission sump case can be checked through a view port on the pylon support structure in the cabin, which has a light with a push-button switch. Transmission oil system capacity is 2.75 gallons. (Figure 1-13 and servicing diagram figure 1-22.)

TRANSMISSION OIL COOLER

A transmission oil cooler (figure 1-13) is incorporated in the transmission oil system. The transmission oil cooler is installed aft of the engine combining gearbox and receives cooling air from the blowers mounted on the combining gearbox. This cooler is combined with the cooler for the engine combining gearbox output section as a single unit (figure 1-14). An oil filter is mounted in the transmission compartment for cleaning the oil of foreign particles. The filter head has a bypass valve to assure oil flow in the event of clogging, with a visual red indicator which pops up if bypass condition is impending. During starts in cool ambient temperatures, a relief valve will open to bypass the cooler until oil warms, then will close for normal flow.

INDICATORS

The TRANS OIL pressure and temperature indicator (figure 1-11) is located in the center of the instrument panel. This instrument displays pressure indications from the transmission oil pressure transmitter on the left side of the indicator. The oil temperature indication is shown on the right side. The indicator incorporates a temperature bulb which controls the temperature indication. Readings on the indicator are provided in psi and degrees centigrade which then must be multiplied by ten for actual value. Electrical power for the temperature portion of the indicator and transmitter is supplied by the 28V DC circuit (figure 1-17). The pressure portion is powered by the 26V AC bus (figure 1-17). Circuit protection is provided by the XMSN OIL TEMP and XMSN OIL PRESS circuit breakers (figure 1-18).

TRANSMISSION OIL PRESSURE CAUTION LIGHTS

A caution light marked XMSN OIL PRESS is located on the CAUTION panel (figure 1-20). This caution light is electrically connected to a transmission-mounted pressure switch which is actuated by the transmission oil pressure. A drop in oil pressure below approximately 30 psi closes the electrical circuit and illuminates the caution light. The circuit is supplied power by the 28V DC essential bus (figure 1-17) and protected by circuit breaker CAUTION LTS (figure 1-18).

TRANSMISSION OIL TEMPERATURE CAUTION LIGHTS

A caution light marked XMSN OIL HOT is located on the CAUTION panel (figure 1-20). The light is connected to a transmission-mounted thermoswitch which, when heated by transmission oil to a temperature above approximately 110° C closes an electrical circuit and illuminates the caution light. The caution light circuit is supplied power by the 28V DC essential bus (figure 1-17) and protected by circuit breaker CAUTION LTS (figure 1-18).

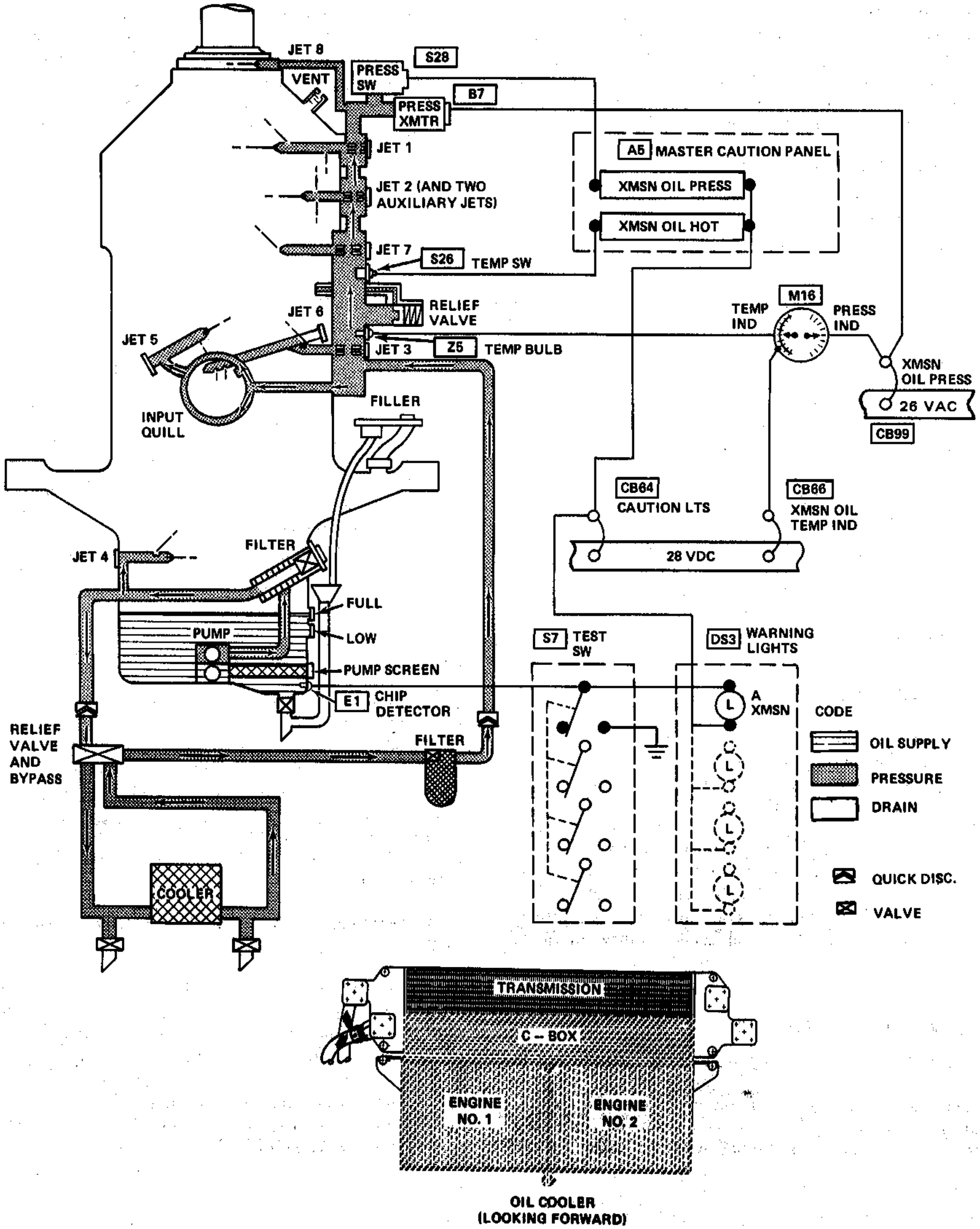
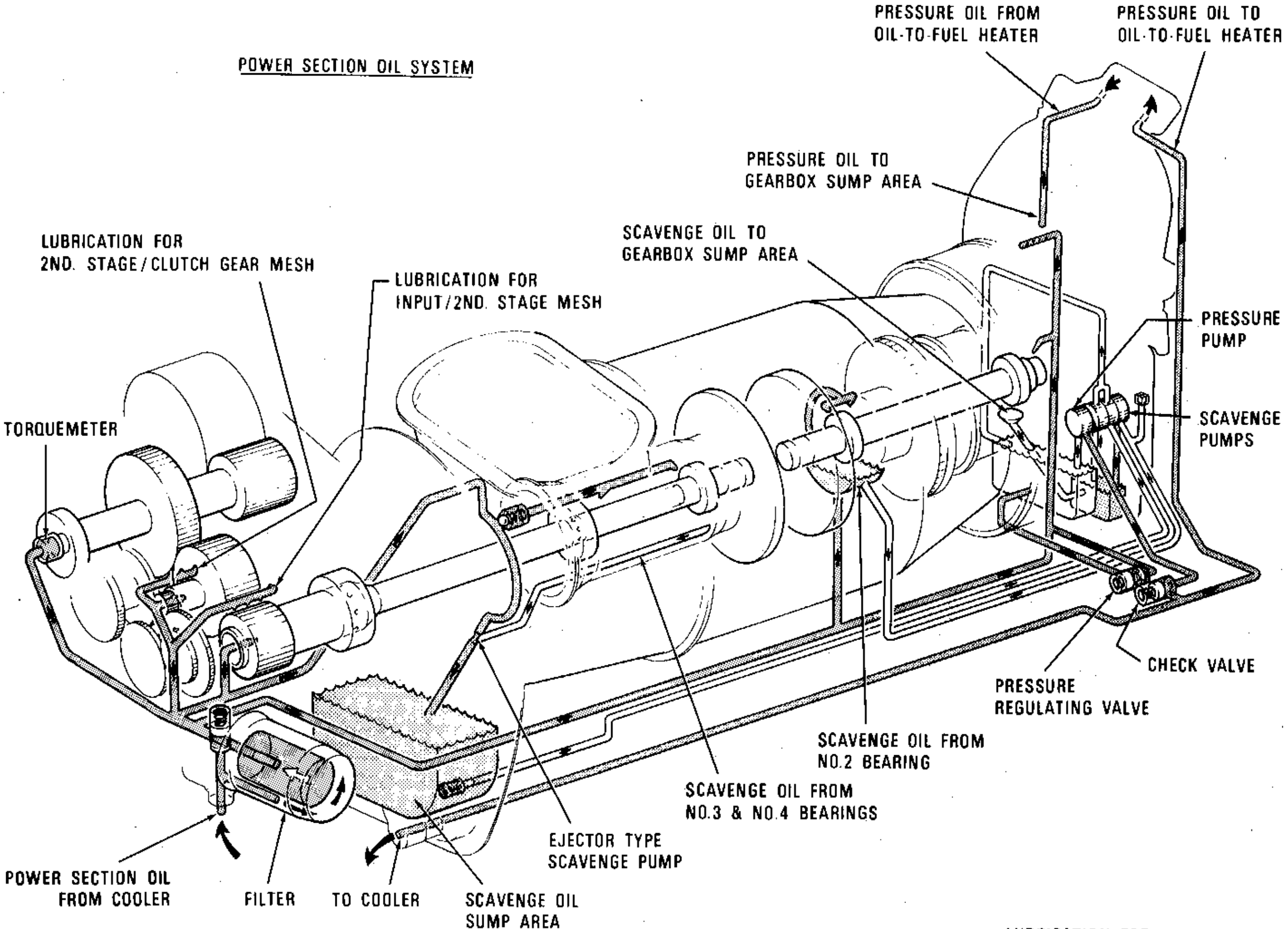


Figure 1-13. Transmission oil schematic



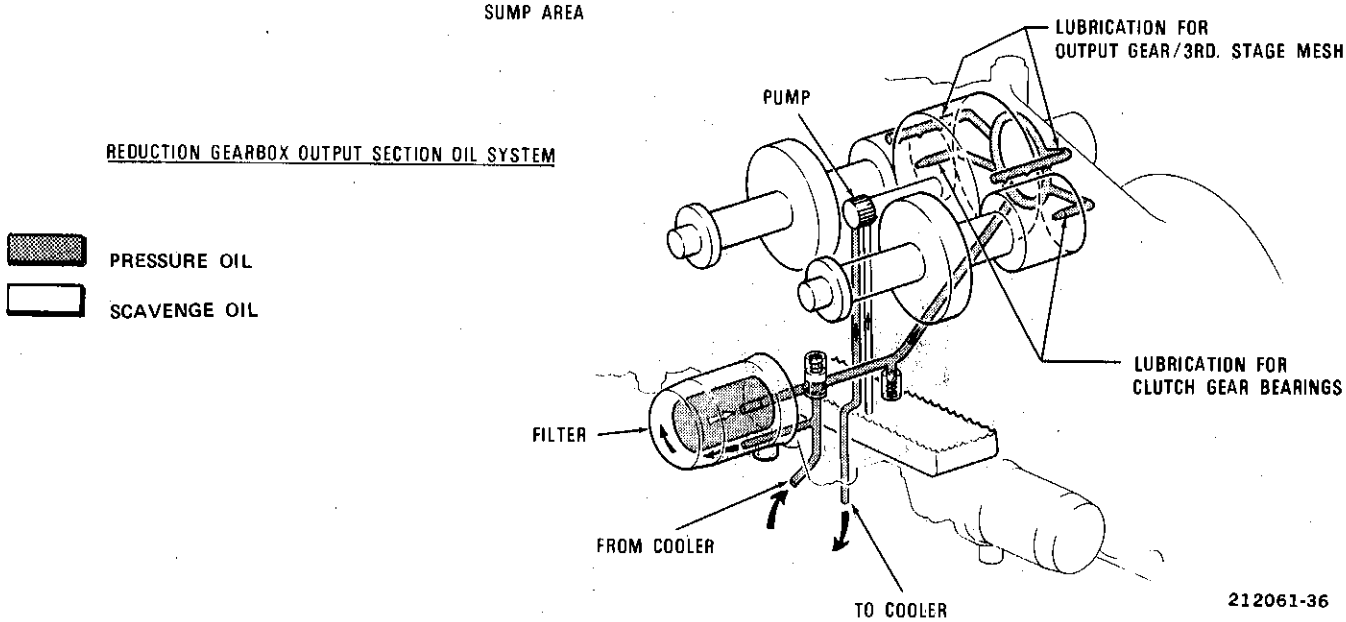


Figure 1-14. Oil system schematic (Sheet 1 of 2)

