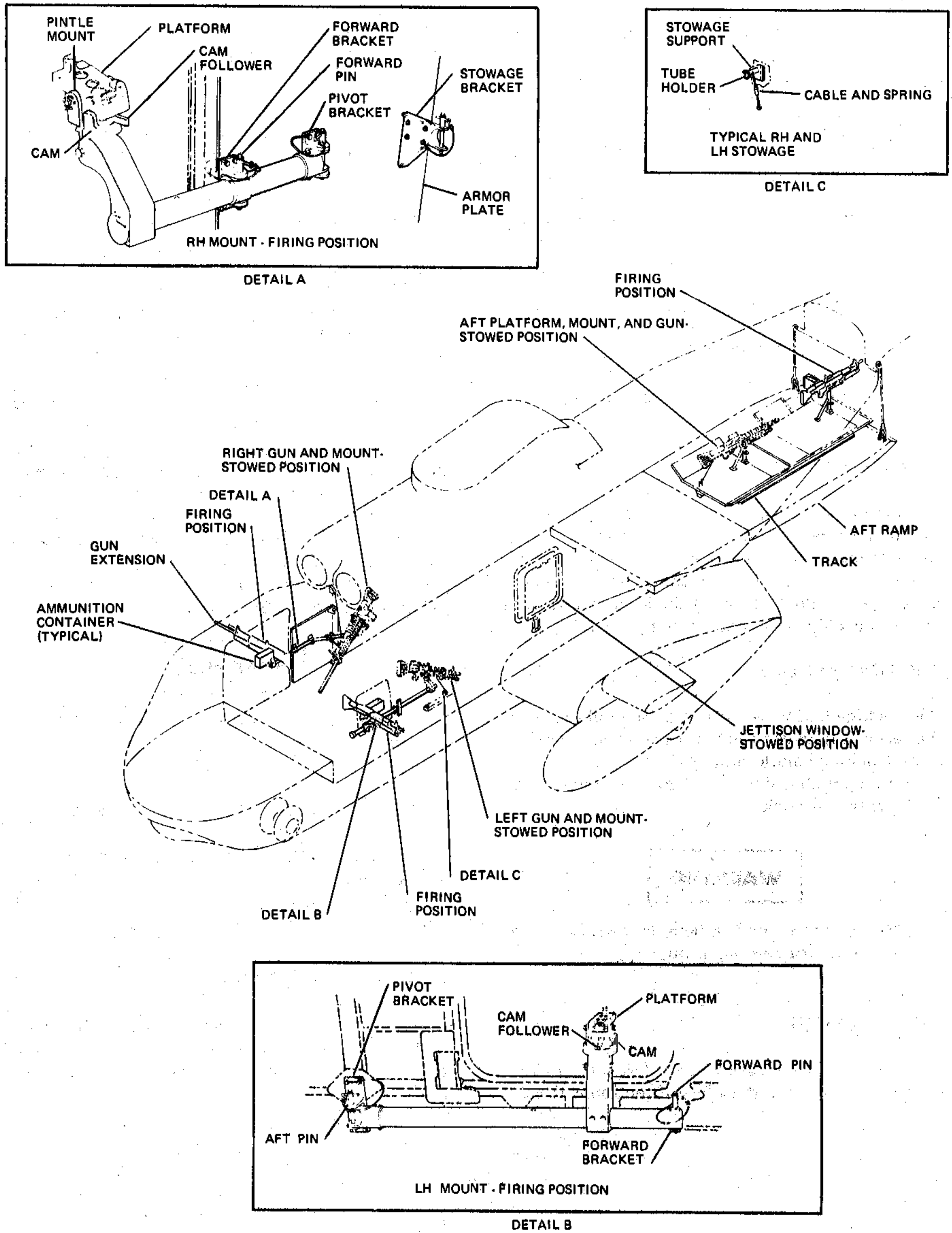


Figure 4-42. Armement System (HH-3E 24▸ 25)

platform provides protection from small arms fire and support for the aft gun mount (see figure 4-42).

LATCH LEVER.

The latch lever actuates the cover latch, which is spring-loaded, and is located at the right rear end of the feed cover. The function of the latch is to secure the cover in the closed position. When the lever is vertical, the cover is locked closed. Turning the lever to the horizontal position unlocks the cover (see figure 4-41).

CAUTION

Do not turn the latch lever more than required to unlock the cover, as damage to the latch spring will result.

BARREL LOCK LEVER.

The barrel lock lever is located on the right front end of the receiver. This lever is secured to the barrel locking shaft and rotates the shaft to lock and unlock the barrel. When the lever is vertical, the barrel is unlocked. When the lever is horizontal, the barrel is locked in place (see figure 4-41).

COCKING HANDLE.

The cocking handle is located on the right side of the receiver between the cover and trigger mechanism. Cocking handle function is to charge the weapon manually. When the handle is pulled to the rear, the bolt is cocked (see figure 4-41).

WARNING

Before firing, the handle must be returned to the forward position.

SAFETY LEVER.

The safety lever is located on the left side of the trigger mechanism. Safety lever function is to prevent the weapon from being fired accidentally. The safety has two marked positions: F (Firing) and S (Safety) (see figure 4-41).

TRIGGER.

The trigger is located below the receiver directly under the feedway. Trigger function is to control firing of the weapon.

GUN STATIONS.

The gun stations are at the personnel door, jettisonable window, and aft ramp. The forward guns are supported on swing-away mounts, providing fields of fire of 70 degrees forward, 50 degrees aft, 65 degrees in depression, and from five to nine degrees in elevation, which is decreased to —30 degrees as the guns are moved toward the auxiliary fuel tank area. The aft gun is supported on a mount and sliding platform attached to tracks on the aft ramp. The tracks permit the platform and mount to be moved into the firing position. This mount provides a rearward field of fire of 80 degrees to each side, 65 degrees in depression, and five degrees in elevation when fired under the pylon, increased to 20 degrees after clearing the pylon. For fields of fire, see figure 4-44.

WARNING

During gunnery missions, the personnel harness for the gun stations will be worn and connected to the rings and supports.

AMMUNITION STORAGE CONTAINERS.

The containers attached to each gun have a capacity of 200 rounds of 7.62MM ammunition, and the containers mounted on the floor each have a capacity of 750 rounds of 7.62MM ammunition.

ARMOR PROTECTION.

The pilot, copilot, crewmen, and vulnerable components of the helicopter are protected from small arms fire by titanium armor plating (see figure 4-45).

