WARNING

If the engine is shut down with a T-handle it should be allowed to coast down to a complete stop before repositioning the T-handle to the FUEL-ON position to avoid a post-shutdown internal engine fire.

FUEL TRANSFER SYSTEM

See Section VII

FUEL QUANTITY INDICATING SYSTEM

The fuel quantity indicating system is of the capacitance type and consists of the following components: four tank sensing units, a fuel quantity indicator, fuel quantity selector switch, a fuel indicator test switch, a low fuel level sensor and a caution panel light. The system accurately measures the fuel quantity in both tanks regardless of temperature changes, provides low fuel level warning, has testing capability and indicates total, forward, or aft fuel quantity as selected by the pilot.

Fuel Quantity Indicator

A fuel quantity indicator, on the instrument panel (15, figure FO-1), indicates the total fuel quantity or the quantity in each tank in pounds. Due to the principle of operation, there is virtually no error in fuel quantity indication arising from volumetric changes of the fuel at different temperatures. The indicator receives power from the AC essential bus or ground inverter through a circuit breaker marked QTY under the heading FUEL on the forward circuit breaker panel.

Fuel Quantity Selector Switch

A fuel quantity selector switch is on instrument panel (figure FO-1). The switch is marked FUEL QTY SEL, and has positions, TOT, FWD, and AFT. When the selector switch is placed on TOT, the fuel quantity indicator will indicate total amount of fuel in both forward and aft fuel tanks. When the switch is placed in either FWD or AFT, the indicator will indicate only the amount of fuel in that specific tank. the selector switch is always left on the FWD position following checks of TOT or AFT fuel quantity because the engine draws fuel from the forward tank.

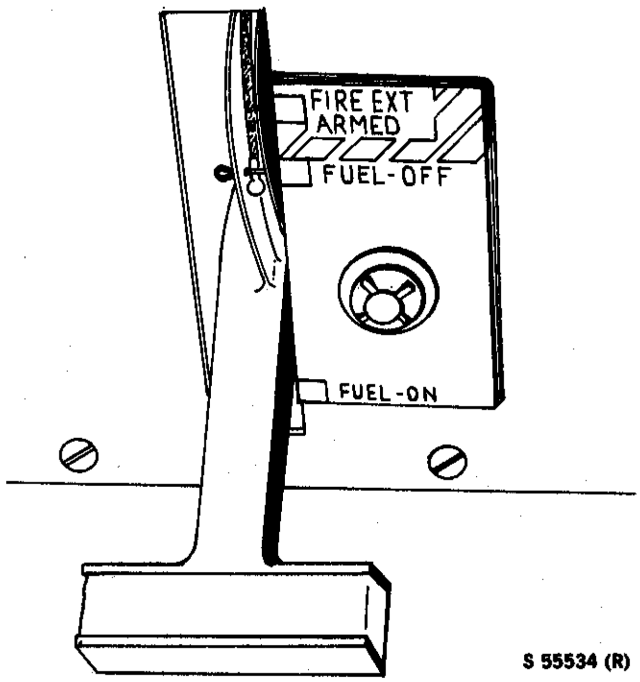


Figure 1-16. Fuel Shutoff T-Handle

Fuel Quantity Indicator Test Switch

A momentary switch (17, figure FO-1), marked FUEL GAGE TEST is on the instrument panel below the fuel quantity indicator. The switch, when depressed, should cause the fuel quantity needle to reflect down scale without hesitation. When the switch is released the needle should return to its original reading.

FUEL LOW LEVEL CAUTION LIGHT

A fuel low level caution light, marked FUEL LOW, is on the caution advisory panel (figure 1-27). The light is connected through a control unit to a sensing element located in the forward fuel tank. The light goes on when approximately 237 to 269 pounds of JP-4 or 255 to 289 pounds of JP-5 (37 to 42 gallons) remain in the forward tank with a 5° nose down attitude. This remaining fuel is enough for approximately 30 minutes of flight at cruising power. The caution light operates on current from the DC essential bus and is protected by a circuit breaker, marked LOW LEVEL, under the general heading FUEL, on the forward circuit breaker panel.

ELECTRICAL SYSTEMS

Two electrical systems (figures FO-3 and FO-4), an

AC system and a DC system, are installed in the helicopter. The AC system supplies and distributes 115/200 volts and 26 volts AC power. The DC system supplies and distributes 28 volts or 24 volts DC power. The primary power sources are two generators. The generators provide 115/200 volts AC thru voltage regulator/supervisory panels to the AC essential and non-essential buses. Two auto-transformers step down 115 volts AC, from the AC essential bus to provide 26 volts AC. 115 volts AC power is also provided by a ground inverter that functions automatically when the DC essential bus is energized and the AC essential bus is de-energized. Two transformer rectifiers (T/R's) rectify 115/200 volts AC from the A/C busses to provide 28 volts DC for the DC system. The T/R's power the start, essential, and non-essential busses thru reverse current cutout relays. When the AC system is not energized, a 24 volt DC nickel-cadmium battery is provided to operate DC powered equipment for a limited time. The electrical systems may be operated from a ground power source thru the AC or DC external power receptacles. With AC external power, all AC and DC equipment may be utilized. With DC external power, only the DC buses are powered. Both systems are controlled by various switches in the cockpit. Each item of electrical equipment is protected from overload by a circuit breaker on one of three circuit breaker panels. A failure of the AC or DC power source is indicated by an appropriate caution light on the caution advisory panel.

GENERATORS

The two brushless 10KVA, 115/200 volts, 400 cycle three-phase generators are mounted on, and driven by, the accessory drive section (figure 1-13) of the main gear box. Each generator has its own voltage regulator supervisory panel and each generator maintains separate loads. The generators operate whenever the main gear box is operating, supplying AC electrical power to the AC distribution system when the generator switches are in the ON position and all supervisory panel functions are satisfied.

Voltage Regulator—Supervisory Panels

The voltage regulator-supervisory panels (figure FO-3) in the transition section, provide voltage regulation, overvoltage, undervoltage, ground or line-to-line faults and under frequency protection. Output voltage of each generator is controlled at 115 volts by a voltage regulator. The supervisory panels sense proper voltage, overvoltages, undervoltages line faults, under frequencies and when all conditions are proper, allow power to flow through the No. 1 and No. 2 line contactors to the AC essential and non-essential busses. If any improper conditions are sensed, the supervisory panel drops the generator from its load by de-energizing the respective line contactor. Underfrequency occurs at 90% Nr or less. The generator is automatically returned to the line when rotor speed is increased above 90% Nr. A generator disabled due to over voltage, undervoltage or line faults may be regained by correcting the discrepancy and moving the appropriate generator switch to the OFF RESET position, then to the ON position.

Generator Switches

The generator switches, on overhead switch panel (figure FO-2) under general heading GENERATOR, have marked positions ON, OFF RESET, and TEST. When the switches are placed in ON, generator power is connected through the respective line contactor to the appropriate bus. When No. 1 generator switch is placed in OFF RESET, the No. 1 main line contactor relay and the bus-tie relay are de-energized. When the No. 2 generator switch is placed in OFF RESET, only the No. 2 line contactor is de-energized. The momentary TEST position provides a means to test the operation of the generator and supervisory panel. By placing the generator switch in the TEST position, power that would normally be used to energize the line contactor is diverted to a test relay. If the generator and supervisory panel are operating correctly, the respective generator caution light on the caution advisory panel will go off.

Generator Caution Lights

Two generator caution lights, marked NO. 1 GENER—ATOR and NO. 2 GENERATOR respectively, are on the caution-advisory panel (figure 1-27). These lights will go on whenever the associated generator is taken off the line by the opening of its line contactor. The generator caution lights are powered by the DC essential bus and are protected by circuit breakers 1-GEN-2 under the general heading WARNING LIGHTS on the forward circuit breaker panel.

GROUND INVERTER

The ground inverter is in the transition section and is used when the generators are off to provide AC power for engine instruments prior to and during engine start. The ground inverter is rated at 115 volts AC, 100 VA, single phase and 400 cycles. The ground inverter is automatically energized whenever the DC essential bus is energized and the AC essential bus is not energized. When the AC essential bus is energized, a relay is actuated that disconnects the ground inverter from the DC essential bus and shifts the ground inverter to the AC essential bus. The ground inverter is protected by a circuit breaker marked INV INPUT, on the aft circuit breaker panel.

AUTOTRANSFORMERS

Two autotransformers step down B and C phase 115 volt AC essential bus power to 26 volts for navigation radio and pressure indicators. The ϕB (radio) autotransformer powers navigation indicators. (See Section IV, ELECTRICAL POWER DISTRIBUTION). The ϕC (pressure indicator) autotransformer powers engine, transmission, and hydraulic pressure indicators and the torque indicators. The ϕB autotransformer is protected by a circuit breaker marked AUTO XMFR ϕB, on the forward circuit breaker panel (figure FO-3). The ϕC autotransformer is protected by a circuit breaker marked AUTO XMFR ϕC, on the aft circuit breaker panel (figure FO-3). The ϕC autotransformer also operates off of the ground inverter when it is powered.

AC ESSENTIAL BUS

The AC essential bus, which distributes power to all AC essential equipment, may be powered by the No. 1 generator, the No. 2 generator, or external power. The No. 1 generator, when operating, will always power the AC essential bus through the No. 1 line contactor. Failure of the No. 1 generator de-energizes the No. 1 line contactor and bus tie relay. The No. 2 generator will then power the AC essential bus through the No. 2 and No. 1 line contactors. External power is supplied to the AC essential bus through the external power relay and the No. 2 and No. 1 line contactors.

AC NON-ESSENTIAL BUS

The AC non-essential bus, which powers the No. 2 TR, is energized by the No. 2 generator or by external power. Power is supplied from the generator to the bus through the No. 2 line contact and bus-tie relay. Failure of either generator will result in loss of the AC non-essential bus. The No. 2 TR will be the only AC powered equipment lost.

PILOT'S VGI

The pilot's VGI is normally powered by the No. 2 generator. If the No. 2 generator fails, the No. 2 generator caution light goes on and the pilot's VGI is automatically switched to the AC essential bus. (See Section I ASE Vertical Gyro).

ALTERNATING CURRENT CIRCUIT BREAKERS

Alternating current circuit breakers, protecting various AC circuits, are on the forward and aft circuit breaker panels (figure FO-3). The forward circuit breaker panel is to the left of the cockpit access door. The aft circuit breaker panel is on the cabin compartment aft bulkhead by the entrance door to the transition section. All circuit breakers are marked to indicate the circuit they protect and are of push-pull type that may be reset. Any malfunctioning circuit may be isolated from the AC power supply system by pulling out its circuit breaker. Circuit breaker guards are installed on the forward circuit breaker panel. The guards provide protection against accidental tripping or breakage of the circuit breakers.

TRANSFORMER RECTIFIERS

Two transformer-rectifiers rated at 200 amperes, 28 volts DC, are used to convert 115/200 volts AC to 28 volts DC. The TR's are on the right side of the transition section (figure FO-3). Two loadmeters, one for each T/R indicate the output of the respective T/R. The No. 1 transformer-rectifier is powered by the No. 1 generator through the AC essential bus, and the No. 2 transformer-rectifier is powered by the No. 2 generator through the AC non-essential bus. The No. 1 transformer-rectifier is protected by a ganged circuit breaker, marked No. 1 TR, on the aft circuit breaker panel, and the No. 2 transformer-rectifier is protected by a ganged circuit breaker, marked No. 2, also on the aft circuit breaker panel. The loss of either generator will result in the loss of the No. 2 transformer-rectifier. The loss of either TR will result in the loss of the DC non-essential bus. The non-essential bus may be regained by placing the DC non-essential override switch to the ON position.

Transformer-Rectifier Caution Lights

Two transformer-rectifier caution lights marked No. 1 RECTIFIER and No. 2 RECTIFIER, are on the caution advisory panel (figure 1-27). Failure of a transformer-rectifier, or reverse current cutout relay will light the associated caution light. The lights are powered by the start bus and do not have circuit breakers.

Reverse Current Cutout Relays

Two reverse current cutout relays, one for each TR, monitor the output of the TR's. These relays automatically connect their respective TR to the DC start bus when TR output voltage is greater than that of the bus. Conversely, they automatically disconnect their respective TR from the DC start bus when TR voltage drops to a value lower than that of the bus supply. A caution light on the caution-advisory panel, marked NO. 1 RECTIFIER or NO. 2 RECTIFIER goes on, indicating which TR has been dropped from the line.

BATTERY

The 24-volt, 22-ampere hour nickel cadmium battery, on the right side of the transition section, is accessible from inside the helicopter. Battery power is used for limited ground operations, including starting of the engine, when no external power is available. The battery is used as an emergency source of power in event of failure of both generators and/or transformer-rectifiers, or failure of the No. 1 TR and either generator. The battery is charged whenever the start bus is powered by a T/R or external power and the battery switch is in the START or ON position. The battery is protected by an overtemperature circuit which limits the charge to the battery when the battery temperature reaches approximately 135°F. Then, an advisory light on the caution-advisory panel, marked BAT OVTEMP, goes on. This advisory light will remain on until the battery cools to approximately 115°F even, though the aircraft may be shut down with all power secured.

CAUTION

> If the BAT OVTEMP advisory light goes on, monitor the battery for possible thermal runaway. Refer to Section III for procedures.

The overtemperature circuit is protected by a circuit breaker marked BAT OVTEMP, on the aft circuit breaker panel.

Battery Switch

The three position battery switch is on the overhead panel (figure FO-2) with marked positions START, OFF, and ON. In either the START or ON positions, the battery is connected to the start bus. The START position also provides a convenient way to limit DC loads by disconnecting the DC essential bus from the start bus. This position is used during battery starts.