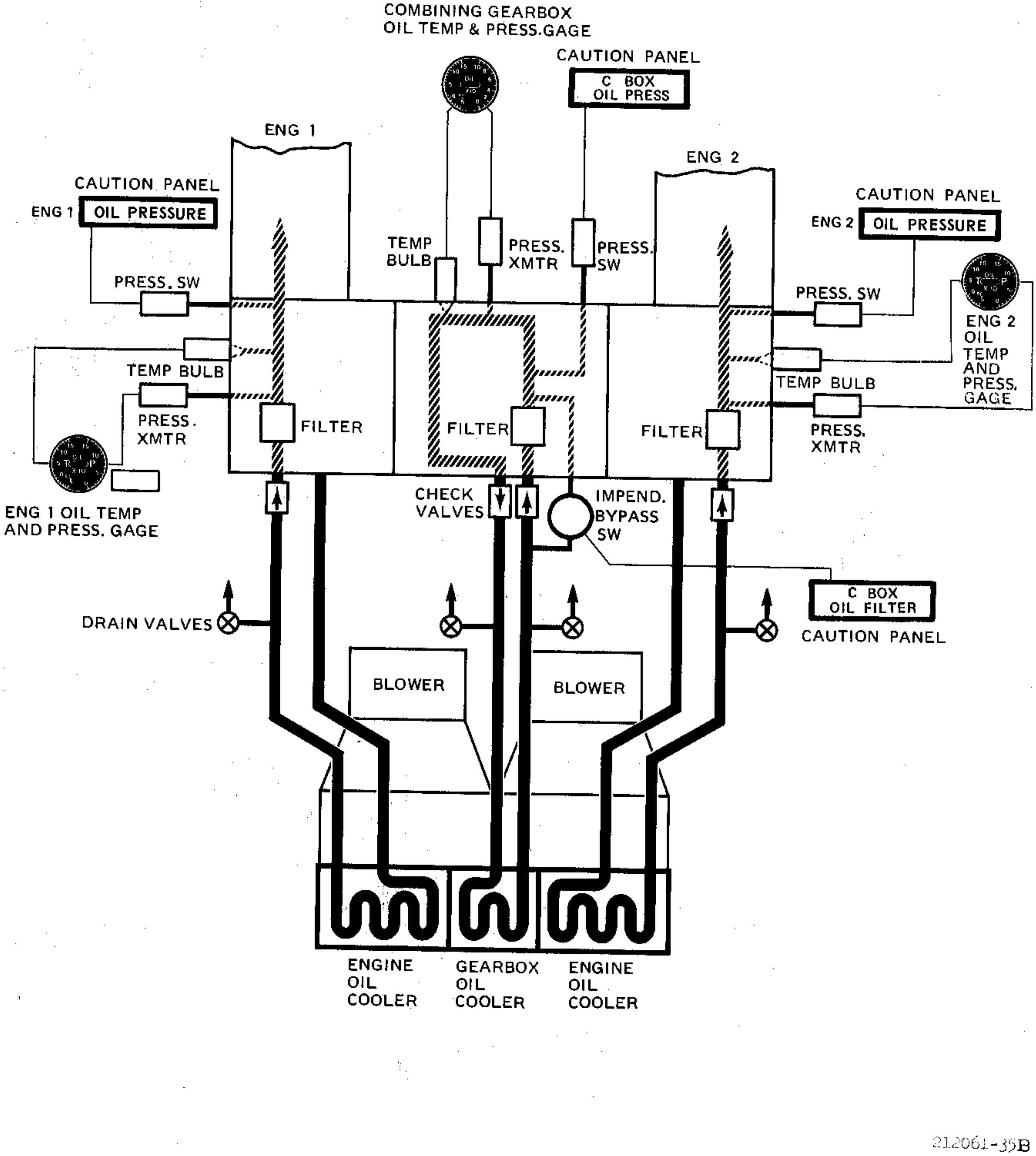


Figure 1-14. Oil system schematic (Sheet 2 of 2)

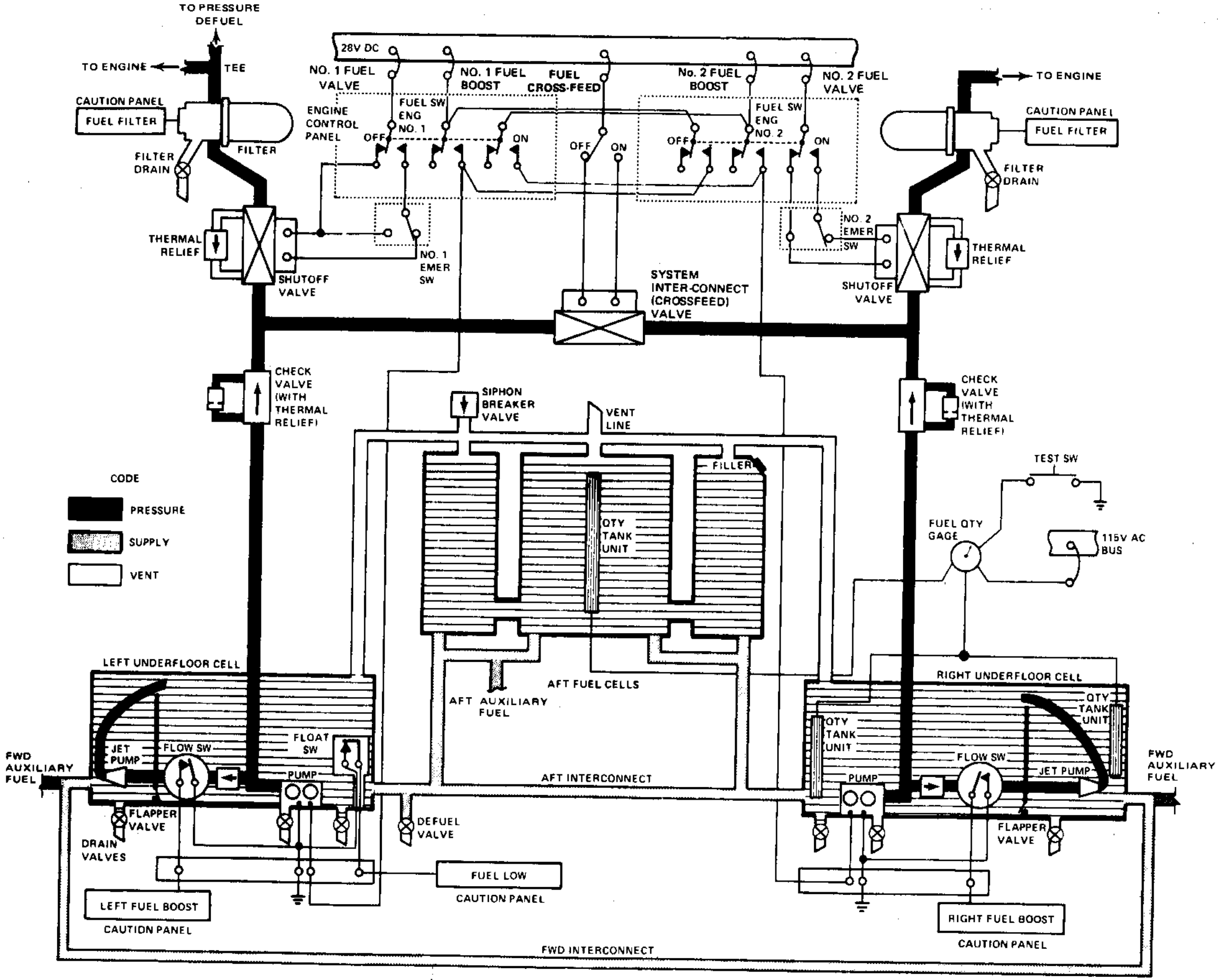


Figure 1-15. Fuel system schematic

FUEL SUPPLY SYSTEM

The fuel is contained in five crashworthy self-sealing cells which are interconnected to act as a single unit. This system replaced the main fuel system with self-sealing fuel cells containing explosion suppression foam, self-sealing interconnect lines between cells and main lines to engine, flexible hoses, and certain fittings designed to break between self-closing valves under high impact load. Frangible clips secure the fuel cells and fittings to adjacent structure. The clips will break under crash loads to permit relative movement of the fuel system components and resist spillage of fuel in crash impact, reducing fire hazard under those conditions. The crashworthy fuel system is designed to reduce fire hazard in the event of crash loads and to increase the overall ballistic explosion protection level. There are no changes in system operation. However, fuel quantity is decreased. (Refer to figure 1-16). Three cells (figure 1-15) are located aft of the pylon structure and the cabin bulkheads, between the engine deck and cabin floor levels. Two cells are located under the cabin floor at each side, ahead of the cabin bulkhead and outboard of the main longitudinal structural beams. Each under-floor cell is equipped with: a sump; a submerged electric motor-driven boost pump; a flow-actuated switch connected to a caution panel to signal if pump is inoperative; a lateral baffle fitted with a flapper valve to allow front-to-rear fuel flow; and an ejector-type pump mounted on the front wall and using booster pump flow to send fuel continuously back over the baffle to the rear portion of the cell to assure maximum usability of fuel in all flight positions. The filler neck and cap for the system are located on the right-hand aft cell. A defueling valve is located in the aft interconnect line directly aft of the left-hand under-floor cell. Vent lines from all cells are connected to a common line leading overboard through the fuselage lower skin. A siphon breaker valve prevents fuel from being siphoned out through vent lines. Fuel quantity gage transmitters are located in the center aft cell and in front and rear compartments of the right hand forward cell. The rear compartment of the left forward cell contains a float switch connected to the FUEL LOW (figure 1-20) light on the caution panel.

Fuel pressure lines from boost pumps are interconnected through an electrically controlled crossfeed valve, so that output of either or both pumps is available for each engine. Lines to each engine pass through an electrically controlled shutoff valve and an airframe-mounted filter. Shut-off valves and directional flow check valves have internal bypass provisions to relieve thermal expansion of trapped fuel when the system is inoperative. Each filter has internal bypass provisions to assure flow if the filter becomes clogged, and is connected to FUEL FILTER caution light (figure 1-20) to give visual signal of bypass condition. The helicopter is provisioned for closed circuit refueling.

The center aft fuel cell has an access door in its rear wall. A fuel quantity transmitter is supported on brackets on the inside of the door. Two float switches mounted at the same location are used only when auxiliary fuel tanks are installed. (Refer to Section IV for auxiliary fuel tanks). A return line and check valve connected to the door serve the governor bleed lines from the engines. Fuel specificaton and grade are shown in the Servicing Diagram (figure 1-22).