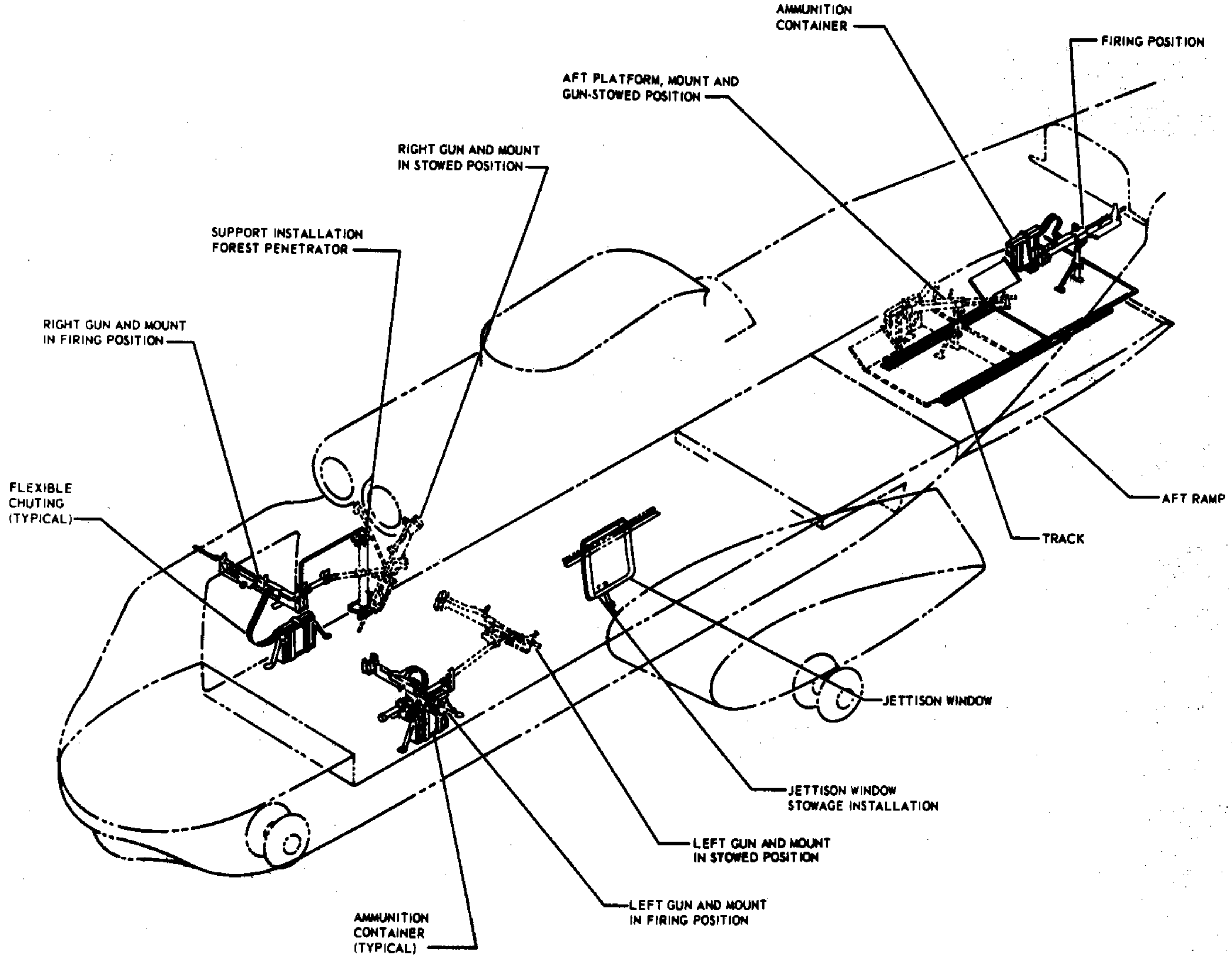


Figure 4-43. Armament System (HH-3E)