START BUS

The start bus functions to distribute power to the essential and non-essential buses and to certain equipment required for engine starting. It may be powered by the battery, transformer-rectifiers or external power. During battery starts with the battery switch in the START position, the start bus supplies power to the No. 1 fuel pump caution light, nonflight instrument lights No. 1 fuel boost pump, rectifier caution lights, and the starter and ignition systems.

DC ESSENTIAL BUS

The DC essential bus supplies power to operate all DC equipment necessary for safety of flight and limited mission accomplishment. The DC essential bus receives power from the start bus and may be energized by either T/R, external power or the battery.

DC NON-ESSENTIAL BUS

The DC non-essential bus supplies power for DC equipment that is not essential to safety of flight or limited mission accomplishment, and receives power from the start bus. The DC non-essential bus is automatically energized any time both T/Rs are operating or when DC external power is applied. Loss of either T/R automatically reduces the DC electrical load by de-energizing the DC non-essential bus. The non-essenial bus may be regained through the use of the DC non-essential bus override switch.

DC NON-ESSENTIAL BUS OVERRIDE SWITCH

The non-essential bus override switch on the overhead switch panel, (figure F0-2) under the general heading DC NON ESS BUS OVD, provides a means for the pilot to manually connect the DC non-essential bus to the start bus. The switch has two positions, one of which is marked ON. In the ON position, the non-essential bus will remain energized as long as the DC essential bus is energized.

Monitor Ammeter Advisory Light

A green MONITOR AMMETER advisory light is on the caution advisory panel (figure 1-27). This light will go on whenever the DC non-essential bus override switch is placed in the ON position and a generator or transformer-rectifier is off the line. This light is connected to the start bus and does not have a circuit breaker.

LOADMETER

Two loadmeters on the forward circuit breaker panel (figure FO-3), visually indicate the power output of its respective transformer-rectifier. The face of both loadmeters read from 0 to 1.0. The 1.0 indication is equivalent to 100% of the rated output of the T/R or 200 amps. During engine start, the loadmeters are dropped from the circuit to protect them from the high DC loads encountered during start.

DC UTILITY RECEPTACLE

A utility receptacle labeled UT RECEPT 28 VDC, is in the cabin, aft of the cargo door on the right side. Power to the receptacle is supplied by the DC non-essential bus through the UTILITY RECP circuit breaker, on the aft circuit breaker panel. The receptacle is protected from damage and entrance of foreign matter by a cap.

DIRECT CURRENT CIRCUIT BREAKERS

Direct current circuit breakers, protecting the various DC circuits are on the forward and aft circuit breaker panels (figure FO-3). All circuit breakers are marked as to the circuit they protect and are of the push-pull type that may be reset. Any malfunctioning circuit may be isolated from the DC power supply system by pulling out the circuit breaker. Circuit breaker guards are installed on the forward circuit breaker panel. The guards provide protection against tripping or breakage of the circuit breakers.

EXTERNAL POWER

External Power Switch

The external power switch is on the overhead switch panel (figure FO-2) under the heading EXT PWR with positions marked ON and OFF. In the OFF position, neither external power relay can be energized. With the switch in the ON position and either AC or DC external power applied, the appropriate external power relay will be closed, provided that with AC external power, the BATT switch has been placed in the ON position. The external power switch is protected by a circuit breaker marked EXT PWR on the forward circuit breaker panel.

External Power Advisory Light

The external power advisory light, on the caution advisory panel (figure 1-27) and marked EXT PWR, will go on when the external power switch is ON and external power is being supplied to the aircraft. This light goes on whenever the external power relay is energized.

AC External Power Receptacle

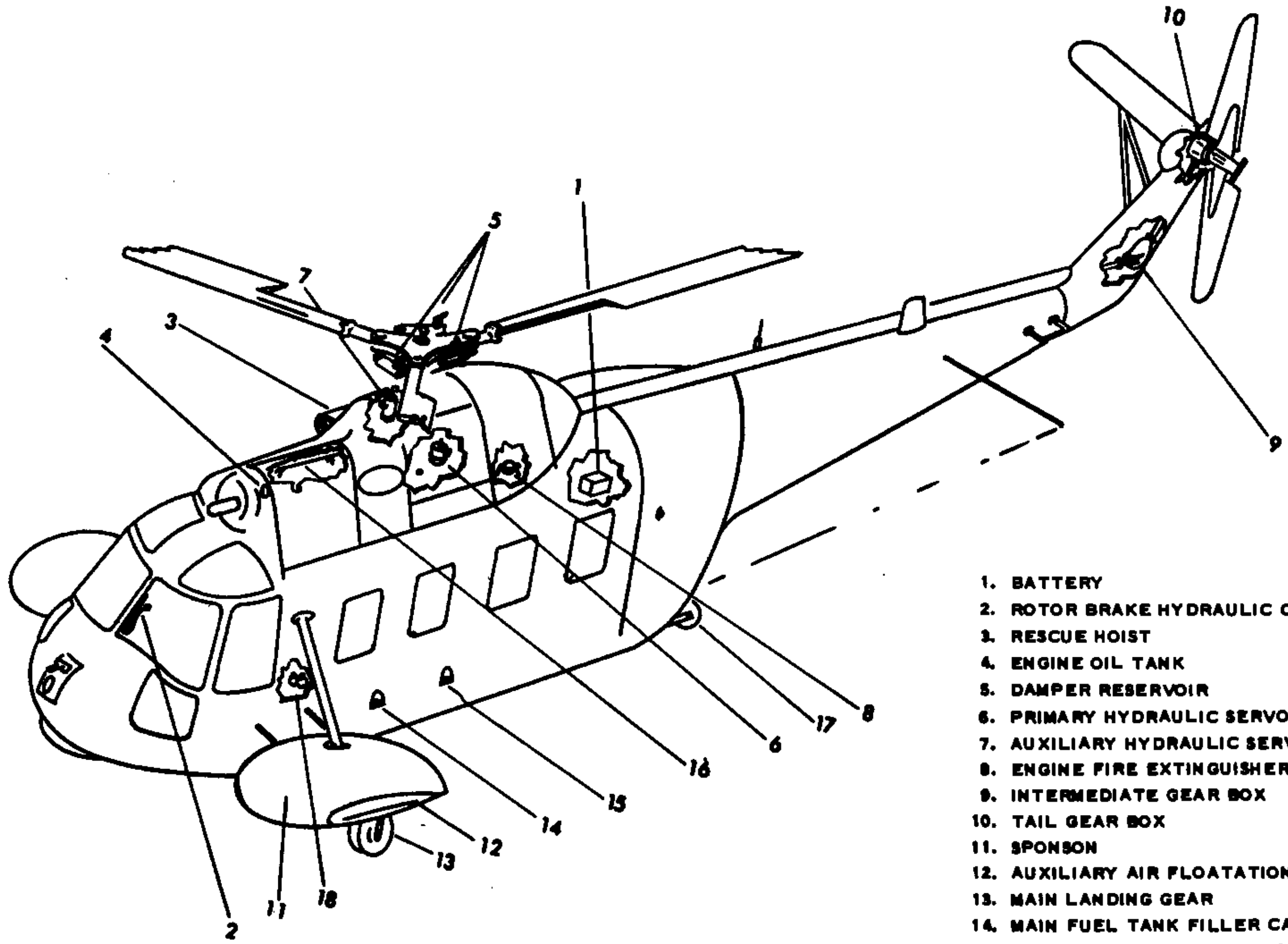
The AC external power receptacle permits connection of an external source of 115/200 volt, 400 cycle AC power to the AC distribution system. The AC receptacle (figure F0-3) is on the right side of the fuselage, aft of the cabin door. With external power connected, the battery and external power switches ON, DC power from the battery passes through the external power plug and receptacle to the phase sensing relay. If proper phasing and voltage of applied external power are sensed by the phase sensing relay, the external power relay will be energized, allowing the AC power through the NO. 2 and NO. 1 line contactors to the AC essential bus. The AC non-essential bus is energized, through the external power relay and bus-tie relay. The EXT PWR advisory light goes on when the external power relay is energized. When AC external power is applied, all buses of both the AC and DC are energized.

DC External Power Receptacle

The DC external power receptacle (figure FO-3) permits connection of an external source of 28 volt DC power to the DC system. The DC receptacle is on the right side of the fuselage aft of the cabin door. The external power switch must be in ON to utilize external power. External DC power is supplied through the start bus to the DC essential and non-essential busses.

FLIGHT CONTROL SYSTEMS

The flight controls are divided into three major systems: The cyclic control system, the collective pitch control system, and the directional control system (figure 1-24). Components of the systems include two cyclic control sticks, two collective pitch levers and two sets of directional control (tail rotor) pedals, all in the cockpit, the auxiliary servo cylinder assembly, the mixer unit, three primary servo units, and various control rods, cranks and cables. The cyclic control system controls the pitch of the main rotor blades by changing the angle of incidence of each blade individually and unequally. Moving either cyclic control stick in the desired direction of flight (fore, aft, left, or right) tilts the tip path plane in that direction causing the helicopter to move in the same direction. A cyclic control stick trim system maintains a selected cyclic position, a requirement for proper operation of the Automatic Stabilization Equipment (ASE). The collective pitch control system controls the angle of the main rotor blades by changing the angle of incidence of each blade simultaneously and equally. Moving either collective pitch



1. BATTERY
2. ROTOR BRAKE HYDRAULIC CYLINDER
3. RESCUE HOIST
4. ENGINE OIL TANK
5. DAMPER RESERVOIR
6. PRIMARY HYDRAULIC SERVO SYSTEM RESERVOIR
7. AUXILIARY HYDRAULIC SERVO SYSTEM RESERVOIR
8. ENGINE FIRE EXTINGUISHER BOTTLE
9. INTERMEDIATE GEAR BOX
10. TAIL GEAR BOX
11. SPONSON
12. AUXILIARY AIR FLOATATION SYSTEM
13. MAIN LANDING GEAR
14. MAIN FUEL TANK FILLER CAP
15. AFT FUEL TANK FILLER CAP
16. ENGINE FIRE EXTINGUISHER
17. TAIL LANDING GEAR
18. AIR BOTTLES (SEE 12)

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Figure 1-17. Servicing Diagram