Helicopters modified by TCTO 1H-1(U)N-609 have the aft auxiliary fuel cells replaced by crashworthy fuel cells. The crashworthy fuel system provides the same crashworthiness and ballistic protection as the main crashworthy fuel system. Flexible hoses are connected with breakaway self-closing valves. The fuel cell panel and pump support are secured with frangible clips. There is no change in system operation. Total tank capacity if 95.9 U.S. gallons (both tanks).

FUEL QUANTITY DATA

U. S. GALLONS AND POUNDS JP-4			
NO. OF TANKS	NORMAL SERVICING	MAXIMUM CAPACITY	USABLE FUEL
1	203 Gallons 1320 Pounds	203 Gallons 1320 Pounds	197.5 Gallons 1284 Pounds
Internal Forward Auxiliary Tanks 2	(Both tanks) 290 Gallons 1885 Pounds	(Both tanks) 290 Gallons 1885 Pounds	(Both tanks) 290 Gallons 1885 Pounds
Aft Auxiliary Tanks 2	(Both tanks) 90 Gallons 585 Pounds	(Both tanks) 90 Gallons 585 Pounds	(Both tanks) 87.7 Gallons 570 Pounds

NOTES

1. Refer to Servicing Diagram for fuel specification.
2. JP-4 density based on standard day 6.5 pounds per gallon.
3. Usable main system fuel is approximately 197.5 gallons (1284 pounds) with a 7 degree nose down attitude.
4. Left Aft Auxiliary Fuel Tank usable fuel is 47.4 gallons. Right Aft Auxiliary Fuel Tank usable fuel is 40.3 gallons with a 7 degree nose down attitude.

Figure 1-16. Fuel Quantity Table (Sheet 1 of 3)

FUEL QUANTITY DATA

U. S. GALLONS AND POUNDS JP-5			
NO. OF TANKS	NORMAL SERVICING	MAXIMUM CAPACITY	USABLE FUEL
1	203 Gallons 1380 Pounds	203 Gallons 1380 Pounds	197.5 Gallons 1343 Pounds
Internal Forward Auxiliary Tanks 2	(Both tanks) 290 Gallons 1972 Pounds	(Both tanks) 290 Gallons 1972 Pounds	(Both tanks) 290 Gallons 1972 Pounds
Aft Auxiliary Tanks 2	(Both tanks) 90 Gallons 612 Pounds	(Both tanks) 90 Gallons 612 Pounds	(Both tanks) 87.7 Gallons 596 Pounds

NOTES

1. Refer to Servicing Diagram for fuel specification.
2. JP-5 density based on standard day 6.8 pounds per gallon.
3. Usable main system fuel is approximately 197.5 gallons (1343 pounds) with a 7 degree nose down attitude.
4. Left Aft Auxiliary Fuel Tank usable fuel is 47.4 gallons. Right Aft Auxiliary Fuel Tank usable fuel is 40.3 gallons with a 7 degree nose down attitude.

Figure 1-16. Fuel Quantity Table (Sheet 2 of 3)

FUEL QUANTITY DATA

U. S. GALLONS AND POUNDS JP-8			
NO. OF TANKS	NORMAL SERVICING	MAXIMUM CAPACITY	USABLE FUEL
1	203 Gallons 1360 Pounds	203 Gallons 1360 Pounds	197.5 Gallons 1323 Pounds
Internal Forward Auxiliary Tanks 2	(Both tanks) 290 Gallons 1943 Pounds	(Both tanks) 290 Gallons 1943 Pounds	(Both tanks) 290 Gallons 1943 Pounds
Aft Auxiliary Tanks 2	(Both tanks) 90 Gallons 603 Pounds	(Both tanks) 90 Gallons 603 Pounds	(Both tanks) 87.7 Gallons 588 Pounds

NOTES

1. Refer to Servicing Diagram for fuel specification.
2. JP-8 density based on standard day is 6.7 pounds per gallon.
3. Usable main system fuel is approximately 197.5 gallons (1323 pounds) with a 7 degree nose down attitude.
4. Left Aft Auxiliary Fuel Tank usable fuel is 47.7 gallons. Right Aft Auxiliary Fuel Tank usable fuel is 40.3 gallons with a 7 degree nose down attitude.

Figure 1-16. Fuel Quantity Table (Sheet 3 of 3)

FUEL SYSTEM CONTROLS AND INDICATORS

Fuel flow is controlled by operation of the switches on the ENGINE control panel (figure 1-8). The fuel system controls consist of an ENG 1 (ON-OFF) switch, ENG 2 (ON-OFF) switch and a CROSSFEED (ON-OFF) switch.

Fuel Valves

The fuel valves include the engine No. 1 fuel shutoff valve, the engine No. 2 fuel shutoff valve, and the fuel crossfeed valve (figure 1-15). These valves open or close to control the flow of fuel to the engines. Power is supplied to these valves by the 28V DC essential bus (figure 1-17) through circuit breakers No. 1 FUEL VALVE, No. 2 FUEL VALVE, and FUEL CROSSFEED (figure 1-18). Engine No. 1 fuel shutoff valve (figure 1-15) is controlled by ENG 1 FUEL switch (figure 1-8) for normal operation. Engine No. 2 fuel shutoff valve is controlled by ENG 2 FUEL switch for normal operation. Engine No. 1 fuel shutoff valve will return to its closed position when the FIRE 1 PULL handle (figure 1-11) is pulled out. This stops the flow of fuel to engine No. 1 in the event of fire in the engine area. Engine No. 2 fuel shutoff valve will close when the FIRE 2 PULL handle (figure 1-11) is pulled out. This stops the flow of fuel to engine No. 2 in the event of fire in that engine area. The crossfeed valve (figure 1-15) is controlled by the FUEL CONT CROSSFEED switch on the engine control panel (figure 1-8). The crossfeed valve connects the fuel feed lines together when the FUEL CROSSFEED switch is placed to ON. This allows both engines to be fed from the same fuel line or conversely, both fuel lines can supply one engine.

Fuel Control Switch

Fuel system control electrical power is furnished thru the 28V DC essential bus (figure 1-17). When FUEL CONT switches (ENG 1 and ENG 2) are ON, both boost pumps operate, both shutoff valves are open to send fuel to both engines thru the filters and the fuel control line heaters are energized. If one switch (figure 1-8) is placed to OFF, the corresponding shutoff valve will be closed but the boost pump will continue to run. Circuit breaker protection is furnished by No. 1 FUEL VALVE, No. 2 FUEL VALVE, No. 1 FUEL BOOST, No. 2 FUEL BOOST, No. 1 FUEL CONT HTR, and No. 2 FUEL CONT HTR (figure 1-18).

NOTE

When a FIRE PULL handle is actuated the corresponding engine fuel shutoff valve will close.

Fuel Crossfeed Switch

The crossfeed valve (figure 1-15) is opened or closed by the FUEL CONT CROSSFEED switch (figure 1-8); when valve is open either pump can supply both engines. Power is supplied through the 28V DC essential bus (figure 1-17) and protected by circuit breaker FUEL CROSSFEED (figure 1-18). Regardless of shutoff valves, when a boost pump is operating there is fuel flowing through the line within the cell to the jet pump, and the flow-actuated switch will prevent caution panel indications of boost pump failure. As long as there is fuel in the aft cells, the underfloor cells will remain full of gravity feed through the aft interconnect line.

Fuel Quantity Indicator

Fuel quantity gage indication, powered by the 115V AC bus (figure 1-17) will be governed by the tank unit on the aft center cell. When system fuel level drops below level of aft cells, the tank units in the right underfloor cell will control the quantity gage (figure 1-11). The indicator is calibrated in pounds. Circuit protection is furnished through circuit breaker FUEL QTY (figure 1-18).

Fuel Gage Test Switch

A fuel gage test switch (figure 1-11) is located on the instrument panel. The switch provides a means of testing the indicator and circuit operation. When the switch button is depressed and held in, the fuel quantity indicator pointer moves to a lesser quantity, and upon release should return to actual reading. Power is supplied through the fuel quantity indicator.

Fuel Filter – Caution

Two fuel filter caution segments labeled FUEL FILTER are incorporated in the caution panel; one for each engine. Illumination of this segment signifies that the fuel filter is clogged and a bypass condition is impending. Power is furnished through 28V DC essential bus (figure 1-17) and protected by circuit breaker CAUTION LTS (figure 1-18).

Fuel Low – Caution

A fuel low caution light is located on the caution panel (fig 1-20) labeled FUEL LOW. Illumination of this light will come on at 150 lbs, plus or minus 40 lbs, when the aircraft is in a stabilized flight atitude (nose down). The light will be erratic in function when the aircraft is subjected to sudden maneuvers, continuous turns or any tail low maneuver. A continuous light above 190 lbs when the aircraft is in a stabilized flight atitude (nose down), is abnormal. The light is illuminated by a float switch in the left forward cell. Power is furnished by the 28V DC essential bus (fig 1-17) and protected by circuit breaker CAUTION LTS (figure 1-18).

Boost Pump – Caution

Boost pump caution lights LEFT FUEL BOOST and RIGHT FUEL BOOST are located on the caution panel (figure 1-20). Caution lights illuminate when the flow switch fails to sense fuel flow. Power is supplied through the 28V DC essential bus (figure 1-17) and protected by circuit breaker CAUTION LTS (figure 1-18).

ELECTRIC POWER SUPPLY SYSTEM

The primary electrical system (figure 1-17) is a 28-volt direct current, single-conductor, negative ground system.

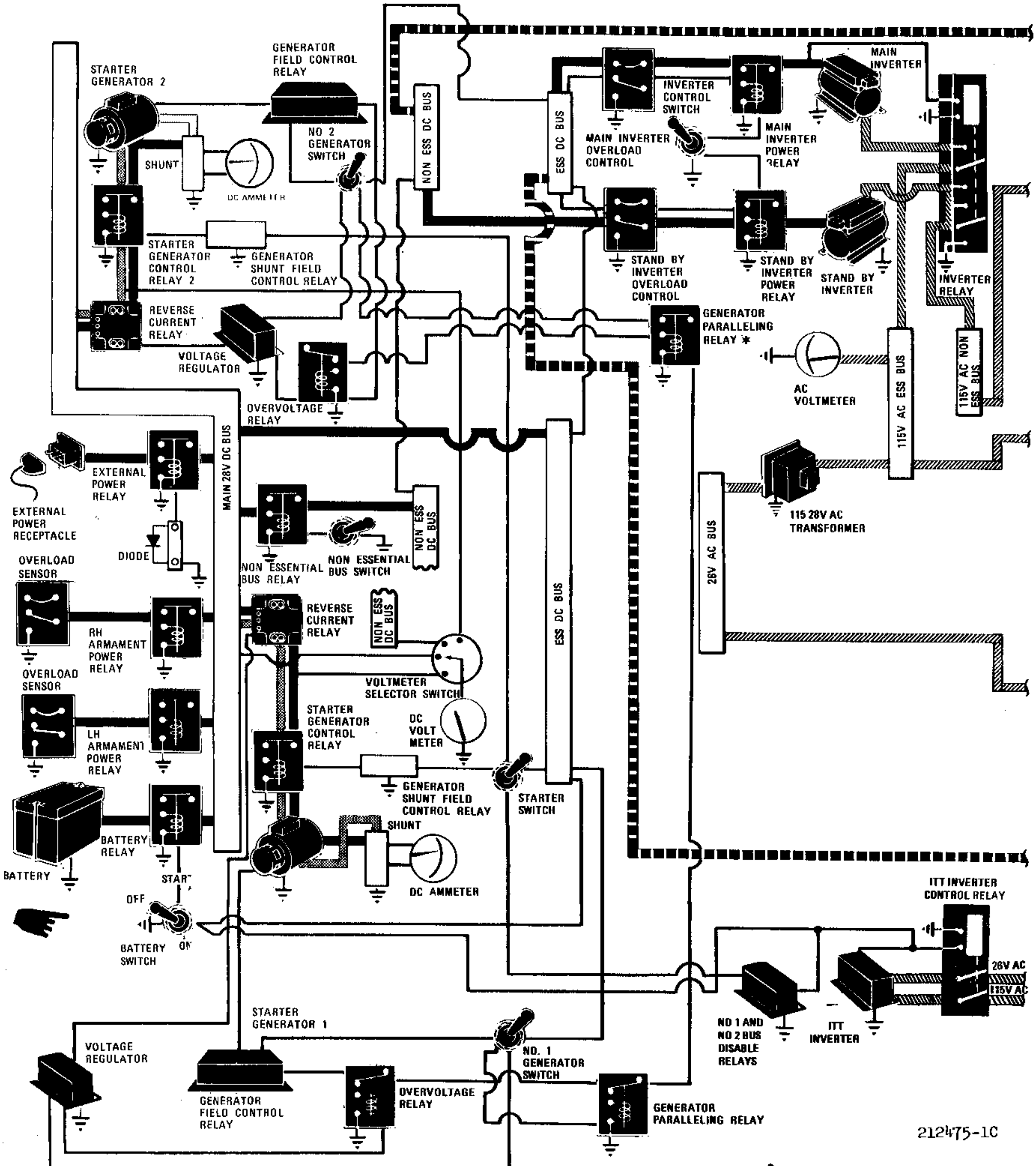


Figure 1-17. Electrical schematic (Sheet 1 of 2)