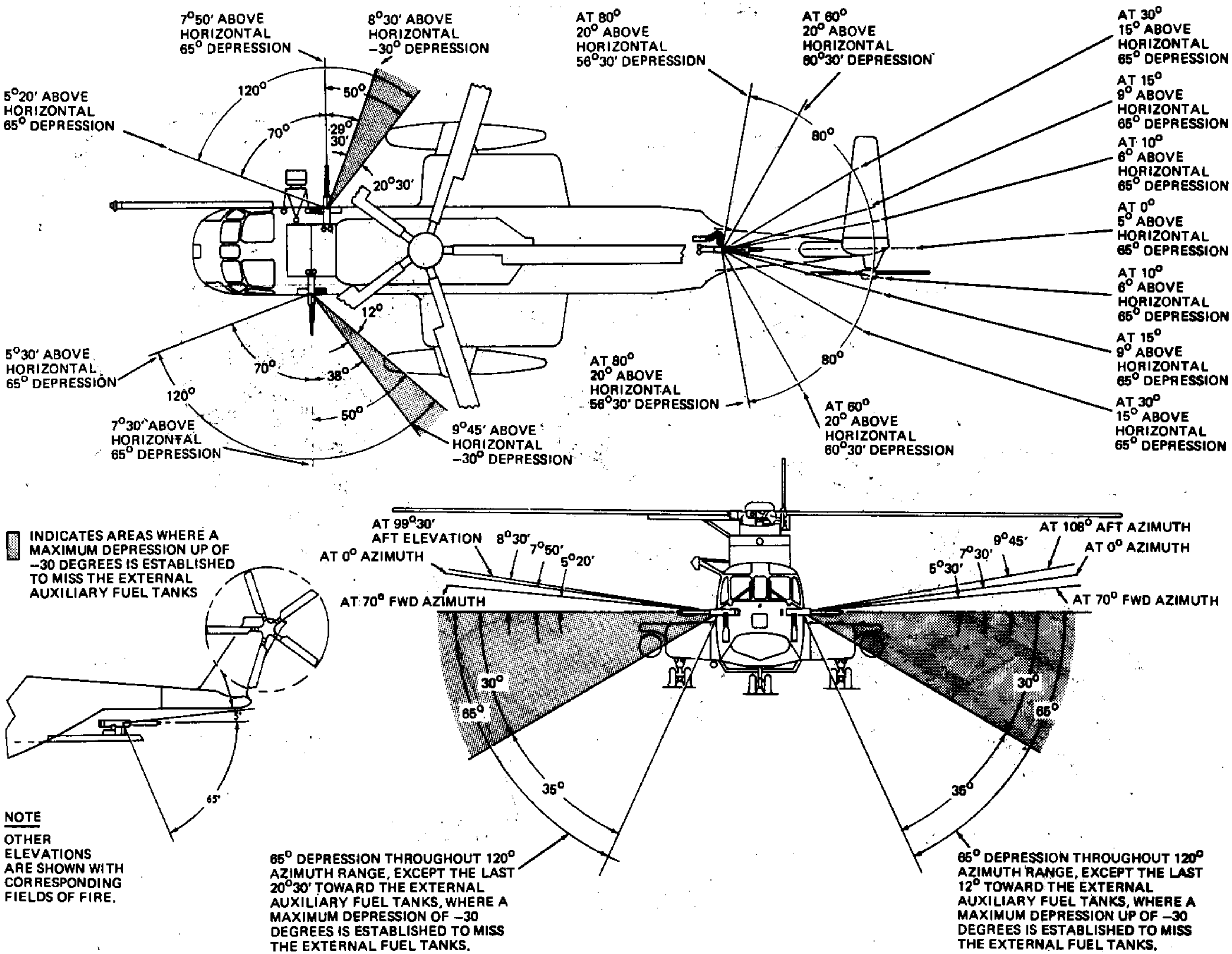


Figure 4-44. Armament Fields of Fire

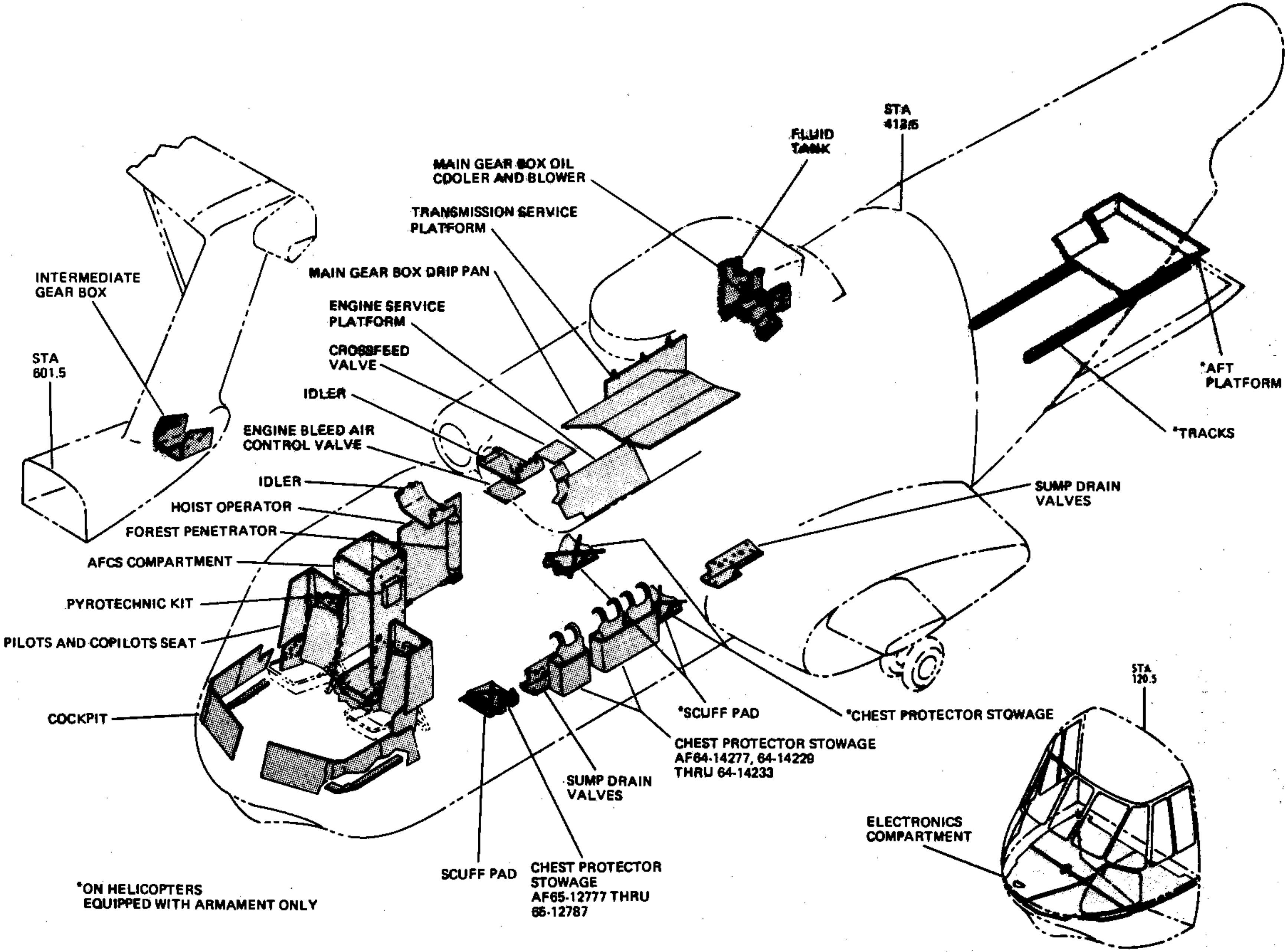


Figure 4-45. Helicopter Armor Protection

KB-18 STRIKE CAMERAS.

Two KB-18 strike cameras are installed on a limited number of helicopters, modified by T.O. 1H-3-577, to give a photographic capability to the mission. One camera is a vertical viewing camera and the other is an aft viewing camera.

Operating Controls.

The cameras are operated by a camera power control panel, located on the lower right side of the cockpit console, an ordnance release panel, located on the lower center of the cockpit console, and two camera operating control panels, located in the middle of the cargo compartment on the right side. The camera power control panel contains two push-to-test CAMERA-ON lights to indicate camera operation, an ON/OFF power switch to supply camera power, and two ON/OFF operate switches for individual operation of the cameras. The ordnance release panel contains a toggle switch with marked positions ARM and SAFE, a pushbutton switch with a RELEASE position, and a frame ID pushbutton switch for frame marking identification. The camera operating control panel has a frame-per-second control knob with positions 4, 2, and 1, and an overrun control knob with positions 1, 2, 3, 4, 5, and 6. The vertical camera has an automatic and manual mode of operation. The automatic mode is controlled by the ordnance release switch and the manual mode is controlled by the operate switch on the camera power control panel. The aft camera has only a manual mode of operation controlled by the operate switch on the camera power control panel.

Operating Procedures.

BEFORE TAKEOFF.

1. Vertical camera operating control panel.
   a. Frame/sec. control knob — 4.
   b. Overrun control knob — 6.
2. Aft camera operating control panel.
   a. Frame/sec. control knob — 1.
   b. Overrun control knob — 2.
3. Camera power switch — ON.
4. Camera on lights — DEPRESS.
   Press to test for light operation.

INFLIGHT.

For manual operation (both cameras).

1. AFT or VERT operate switch — ON.
   When thirty seconds from target, place operate switch ON.
2. AFT or VERT operate switch — OFF, AFTER TARGET RUN.
   For automatic operation on (vertical camera only).

1. Ordnance release button — DEPRESS.
2. Frame ID button — DEPRESS.
   (Hold momentarily when frame identification marking is depressed; depress the frame ID button simultaneously with initial depression of the ordnance release button and hold for at least one second.)

BEFORE LEAVING THE HELICOPTER.

1. Check that the camera power and the AFT and VERT operate switches are in the OFF position.

NOTE

When the KB-18 camera has used its film supply, it will automatically cease operating; however, power must be turned off before removing film.

FLARE EJECTOR SET AN/ALE-20(V).

The AN/ALE-20(V) flare ejector system (figure 4-46) installed on helicopters modified by T.O. 1H-3-646 is an airborne countermeasure system designed to deceive infrared guidance systems used in certain types of missiles. Deception is accomplished by releasing flares having an infrared energy component large in comparison to the helicopters infrared output so that the missile will fly to the flare instead of the helicopters. The system consists of one power panel, one flare programming control panel, five remote release panels, one flare arming switch, one junction box, and two identical stepping switches with companion flare ejector cases. The AN/ALE-20(V) flare ejector system provides a