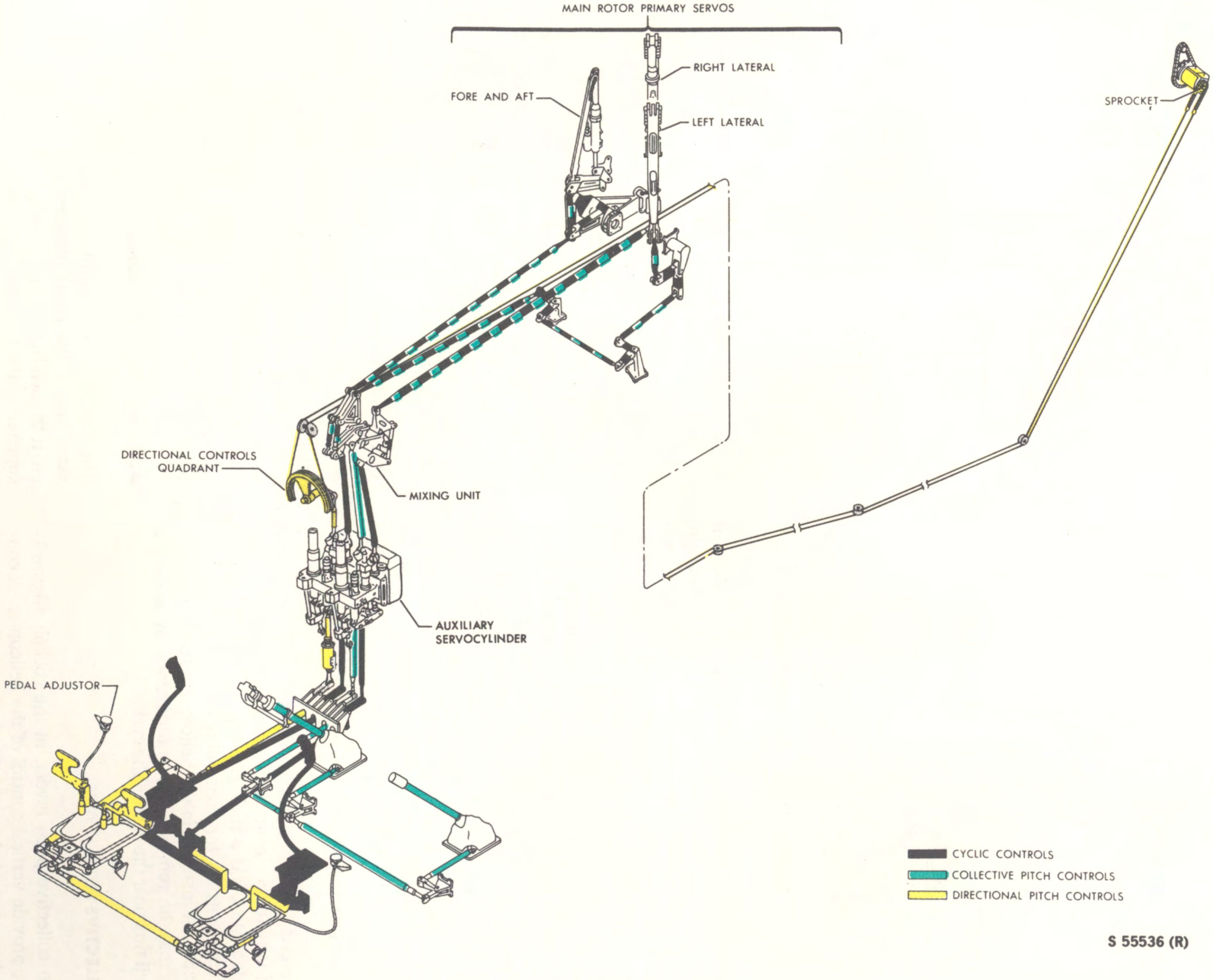


Figure 1-18. Flight Controls

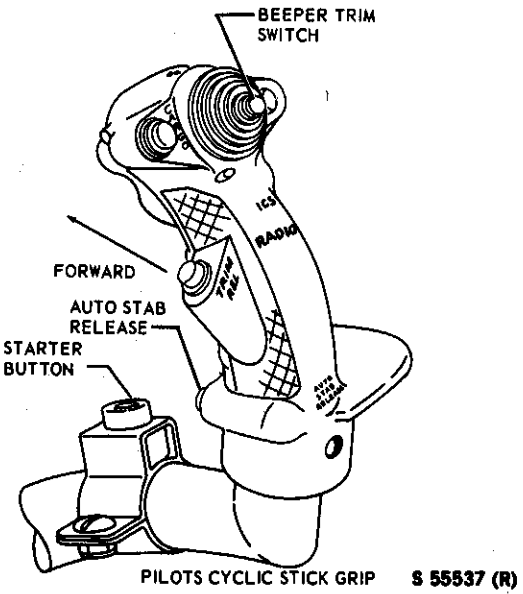


Figure 1-19. Pilot's Cyclic Stick Grip

lever up or down causes the helicopter to climb or descend by increasing or decreasing blade pitch angle. Cyclic control stick and collective pitch lever movements in the cockpit are transmitted by a series of control rods and bell cranks to the auxiliary servo cylinder assembly, the mixing unit and the primary servo units. The directional control system controls the pitch angle of the tail rotor blades. Pushing either left pedal increases blade pitch so the tail rotor over compensates for main rotor torque, causing the helicopter to turn left. Pushing either right pedal decreases blade pitch, allowing torque to turn the helicopter to the right. Tail rotor pedal movements are transferred thru control rods to the auxiliary servo cylinder assembly and cables to the tail rotor pitch change mechanism.

CYCLIC CONTROL STICKS

Two cyclic control sticks in the cockpit provide lateral and longitudinal control of the helicopter. Each cyclic control stick has a stick grip (figure 1-19) containing several switches for controlling various systems and equipment installed in the helicopter. The copilots cyclic may be removed for maintenance by pulling a small ring near the base of the cyclic.

COLLECTIVE PITCH LEVERS

Two collective pitch levers, in the cockpit (figure 1-20), provide vertical control of the helicopter. A control box, mounted on the forward end of the pilot's collective pitch lever, contains several switches for

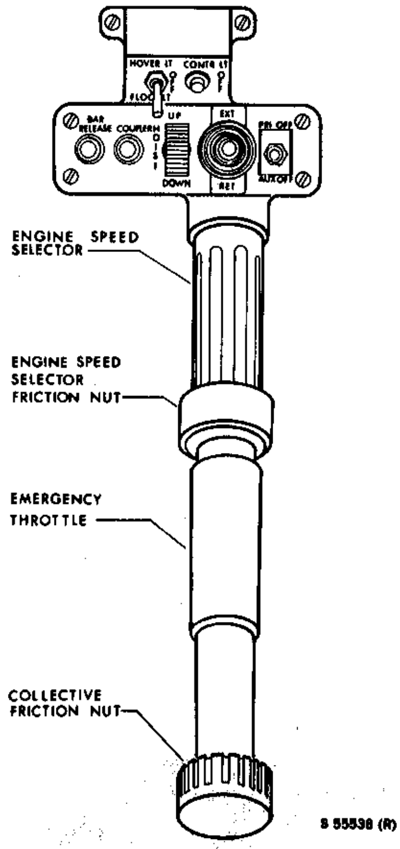


Figure 1-20. Pilot's Collective Pitch Lever

controlling various systems and equipment installed in the helicopter. Speed selector and emergency throttle controls are mounted on both collective pitch levers. The pilot's collective pitch lever also has a friction lock that may be adjusted to prevent the collective from creeping in flight.

TAIL ROTOR PEDALS

Two sets of tail rotor pedals (17 and 25, figure 1-2), in the cockpit, provide heading (yaw) control of the helicopter. Each set of tail rotor pedals may be independently adjusted, fore or aft, by a separate tail rotor pedal adjustment knob marked FWD and AFT with appropriate arrows. Toe brakes are mounted on the pilot's tail rotor pedals.

AUXILIARY SERVOCYLINDER ASSEMBLY

The auxiliary servocylinder assembly, in a compartment behind the pilot, provides a separate hydraulic assist to each flight control system. Hydraulic power is supplied by the auxiliary servo hydraulic system pump. The auxiliary servocylinder assembly will continue to function in case hydraulic pressure is lost. Mounted on the fore-and-aft and lateral banks of the servocylinder assembly are a pair of solenoid-operated cyclic control stick trim system valves. Mounted on the fore-and-aft, lateral, and directional (yaw) banks are flapper valves for the introduction of automatic stabilization equipment (ASE) control signals. There is no flapper valve for ASE control of the collective bank. The directional bank also incorporates a pedal dampening piston which restricts sudden turns of the helicopter preventing overtorque of the tail rotor drive system. Hydraulic pressure to the auxiliary servocylinder assembly may be shut off by actuating the servo hydraulic electrical shutoff system switch.

AUXILIARY SERVO HYDRAULIC SYSTEM PUMP

The auxiliary servo hydraulic system pump (figure 1-21), on the main gear box accessory section, (figure 1-13) is a variable delivery, piston type, constant pressure pump. The pump provides pressurized hydraulic fluid to operate the auxiliary servocylinder assembly, the main landing gear and rescue hoist. The pump is driven by the main gear box.

MIXER UNIT

The mixer unit, above the auxiliary servocylinder assembly, is the junction point where cyclic control stick movements and/or collective pitch lever control movements merge to transmit individual or combined control movements to the primary servo units.

PRIMARY SERVO UNITS

Three primary servo units, mounted on the main gear box, transmit fore-and-aft, lateral and vertical control inputs to the stationary star assembly. The primary servo units are hydraulically assisted by power supplied from the primary servo hydraulic system pump. The primary servo will continue to function if hydraulic pressure is lost. Hydraulic pressure to the primary servo units may be shut off by actuating the servo hydraulic electrical shutoff system switch.

PRIMARY SERVO HYDRAULIC SYSTEM PUMP

The primary hydraulic system pump (figure 1-21), mounted on the accessories section of the main gear box, (figure 1-13) is a variable delivery, piston type, constant pressure pump that provides pressurized hydraulic fluid to operate the primary servo units. The pump is driven by the main gear box.

SERVO HYDRAULIC ELECTRICAL SHUTOFF SYSTEM

The servo hydraulic electrical shutoff system (figure 1-21), interconnects the two servo systems electrically to provide a means of shutting off hydraulic pressure to only the selected servo system. System components include a servo shutoff switch, two servo hydraulic pressure indicators, two pressure switches and two caution advisory panel capsule lights. Electrical power for the system, except the pressure indicators, is supplied from the 28 volt DC essential bus protected by two circuit breakers marked PRI and AUX under the general heading SERVO, on the forward circuit breaker panel.

Servo Hydraulic Electrical Shutoff System Switch

The servo shutoff switch on the pilot's collective control box, is a three-position toggle switch. The marked switch positions are PRI OFF and AUX OFF. Both servo systems are normally in operation with the switch in the unmarked center (ON) position. To turn off the primary servos, the switch is placed in the forward PRI OFF position. To turn off the auxiliary servo, switch is placed in the aft AUX OFF position. Neither servo may be shut off unless normal hydraulic pressure is in the remaining system.

SERVO HYDRAULIC PRESSURE INDICATORS

The servo hydraulic system pressure indicators (24 and 28, figure FO-1), marked AUX HYD PRESS and PRI HYD PRESS, are on the instrument panel. The indicators visually indicate pressure within the auxiliary and primary servo systems and operate on 26 volts AC from the ϕC autotransformer. The indicators are individually protected by circuit breakers marked HYD PRESS AUX and HYD PRESS PRI on the forward circuit breaker panel.

SERVO LOW PRESSURE CAUTION LIGHTS

Two hydraulic servo low pressure caution lights marked AUX HYD PRESS and PRI HYD PRESS are on the caution advisory panel (figure 1-27). If the pressure in the auxiliary servo drops below 1000 PSI or the pressure in the primary servo drops below 750 PSI, the respective caution light will go on.

CYCLIC CONTROL STICK TRIM SYSTEM

The cyclic control stick trim system is an electrically-

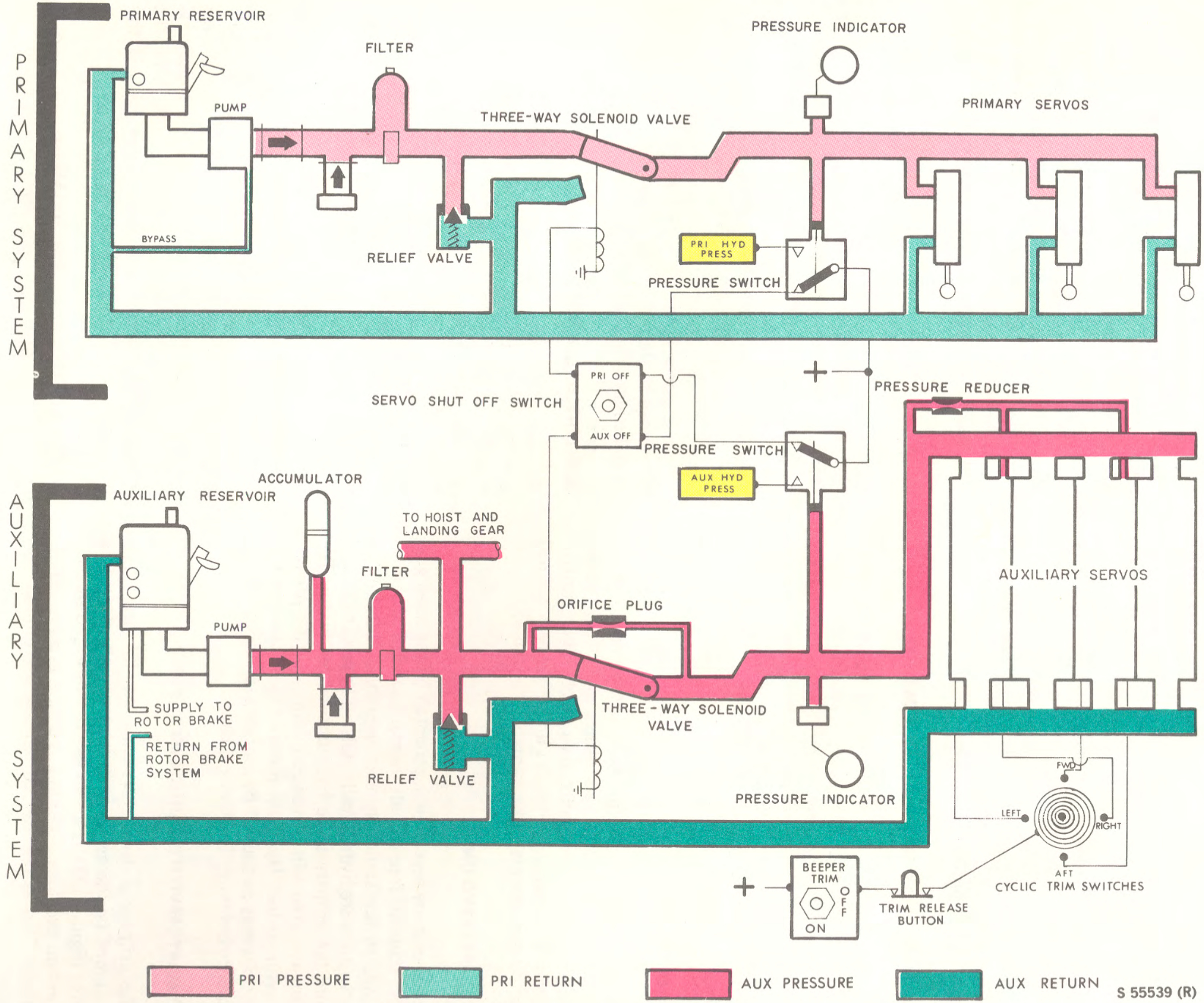


Figure 1-21. Primary and Auxiliary Hydraulic Systems

controlled, hydraulically-actuated system for moving the cyclic control sticks either fore, aft, left or right or maintaining the cyclic sticks in a selected position. Components of the system include a BEEPER TRIM master switch, two TRIM REL switches, two STICK TRIM switches, four solenoid-operated valves, two hydraulically-operated actuators, each containing a force gradient spring, and a pressure reducer to reduce auxiliary servo pressure to 60 PSI. The system is powered by 28 volts from the DC essential bus and is protected by a circuit breaker, marked BEEPER TRIM, on the forward circuit breaker panel.

Beeper Trim Master Switch

A BEEPER TRIM switch, on the overhead console (figure FO-2) with positions marked ON and OFF, provides master control of the stick trim system. With the BEEPER TRIM switch in the OFF position power is supplied to all four solenoids, opening the valves to allow free movement of the cyclic. When the BEEPER TRIM switch is placed in the ON position, electrical power is removed from the four solenoids, closing the valves to hydraulically lock both actuators. The force gradient springs within the actuators allow the cyclic sticks to be manually moved without disturbing the trim setting. When the cyclic stick is released, the force gradient springs will return the cyclic to the trimmed position.