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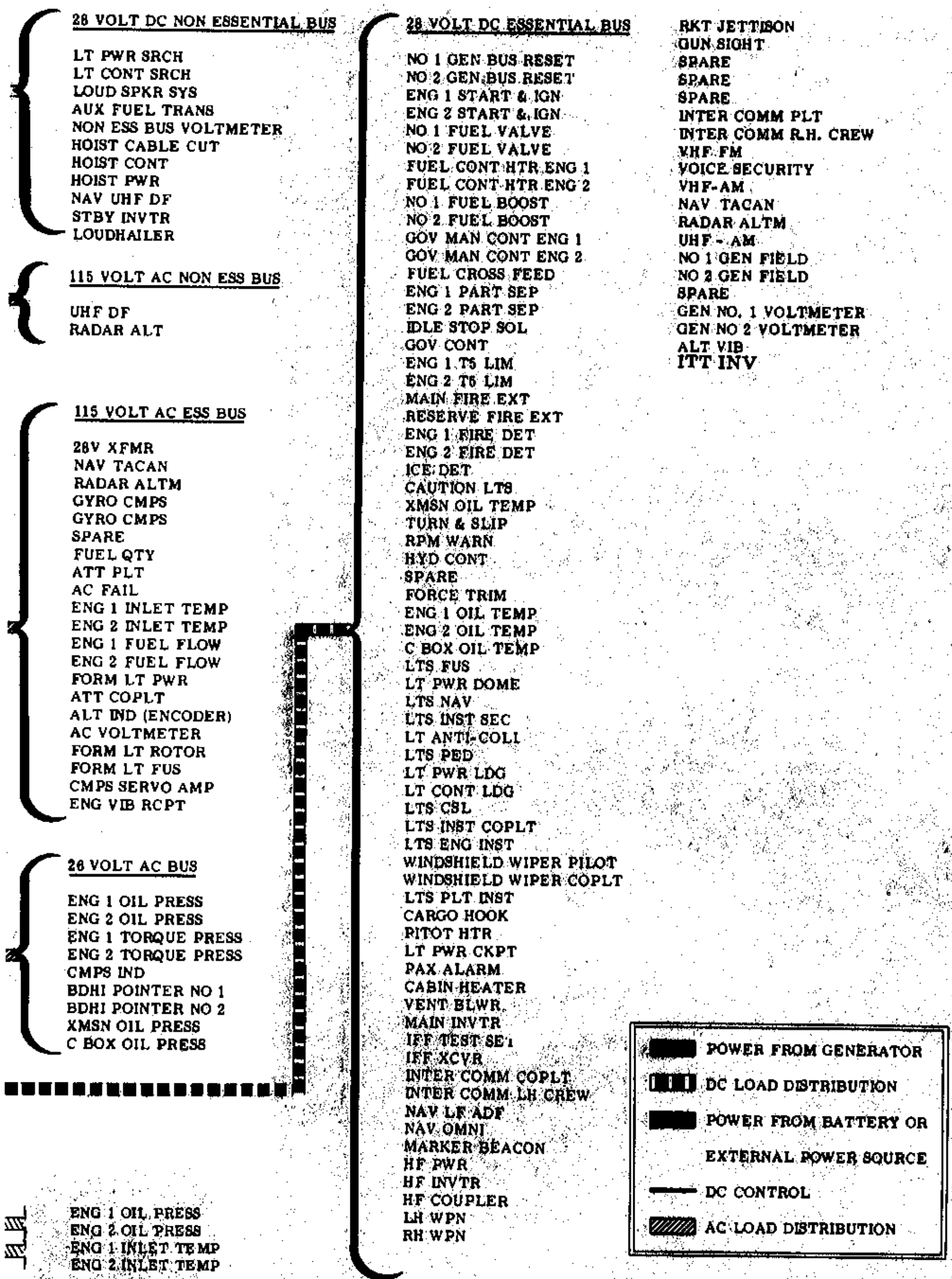

Figure 1-17. Electrical schematic (Sheet 2 of 2)

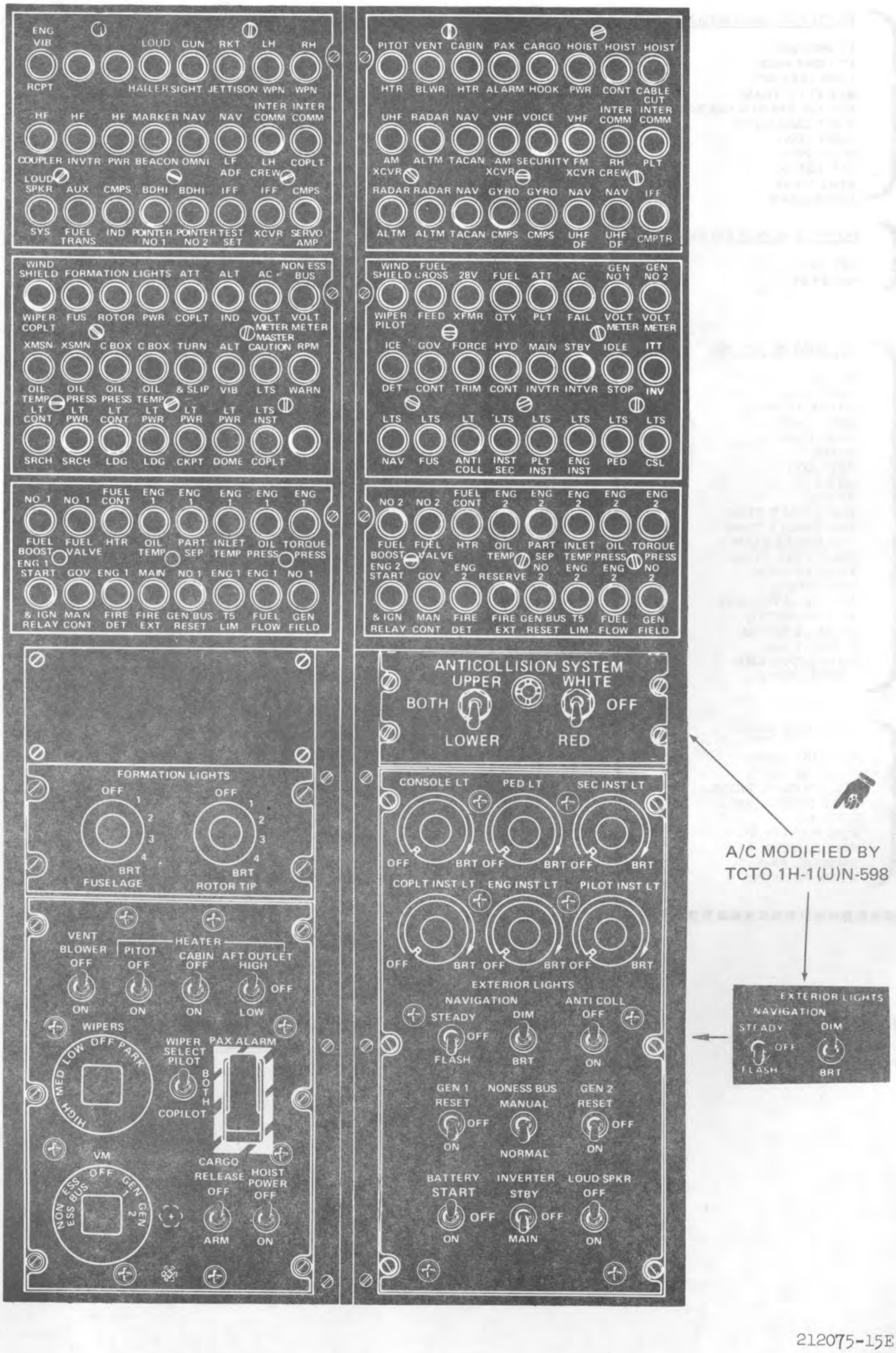

Figure 1-18. Overhead console and circuit breaker panel (typical).

Power is supplied by two 30-volt, 200 ampere starter-generators; one mounted on each engine. AC power is supplied by two 115V, 400 Hz, 750VA inverters. Normally the MAIN inverter powers the 115V AC essential bus and the STBY inverter supplies the 115V AC non-essential bus. Emergency DC power is supplied by the battery.

DIRECT CURRENT POWER SUPPLY

The 28-volt direct current power supply system is a single-conductor system with a negative ground to the helicopter structure. Power is supplied by the two 30-volt, 200 ampere starter-generators mounted on the accessory gearbox of each engine. The generators deliver full regulated power when Ng speed is approximately 71 percent or above. The generators are normally operated in parallel to provide power for all electrical loads on the helicopter. This power is distributed by a dual bus arrangement (figure 1-17) such that in the event of a generator failure, the non-essential bus will be automatically de-energized and the remaining generator will continue to provide power to the essential loads. The non-essential bus can be manually reactivated by the NON ESS BUS switch (figure 1-18). An external power receptacle is provided (figure 1-2) on the front of the helicopter.

All data on page 1-28B deleted.

DC NON-ESSENTIAL BUS SWITCH

The DC non-essential bus switch controls the non-essential bus relay. When in the NORMAL position and upon failure of either generator, the non-essential bus relay will open, dropping the non-essential DC bus from the line. When the switch is in the MANUAL position, regardless of generator availability, the non-essential DC bus will remain electrically connected to the main 28V DC bus (figure 1-17).

BATTERY

A 24-volt, 34 ampere, nickel-cadmium battery located in the nose compartment (figure 1-2), provides power for starting, when a battery start is necessary. The battery also provides a back-up emergency source of power in the event both generators should fail. Assuming 85 percent charged, the battery can supply the essential DC loads for a period of approximately 12 minutes.

CAUTION

Nickel - Cadmium battery failure of an explosive nature can occur as a result of high charging rates being applied to a battery that is in a reduced state of charge. To reduce the possibility of such failures, an external ground power unit should be used, whenever possible, to accomplish ground operation of electrical systems, preflight checks and engine starts.

Battery Switch

The battery switch is located on the front of the overhead console (figure 1-18) and has three positions, labeled START, OFF and ON. When the switch is in the START position, the battery relay is closed and battery power is supplied to the primary bus and the ITT inverter system. With the switch in the ON position, the battery relay closes and power is supplied to the primary bus. When the switch is in the OFF position, the battery relay opens the circuit and no power is delivered from the battery.