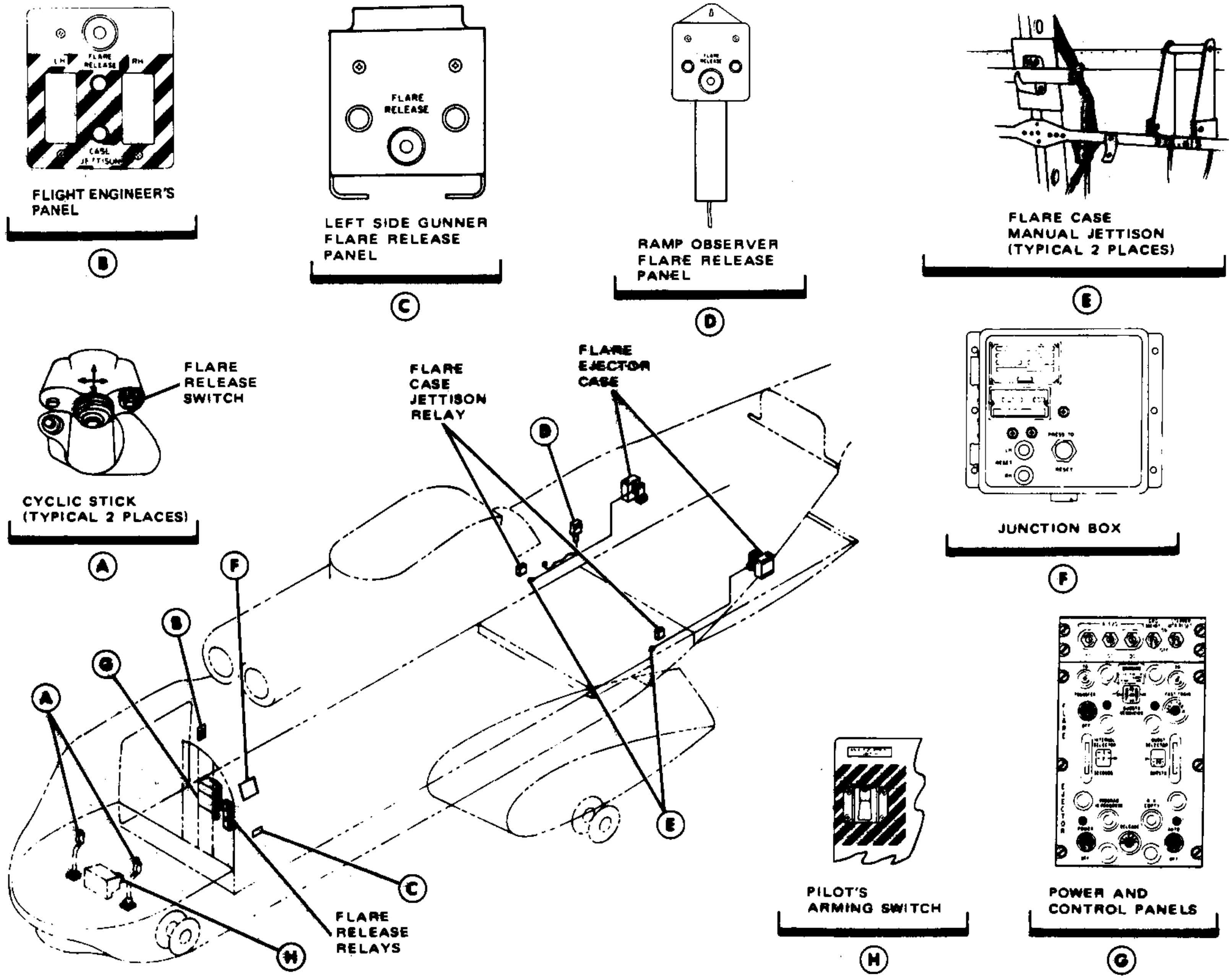


Figure 4-46. Flare Ejector System

means of selecting the number of flares to be released, timing the interval between bursts, and igniting and releasing the flares from the helicopter. The AN/ALA-17 flare set is used with the system and provides 16 single flare bursts per system with a maximum of 32 single flare bursts per system with two sets of ejector cases installed in the helicopter. The flares may be programmed to release in bursts of one, two, or three flares at intervals of 2 to 20 seconds between bursts. The flares may also be released at a rate of one flare every 65 milliseconds by depressing the FAST TRAIN switch and all flares may be released in less than 3 seconds. The system electrical dc power is wired through the helicopter squat switch and prevents inadvertent releasing of flares when the helicopter is on the ground. The system is provided with a GRD MAINT switch located above the flare programming control panel. The system is powered by 28-volt dc power from the essential dc bus through a 15-ampere circuit breaker and a 10-ampere circuit breaker, and 115-volt ac power from the essential ac bus through a 5-ampere circuit breaker. The pilot's flare arming switch must be on to enable ejector set operation.

WARNING

> Under normal operating conditions, during inflight refueling or fuel dumping, the AN/ALE-20 flare ejector system will be dearmed at the PILOT'S AN/ALE-20 ARMING SWITCH.

The ignition of a flare within the loading case presents an extreme fire hazard to the helicopter. The following safety features provide for jettisoning the flare case should such ignition occur. A heat sensor within each case will initiate automatic firing of explosive bolts holding the case when temperature in the case rises above the preset limit. The same explosive bolts can be electrically fired by engaging either the RH CASE JETTISON switch or the LH CASE JETTISON switch on the flight engineer's panel at the right side flare release position. If the explosive bolts fail to jettison the case, the backup manual system can be used. The manual jettison device pulls pins holding the case when the manual release lever, just forward of the ramp, is pulled.

FLARE PROGRAMMING CONTROL PANEL.

(Figure 4-47.)

The flare programming control panel is located on the forward cabin bulkhead. The control panel supplies the junction box with command pulses at the right times and rates. The panel contains power and automatic signal switches, transfer switch, release switch, fast train switch, flares-per-burst switch, program-in-progress light, RH empty light, bursts-remaining counter, burst interval selector, and a burst selector. Flare releasing may be controlled from the flare programming control panel or any of the five remote release positions.

Power and Automatic Signal Switches.

The power switch is a two-position (POWER and OFF) toggle type switch that controls 28-volt dc and 115-volt ac, 400-Hz power to the equipment. The auto switch is a two-position (AUTO and OFF) toggle type switch that allows the system to accept a release signal from a remote signal source, such as a remote firing panel.

Transfer Switch.

The transfer switch is a two-position (TRANSFER and OFF) toggle type switch that controls the operation of the relay in the junction box. The purpose of the switch in the TRANSFER position is to transfer electrical power to the right bank of flares should a malfunction jettison occur in the left bank.

Release Switch.

The release switch is a momentary-contact pushbutton switch, placarded RELEASE. The switch, when actuated, initiates the firing program established by other control panel settings.

Fast Trein Switch.

The fast train switch is a momentary-contact pushbutton switch, placarded FAST TRAIN. When the switch is actuated, one flare is released every 65 ± 10 milliseconds until cases are empty.