Cyclic Stick Trim Release Switches

A cyclic stick trim release momentary switch, marked TRIM REL, is on each pilot's cyclic control stick grip (figure 1-19). By depressing the TRIM REL switch on either cyclic stick momentary disengagement of the system occurs by electrically opening all four solenoid valves. This allows free movement of the cyclic sticks to a new position. Releasing the switch will allow the cyclic sticks to be retained in the newly selected position. The cyclic sticks may also be moved against the tension of the force gradient springs and, when the stick is in the desired position, the TRIM REL switch may be momentarily depressed to allow followup of the actuators to hold the stick in the new position.

Cyclic Stick Trim Switches

Two thumb-operated cyclic stick trim four-way momentary switches, one on each cyclic control stick grip (figure 1-19) marked FWD, AFT, L, and R, allows the stick trim system to be operated with a fine degree of control. Both cyclic sticks may be repositioned at a controlled rate of displacement by momentarily actuating either switch in the desired direction. This enables the pilot to make gradual changes in cyclic stick position without a tendency to over control. It also prevents inadvertent coupling of lateral and longitudinal cyclic movements.

AUTOMATIC STABILIZATION EQUIPMENT (ASE)

The ASE used in this helicopter differs from the autopilot used in fixed wing aircraft in that it may be engaged at all times, has less control authority than the primary flight control system, and may be easily overridden through normal use of the flight controls. The pilot has direct control of the system at all times and can engage or disengage the entire ASE or any channel, as desired, by means of switches on the ASE control panel, channel monitor panel, and cyclic sticks. The flight directors in the ASE mode provides the pilot and copilot with visual indications of all ASE signals. Attitude and directional stabilization are controlled through pitch, roll and yaw channels. A fourth channel for barometric altitude stabilization is incorporated in the ASE but BAR ALT HOLD components are not installed in this helicopter. In the pitch and roll channels, fuselage attitude is held constant by comparing the actual attitude signal received from the selected vertical gyro with the desired attitude signal received from the cyclic stick position sensor. Automatic pitch and roll attitude stability correction occurs any time the helicopter is displaced from the desired attitude. In the yaw channel, the helicopter heading is maintained by comparing actual heading signals received from the MA-1 compass system with desired heading signals. The helicopter can be flown to the desired heading by use of the yaw trim knob on the ASE control panel or by use of the tail rotor pedals. When the pilot flies the helicopter to the desired heading by use of the tail rotor pedals, the yaw channel is placed in a synchronizing mode and no heading correction signal is developed until his feet are removed from the pedals. During the synchronizing mode, the yaw rate gyro develops a signal proportional to the rate of turn of the helicopter. This signal initiates an open-loop spring condition that produces a feedback force at the pedals. As the pilot presses either pedal to turn the helicopter, he feels an opposing force proportional to the rate of turn. The feedback force remains until the pilot rolls out on the new reference heading. Heading stability correction occurs any time the helicopter is displaced from the desired reference heading. The ASE system utilizes power from the AC essential bus, the No. 2 generator if the STBD gryo is selected, and the DC essential bus. Switches in the control circuit enable the pilot to engage and disengage the ASE. A thermal time delay

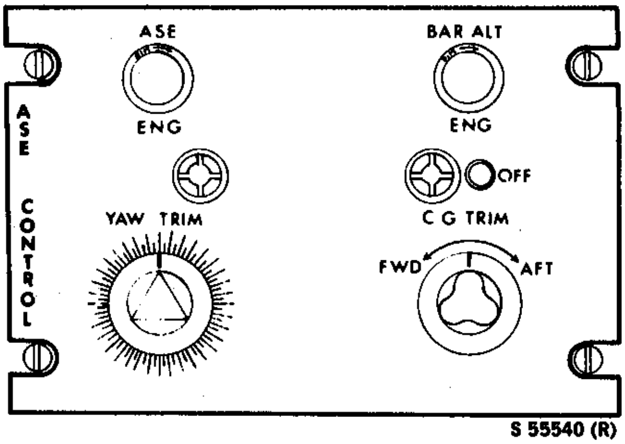


Figure 1-22. ASE Control Panel

relay is incorporated in the ASE engage circuit to allow approximately 120 seconds for AC power to bring the vertical gyros to a stabilized state before the ASE can be engaged. DC power is applied to the system when the ASE is engaged. The ASE is comprised of the following components: stick position sensors, vertical gyros, vertical gyro indicators, force link assembly micro switches, ASE amplifier, dual channel lag amplifier, ASE control panel, ASE disengage switches, auxiliary servo, channel monitor panel, flight director and caution advisory panel lights.

STICK POSITION SENSOR

Cyclic stick position sensors, in the auxiliary servo enclosure, provide desired attitude signals in pitch and roll channels.

VERTICAL GYRO

Two vertical gyros, copilots (port) and pilot's (starboard), provide actual attitude signals in pitch and roll. Either gyro may be selected for ASE operation by a switch on the channel monitor panel. Three-phase 115V AC electrical power is supplied to the pilot's (starboard) vertical gyro by the No. 2 generator through a ganged circuit breaker marked No. 2 GEN under the heading PILOT'S VGI. In the event of No. 2 generator failure, the pilot's vertical gyro is automatically switched to the AC essential bus and is then protected by a ganged circuit breaker marked No. 1 GEN under the heading PILOT'S VGI, on the forward circuit breaker panel. The copilot's (port) vertical gyro is powered by the AC essential bus through a ganged circuit breaker marked CO-PILOT'S VGI, on the forward circuit breaker panel.

Vertical Gryo Indicator

The actual attitude of the helicopter, sensed by the vertical gyros, is displayed on two vertical gyro indicators (7 and 39, figure FO-1). The output from the copilot's (port) vertical gyro is fed to the copilot's vertical gyro indicator and the output from the pilot's (starboard) vertical gyro is fed to the pilot's vertical gyro indicator.

Off Flag and Caution Panel Lights

Power failure detectors mounted on the tilt table monitor the three-phase AC power provided to each of the vertical gyro assemblies. It detects either a power failure or a voltage unbalance and causes the OFF flag to appear on the vertical gyro indicator of the affected assembly. In addition, a loss or unbalance of power to the pilot's (starboard) vertical gyro assembly will put on the GYRO ERECT light on the caution advisory panel (figure 1-27). The vertical gyro indicator OFF flags are displayed and the GYRO ERECT light remains ON for 60 seconds after power is applied to their respective vertical gyros. This indicates that they are unreliable until the gyros have had enough time to come up to speed. The GYRO ERECT caution light is protected by a circuit breaker marked GYRO ERECT, under the heading WARNING LIGHTS, on the forward circuit breaker panel. Additionally, a caution light (figure 1-27) marked ASE OFF comes on when the ASE is disengaged while the DC essential bus is energized. The ASE OFF caution light is protected by a circuit breaker marked ASE OFF WARN, on the forward circuit breaker panel.

YAW CHANNEL

The force link assembly microswitches actuate when 6-8 pounds of force is applied to either right or left tail rotor pedal. This nulls actual heading signals to the yaw channel while the pilot selects a new heading using the tail rotor pedals. Artificial feel in the tail rotor pedals is provided by feeding signals from the yaw rate gyro to the auxiliary servo which tends to move the pedals in a direction opposite to the direction of turn. Heading selection can also be made using the YAW TRIM knob on the ASE control panel.

ASE AMPLIFIER

The ASE amplifier compares desired attitude signals to actual attitude signals and derives an error signal which is fed to the auxiliary servo flapper valves. The ASE amplifier consists of a chassis assembly, four plug-in transistorized modules, and a yaw rate gyro. Three of the plug-in modules are the pitch, roll, and yaw channel amplifier modules which serve to amplify, modify, and compare the attitude signals received from the various attitude sensors. The fourth, the yaw synchronizer module, serves to keep the yaw channel nulled when the pilot is making a manual turn. The ASE amplifier utilizes both DC and AC electrical power. The DC essential bus provides DC power through a circuit breaker marked AUTO STAB, on the forward circuit breaker panel. The AC essential bus provides single-phase power through two circuit breakers marked AUTO STAB ϕA and AUTO STAB ϕB, on the forward circuit breaker panel.

DUAL CHANNEL LAG AMPLIFIER

The dual channel lag amplifier is on the ASE component rack. Helicopter response to control movement is more rapid in roll than in pitch. To prevent overcontrolling in roll, the desired attitude signal to the ASE amplifier is lagged by 0.5-0.8 seconds in the dual channel lag amplifier. In this way a new desired attitude signal does not combine with lateral cyclic movement to produce an excessive roll rate and subsequent overcontrolling by the pilot.

AUXILIARY SERVO

The auxiliary servo is a component of the flight control system. Its construction features permit either ASE or manual flight control inputs to be made at the auxiliary servo. Electrical input signals from the ASE amplifier actuate auxiliary (flapper) valves, on three of four auxiliary servo power pistons. The collective auxiliary servo valve is not used since altitude stabilization components are not installed in this aircraft.

ASE CONTROL PANEL

The ASE control panel, on the lower console (figure FO-2), contains the ASE system engage switch and trim controls for the pitch and yaw channels.

ASE Engage Switch

The ASE ENG switch is a normally open, momentary contact, push-button switch with a green self-contained indicating light, provided with a bezel-operated dimming iris. Pressing the ASE ENG switch engages the pitch, roll and yaw channels, lights the green indicating light, and turns off the ASE OFF caution light. The ASE is held engaged by power from the DC essential bus.

CG Trim Control

The CG TRIM control knob is cloverleaf-shaped. It allows the pilot to adjust the center point of pitch channel authority. This compensates for changes in the center of gravity of the helicopter which would shift cyclic stick position away from the center point of ASE authority.

YAW Trim Control

The YAW TRIM control knob is triangular-shaped. It permits the pilot to select a new desired heading without using the tail rotor pedals. The control dial is engraved with 72 units of 1° each so that one full rotation of the knob initiates a turn of 72°. Turning the clock clockwise initiates a right turn; turning the knob counterclockwise initiates a left turn. The actual position of the dial pointer is not important since the yaw channel is self-nulling.

BAR ALT Engage Switch

The BAR ALT ENG switch is of the same type as the ASE ENG SWITCH. When depressed, the BAR ALT ENG indicator light will go on, but does not affect ASE operation since barometric altitude stabilization is not incorporated in this installation.

CHANNEL MONITOR PANEL

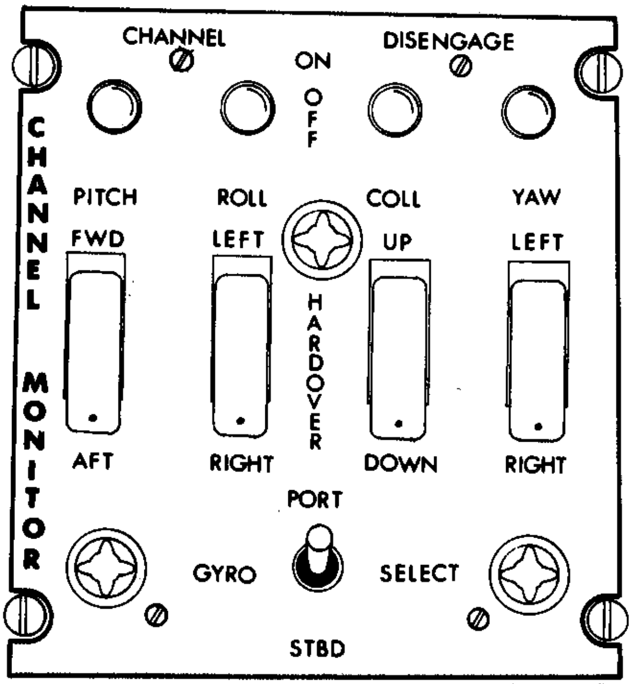
The channel monitor panel (figure 1-23) contains controls which permit disengaging individual channels and introducing test signals. It also provides a means of selecting either vertical gyro as a source of actual pitch and roll attitude signals for the ASE amplifier. The channel monitor panel will be fitted with a removable plastic cover which protects the panel from water. All switches, except the hardover switches may be actuated with the plastic cover in place.

Channel Disengage Switches

The channel disengage switches are in the electrical circuit between the hardover switches and the auxiliary servo flapper valves. Four CHANNEL DISENGAGE switches marked PITCH, ROLL, COLL, and YAW permit disengagement of individual channels. Actuation of the COLL CHANNEL DISENGAGE switch does not affect operation of the ASE.

Hardover Switches

The hardover switches are in the electrical circuit between the ASE amplifier and the channel disengage



S 55541 (R)

Figure 1-23. Channel Monitor Panel

switches. Four HARDOVER switches marked PITCH, ROLL, COLL, and YAW, when actuated, apply hardover signals to the individual channels for testing. Actuation of the COLL HARDOVER switch does not affect operation of the ASE.

WARNING

Do not operate the HARDOVER switches in flight. Operation of one or more HARDOVER switches in flight will cause sudden and violent displacement of the helicopter.

Gyro Select Switch

The GYRO SELECT switch with marked positions PORT and STBD is used to select the reference vertical gyro for the ASE. In PORT position the copilot's vertical gyro provides actual attitude signals for the ASE; in STBD position the pilot's vertical gyro provides the signals.