GENERATOR

The starter-generator is located on the front of the engine accessory gearboxes, one on each engine. The generators are 30-volt, 200 amperes that deliver regulated power when Ng speed is approximately 71 percent or above. Electrical power is distributed by a dual bus arrangement (figure 1-17) such that in the event of a generator failure, the non-essential bus will be automatically de-energized and the remaining generator will continue to provide power to the essential bus. The generator voltage regulator automatically controls the generator field current to maintain the proper generator output voltage of 27-28.5V DC. The generator reverse current relay automatically opens the circuit from the generator to primary bus when battery voltage is greater than generator voltage, thus preventing discharge of the battery through the generator. The generator is protected by the generator field and over-voltage relays.

On aircraft modified by T.C.T.O. 1H-1(U)N-568, an improved generator paralleling system is installed consisting of a paralleling relay and circuitry for each generator (figure 1-17). The paralleling relays and circuit relieve intermittent oscillations under low electrical load conditions and at low generator speeds when the engines are operating below 72% Ng. The paralleling relays are controlled by and actuated from the generator switches. When the generator switches are both in the ON position, the paralleling relays are actuated and both voltage regulators are connected together through the over-voltage relays and the paralleling relays to equalize the electrical load between the two generators.

Generator Switch

The generator switches are located on the overhead console (figure 1-18) and labeled GEN 1 and GEN 2. The switches have three positions, RESET, OFF and ON. It controls the flow of current to the primary bus. When the switch is in the ON position, it completes the generator output circuit to the reverse current relay and primary bus. The RESET position is spring loaded to return to OFF when released. RESET position is used to attempt to restore generator power to the system if the generator caution light is on. The switch should be held in RESET momentarily and then moved to ON. If the light goes out, system has returned to normal.

Generator Caution Lights

The generator caution lights are located on the instrument mounted caution panel (figure 1-11) and labeled DC GENERATOR. The lights are controlled by the reverse current relays for each generator. When the relays are open the lights glow and the generator switches in the overhead console (figure 1-18) should be held in the RESET position momentarily and then moved to ON. When the generators start operating, lights will extinguish if system is functioning properly. Power is supplied through the 28V DC essential bus (figure 1-17) and protected by circuit breaker CAUTION LTS (figure 1-18).

VOLTMETER (DC PORTION) AND SELECTOR SWITCH

The voltmeter selector switch is a rotary type switch located on the overhead console (figure 1-18) labeled VM. The switch has five different positions for monitoring voltage, which is indicated on the voltmeter (figure 1-11). The dual AC and DC voltmeter monitors and indicates voltage simultaneously. Voltage will be indicated on voltmeter in selected position of GEN 1, GEN 2, NON ESS BUS, and ESS BUS. Circuit protection is provided by circuit breakers GEN No. 1 VOLTMETER, GEN No. 2 VOLTMETER, NON ESS BUS VOLTMETER, and GOV MAN CONTROL ENG 1 (figure 1-18).

AMMETER

The ammeter is located in the center of the instrument panel (figure 1-11). The dual DC ammeter indicates the output in amperes of the generator portions of starter-generator No. 1 and No. 2 (Refer to figure 1-17).

EXTERNAL POWER RECEPTACLE

An external power receptacle (figure 1-2) is located adjacent to the nose compartment door, just inside an access door labeled 28V DC. The receptacle provides for connecting an external source of power for engine starting and ground servicing of all electrical equipment.

External Power Caution Light

An external power caution light located on the caution panel (figure 1-20) labeled EXTERNAL PWR illuminates when power is applied to the essential bus and the external power access door is opened. The switch is actuated by the access door. Caution light will extinguish when door is closed. Power is supplied through the 28V DC essential bus (figure 1-17) and protected by circuit breaker CAUTION LTS (figure 1-18).

CIRCUIT BREAKER PANEL

The circuit breaker panel (figure 1-18) is a combination AC and DC panel located on the overhead console. The circuit breakers are ON when pushed in, OFF when pulled out. The breakers automatically pop out when the circuit they protect becomes overloaded or may be pulled out by hand to disconnect their circuit from the bus.

NOTE

All circuit breakers are trip-free, i.e., manually pushing the breaker will not restore power if fault still exists in the circuit.

ALTERNATING CURRENT POWER SUPPLY

Alternating current power is supplied by the two 115 volt, 400 Hz, single-phase, 750 volt-ampere inverters which are powered by the 28V DC power supply. The AC power from the inverters is distributed to the AC buses (figure 1-17). The 26V AC bus is supplied power through the 115/28V AC transformer. An alternate source of 115/26V AC power is available from the ITT inverter to power the inner turbine temperature and oil pressure indicating systems during battery starts when the aircraft main/spare inverters are off.

INVERTERS

The main inverter normally supplies AC power to all systems requiring AC power except for the radar altimeter and the UHF-DF. DC power is supplied to the main inverter from the 28V DC essential bus and is controlled by the INVERTER MAIN-STBY switch (figure 1-18) located on the overhead console. It is protected by an overload sense control and MAIN INVTR circuit breaker (figure 1-18). The standby inverter is also made operational when the INVERTER switch is in MAIN position and the radar altimeter and/or the UHF-DF receiver is energized. DC power is supplied to the standby inverter from the 28V DC non-essential bus. It is protected by an overload sense control and STBY INVTR circuit breaker.

Inverter Caution Light

The caution panel (figure 1-20) located on the instrument panel, has a segment labeled AC FAIL which will illuminate in the event of loss of power to AC essential bus. Upon illumination, the inverter switch (figure 1-18) should be placed in the STBY position. Power is supplied to the light through the 28V DC essential bus (figure 1-17) and protected by circuit breaker CAUTION LTS (figure 1-18).

VOLTMETER (AC PORTION)

The voltmeter, located in the lower center area of the instrument panel (figure 1-11) monitors and indicates voltage simultaneously. The AC portion of the dual voltmeter indicates voltage of 115V AC essential bus (figure 1-17) and is protected by circuit breaker AC VOLTMETER.

ITT INVERTER SYSTEM

The ITT inverter system provides an independent source of 115V AC power to the inter turbine temperature indicating system and 26V AC power to the engine oil pressure indicating system during battery starts when it is desirable to leave the aircraft inverters off. The 28V DC powered system consists of a series of relays and an ITT inverter, and is protected by the ITT INV circuit breaker (figure 1-18). When the battery switch is positioned to START, or when the battery switch is ON and either pilot's or instructor's start switch is positioned to ENG 1 or ENG 2, the ITT inverter control relay will be energized, and 26V AC and 115V AC will be routed to the inter turbine temperature and engine oil pressure indicating systems from the ITT inverter.

FLIGHT CONTROL SYSTEM

The flight control system is a positive mechanical type, actuated by dual conventional helicopter controls which, when moved, directs the helicopter in various modes of flight. The system includes a cyclic control stick, the collective pitch (main rotor) control lever, tail rotor control pedals, and a synchronized elevator.

The control forces of the flight control system are reduced to a near-zero-pounds force, by hydraulic servo cylinders which are connected to the control system mechanical linkage and powered by the transmission driven hydraulic pumps. Force trims (force gradient) connected to the cyclic and directional controls are electrically operated mechanical units used to induce artificial control feeling into cyclic and directional controls and prevents movement of their own accord.

FORCE TRIM (FORCE GRADIENT)

Force trim centering devices are incorporated in the cyclic controls and direction pedal controls. These devices are installed between the cyclic stick and associated hydraulic servo cylinders, and between the tail rotor control pedals and the associated hydraulic servo cylinder. The devices act to furnish a force gradient or "feel" to the cyclic control stick and directional control pedals; however, these forces can be reduced to near zero by depressing and releasing the force trim pushbutton switch (figure 1-10) on top of the cyclic control stick. This gradient is accomplished by means of springs and magnetic brake release assemblies which enable the pilot to trim the controls as desired, for any condition of flight. A FORCE TRIM ON/OFF switch is installed on the miscellaneous control panel (figure 1-12) to deactivate the force trim.

STABILIZER BAR

The stabilizer bar is mounted on the main rotor mast above and at 90 degrees to the main rotor blades (figure 1-2). The bar is connected into the main rotor system in such a manner that the inherent inertia and gyroscopic action of the bar is induced into the rotor system and produces a measure of stability for all flight conditions. If the helicopter attitude is disturbed while hovering, the bar, due to its gyroscopic action tends to remain in its present plane. The relative movement between the bar and mast causes the hub and blade assembly to feather and return the main rotor to its original plane of rotation. Due to a restraining and damping action, the bar possesses a mast-following characteristic. The following time is regulated by two hydraulic dampers connected to the bar in such a manner that a movement of the mast is transmitted to the bar through dampers at a rate determined by the adjustment of the dampers. A compromise is met in which the bar provides the desired amount of stability and still allows the pilot complete responsive control of the helicopters. The dampers are cam actuated to increase damping as the stabilizer bar becomes displaced.

CYCLIC PITCH CONTROL STICK.

The cyclic pitch control stick (figure 1-9) operates the fore and aft and lateral control systems. When moved in any direction, it will produce a corresponding directional movement of the helicopter which is the result of a change in the plane of rotation of the main rotor. Cyclic stick fore and aft movement changes the synchronized elevator attitude by means of mechanical linkage and connecting control tubes. The pilot's cyclic stick grip contains the cargo release switch, a trigger-type two-position, intercomm/radio transmitter switch, armament fire control switch, force trim release switch and rescue hoist switch. Desired operating friction can be induced into the cyclic stick by hand tightening the friction adjuster located on the stick. The copilot's cyclic control stick is the same as the pilot's cyclic stick except it does not have an operable rescue hoist switch or a friction adjuster.

COLLECTIVE PITCH CONTROL LEVER

The collective pitch control levers (figure 1-10) are located to the left of the pilots position and control vertical mode of flight. Desired operating friction canbe induced into the pilot's control lever by hand tightening the friction adjusters (figure 1-10)

Two rotating grip-type throttles with dissimilar surfaces and a switch box assembly are located on the upper end of the collective pitch control levers. The pilot's switch box assembly contains the three-position starter switch, governor rpm increase-decrease switch, engine idle stop release switch, and landing light and searchlight switches. A springloaded pitch lever down lock (figure 1-10), is located on the floor at the approximate center and inboard of the pitch control lever. The copilot's collective pitch control lever contains only the rotating grip-type throttles with dissimilar grip surfaces and beep switch.

NOTE

The collective pitch control system has a built-in friction which requires a force to move the stick up from the neutral (center of travel) position of 8 to 10 pounds with boost ON.

TAIL ROTOR CONTROL PEDALS

The tail rotor control pedals (figure 1-10) through pushpull tubes, bellcranks, pitch links, and a hydraulic actuator alter the pitch of the tail rotor blades and provide the means of directional control. Pedal adjusters (figure 1-10) are located at the base of the instrument panel directly forward of the pilot and copilot positions. The force trim system is connected to the directional controls and is operated by the force trim switch on the cyclic control stick grip. The copilot's tail rotor control pedals are identical to the pilot's pedals.

SYNCHRONIZED ELEVATOR

The synchronized elevator (figure 1-2) is located near the aft end of the tail boom. The pitch angle of the left side of the elevator is 3 degrees lower than the right side. The difference in pitch is to equalize the loads on the elevators. The synchronized elevator is connected by control tubes and mechanical linkage to the fore and aft cyclic control system at the swashplate. Fore and aft movement of the cyclic control stick produces a change in the synchronized elevator attitude, thus increasing controllability and lengthening CG range.