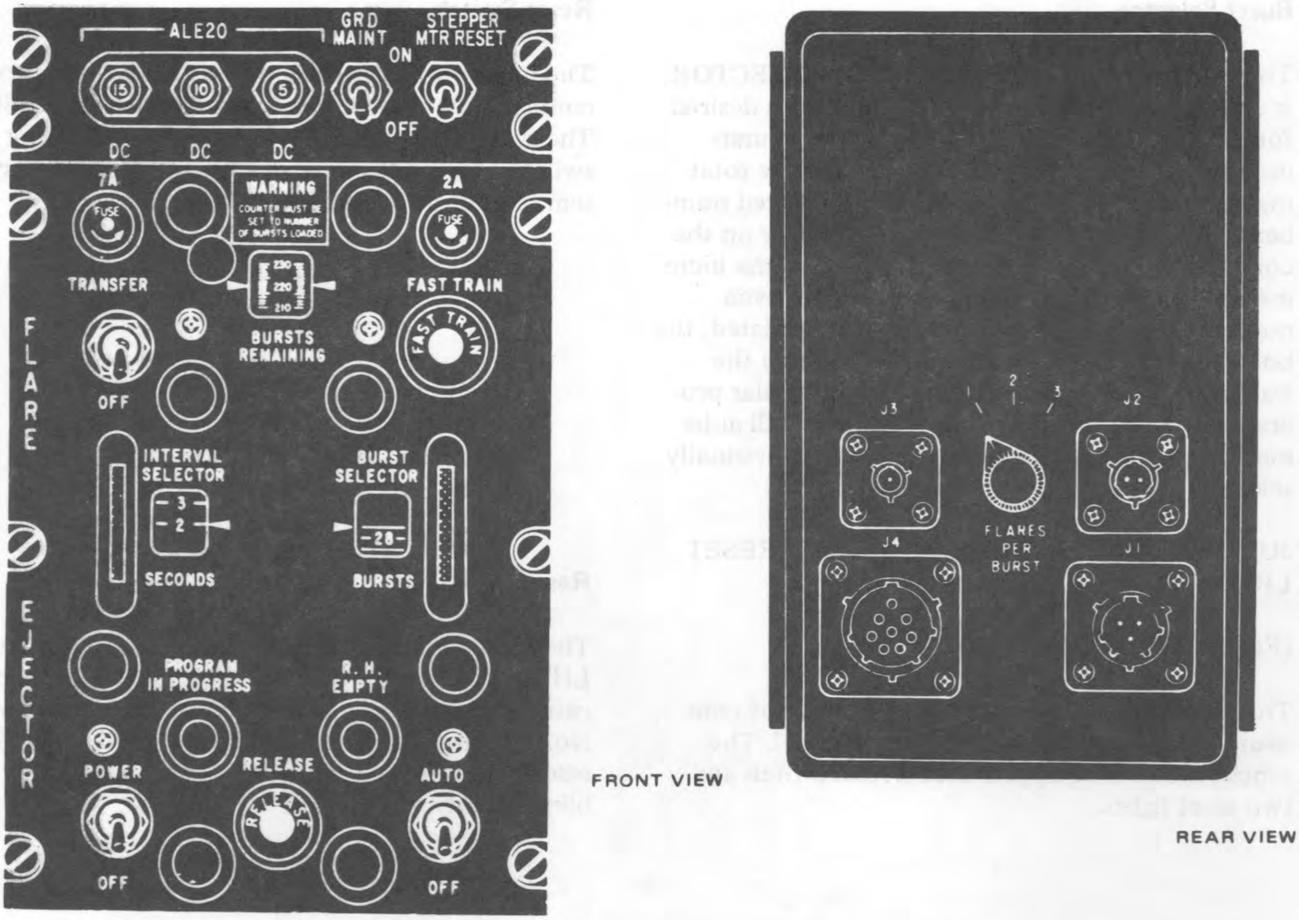


Figure 4-47. Flare Programming Control Panel and Power Panel

Program-in-Progress.

The program-in-progress light is a 28-volt dc press-to-test light that is placarded PROGRAM IN PROGRESS. This light acts as a reminder to the operator that the flare release program established by the other controls is in progress. The light glows when the fast train switch or the release switch is depressed. The light goes off upon completion of the program, but not in fast train. The light also goes off if power to the system is momentarily lost, thus indicating that the release program must be reinitiated when flares are required.

RH Empty Light.

The RH empty light is a press-to-test light that is placarded R.H. EMPTY. This light serves as an indication to the operator that all flares in the right-hand bank have been released.

Bursts Remaining Counter.

The counter, placarded BURSTS REMAINING, displays the total number of bursts remaining on the helicopters. The counter is equipped with a mechanical reset control mounted on the control panel.

Burst Interval Selector.

The burst interval selector, placarded INTERVAL SELECTOR, controls the burst interval generator. The thumbwheel is detented for positive, easy setting in 1-second increments from 2 seconds to 20 seconds. The drum is marked in 1-second increments and numerals for all even numbered settings.

Burst Selector.

The burst selector, placarded BURST SELECTOR, is used to select the number of flare bursts desired for a particular program. The number of bursts used for a particular program is selected by rotating the thumbwheel control until the desired number of bursts is shown through the window on the control panel. The drum is marked in 1-burst increments from 1 to 29, and numerals for all even numbered settings. When a program is initiated, the burst selector counts subtractively to show the number of bursts remaining in that particular program. When the drum counts to zero, it will automatically return to the number of bursts originally selected.

JUNCTION BOX RESET SWITCH AND RESET LIGHTS.

(Figure 4-48.)

The junction box operates on the receipt of command pulse signals from the control panel. The junction box is equipped with a reset switch and two reset lights.

Reset Switch.

The junction box is equipped with a momentary-contact type switch, placarded PRESS TO RESET. The switch when activated returns the stepping switches to the No. 1 positions and allows the system to start at its initial settings.

WARNING

The junction box PRESS TO RESET switch is not be be actuated during flight. The switch shall be reset by ground-servicing personnel.

Reset Lights.

There are two press-to-test reset lights, placarded LH and RH, on the junction box. These lights indicate which stepping switches have been reset to the No. 1 position and are ready to be fired again. The reset circuit is such that the LH and RH lamps blink rapidly when reset.

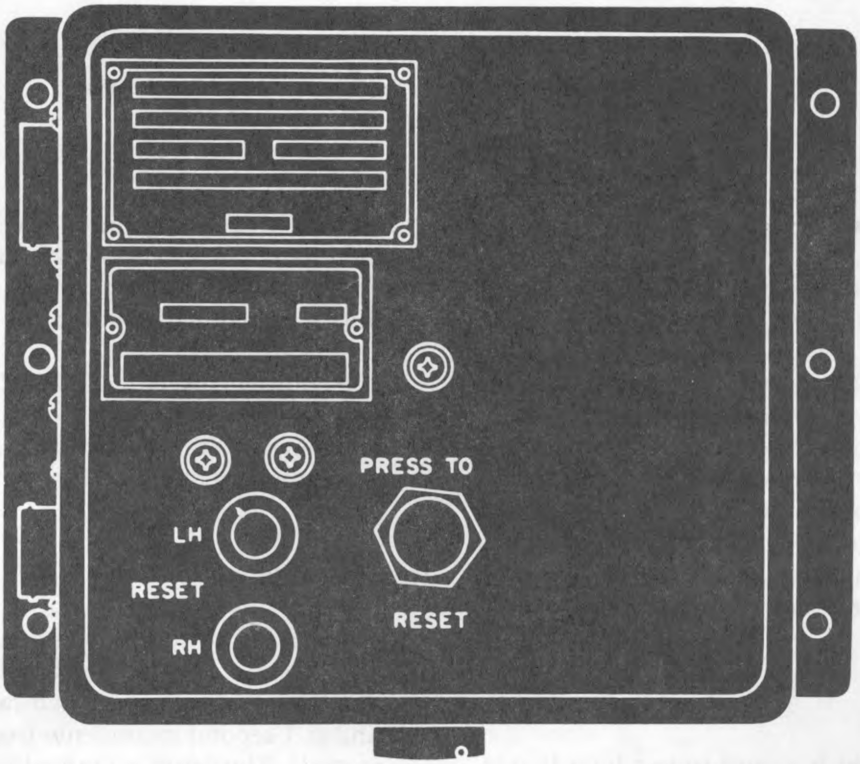


Figure 4-48. Junction Box

FLIGHT ENGINEER'S PANEL.

The flight engineer's panel (figure 4-49) contains switches to release flares and to jettison the flare case assemblies. The FLARE RELEASE pushbutton initiates a flare release signal to the control panel to release the preprogrammed flare burst. The CASE JETTISON switches initiate a signal to the explosive bolt holding the case assembly to the helicopter. The RH switch jettisons the case assembly on the right side of the helicopter. The LH switch jettisons the case assembly on the left side of the helicopter.

Figure 4-49. Flight Engineer's Panel

CYCLIC STICK FLARE RELEASE SWITCH.

A flare release switch is located on the top right of the pilot's and copilot's cyclic stick to enable the pilot or copilot to release flares. The switches initiate a signal to the control panel to release the preprogrammed flare burst.

RAMP OBSERVER FLARE RELEASE PANEL.

The ramp observer flare release panel is mounted on a bracket and can be removed and hand held. Movement is limited by the length of the connecting electrical cord. The FLARE RELEASE switch initiates a signal to the control panel to release the preset flare burst.

LEFT SIDE FLARE RELEASE PANEL.

The left side flare release panel is mounted to the right of the left forward cargo window. The panel contains a FLARE RELEASE switch that initiates a signal to the control panel to release the preprogrammed flare burst.

FLARE CASE MANUAL JETTISON LEVER.

The flare case manual jettison lever is used to release the case assembly containing the flares from the helicopter in an emergency. The lever pulls two locking pins on the case assembly mount to jettison the case assembly and flares.