FLIGHT DIRECTOR ASE MODE

Two OFF flags come into view when the ASE is disengaged or when electrical power to the ASE amplifier fails. When the ASE is engaged, the horizontal and vertical indicator bars indicate the amount and direction of the ASE correction signals fed to the pitch and roll auxiliary servo (flapper) valves. The lower pointer indicates the ASE signal input to the yaw servo valve. The horizontal bar monitors pitch channel correction signals; upward deflection of the bar indicates a nose-down signal and downward deflection of the bar indicates a nose-up signal. The vertical bar monitors roll channel correction signals: left deflection of the bar indicates a left roll signal and right deflection of the bar indicates a right roll signal. The power pointer monitors yaw channel correction signals; left deflection of the pointer indicates a left tail rotor pedal signal and right deflection of the pointer indicates a right tail rotor pedal signal. This indicator is used in flight to monitor ASE functioning and the helicopter's center of gravity. It may also be used during testing to determine whether the various channels are functioning properly. During the ASE ground check the fuselage cannot change attitude to follow cyclic control inputs. Consequently the actual attitude signals to the ASE amplifier remain the same. However the desired attitude signals change as the pilot moves the cyclic and yaw trim knob. Forward cyclic causes the horizontal bar to rise; aft cyclic causes the horizontal bar to fall. Left cyclic causes the vertical bar to move left, right cyclic causes the vertical bar to move right. Counterclockwise rotation of the yaw knob causes the lower pointer to move left. Clockwise rotation of the yaw trim knob causes the lower pointer to move right. Flight director response in flight is opposite to the above, due to the negative stability of the helicopter; for example, shortly after selecting a lower nose attitude with cyclic the horizontal bar will fall because ASE is feeding a nose-up correction signal to the pitch channel auxiliary servo valve.

ASE DISENGAGE SWITCHES

The ASE disengage switches are pushbutton, momentary-type, mounted on each cyclic control (figure 1-19) and are marked AUTO STAB RELEASE. Depressing either switch disengages the ASE which can be observed by the green ASE ENG light going off, lighting of the ASE OFF caution light and the display of OFF flags in the flight directors when ASE mode is selected.

TURN SWITCH

The TURN engage switch is a momentary-type on the pilot's and copilot's cyclic grips. The coordinated turn function is not incorporated in this aircraft.

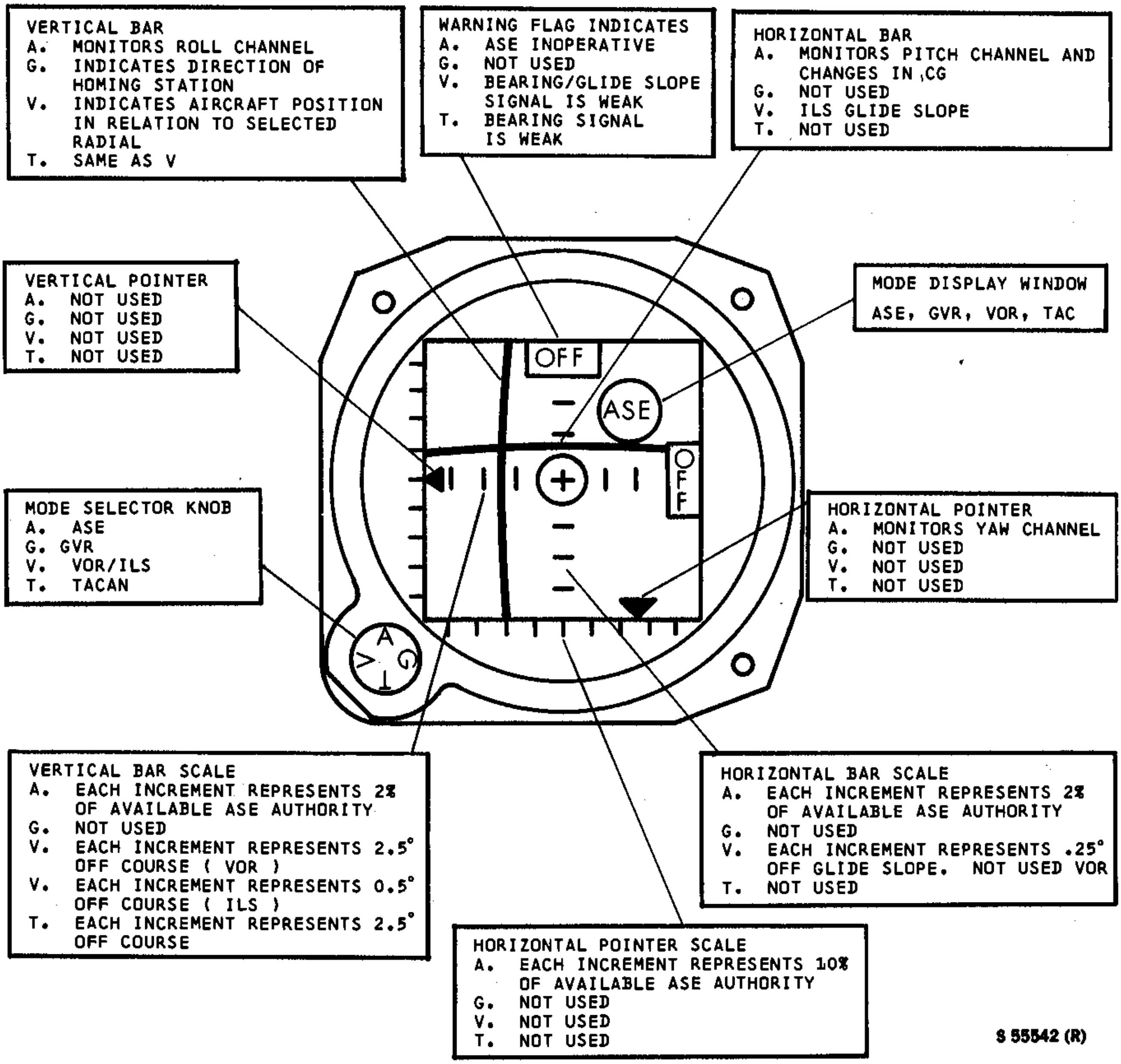


Figure 1-24. Flight Director

BAR ALT OFF SWITCH

Depressing the momentary OFF switch, on the collective, will put off the BAR ALT ENG indication bulb if it is on. This does not effect the operation of the ASE.

INSTRUMENTS

STANDBY COMPASS

A magnetic standby compass is mounted on the windshield center support (7, figure 1-2). The compass indicates helicopter heading in relation to magnetic north pole. Its dial is graduated with cardinal heading indicated by enlarged letters with subordinate heading numerically indicated in 30° units each, with finer linear graduations of 5° each. A compass correction card is provided so that compass readings may be compensated for deviation.

FREE-AIR TEMPERATURE GAGE

A bimetallic free-air temperature gage is in the center

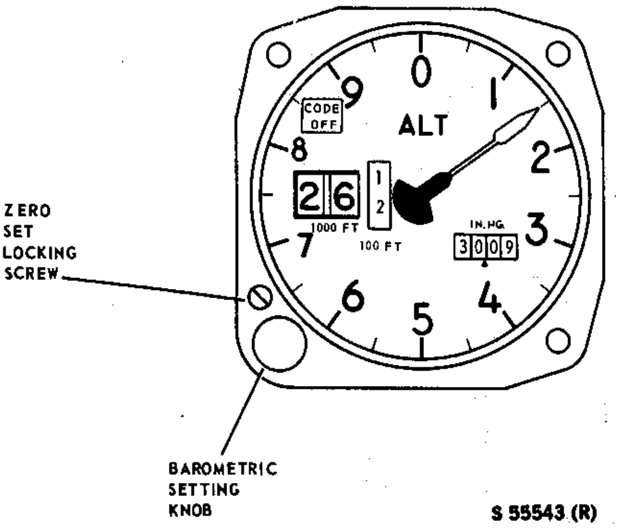


Figure 1-25. Altimeter-Encoder AAU-21/A or AAU-32A

of the windshield. It indicates temperatures from −70°C to +50°C in units of 2° each.

CLOCKS

An 8-day, elapsed-time, 12-hour clock is on both the pilot's and copilot's instrument panel (3 and 34, figure FO-1). The control knob for the elapsed-time mechanism is in the upper right corner. Three successive depressions of this knob start, stop and return to the starting position, the sweep-second and totalizer hands.

PITOT-STATIC SYSTEM

An electrically-heated pitot tube is on the right side of the fuselage above the pilot's compartment. The static ports, on the left and right-hand sides of the fuselage transition section, are connected through tubing to the pilot's and copilot's vertical speed indicators and the altimeters. The pitot and static ports are connected through tubing to airspeed indicators.

PRESSURE ALTIMETERS

Altimeter-Encoder AAU-21/A or AAU-32/A

One altimeter-encoder (figure 1-25) is installed in the pilot's instrument panel. The altimeter-encoder combines a conventional barometric type altimeter, possessing a counter-drum-pointer display, with an altitude reporting encoder in one self-contained unit. The 10,000- and 1,000-foot counters and the 100-foot drum provide a readout of altitude in units of 100 feet, from −1,000 to 38,000 feet. The pointer repeats the indications of the 100-foot drum, and serves both as a vernier for the drum and as a quick indication of the rate and sense of altitude changes. Two methods may be used to read indicated altitude on the counter-drum-pointer altimeter: (1) read the counter-drum window, without reference to the pointer as a readout in thousands and hundreds of feet; or, (2) read the thousands of feet on the two counter indicators, without refering to the drum, and then add the 100-foot pointer indication. The self-contained servo driven encoder provides altitude encoded in 100-foot units for automatic transmission when the AN/APX-99 transponder is interrogated in the altitude mode. The digital output to ground radar is referenced to 29.92 in Hg and not the pilot-selected altimeter setting. In case of power loss to the encoder servo system, a CODE OFF flag appears automatically in a window in the upper left portion of the display, indicating that altitude information is no longer being transmitted to the ground. In this condition, the instrument continues to function as a normal barometric altimeter. The barometric pressure is entered by use of a barometric set knob in the lower left front of the instrument case. The altimeter setting appears on counters in the window at the lower right of the display and has a range of settings from 28.10 to 31.00 in Hg. On those helicopters with an altimeter-encoder AAU-21/A installed, an internal vibrator operates continuously whenever power is supplied to the DC Essential Bus. The vibrator minimizes internal mechanical friction, allowing the instrument to provide a smoother display during changing altitude conditions. Should vibrator failure occur, the altimeter will continue to function pneumatically, but a less-smooth movement of the instrument display will be evident with changes in altitude. The altitude-encoder AAU-32/A does not require an internal vibrator due to its solid-state construction.

Altimeter AAU-24/A

One altimeter (figure 1-26) is installed in the copilot's flight instrument panel. The instrument is similar to the altitude-encoder, except that it does not have an altitude-encoder nor the CODE OFF display mechanism. The indicated altitude on the altimeter is from −1000 to 38,000 feet. The altitude display, altimeter setting, and vibrator considerations described for the altimeter-encoder also apply to the copilot's altimeter.

WARNING

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

NOTE

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

Vertical Speed Indicators (VSI)

Two vertical speed indicators are on the instrument panel (12 and 44, figure FO-1). The indicators give climb or descent speed information on two logarithmic scales. This allows widely spaced markings near zero and a condensed scale at higher vertical speeds. A zero setting screw permits the pointer to be reset to the zero graduation. The indicators are connected to the static pressure line through tubing.

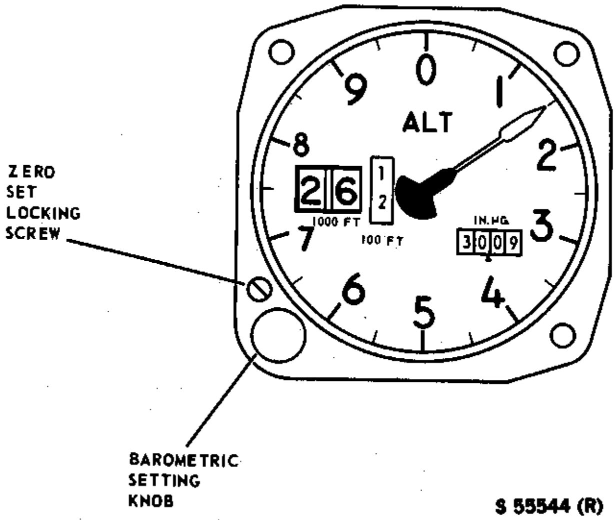


Figure 1-26. Altimeter AAU-24/A

Airspeed Indicators

Two airspeed indicators are on the instrument panel (4 and 36, figure FO-1). The indicators give forward speed information. The indicators are connected to both the pitot pressure system and the static pressure system.

Pitot Heat Switch and Advisory Panel Light

A switch marked PITOT HEAT, with positions ON and OFF, is on the overhead switch panel (figure FO-2). When the switch is placed ON, the pitot head is electrically-heated to prevent formation of ice on or within the tube. The ON position provides power from the DC essential bus to two circuits. One circuit includes a circuit breaker marked PITOT HEAT and the pitot tube heater. The other circuit includes a circuit breaker marked PITOT HEATER under the general heading WARNING LIGHTS and the light capsule marked PITOT HEAT, in the caution advisory panel (figure 1-27). Both circuits pass through a relay which allows the PITOT HEAT advisory light to go on only if the heater is operating. Lighting of the light indicates that the switch is ON and the pitot tube heater is working. Both circuit breakers are on the forward circuit breaker panel.