HYDRAULIC POWER SUPPLY SYSTEM

The main rotor cyclic and collective controls are powered by two separate and completely independent hydraulic systems; System 1 and System 2. In addition to powering the cyclic and collective controls, System 1 also supplies power to the tail rotor pitch control. The two systems are the same except for the difference noted above and for pump operating speed (figure 1-19). Each system includes a reservoir, a transmission-driven pump, an integrated valve and filter assembly, directional flow check valves, pressure switch, accumulator, and connecting hoses and tubes. Two quick-disconnect couplings are also provided, in each system, to allow use of an external hydraulic power source for ground operation. The transmission-driven pump draws fluid from the reservoir for delivery under pressure through an external line and check valve to the integrated valve and filter assembly. The pump has internal pressure compensation and supplies flow according to system demand. In the valve and filter assembly, fluid passes through a pressure line filter to the solenoid operated shut-off valve and system relief valve. The system shut-off valve is normally open, except when energized to a closed position by a 28V DC circuit through the HYDR CONTROL selector switch or the HYDR CONTROL MASTER switch in the OFF position. When the shut-off valve is open, normal pressure of 1000 psi is available to flight control actuators, and flow will occur as actuator servo valves are moved by linkage from the control sticks or pedals. Fluid returns to the reservoir through the shut-off valve, and a return line filter. When the solenoid shut-off valve is closed, or after shut down, the pressure operated shut-off valve will close as the pressure drops below 600 psi. The appropriate HYD SYS caution panel segment will illuminate when the pressure drops below 650 psi. The closed pressure operated shut-off valve will now have isolated the control actuators and attached lines from the rest of the system, retaining fluid pressure in the actuators (irreversibility) to maintain a stiff control column. The spring loaded accumulators serve to make up fluid lost through leakage. When system 1 is shut off, the directional control cylinder will not be powered. The HYDR CONTROL MASTER switch can be utilized to turn both systems off simultaneously to check irreversibility. When pressure is again applied to the system, the worded segment will extinguish and the pressure operated shut-off valve will open at 750 psi. There are two integral non-electric devices (one on the pressure filter, one on the return filter of each system) that will give visual warning of partial filter clogging by pop-out of red indicator buttons. In addition, an electrically operated indicator mounted in the nose compartment will indicate when any of the four filter elements are partially clogged. The indicators are actuated at 70 psi differential across the filter element and once activated, remain extended until manually reset.

FLIGHT CONTROLS AND ACTUATOR UNITS

The hydraulic actuator system (figure 1-19) reduces the operational loads of the helicopter cyclic, collective, and directional control systems. Movement of the controls in any direction causes a power cylinder servo valve, in the appropriate system, to open and admit hydraulic pressure to actuate the cylinder for control movement.

HYDRAULIC SYSTEM CONTROL SWITCH

A three-position, hydraulic control toggle switch (figure 1-12) labeled SYS 1 ON, SYS 2 ON, and BOTH is located on the pedestal. When the switch is in the BOTH position or both systems ON position, the solenoid operated valve pressure port, in both systems, is connected to the cylinder port and hydraulic power is supplied to the systems. When the switch is in one of the "SYS ON" positions, the solenoid operated valve pressure port, in the other system, is closed, and the cylinder port is connected to the return port, removing pressure from the system. In addition, the HYDR CONTROL MASTER switch, when OFF, will energize both solenoid operated valves turning both systems off. Electrical power for each system is supplied through the 28V DC essential bus (figure 1-17) and protected by circuit breaker HYD CONT (figure 1-18).

HYDRAULIC SYSTEM RESERVOIRS

The hydraulic system reservoirs are mounted in front of the pylon on the cabin roof (figure 1-21). The system 2 reservoir is on the left-hand side, the system 1 reservoir is on the right-hand side. The reservoirs are identical. Each has a capacity of 5.25 pints. The cowling has a hole on either side through which the quantity of fluid in the reservoir can be read.

HYDRAULIC SYSTEM INDICATORS

Two low pressure caution lights labeled HYD SYS 1 and HYD SYS 2 are located on the caution panel (figure 1-20). Illumination of the lights indicate hydraulic pressure has dropped below 650 psi. Power is furnished through the 28V DC essential bus and protected by circuit breaker CAUTION LTS.

An electrically operated filter clog indicator is located in the nose compartment. This indicator will change from solid black to white clover leaf if any of the four filters is clogged. To determine which filter is clogged the indicators on each individual filter must be checked. Power is supplied through 28V DC bus (figure 1-17) and protected by circuit breaker HYD CONT (figure 1-18).

LANDING GEAR SYSTEM

The main skid landing gear employs two parallel tubular aluminum alloy skids and two curved aluminum cross tubes attached to the fuselage beams at four points. The cross tubes are flexible and provide limited cushioning on landing. Replaceable skid shoes cover the portion of the skids normally in contact with the surface. Manually retractable and quickly removable wheel assemblies have been provided to facilitate helicopter ground handling operations.

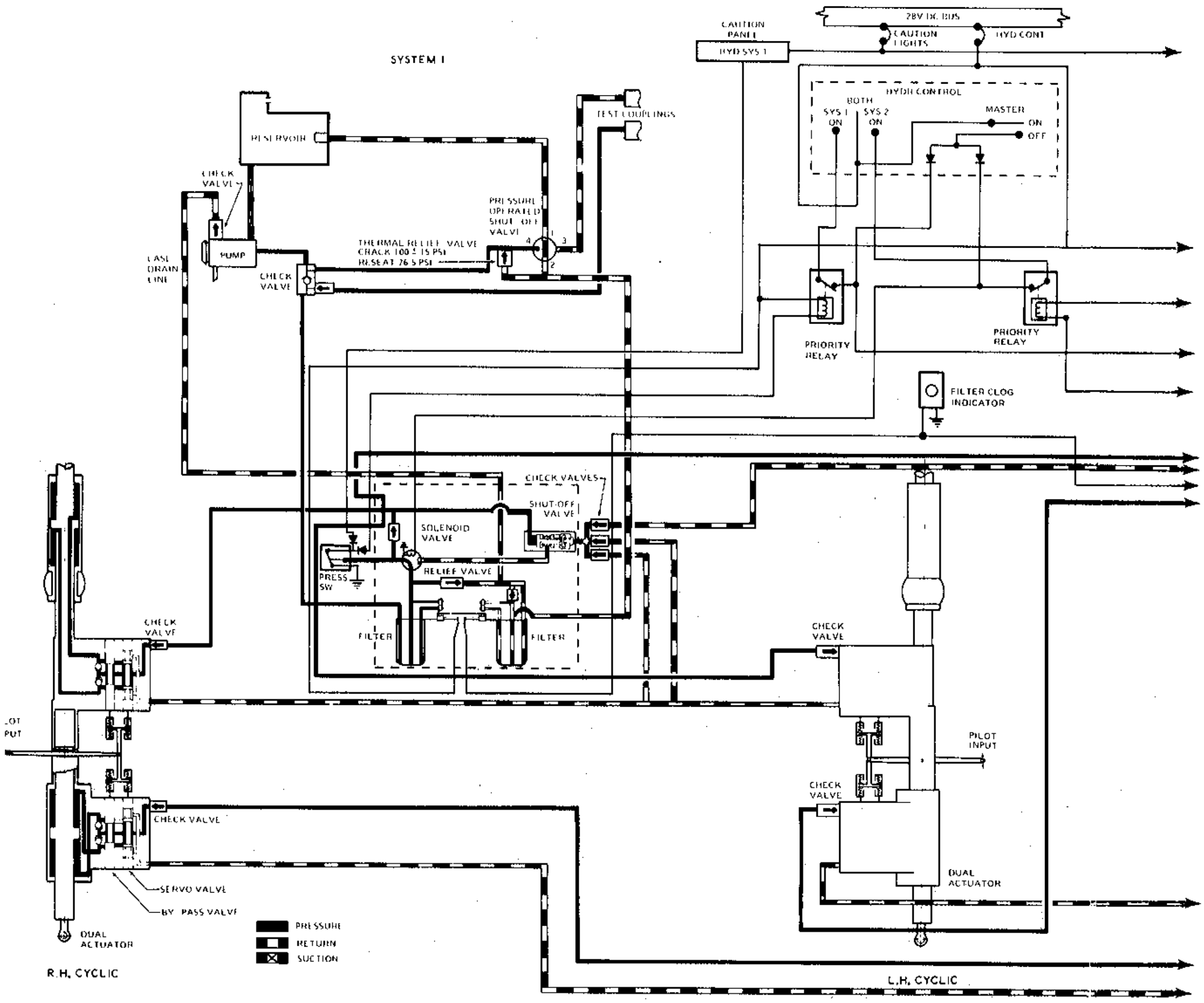

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Figure 1-19. Flight controls hydraulic system schematic (Sheet 1 of 2)

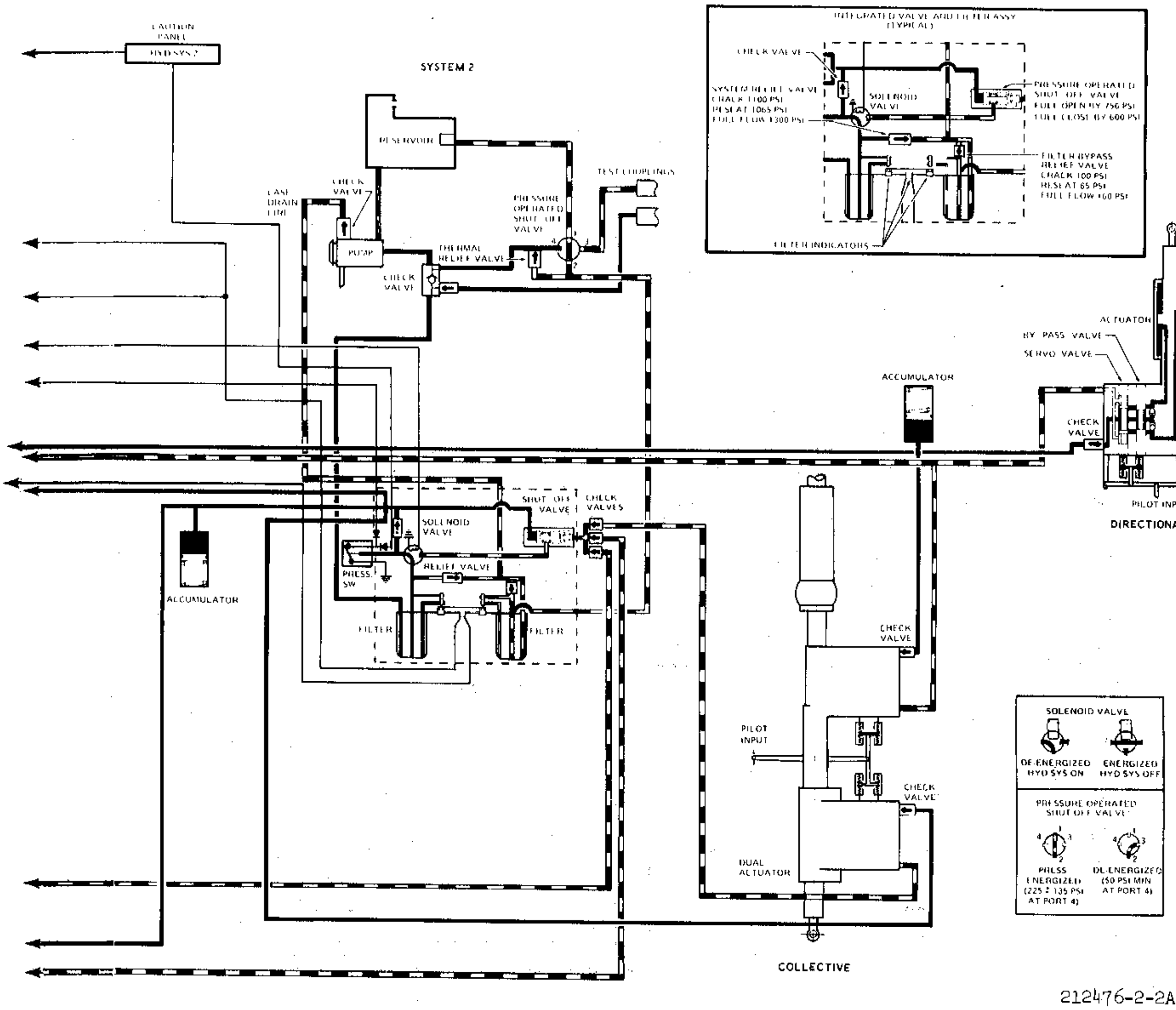

Figure 1-19. Flight controls hydraulic system schematic (Sheet 2 of 2)

TAIL SKID

A steel, tubular type, tail skid is installed on the aft end of the tail boom. The skid provides limited protection to the tail rotor during nose high landings.