FLARE EJECTOR SET POWER PANEL.

The flare ejector set power panel contains two 28-volt dc circuit breakers, the 115-volt, 400-Hz circuit breakers, and the two maintenance switches. The STEPPER MTR RESET switch removes power from the case assembly stepping switch assembly reset relay to prevent stepping switch reset until all flares have been released. The GRD MAINT switch applies power to the system when on the ground by bypassing the landing gear interlock.

CASE ASSEMBLY.

Each case assembly contains an AN/ALA-17 flare set containing 8 tubes with 2 flares in each tube, giving a total of 16 flares for each assembly. The helicopter installation contains two fixed-mounted case assemblies for a total of 32 flares. A stepping switch assembly is mounted on the case assembly and is used to supply 28-volt dc igniting power to the squib of a flare pellet. The flare is released from the case assembly upon ignition.

AN/ALA-17 FLARE SET.

The AN/ALA-17 flare set consists of a CY-2617/ALA-17 rack and eight M-3529/ALA-17 cartridges. Each cartridge consists of two RR-108/ALA-17 flares (an upper and lower flare crimped together). The cartridges are covered with polyethylene sleeves, which act as a buffer and as a lubricant between cartridges. The cartridges are positioned in the rack and held in place by a locking bar. An assembled flare set is 12-1/2 inches in width, 12-1/2 inches in height, and weighs 41 pounds. The flare set is installed in the case assembly located on each side of the helicopter.

RESCUE HOIST CABLE QUICK SPLICE PLATE

The hoist quick splice plate assembly is a device to be used by rescue personnel in the event that the hoist cable has been inadvertently cut or has broken. It is to be used only when time is a factor and no other means is available for safe rescue.

NOTE

Cable quick splicing can be used only when there is adequate cable left on the hoist drum to accomplish the rescue.

Pull out sufficient cable from the hoist drum to allow installation of the quick splice plate.

Leaving approximately 6 to 10 inches of the broken end of the cable extending beyond the end of the plate, position the cable at the start point on the front of the plate. (Figure 4-50 of this supplement). Following the direction indicating arrows, position the cable under the retainer clip, proceed to slot 1 and thread cable thru the cable slots in numerical order. While going thru the cable threading process, keep the cable as tight as possible.

WARNING

Failure to follow the proper cable threading sequence on the splice plate will result in cable slipping out of the splice plate when tension is applied to the rescue hook.

After quick splice plate installation, place cable on floor, carefully step on it and pull on the hook assembly sufficiently to remove all possible cable slack and to assure that the splice is secure. Exercise care not to kink the cable while standing on it. An alternate method is to have another crew member hold the cable while the hook is being pulled.

The spliced cable is ready for use.

WARNING

The quick splice plate will not engage the hoist upper limit switch and will probably result in cable separation when the plate engages the boom head. Exercise extreme care to keep the splice plate well below the boom head when the cable is being retracted.

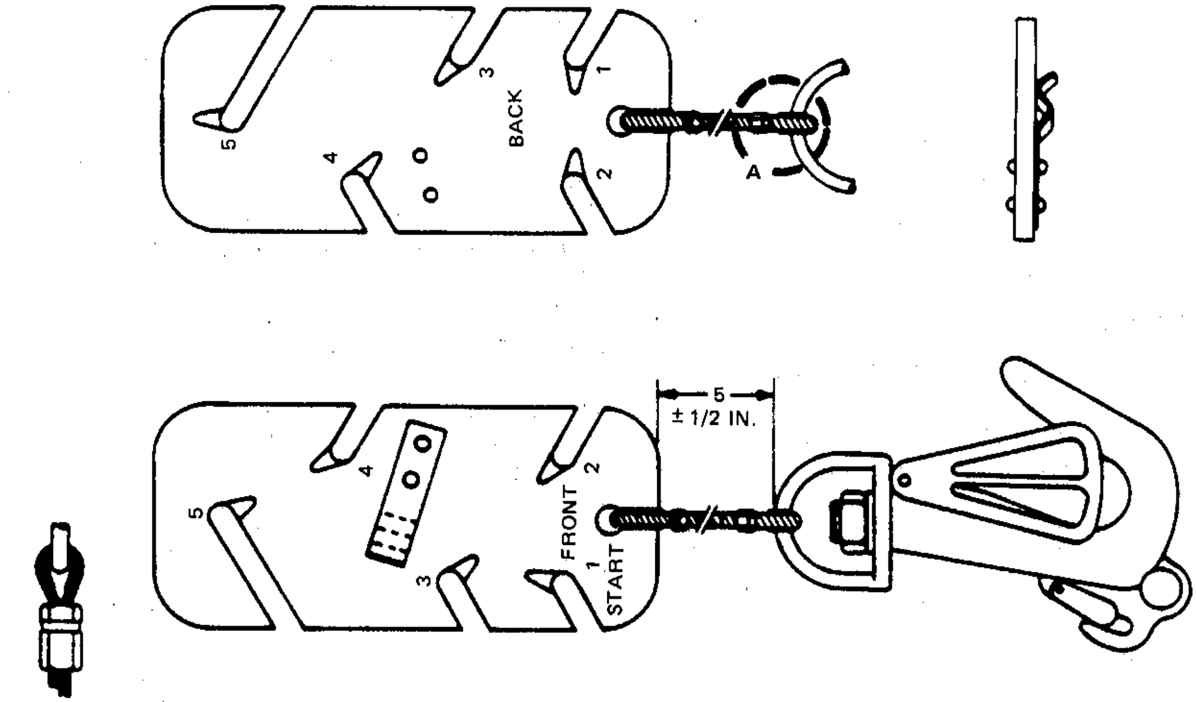


Figure 4-50. Rescue Hoist Cable Quick Splice Plate Assembly

SECTION V

OPERATING LIMITATIONS

TABLE OF CONTENTS

INTRODUCTION.

The operating limitations contained in this section are derived from experience gained during the design, production, and flight test of the helicopter. These limitations, which must be observed if safe and efficient operation are to be attained, should be studied carefully to familiarize the pilot with proper operation of the helicopter and associated equipment. The instruments in the helicopter are marked as shown in figure 5-1 to indicate to the pilot that flight operation is being accomplished in a safe, desirable or unsafe region. Appropriate explanations are provided where the markings are not self-explanatory. In addition, other limitations on operational procedures, maneuvers, and loading are covered.

NOTE

If any of the operating or red line limits included in any section of the Flight Manual are exceeded, remarks concerning the degree to which the limits were exceeded and the time duration will be entered in the Form 781.

MINIMUM CREW REQUIREMENTS.

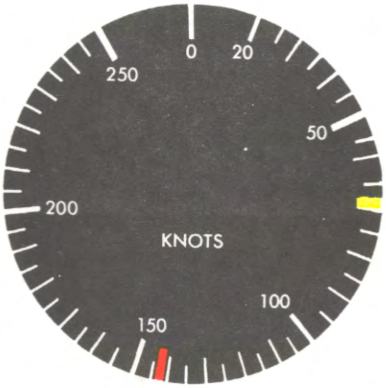
The minimum crew required to operate the helicopter is a pilot and a copilot. The basic crew is a pilot, copilot, and flight engineer. Additional crewmembers may be added at the discretion of the operational commander.

INSTRUMENT RANGE MARKINGS.

Instrument markings shown in figure 5-1 and other operating limitations in this section are not repeated elsewhere in the manual. Unmarked or blank areas between upper and lower radials, or between a green arc and a red radial, indicate regions that should be avoided except for transient conditions such as starting, ground operation, etc.