TURN AND SLIP INDICATORS

Two turn and slip indicators (6 and 38, figure FO-1) are installed on the instrument panel; one in front of the pilot and one in front of the copilot. The indicators visually indicate helicopter rate-of-turn and balanced

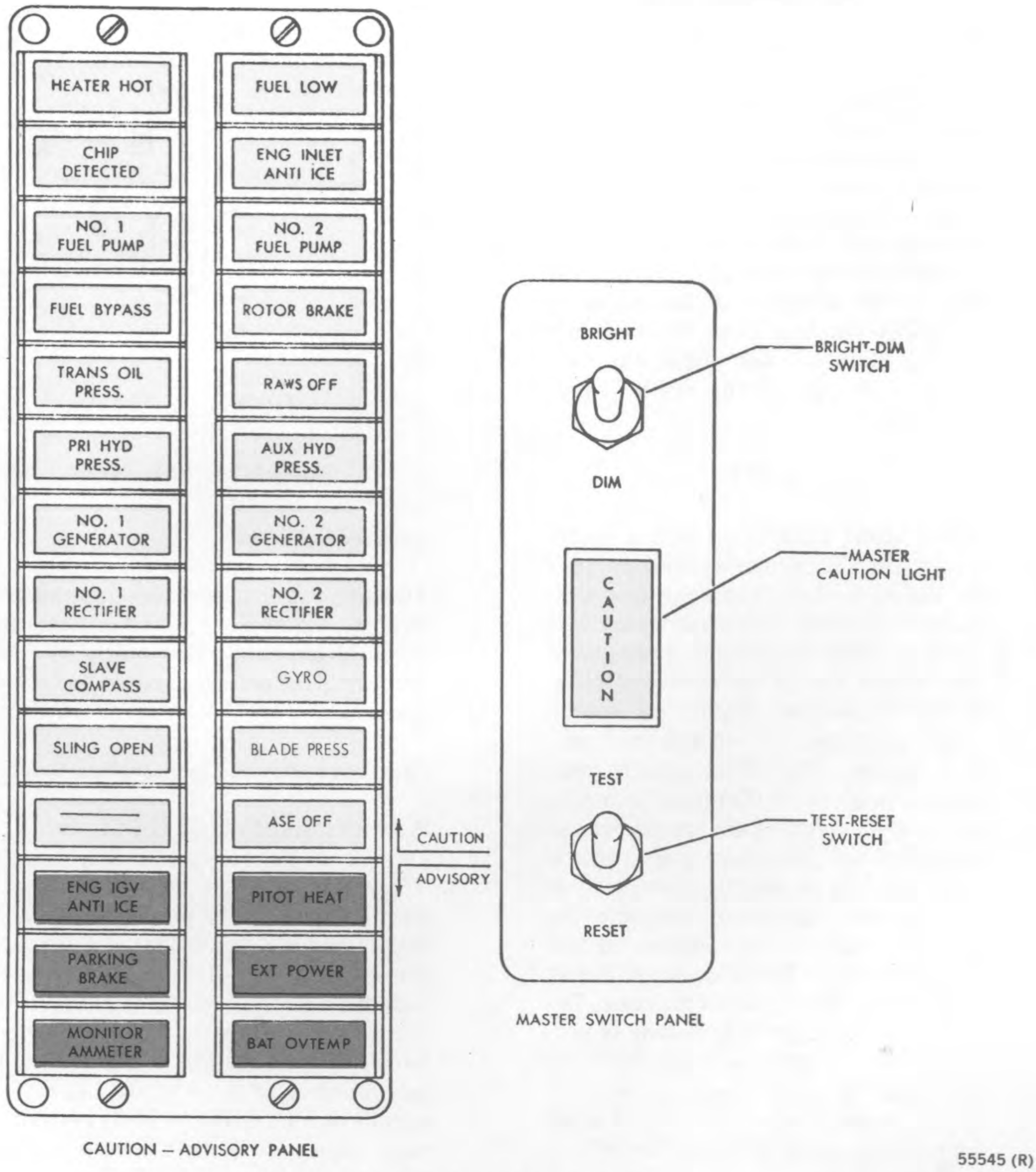


Figure 1-27. Caution-Advisory Panel

flight. Single needle width deflection indicates a 4 minute turn. A standard rate turn will require a two needle width deflection. The gyros of the indicators operate on DC power from the essential bus and are protected by circuit breakers, marked TURN & SLIP, PILOT-COPILOT, on forward circuit breaker panel.

CAUTION ADVISORY PANEL

The HH-52A caution advisory panel, (figure 1-27) to the left center of the instrument panel, is composed of 28 different capsules, 22 amber and 6 green. The amber caution capsules offer information of an emergency nature such as generator failures, low fuel, hydraulic and oil pressure failures, etc. The green advisory capsules provide information of an advisory nature such as external power on, parking brake on, etc. When an irregularity is detected in any system serviced by the caution portion of the caution advisory panel, a brilliant master CAUTION light in the center of the master switch panel will go on. Also, an amber capsule on the caution advisory panel will light, giving a printed statement of the irregularity detected. The capsule and the master CAUTION light will remain on until the difficulty is corrected. Conditions of the aircraft systems which light any of the advisory lights on the caution advisory panel will cause a green printed statement to appear in the capsule but will not light the master warning light. The caution advisory panel capsules are individual, swivel-mounted units containing two bulbs each. Depressing either side of the capsules will turn the capsule 180° so that its aft side is visible, and the face of the capsule is facing into the unit itself. This exposes the bases of the two bulbs within the capsule, which may be removed and replaced as needed. Each caution and advisory capsule has its own operating electrical circuit and receives power through the system it serves.

MASTER SWITCH PANEL

The master switch panel, (figure FO-2) to the right of the caution-advisory panel, contains a test-reset switch, a bright-dim switch, and a master caution light that goes on when an irregularity is indicated by any caution capsule on the caution-advisory panel. The test-reset switch is spring-loaded to the center position with two momentary positions marked TEST and RESET. The pilot may put off the master caution light by depressing the TEST-RESET switch to the RESET position. This may be done even though the caution condition which caused the amber light to go on has not been corrected. This feature is provided so that the pilot's attention may be attracted to any new caution condition which may arise and cause the re-lighting of the master CAUTION light. All lamps, the caution advisory panel and the master CAUTION light, may be tested by placing the spring-loaded TEST-RESET switch to the test position. The bright-dim switch, spring-loaded to the center position, has two momentary positions marked BRIGHT and DIM. Dimming of all lamps on the caution advisory/panel and the master CAUTION light may be done by placing the bright-dim switch in the DIM position. Also dimmed is the indicator light in the controllable landing light master switch. The master switch panel receives power from the DC essential bus and is protected by a circuit breaker on the forward circuit breaker panel under the heading CAUTION PANEL.

Conditions causing the lighting of a caution or advisory light are discussed in Sections I or IV, under the subject heading of the affected aircraft system. Correct pilot response to a caution light is covered in Section III.

LANDING GEAR SYSTEM

The landing gear system consists of two partially retractable main landing gear assemblies, a non-retractable full swiveling tailwheel and an actuating system. The main landing gear retracting system operates on hydraulic pressure from the auxiliary hydraulic system. The necessary electrical power is provided from the DC essential bus through a circuit breaker marked LDG GEAR, on the forward circuit breaker panel.

MAIN LANDING GEAR

The two main landing gear assemblies are attached to the sponsons by oleo struts and support struts, and may be partially retracted into the sponson for water landing. Each main landing gear is equipped with hydraulic brakes. Round inspection windows made of transparent acrylic are on the outboard surface of each sponson strut fairing panel. The inspection windows aid inspection of the shear bolts' indicator tape.

Main Landing Gear Switch

The main landing gear switch, marked LANDING GEAR, with marked positions UP and DOWN, is on the instrument panel (18, figure FO-1). When the main landing gear switch is placed in UP, a control valve is energized and allows the auxiliary hydraulic system pressure to retract the main landing gear. When the main landing gear switch is placed in DOWN, the control valve is de-energized and hydraulic pressure is released. The normal air charge in the oleo strut of each main landing gear, plus weight of the wheels, forces the struts to move to the extended position.

Main Landing Gear Position Indicator

The main landing gear position indicator is on the instrument panel by the main landing gear switch. When the main landing gear is in the retracted position, the indicator shows an UP indication. When the main landing gear is in the extended position during flight, the indicator shows a symbol representing the wheels. While on the ground, a barber pole type indication will be visible. A barber pole indication also shows when the gear are in transit.

TAILWHEEL

The non-retractable tailwheel is underneath the aft section of the hull. The tailwheel is the full-swiveling and self-centering type, and may be mechanically locked in the center (fore-and-aft) position.

Tailwheel Lock Handle

The tailwheel lock handle, marked PULL UP TO LOCK, is at the aft end of the lower radio console. When the handle is pulled up to the LOCKED position, the control cable slackens, allowing the spring-loaded lockpin to engage after the tailwheel centers. When the handle is pushed down to the UNLOCKED position, the control cable pulls the lockpin from the swivel joint, permitting the tailwheel to swivel through 360°. A button in the center of the handle must be pressed to release a rachet-type lock, before the tailwheel lock handle can be pushed down to the unlocked position.

WHEEL BRAKE SYSTEM

The main landing gear wheels are equipped with hydraulic brakes. The self-contained brake system is operated by toe pedals (9, figure 1-2). A parking brake valve and handle permits locking the brakes when the helicopter is parked.

BRAKE PEDALS

Each main landing gear wheel is individually braked by depressing the corresponding brake pedal, mounted on the pilot's tail rotor pedals. The copilot's tail rotor pedals do not have brake pedals.

PARKING BRAKE

The parking brake handle marked PARKING BRAKE is on the left side of the lower radio console (22, figure 1-2). The brakes are locked by depressing the brake pedals pulling the parking brake handle up, and releasing the brake pedals before releasing the parking brake handle. Pressing the right brake pedal releases the parking brakes, causing the parking brake handle to return to the unlocked position. The parking brake handle should be held and lowered gently when the brakes are released.

Parking Brake Advisory Light

A light marked PARKING BRAKE, on the caution advisory panel (figure 1-27), goes on any time the parking brake handle is in the UP position, regardless of whether the brakes are set. The light receives electrical power from the DC Essential bus through a circuit breaker marked PARK BRAKE, on the forward circuit breaker panel.

EMERGENCY EQUIPMENT

LIFE RAFTS

Four one-man life rafts (figure 1-28) designated LR-1 are on the Standard SAR Board. The rafts have a single compartment flotation tube that is inflated by CO2. The floor is non-inflatable. The rafts are equipped with a sea anchor and weathershield, and have a sea anchor and retaining line pocket. The rafts are inflated by pulling the inflation assembly actuating lanyard. After boarding, additional inflation of the LR-1 is possible by use of an oral inflation valve. There are no survival items in the raft. The rafts may be air-dropped to a survivor in the water.

FIRST AID KITS

There are two first aid kits (figure 1-28) carried aboard the helicopter. One is installed on the aft cabin bulkhead, and one on the bulkhead behind the copilot's seat. The kits are mounted to the bulkheads by web straps and snaps. They may be removed from the bulkhead for use. The kits contain adhesive bandages, iodine swabs, ammonia inhalants, compresses, tourniquets, forceps, petrolatum, and scissors.

SURVIVAL RADIOS (AN/PRC-63/90) AND PERSONAL LOCATOR BEACON (URT-33A)

One of either of the above is in a stowage pocket on each SRU-21/P Survival Vest.

CRASH AXE (figure 4-24)

Located on the Standard SAR Board.

CABLE CUTTER (figure 4-24)

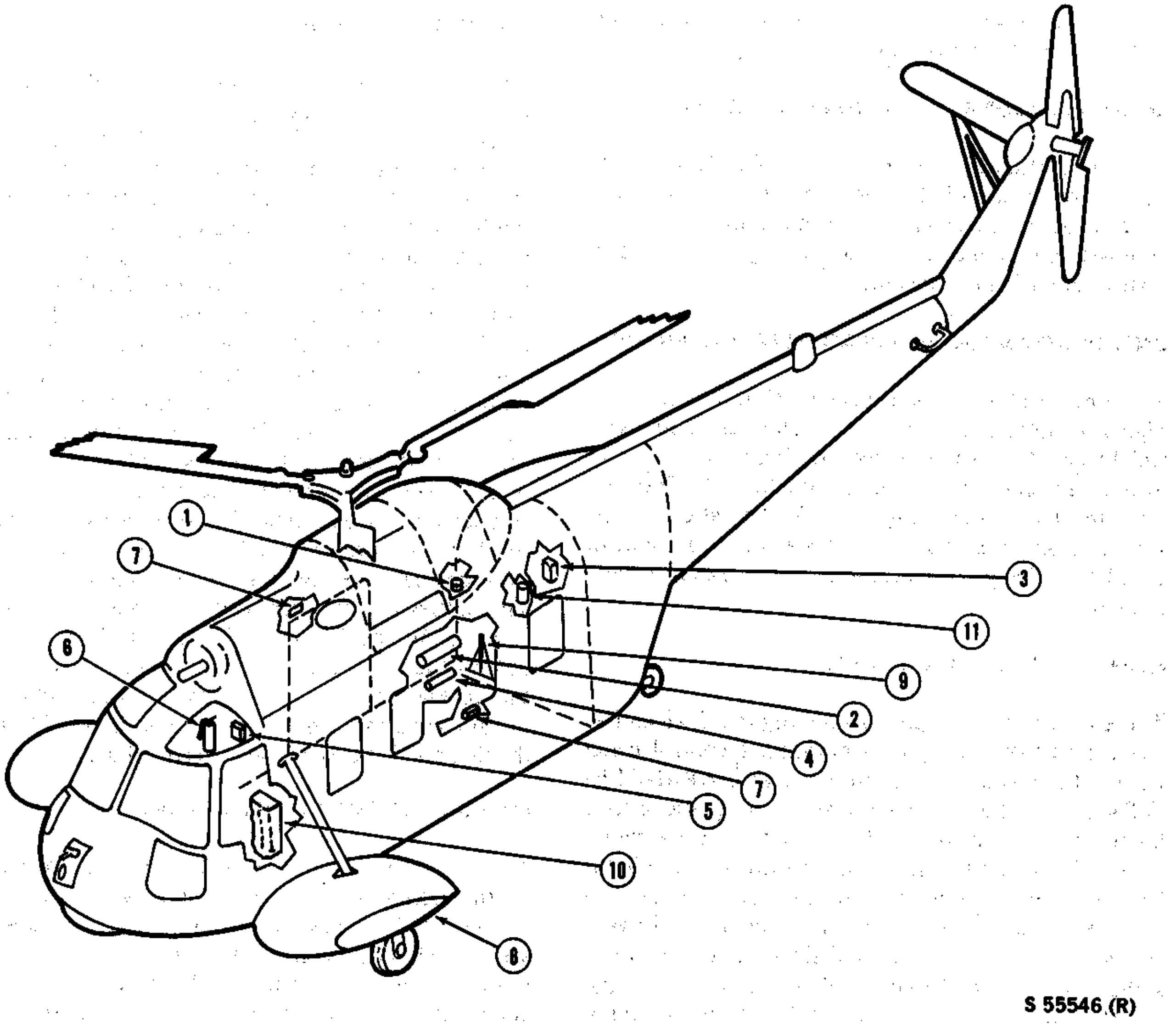
Located on the Standard SAR Board.