FLIGHT INSTRUMENTS

The flight instruments include pilot's and copilot's attitude indicators, airspeed, turn and slip, vertical velocity, altimeter, standby compass, free air temperature and clock.

ATTITUDE INDICATORS

The MM-1 pilot's and copilot's attitude indicators (figure 1-11) display flight attitude of the helicopter in relation to the earth's horizon. Pitch attitude is indicated by motion of the sphere in relation to the miniature airplane. Roll attitude is indicated by motion of the roll pointer with respect to the fixed roll scale located at the top of the display. The indicator sphere can be adjusted to zero indication by the pitch and roll trim knobs which are located on the face of the instrument. The power OFF flag, located in the lower left-hand portion of the display will be exposed with any interruption of indicator power. Horizontal markings indicate the degree of dive or climb, while bank (roll) angles are read from the semicircular scale located on the upper half of the indicator face. Power is supplied through the 115V AC essential bus (figure 1-17) and protected by circuit breakers ATT PLT, and ATT COPLT (figure 1-18). The OFF flag (power warning) should disappear within 90 seconds after power is initially supplied and display will erect and remain stable.

AIRSPEED

The two airspeed indicators (figure 1-11) are standard pitot-static instruments. The single scale airspeed indicators are calibrated in knots and provide an indicated airspeed of the helicopter at any time during forward flight by measuring the difference between impact and static air pressure from the roof mounted combination pitot and static air pressure tube (figure 1-2).

TURN AND SLIP INDICATOR

The pilot's and copilot's turn and slip indicators (figure 1-11) consists of a rate of turn pointer and a ball (slip indicator) which operate independently of each other. The turn indicator is controlled by electrically actuated gyros. Needle and ball are combined in one instrument and are normally read and interpreted together. Power is supplied through the 28V DC essential bus (figure 1-17) and protected by circuit breaker TURN & SLIP (figure 1-18).

ALTIMETER

The altimeters (figure 1-11) furnish direct readings of height and are actuated by the static pressure system. The altimeters are connected through piping to static pressure ports of the combination pitot-static tube, mounted on the cabin roof.

VERTICAL VELOCITY INDICATOR

Two vertical velocity indicators are provided, one for the pilot and one for the copilot. Mounted on the instrument panel (figure 1-11), the indicators register ascent and descent speed of the helicopter in feet per minute. The instrument is connected to the static air system to sense the rate of atmospheric pressure change.

STANDBY COMPASS

A standard, magnetic type compass is mounted in a bracket above the center of the pilot's windshield. The compass correction card, located in a card holder on the center brace between pilot's and copilot's windshield, is provided for use in conjunction with the magnetic compass.

NOTE

When only one generator is ON the standby compass will be in error as much as 30 degrees.

FREE AIR TEMPERATURE

The bi-metal free air temperature indicator is in the upper, inboard corner of the pilot's windshield, and provides a direct reading of the outside air temperature calibrated in degrees centigrade.

CLOCK

The clock (figure 1-11) is a standard eight-day clock with a start stop sweep-second and recording hand for up to one hour elapsed time. A control knob on front of case winds and sets the clock. A second knob on the front case starts, stops, and resets the elapsed time section without affecting the hour and minute hand.

PITOT STATIC SYSTEM

The pitot-static system consists of the electrically heated pitot-static tube, static lines, static manifold, pitot lines and necessary piping to connect to altimeters, vertical velocity and airspeed indicators.

PITOT TUBE

The pitot-static tube (figure 1-2) is mounted on the cabin roof and supplies impact air through the pitot lines to the airspeed indicators. The static air pressure is sensed from the pitot-static tube through the static lines and static manifold to the altimeters, vertical velocity, and airspeed indicators.

MASTER CAUTION SYSTEM

The master caution system is a segment wording type, consisting of a segmented word warning caution panel and two remote master caution indicator lights. A chip detector caution panel is located above the main caution panel.

MASTER CAUTION LIGHTS

The master caution lights (figure 1-11) are located at top of pilot's and copilot's instrument panel. Their purpose is to serve as an additional warning when any section of the caution panel is illuminated. When the master caution lights illuminate the pilot and copilot are alerted to check the caution panel (figure 1-20) for the fault. The Master Caution lights will illuminate each time a caution panel segment illuminates. (They are reset by pushing and releasing the Master Caution lights each time a fault indication occurs.)

CAUTION PANEL

The caution panel is located on the instrument panel (figures 1-11 and 1-20) and provides a visual indication by means of individually backlighted, worded segments, that any of twenty-nine fault conditions has occurred. When illuminated the lettering in the segments will be visible day or night. A BRIGHT/DIM switch is provided and controlled by a toggle switch on the caution panel. Electrical power is supplied from the 28V DC ESS Bus and protected by a MASTER CAUTION LTS circuit breaker (figure 1-18). The caution panel will indicate the following fault conditions:

CAUTION PANEL SEGMENT WORDING	FAULT CONDITION
OIL PRESSURE	Below 30 psi
CHIP DETECTOR	Metal Particles In Engine
DC GENERATOR	DC Generator Failure
GOV MANUAL	Fuel Control Solenoid Valve Positioned to Manual
FUEL FILTER	Fuel Filter Clogged

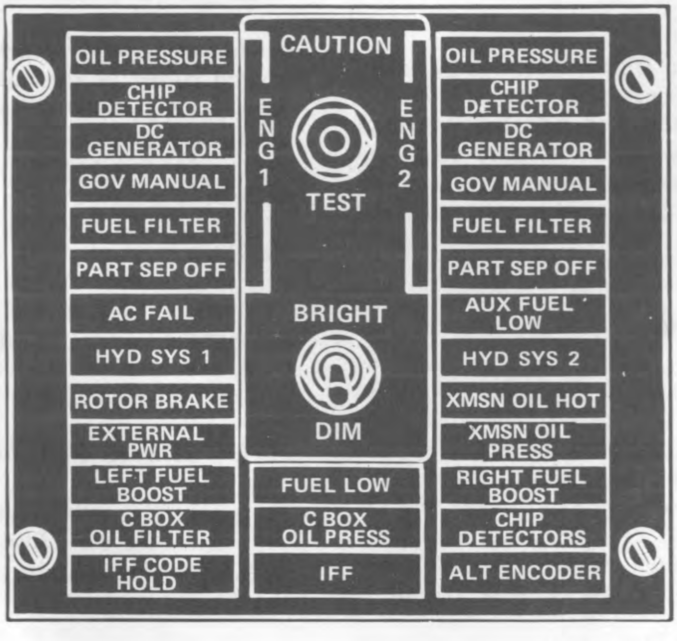

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Figure 1-20. Caution panel

CAUTION PANEL SEGMENT WORDING	FAULT CONDITION
PART SEP OFF	Particle Separator Bypassed
AC FAIL	Failure of AC Inverter
HYD SYS 1	Hydraulic Pressure Below 650 psi
HYD SYS 2	Hydraulic Pressure Below 650 psi
FUEL LOW	Fuel Quantity Below 150 ± 40 lbs (at normal flight attitude)
RIGHT FUEL BOOST	Right Fuel Boost Pump Flow Stopped
LEFT FUEL BOOST	Left Fuel Boost Pump Flow Stopped
AUX FUEL FLOW	Auxiliary Fuel Tank Empty
XMSN OIL PRESS	Transmission Oil Pressure Below 30 psi
XMSN OIL HOT	Transmission Oil Temperature Above 110° C
C BOX OIL PRESS	Combining Gear Box Oil Pressure Below 30 psi
C BOX OIL FILTER	Combining Gearbox Oil Filter Clogged
ROTOR BRAKE	Rotor Brake On
EXTERNAL PWR	External Power Door Open
CHIP DETECTORS	Metal Particles in Oil
IFF CODE HOLD	IFF Code Hold
IFF	IFF Mode 4 Inoperative
ALT ENCODER	Altimeter Encoder Inoperative

CHIP DETECTOR CAUTION PANEL

The chip detector caution panel (figure 1-11) is located above the main caution panel. The chip detector caution panel illuminates to indicate which gearbox has metal particles. When a chip is detected the MASTER CAUTION light illuminates the chip detector light on the caution panel and the appropriate light on the chip detector caution panel (figure 1-21). Pressing the chip detector panel will illuminate the master caution light segment and chip detector portion on caution panel.

SEGMENT WORDING	FAULT CONDITION
90°	Metal Particles in 90° Gearbox
42°	Metal Particles in 42° Gearbox
XMSN	Metal Particles in Transmission
CBOX	Metal Particles in Combining Gearbox

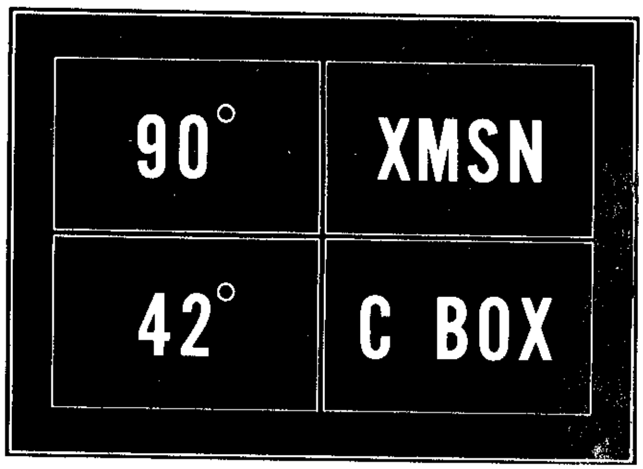


Figure 1-21. Chip detector panel

BRIGHT DIM SWITCH

The BRIGHT/DIM switch on the caution panel (figure 1-20) permits the pilot to manually select a bright or dimmed condition for the worded segments, master caution, RPM warning and fire detector warning lights on the instrument panel. After each initial application of power the lamps will come on in the bright condition. Momentarily placing the switch in the up position selects the bright condition, and down selects the dim condition.

NOTE

The system may only be dimmed if the pilot instrument light, located in the overhead console, is ON.

TEST SWITCH

The caution panel (figure 1-20) is provided with a TEST SWITCH enabling the pilot to manually test the caution light system. The switch is labeled TEST. Momentarily pushing down test switch to TEST position will cause illumination of all the segments on the caution panel, the master caution light, and RPM warning light. Testing of the system will not change any of the fault conditions which existed prior to testing. A malfunction can exist in the caution panel system without knowledge of the aircrew. Pushing the Test Switch only verifies the lights will illuminate.

WARNING SYSTEM

The warning system consists of individually illuminated warning lights mounted on the instrument panel. The purpose of this system is to provide visual indication of the conditions or combinations that have occurred.

RPM — LIGHT WARNING
- Rotor drops below 92% ± 2% or exceeds 103% ± 2
  Either engine Ng speed drops below 52% ± 2

AUDIO WARNING
- Rotor rpm drops below 92% ± 2

FIRE — Fire in engine 1 or 2 as indicated by light in handle

Fire Detector/Warning System

Two fire detector warning lights are located on each FIRE PULL handle (figure 1-11) on the instrument panel. The lights inform the pilot of an engine fire by illumination of the light on the appropriate FIRE PULL handle. To test the lights press fire detector PRESS to TEST switch (figure 1-11). Two lights in each handle should illuminate when pressed, extinguish when released. The detector elements are heat sensing, connected in a series loop across the fire detector control unit. These heat sensing elements exhibit high electrical resistance at normal ambient temperatures, but their resistance drops rapidly when heated. The fire detector unit senses this low resistance from any fire or overheat condition which causes the appropriate FIRE PULL handle to illuminate. Power is supplied by the 28V DC essential bus.