NOTE

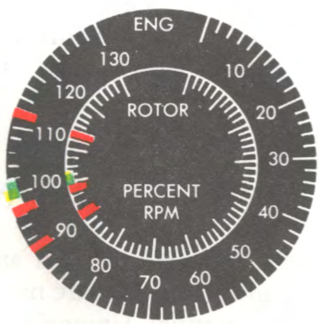
On the power turbine inlet temperature and torque gages, the 30 minute operating limitation between normal and military power is represented as an unmarked area.



FUEL GRADE - JP4/JP5

AIRSPEEDS

YELLOW RADIAL	70 KNOTS	LANDING GEAR - DOWN
RED RADIAL	142 KIAS MAXIMUM	

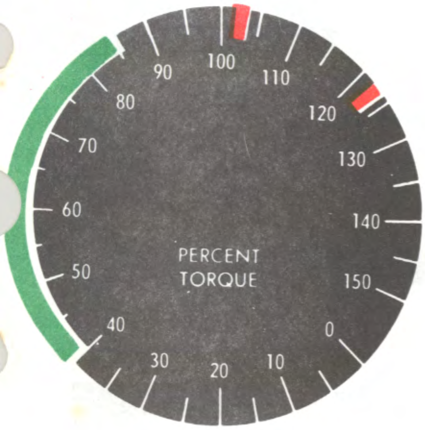


TRIPLE TACHOMETER (Nf/Nr)

RED RADIAL	91%	MINIMUM POWER - OFF
RED RADIAL	98%	MINIMUM POWER - ON
YELLOW ARC	98 - 100%	PRECAUTIONARY, INDICATES APPROACH TO MINIMUM N_r FOR YAW CONTROL AUTHORITY WITH POWER ON
GREEN ARC	100 - 103%	NORMAL OPERATING RANGE
RED RADIAL	112%	MAXIMUM POWER - OFF OR NO LOAD CONDITION

Figure 5-1. Range Markings (Sheet 1 of 4)

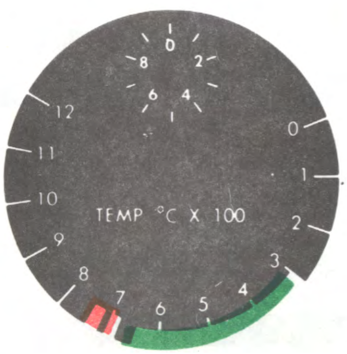
FUEL GRADE - JP4/JP5



TORQUEMETER

GREEN ARC	40 – 86% Q	NORMAL OPERATING RANGE
RED RADIAL	103% Q	MAXIMUM TWIN ENGINE POWER
RED RADIAL	123% Q	MAXIMUM SINGLE ENGINE POWER

NOTE: The 86% limit is established by the main gear box maximum continuous input limit. The main gear box is rated @ 2100 HP maximum continuous power.

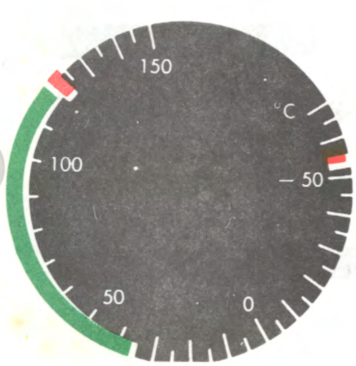


POWER TURBINE INLET TEMPERATURE T_5

GREEN ARC	300° TO 660°C T_5	NORMAL OPERATING RANGE
RED RADIAL	696°C T_5	END OF 30 MINUTE LIMIT RANGE
RED RADIAL	721°C T_5	END OF 5 MINUTE LIMIT RANGE (MAXIMUM EXCEPT FOR STARTING AND TRANSIENT CONDITIONS

GAS GENERATOR (N_g) TACHOMETER

YELLOW RADIAL	56% Ng	(GROUND IDLE ± 3%)
GREEN ARC	70 – 102.7% Ng	NORMAL OPERATING RANGE
RED RADIAL	102.7% Ng	MAXIMUM



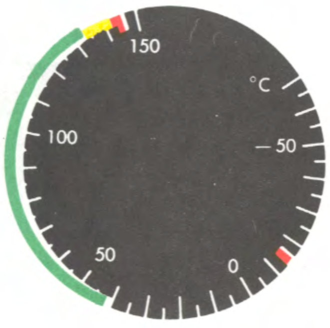
* ENGINE OIL TEMPERATURE

RED RADIAL	-54°C	MINIMUM
GREEN ARC	35° TO 121°C	NORMAL RANGE
RED RADIAL	121°C	MAXIMUM

Figure 5-1. Range Markings (Sheet 2 of 4)

* **ENGINE OIL PRESSURE**

RED	RADIAL	8 PSI	MINIMUM
GREEN	ARC	19 TO 60 PSI	NORMAL RANGE
RED	RADIAL	75 PSI	MAXIMUM



* **TRANSMISSION OIL TEMPERATURE**

THE TRANSMISSION OIL TEMPERATURE INDICATOR IS CONNECTED TO AN OIL TEMPERATURE BULB ADJACENT TO THE MAIN GEAR BOX OIL OUTLET PORT.

RED	RADIAL	–15°C	MINIMUM
GREEN	ARC	40° TO 135°C	NORMAL
YELLOW	ARC	135° TO 145°C	PRECAUTIONARY, MAXIMUM TIME LIMIT 60 MINUTES
RED	RADIAL	145°C	MAXIMUM

CAUTION LIGHT

THE TRANSMISSION OIL TEMPERATURE CAUTION LIGHT WILL COME ON WHEN THE TRANSMISSION OIL TEMPERATURE REACHES 120°C AT THE MAIN GEAR BOX OIL INLET PORT.

* **TRANSMISSION OIL PRESSURE**

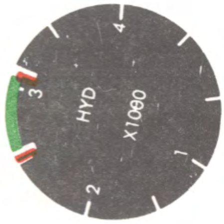
TRANSMISSION OIL PRESSURE INDICATOR IS ACTUATED BY A PRESSURE TRANSMITTER CONNECTED TO MAIN GEAR BOX OIL INLET PORT.

RED	RADIAL	12 PSI	MINIMUM
YELLOW	ARC	12-35 PSI	PRECAUTIONARY INDICATES PRIMARY OR SECONDARY PUMP FAILURE OPERATING RANGE
GREEN	ARC	35-90 PSI	NORMAL OPERATING RANGE
RED	RADIAL	100 PSI	MAXIMUM

CAUTION LIGHT

TRANSMISSION OIL PRESSURE CAUTION LIGHT COMES ON WHEN THE MAIN GEAR BOX OIL PRESSURE DROPS BELOW 4 PSI AS IT ENTERS THE LAST OIL PRESSURE JET IN THE GEAR BOX.

Figure 5-1. Range Markings (Sheet 3 of 4)



*

UTILITY HYDRAULIC PRESSURE

RED RADIAL	2600 PSI	MINIMUM
GREEN ARC	2600 TO 3150 PSI	NORMAL RANGE
RED RADIAL	3150 PSI	MAXIMUM

*

AUXILIARY HYDRAULIC SERVO PRESSURE

RED RADIAL	1300 PSI	MINIMUM
GREEN ARC	1300 TO 1600 PSI	NORMAL RANGE
RED RADIAL	1600 PSI	MAXIMUM

CAUTION LIGHT

THE AUXILIARY SERVO PRESSURE CAUTION LIGHT WILL COME ON WHEN PRESSURE DROPS BELOW 1000 PSI.

*THESE INSTRUMENTS ARE POSITIONED SO THAT UNDER NORMAL OPERATING CONDITIONS THEIR NEEDLES ARE AT APPROXIMATELY THE NINE O'CLOCK POSITION.