HOIST CABLE QUICK SPLICE

A hoist cable quick splice (figure 4-22) is on the Standard SAR Board.

PORTABLE FIRE EXTINGUISHERS

One hand-held portable CO2 fire extinguisher (figure 1-28) is in the cockpit on the bulkhead behind the pilot's seat. A second CO2 fire extinguisher is on the aft bulkhead in the cabin.



1. ENGINE COMPARTMENT FIRE EXTINGUISHER
2. LIFE RAFT LR-1
3. FIRST AID KIT (CABIN)
4. FIRE EXTINGUISHER (CABIN)
5. FIRST AID KIT (COCKPIT)
6. FIRE EXTINGUISHER (COCKPIT)
7. EMERGENCY EXIT LIGHTS
8. AUXILIARY FLOATATION
9. ANCHOR
10. DROGUE
11. UNDERWATER ACOUSTIC LOCATOR BEACON

Figure 1-28. Emergency Equipment

LIFEJACKET STOWAGE

SRU-21/P Survival Vests with LPU-10/P Life Preservers are stowed in the cabin. The location is determined by the operating unit.

COCKPIT EMERGENCY INSTRUMENT LIGHT RHEOSTAT

A rheostat (figure 4-19) marked INST EMER LTS, is on the overhead switch panel. The instrument emergency light rheostat controls the red lamp of the pilot's compartment dome light if an emergency source of instrument lighting is desired.

UNDERWATER ACOUSTIC LOCATOR BEACON (PINGER)

The underwater acoustic beacon (figure 1-28) is a highly reliable, impact-resistant, water-activated, lightweight unit that will enhance locating crashed aircraft in a water environment of any depth to 20,000 feet. This unit has an operating life of 30 days after immersion in fresh or salt water and has a detection range of 2,000 to 4,000 yards, depending upon exposure and sea state.

EMERGENCY EXIT LIGHT SYSTEM

The emergency exit light system (figures 1-28 and 4-19) consists of two light assemblies, a control panel with a three-position switch marked ARM-OFF-DISARM, a relay, and associated wiring and circuit breakers. The control panel is mounted on the pilot's instrument panel to the left of the fire warning light. One emergency exit light is below the port cabin emergency exit hatch and the other is at the upper forward corner of the cabin entrance. They light up the emergency exit release handles. With the switch in the ARM position, the batteries will receive a trickle charge from the DC essential bus. The charging circuit is protected by a circuit breaker marked EMERG EXIT LIGHT under the general headings LIGHTS on the forward circuit breaker panel. Two small lights glow within the lens of the light assemblies when the batteries are being charged. Automatic activation of the emergency exit lights depends on interruption of the charging current. This may occur in two ways: when transmission oil pressure drops to 6-8 psi or less and following the loss of DC essential power. The system is deactivated by moving the switch to the momentary DISARM position then to OFF. The DISARM position does not require a power source. The lights may be removed from their retainers for use as an emergency flashlight by pulling the red PULL EMERGENCY LIGHT handle. This handle then functions as an ON-OFF switch for the light. A full-charged battery will provide sufficient power to operate the light for about 30 minutes.

AUXILIARY FLOTATION SYSTEM

An auxiliary flotation system (figure 1-29) is installed to improve lateral stability in adverse sea conditions when the helicopter is resting on the water with the rotor shut down. The manually-operated system consists of two dual compartment inflatable floats, two air bottles, two gages, two manually-operated valves and an actuating handle with control cables. The floats are made from neoprene-impregrated nylon and are stowed in rubberized nylon enclosures attached to the out-board side of each sponson. Each float is divided into two airtight compartments. Two compressed air bottles, stowed beneath the vertical gyro tilt table in the cabin aft of the copilot, supply air for inflation of the floats; the left bottle inflates the forward compartment of each float while the right bottle inflates the aft compartment of each float. Pressure gages are provided for both air pressure spheres and can be viewed through the protective screen surrounding the bottle. The bottles are fully serviced if the gages indicate between 2650 and 3000 psi. The system is activated by pulling the manual release handle, on the left side of the cockpit entrance, aft of the copilot. Pulling the handle actuates both floats simultaneously. Normally the floats will take 6 to 7 seconds to inflate.

BOW LINE ASSEMBLY (ANCHOR ROPE)

Two bow line assemblies form a bridle around the nose of the helicopter. For stowage, each assembly is secured to the shackle at the bow. The opposite ends are stowed under each cockpit sliding window to a stud mounted on the helicopter. The bow line assembly, (line, thimble, ring, and retrieving line) is secured to the stud with a short length of shock cord to hold the rope assemblies secure and taut. The upper end of the retrieving line is secured to a spring clip. The retrieving line is coiled and lashed to the anchor rope assembly. When deployed, the snap hook on the end of the drogue or anchor line is attached to the thimble only.

PARACHUTE SEA DROGUE

The sea drogue (figure 1-29) consists of a parachute. The risers for the chute are joined at the base to a large drogue line. Attached to the end of the drogue line is a large snap hook which is hooked to the thimble of the bow line during drogue deployment. For stowage, the drogue chute is packed in a protective canvas cover with the drogue line wrapped around it. Also attached to the canvas cover is a rip cord line, with a small red

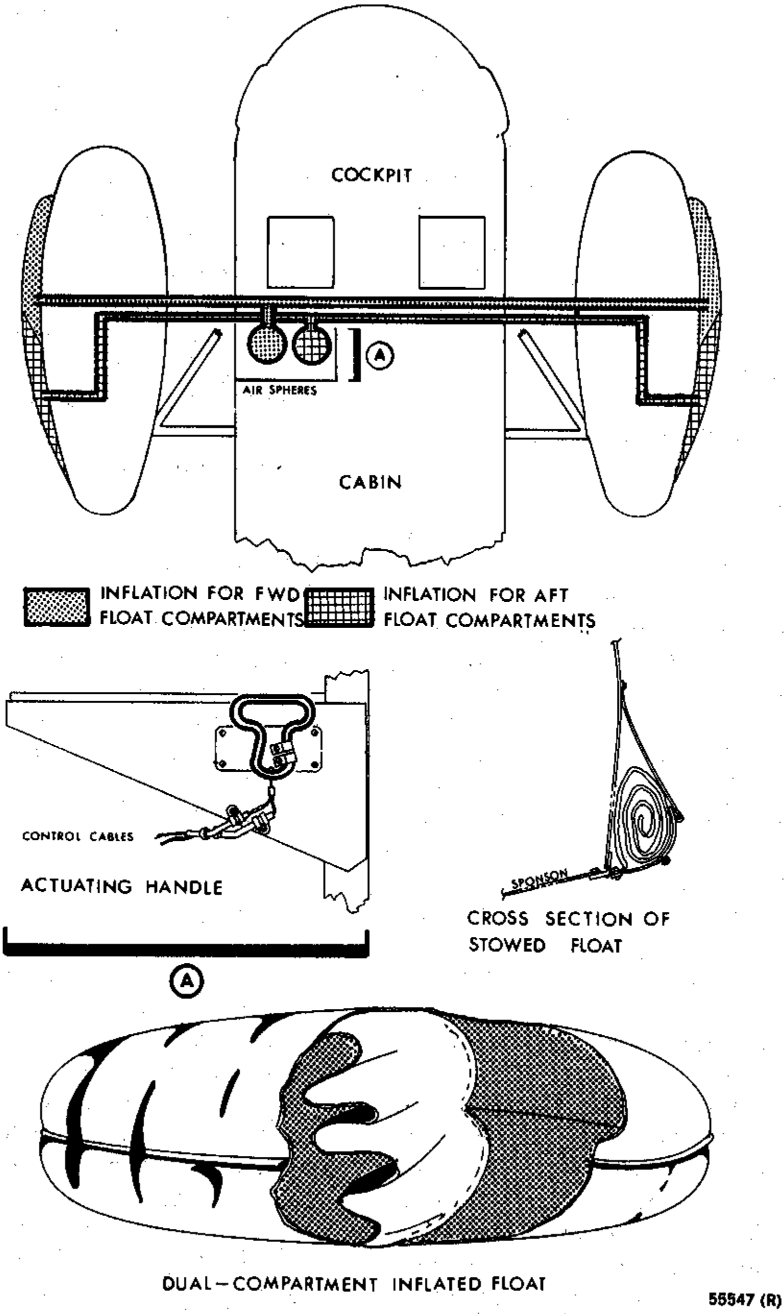


Figure 1-29. Auxiliary Flotation System

snap hook, that allows the chute to be deployed after the pack is in the water. The entire assembly is mounted on the bulkhead in the cockpit behind the copilot seat.

DANFORTH ANCHOR

The helicopter is equipped with a shallow water anchor, of Danforth design (figure 1-29), with 150 feet of anchor line and a snap hook. The anchor is stowed on the aft cabin bulkhead in a protective canvas bag. Use of the Danforth anchor in water depth exceeding the length of the anchor line makes the anchor ineffective. Should the anchor drag, its effect will be similar to that of the sea drogue and should tend to hold helicopter into the wind and waves. To be most effective in stopping all drift, the anchor line should be at least 6 times the depth of the water.

ENGINE COMPARTMENT FIRE DETECTOR SYSTEM

The engine compartment fire detector system visually indicates a fire or an overheat condition within the engine compartment. The system consists of four sensing elements, a control unit, three warning lights, a test switch, and three circuit breakers. The engine compartment fire detector system operates on 115 volts from the AC essential bus or the ground inverter when energized. A circuit breaker marked FIRE DET located on the forward circuit breaker panel protects the system. The fire detector system's four sensing elements, within the engine compartment, actuate the control unit in the event of an overheat condition or fire. When actuated, the control unit lights the engine fire warning light on the instrument panel and two engine fire warning lights in the emergency fuel shutoff fire extinguisher arm T-handle.

Engine Compartment Fire Detector Sensing Elements

Four sensing elements in the engine compartment are connected in series to the control unit. The elements consist of Inconel tubing with a nickel wire center conductor. The hermetically sealed tubing contains an insulation and a special impregnation of a selected inorganic salt compound. When an overheat condition or fire occurs, the resistance of the inorganic salt drops sharply, changing the impedance. The change is sensed by the control unit which transmits the signal to the engine fire detector warning lights. The elements in this system have a critical temperature of 575°F (301.6°C). The sensing element could be broken into two pieces and both pieces would still be able to sense an overheat, providing that there is no shorting out of the broken ends of either piece. If a broken end is shorted out, the fire warning lights will go on. If neither end is shorted out, the sensors will detect an overheat, but the fire warning lights will not go on when the engine fire detector warning test switch is depressed.

Engine Compartment Fire Detector Warning Lights

A red, press-to-test type warning light, (35, figure FO-1) marked "FIRE WARN" is installed on the pilot's side of the instrument panel and two additional engine fire warning lights are installed in either end of the T-handle (figure 1-16). The lights are put on by the control unit in case of overheat or fire within the engine compartment. When testing the panel light, current is drawn from the DC essential bus. The press-to-test circuit is protected by a circuit breaker marked FIRE DET TEST under the general heading ENGINE on the forward circuit breaker panel.

Engine Compartment Fire Detector Warning Light Test Switch

A momentary push button switch (35, figure FO-1) marked FIRE TEST, on the instrument panel next to the engine fire warning light, provides a means for testing the engine compartment fire detector system. When the switch is closed, a relay in the control unit is actuated which causes a condition similar to fire or overheat. The warning lights will go on with this test. The test switch circuit is powered by the DC essential bus and is protected by a circuit breaker marked FIRE DET TEST under the general heading ENGINE.

ENGINE FIRE EXTINGUISHING SYSTEM

The engine fire extinguisher system provides a means of extinguishing fires in the engine compartment. The system consists of a fire extinguisher container, a T-handle marked FUEL ON-FUEL OFF-FIRE EXT ARMED, two microswitches, a fire extinguisher switch, a circuit breaker, a forked engine compartment discharge tube, an overboard discharge tube, and a thermal discharge indicator. With DC essential power available, when the T-handle (figure 1-16) is pulled to the FIRE EXT ARMED position and the fire extinguishing switch is depressed, a liquid extinguishing agent is discharged from the container through the engine compartment discharge tubes and vaporizes.

Fire Extinguisher Container

The fire extinguisher container is charged with 2.5 pounds of bromotrifluoromethane (CF_3BR), and is pressurized with nitrogen (N_2) to 350 psi at 21.1°C.