EMERGENCY EQUIPMENT

FIRE EXTINGUISHING SYSTEM

Portable Fire Extinguishers

Two portable fire extinguishers are provided with the helicopter, one is located to the right of pilot's seat on the floor, and the other, on the left side of the pedestal. The mounting brakets are of the quick opening type for rapid removal by pilot or crew (figure 3-1).

NOTE

For the fire extinguisher to discharge properly, it must be held in a near upright position.

Engine Fire Extinguishers

The engine fire extinguishing system includes the MAIN and RESERVE fire extinguisher bottles, engine No. 1 and engine No. 2 fire extinguisher relays, MAIN FIRE EXT and RESERVE FIRE EXT circuit breakers, a three-position level-lock FIRE EXT selector switch, FIRE 1 PULL handle and FIRE 2 PULL handle. The MAIN fire extinguisher bottle, located aft of engine No. 2, may be activated at any time either FIRE PULL handle is pulled and the selector switch is then positioned to MAIN. However, with both FIRE PULL handles pulled the main bottle will not fire and if reserve is selected, the agent will be discharged into both engines. The RESERVE bottle, located aft of the No. 1 engine, may be activated only when a fire is detected by one of the fire detector units or the test switch is depressed, and the appropriate FIRE PULL handle is pulled and the selector switch is then placed in the RESERVE position. The FIRE EXT selector switch must be pulled out in order to select RESERVE position.

Engine Fire Extinguisher Controls

When a fire occurs in an engine compartment, appropriate FIRE PULL handle (figure 1-11) illuminates. Pulling FIRE PULL handle closes the respective particle separator door and engine fuel shut-off valve in preparation for firing extinguisher bottles. FIRE EXT selector switch (figure 1-11) is then positioned to MAIN which fires the main bottle to the engine compartment. If an indication of fire still exists, the reserve fire extinguisher bottle can be fired into the selected engine compartment by placing FIRE EXT selector switch to RESERVE. Power is supplied by the 28V DC essential bus (figure 1-17) and protected by circuit breakers MAIN FIRE EXT, RESERVE FIRE EXT, ENG 1 FIRE DET and ENG 2 FIRE DET on the circuit breaker panel (figure 1-18).

NOTE

When the fire or overheat condition is eliminated, the light extinguishes on appropriate FIRE PULL HANDLE.

FIRST AID KITS

Four first aid kits are provided in the cabin area, located, two each, on inboard side of each hinged panel aft of pilot, co-pilot (figure 3-1).

NOTE

If the hinged panels are removed for flight, remove the first aid kits from the panels and secure them in the cabin where they will be immediately available.

PAX ALARM SYSTEM

The PAX alarm system consists of the PAX alarm switch (a guarded switch) located on the overhead console panel (figure 1-18), and the PAX alarm horn. The system receives 28V DC from the essential bus (figure 1-17) and protected by the PAX ALARM circuit breaker (figure 1-18). When the PAX alarm switch is activated, the horn emits a warning signal.

DOORS

PILOT'S AND COPILOT'S DOORS

The pilot's and copilot's entrance doors (figure 1-2) are formed aluminum frames with plastic windows in the upper section. Ventilation is increased by lowering the windows (figure 1-10). Cam type door latches are used and both pilot and copilot doors (figure 2-3) are jettisonable internally or externally.

CARGO DOORS

The cargo doors (figure 2-3) are formed aluminum frames with removable plastic windows in the upper section which can be jettisoned by pulling the release handle and pull in the window. These doors are on rollers and slide aft to the open position allowing access to the cargo area. The helicopter can be flown with doors closed, open, or off. See Section V for airspeed limitations.

HINGED PANELS

Two hinged panels (figure 1-2) forward of the cargo doors, provide a larger entrance to the cargo-passenger compartment. The panels may be remove prior to flight, provided the cargo door is open.

PILOT'S AND COPILOT'S SEATS

The pilot's and copilot's seats (figure 1-10) are adjustable and each seat is mounted on two fixed tracks. Vertical height adjustment is accomplished by a lever on the right hand side. Forward and aft adjustment is accomplished by the lever on the left hand side. Each seat is equipped with a lap safety belt and inertia-reel shoulder harness.

SHOULDER HARNESS

An inertia-reel shoulder harness is incorporated in the pilot's and copilot's seats with manual lock-unlock handle (figure 1-10) on the right side of the seat. Movement of the handle to the forward position locks the inertia reel, movement of the handle to the aft position unlocks the reel. With the control in unlock position, the reel will automatically lock when the helicopter encounters an impact force of two or three "G" deceleration. Locking of the reel can be accomplished in any position and the reel will automatically take up the slack in the harness. To release the lock it is necessary to lean back slightly releasing tension on the lock, move the control handle first to the lock position and then to the unlock position. It is possible to have pressure against the seat back whereby no additional movement can be accomplished and the lock cannot be released. If this condition occurs, it will be necessary to release shoulder harness.

AUXILIARY EQUIPMENT

The following systems and equipment are covered in Section IV:

Loudspeaker System

Internal Rescue Hoist

External Cargo Installation

Armament Systems

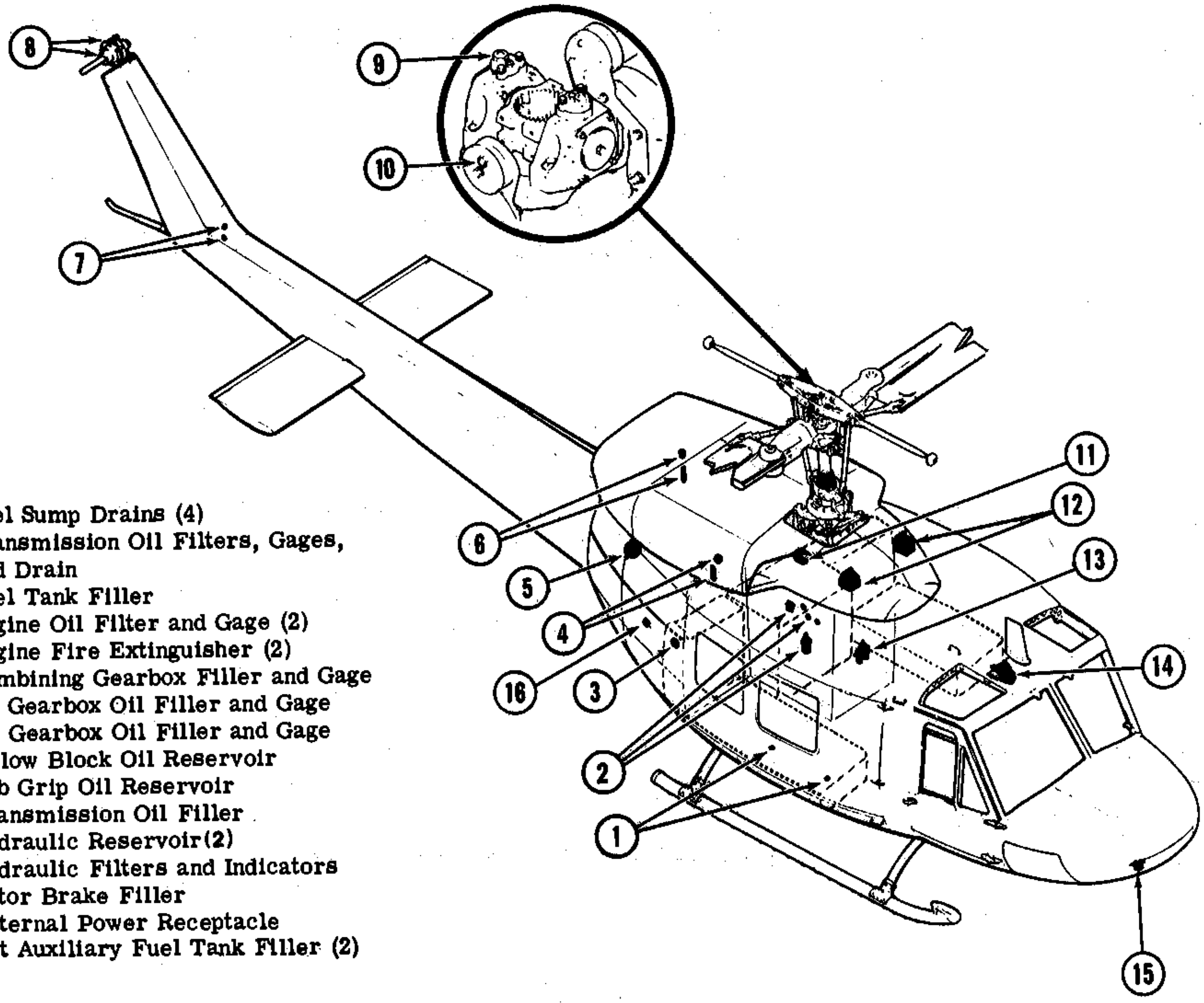
Gunner ICS

Passive Armor

Auxiliary Fuel Tanks

Communications and Associated Electronic Equipment

Heating and Ventilation System



1. Fuel Sump Drains (4)
2. Transmission Oil Filters, Gages, and Drain
3. Fuel Tank Filler
4. Engine Oil Filter and Gage (2)
5. Engine Fire Extinguisher (2)
6. Combining Gearbox Filler and Gage
7. 42° Gearbox Oil Filler and Gage
8. 90° Gearbox Oil Filler and Gage
9. Pillow Block Oil Reservoir
10. Hub Grip Oil Reservoir
11. Transmission Oil Filler
12. Hydraulic Reservoir (2)
13. Hydraulic Filters and Indicators
14. Rotor Brake Filler
15. External Power Receptacle
16. Aft Auxiliary Fuel Tank Filler (2)

FLUID SPECIFICATIONS	USAF	NATO SYMBOL	CIVILIAN GRADES
Engine Fuel	MIL-T-5624 Grade JP-4	F-40	Jet-B
Engine Alternate Fuel	MIL-T-5624 Grade JP-5 MIL-T-83133 Grade JP-8	F-44 F-34 F-30	Kerosene Jet-A-1 Jet-A
Engine Oil and Combining Gearbox	MIL-L-23699	0-156	
Alternate Engine Oil	MIL-L-7808	0-148	
Pillow Block Oil	MIL-L-23699	0-156	
Transmission Oil	MIL-L-23699	0-156	
Tail Rotor Gearbox Oil, 42° and 90°	MIL-L-23699	0-156	
Hydraulic Fluid	~~MIL-H-5606~~	H-515	

SEE 15-*100 MIL-H-83282

Figure 1-22. Servicing diagram (Sheet 1 of 2)

Some fuels serviced by U.S. INTO-PLANE contracts, as listed in Enroute Supplement, no longer contain icing inhibitor. When using alternate and/or commercial fuels, the aircrews must determine if the fuel contains fuel system icing inhibitor. Refer to T.O. 42B1-1-14 for additional fuel usage data.

WARNING

When using fuels without anti-icing additives avoid flying at altitudes where indicated OAT is below -32°C (-25°F) to preclude fuel system icing.

CAUTION

When changing from JP-4 to JP-5 or reverse, a specific gravity adjustment to the fuel control may be required. (Perform acceleration check in accordance with Section VII.)

CAUTION

When utilizing JP-8 fuel, maintain acceleration parameter. Fuel control adjustment may be required.

Figure 1-22. Servicing diagram (Sheet 2 of 2)