The container is on the left side of the transmission deck, just aft of the transmission. A pressure gage is secured to the lower surface of the container. A safety outlet on the lower surface of the container contains a fusible plug which provides for release of the contents when the internal pressure becomes excessive due to high temperature. The contents are released through the overboard discharge tube and thermal discharge indicator, when the ambient temperature reaches 208° to 220°F (97.8° to 104.4°C). The extinguishing agent is retained by a frangible disc within the neck of the container.

WARNING

Bromotrifluoromethane (CF_3BR) is very volatile but is not easily detected by odor. It is non-toxic and can be considered to be about the same as other freons and carbon dioxide, causing danger to personnel primarily by reduction of oxygen. The liquid should not be allowed to contact the skin as it may cause frostbite or low temperature burns because of its very low boiling point.

Thermal Discharge Indicator

The thermal discharge indicator consists of a red disc and retaining ring and is outside, just forward of the escape hatch, on the left side of the helicopter. The thermal discharge indicator provides an immediate visual check on whether or not the container has discharged due to abnormally high container pressure.

Fire Extinguisher Arm T-handle

(See Section I, **FUEL SHUTOFF VALVE—T HANDLE**)

The T-handle, when moved to the FIRE EXT ARMED position actuates two microswitches simultaneously. One microswitch arms the fire extinguisher system. The other microswitch closes the engine cowling shutters.

Engine Compartment Fire Extinguisher Switch

This switch is on the overhead switch panel (figure FO-2) and is labeled FIRE EXT. When the fire extinguisher system is armed, by placing the T-handle in the FIRE EXT ARMED position, and the switch is depressed, the contents of the fire extinguisher container are discharged through the forked engine compartment discharge tube. Power from the DC essential bus fires an explosive cartridge which ruptures a disc in the fire extinguisher container discharger valve, releasing the fire extinguishing agent. The firing circuit is protected by a circuit breaker marked FIRE DET EXT under the general heading ENGINE.

Engine Cowling Fire Shutter System

The engine cowling fire shutter system provides a means of closing the engine cowling shutters instantly, thereby confining a fire to the area within the engine compartment. The system consists of four shutter assemblies on the right engine cowling (figure 1-30), a shutter control solenoid, and a T-handle microswitch. Placing the T-handle in the FIRE EXT ARMED position closes the microswitch and allows power from the DC essential bus to actuate the shutter control solenoid. Solenoid actuation allows the shutter assemblies to close. The shutter control solenoid circuit is protected by a circuit breaker marked SHUTTER CONT on the forward circuit breaker panel.

EMERGENCY EXITS

For emergency routes of escapes and exits, see figure 1-31.

PILOTS' COCKPIT SLIDING WINDOWS

The pilots' cockpit sliding windows are normally opened or closed by actuating the handle on the bottom of each window. The windows will lock in any detent position when the handle is released. The manual emergency release handles, marked EMER EXIT PULL, are on the lower edge of each window inside the cockpit. The windows can be jettisoned outward and downward by pulling the release handle aft and pushing out the window. The windows can also be released from the outside by turning the handle marked EMERGENCY RESCUE-BUTTON TURN HANDLE PULL WINDOW OUT.

CABIN DOOR EMERGENCY EXIT

The cabin door can be jettisoned for emergency exit by pulling down on the handle marked EXIT RELEASE-TURN, which is on forward upper corner of cabin door. When proper conditions exist this handle is lit by an emergency exit light. A similar handle is provided to open cabin door from outside of helicopter. An orange-yellow stripe is painted on the guard assembly of the inside door handle to indicate when the emergency handle is aligned and locked. The guard lever is wired with breakaway copper wire to prevent accidental release of the cabin door.

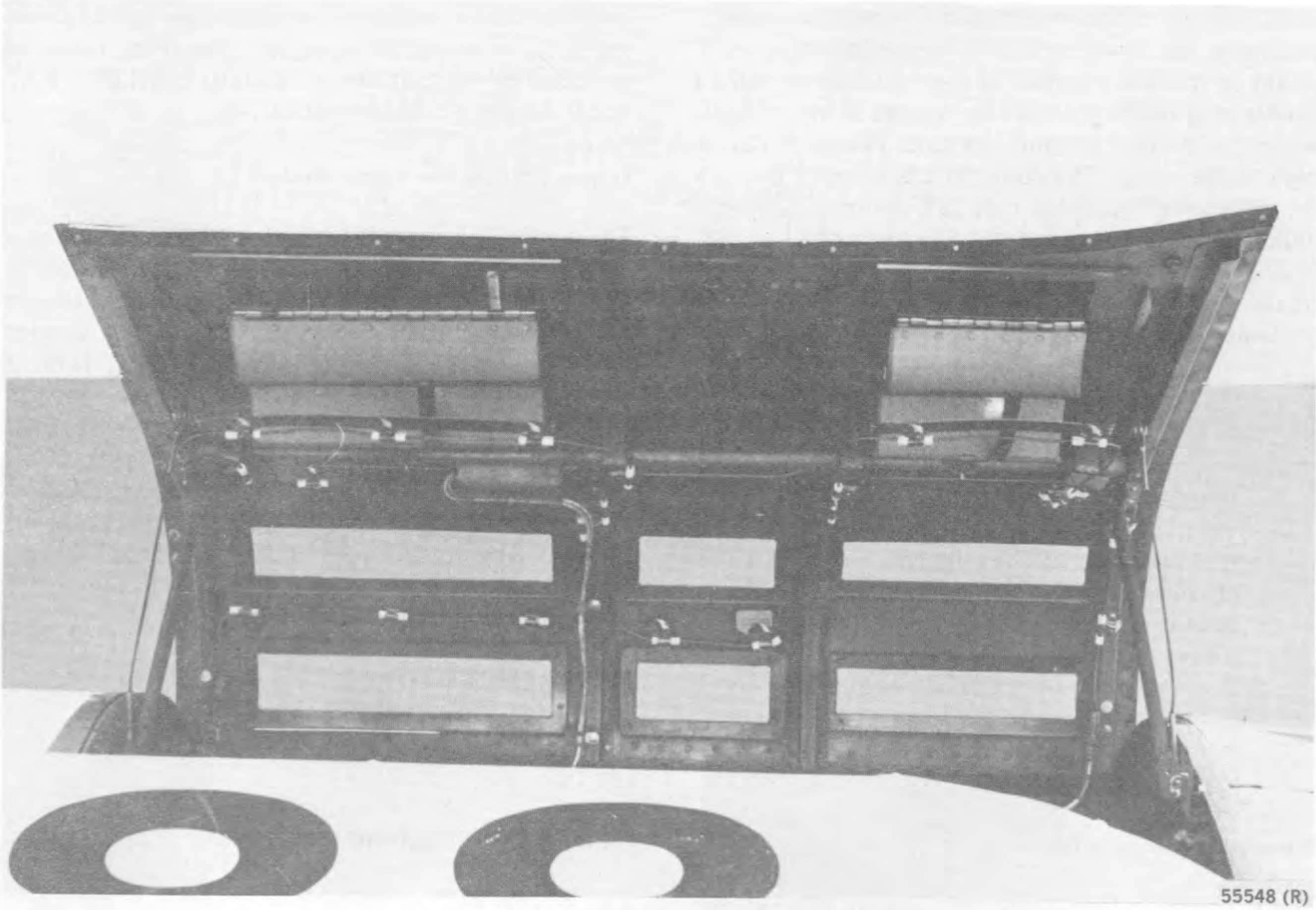


Figure 1-30. Engine-Cowling Fire Shutters

CABIN EMERGENCY HATCH

The panel surrounding the next to last window on left side of cabin can be jettisoned to provide a cabin emergency exit or entrance. An emergency release handle, marked EMERGENCY EXIT-TURN, is on forward lower corner of the panel. To jettison the cabin emergency hatch, the release handle is turned in direction of the arrow and the hatch is pushed out. A similar release handle is provided to open the hatch from outside the helicopter.

CABIN COMPARTMENT WINDOWS

The four remaining cabin windows may be pushed out to provide emergency exits. EMERGENCY EXIT, PUSH OUT WINDOW is stenciled above each window. Each window has an externally located pull tab, marked PULL TAB EXIT RELEASE. Pulling the tab pulls the locking strip out of the rubber seal surrounding the window to aid window removal.

PILOT'S AND COPILOT'S SEATS

The pilot's and copilot's seats are track-mounted in the cockpit. The pilot's seat is on the right. The track-mounted seats are designed to accommodate a back pack parachute, pararaft and seat pan if so desired. Standard configuration prescribes outfitting with custom made seat cushions. Both seats are interchangable and have an approximate 4-inch range for fore-and-aft and height adjustment.

SEAT HEIGHT ADJUSTMENT LEVER

The seat height adjustment levers (14 and 23, figure 1-2) are the rear levers at the right of the pilot's and copilot's seat. The spring loaded levers are pulled up to release the height adjustment lockpins.

SEAT FORE-AND-AFT ADJUSTMENT LEVER

The fore-and-aft adjustment levers (13 and 24, fig-

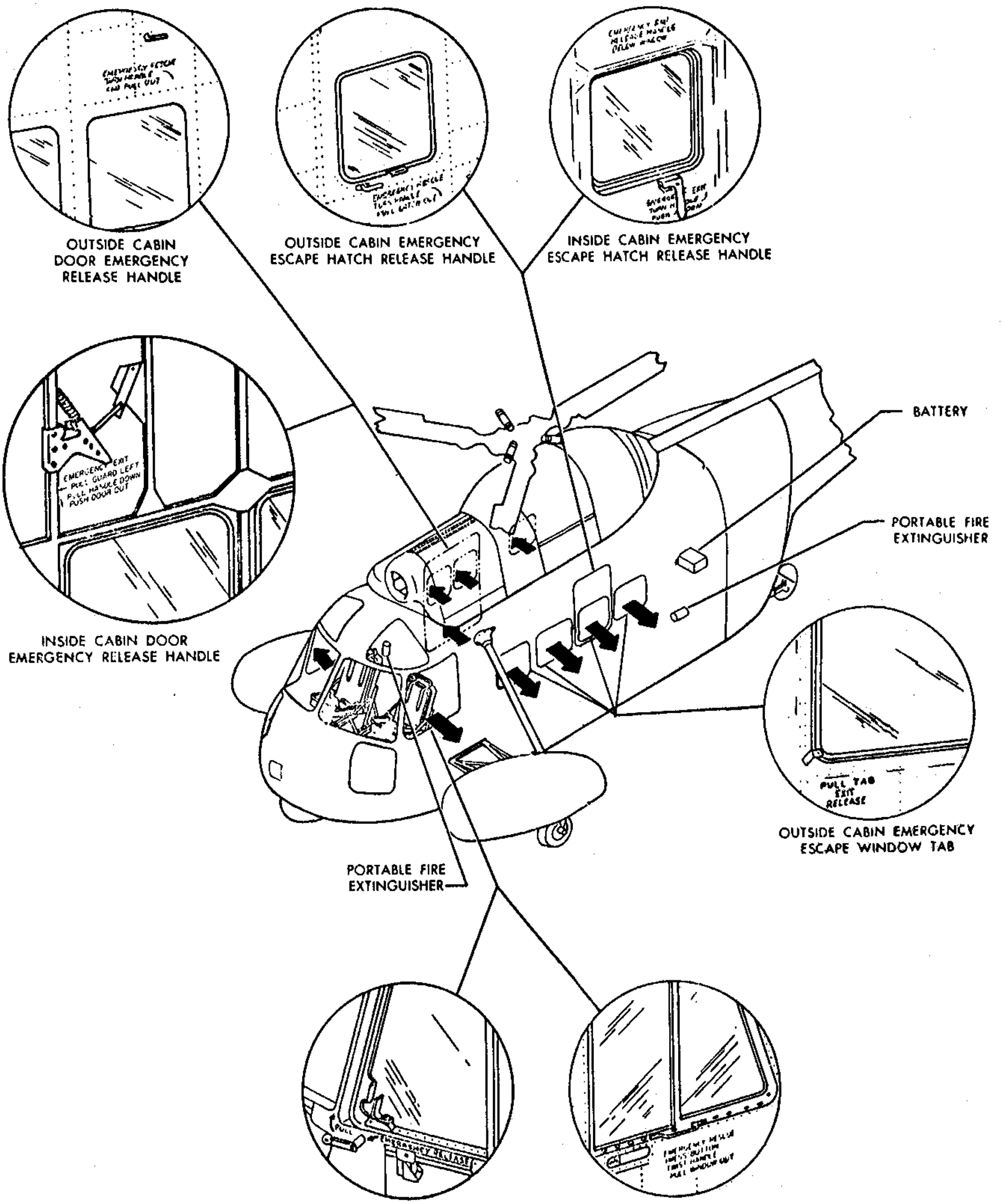


Figure 1-31. Emergency Entrances and Exits

ure 1-2) are the front levers on the right side of the pilot's and copilot's seats. The spring loaded levers are pulled up to release the fore-and-aft seat adjustment lockpins.

SHOULDER HARNESS LOCK LEVER

A two-position shoulder harness inertia reel lock lever (18 and 28, figure 1-2) is at the left side of each seat. When the lever is in the unlocked (aft) position, the shoulder harness cable will extend to allow the occupant to lean forward; however, the inertia reel will automatically lock if an impact force between two and three Gs is encountered. When this occurs, the inertia reel will remain locked until the lever is cycled. When the lever is placed in the locked, forward position, the shoulder harness cable is locked so that the occupant is prevented from leaning forward. The lock position is used to provide an added safety precaution over that of automatic lock on the inertia reel.