*

PRIMARY HYDRAULIC SERVO PRESSURE

RED RADIAL	1300 PSI	MINIMUM
GREEN ARC	1300 TO 1600 PSI	NORMAL RANGE
RED RADIAL	1600 PSI	MAXIMUM

CAUTION LIGHT

THE PRIMARY SERVO PRESSURE CAUTION LIGHT WILL COME ON WHEN PRESSURE DROPS BELOW 1000 PSI.

Figure 5-1. Range Markings (Sheet 4 of 4)

POWER LIMITATIONS

T58-GE-5 Engine

	Gas Generator Speed - % N_g	Power Turbine Inlet Temp °C T_5	Torque % Q	Time Limit
Maximum Power	102.7	721°C	103	5 minutes
Military Power	N/A	696°C	103	30 minutes
Maximum Continuous Power	N/A	660°C	86	None

T58-GE-100 Engine

	Gas Generator Speed - % N_g	Power Turbine Inlet Temp °C T_5	Torque % Q	Time Limit
Maximum Power	102.7	745°C	103	10 minutes
Military Power	N/A	721°C	103	30 minutes
Maximum Continuous Power	N/A	685°C	86	NONE

NOTE

Torque may exceed 103 percent Q on one engine to a maximum of 123 percent, provided that the power of the other is reduced so that total torque for both engines does not exceed 206 percent Q for 30 minutes or 172 percent continuously, and that the single engine N_g, T_5 and Q limits are not exceeded. The governing parameter is the limit which occurs first.

Figure 5-2. Power Limitations Table

INDEX MARK.

A white index mark appears on all instruments having range markings to indicate possible movement of the glass and subsequent incorrect interpretations of the markings.

ENGINE OPERATING LIMITATIONS.

Engine operating limits are illustrated in figure 5-1.

CAUTION

- Dependent upon free air temperature and pressure altitude, the N_g, T_5, or Q power limits may be reached first. This parameter determines whether the 30-minute operating limit or the 5-minute operating limit are effective. For variation of power available with temperature and altitude, refer to the Appendix.
- Extended use of maximum power will reduce engine life. Therefore, maximum power will not be used more than 5 minutes at one time. Immediately after operating at maximum power for a cumulative total of 5 minutes, operate engines 15 minutes at continuous power/ 660°C T_5 or below for cooling in event maximum power is again required.

ENGINE OVERSPEED AND OVERTEMPERATURES.

Exceeding the normal engine operating limitations will result in reduced engine life. These critical operating limitations reflect the absolute limitations of the engine and should only be used when required by an extreme operational situation.

Power Turbine Speed (N_f).

The lower unmarked area on the N_f tachometer indicates a precautionary range for transient operation with engine power on. The upper unmarked area is a transient power range used during ground checks, reduced power operations, and prior to takeoff to allow for engine droop.

CAUTION

During reduced or partial power descents and practice autorotations with minimum collective pitch, N_f may exceed 103% up to a maximum of 112%, at which time the rotor (N_r) and N_f pointers should split if the rotor speed is increased above limits. If the N_r pointer does not split at this point indicating failure of the engine drive shaft to properly disengage from the transmission, power should be applied to bring N_f within limits.

Gas Generator Speed (N_g).

The following overspeed limits are provided for information in case of fuel control malfunction or improper settings. Operation between N_g topping limit and 106% N_g; no action is required.

Power Turbine Inlet Temperature (T_5).

If the maximum T_5 attained during compressor stalls or any other overtemperature is not observed, it is to be assumed that the limits have been exceeded.

Limiting T_5 Rise for Salt Water Operation.

Hovering over salt water is limited by the T_5 rise experienced at the same N_g and torque values as adjusted for the change/variance from the established N_g/T_5 baseline. This limit is 35°C. See Section II for computation instructions.

TRANSMISSION LIMITATIONS.

Main gear box oil temperatures and pressures are shown in figure 5-1. In addition, the transmission system has input torque limits which require certain actions when exceeded. These are provided for information.

MAIN GEAR BOX OIL TEMPERATURE.

The normal operating temperature is 40°C to 135° C. Temperatures above 120°C on the indicator generally result from operating in high ambient temperature, unusual flight attitudes, malfunction of the cooling system or an overserviced condition. Gear boxes operated between 135°C and 145°C for periods not exceeding 60 minutes require a serviceability check. Gear boxes operated between 135°C and 145°C for periods exceeding 60 minutes or above 145°C for any length of time must be replaced.

MAIN GEAR BOX OIL PRESSURE.

Main gear box oil pressure fluctuations of 2 to 3 psi are normal; however, fluctuations up to 10 psi (± 5 psi from a steady position) are allowable provided no indications of a malfunction are apparent.

TWO ENGINES OPERATING.

Transient operation between 206 and 240% total torque is permissible for periods not exceeding 5 seconds. Operation between 206 and 240% total torque for more than five seconds or over 240% torque for any time period requires removal and overhaul of the main gear box.

ONE ENGINE OPERATING.

Transient operation between 123 and 130% for more than 5 seconds or over 130% for any time period requires removal and overhaul of the main gear box.

ROTOR LIMITATIONS.

Normal rotor limitations are shown in figure 5-1. When operating with power on, do not operate continuously below 100% or above 103% N_r. Operations above the continuous limit for significant percentages of mission flight time will degrade the service life of main rotor components. It is recommended that during extended instrument approaches and repeated closed traffic patterns, the continuous limit be observed by beeping throttles to maximum during the final phase of the approach if maximum N_r is desired. During autorotation, do not operate below 91% or above 112% N_r.

CAUTION

Should the lead lag or droop stops be contacted in flight during a main rotor droop, the main rotor head dampers and rotating scissors assemblies must be removed and returned to overhaul for inspection. The main rotor blades must be inspected in accordance with existing inspection requirements.

NOTE

Rotor speeds in excess of 117% N_r subject components to abnormal forces which may cause damage.

AIRSPEED LIMITATIONS.

The maximum permissible airspeed is 142 KIAS. Figure A-38 shows the maximum airspeed at various gross weights, density altitudes, and rotor rpm. Sideward flight is limited to 35 knots. Rearward flight is limited to 30 knots. Maximum airspeeds for maneuvering flight are determined by using the Blade Stall Chart and Blade Tip Mach Chart in the Appendix.

CROSSWIND LIMITATIONS.

The maximum cross wind component for landing is 35 knots.

MANEUVERS.

The helicopter is restricted to normal flying maneuvers. No aerobatic maneuvers are permitted and flight controls should not be moved abruptly. Hovering turns should not exceed a rate of 360 degrees in 15 seconds. Maximum angles of bank, dependent on airspeed and blade load factors are determined using the blade stall chart in the Appendix. The maximum angle of bank is 50 degrees.

JETTISON OF EXTERNAL FUEL TANKS LIMITATIONS.

The following limitations apply to either empty or full tanks:

WARNING

- Do not jettison tanks when both engines have failed.
- To insure effective operation of the manual jettison system, a force of approximately 65 pounds may be required to actuate the manual release handle. A minimum of 2¼ inches of cable travel is necessary to insure simultaneous jettison of both tanks.

1. The recommended airspeed for jettisoning the external fuel tanks is 70 KIAS or less.

CAUTION

Do not jettison external fuel tanks above level flight speeds of 75 KIAS, above rates of descent of 300 feet-per-minute, during autorotation, or asymmetrically during climb.

2. The external tanks should not be jettisoned at rates-of-descent of 300 feet-per-minute or greater, due to the possibility of released tanks striking the main or tail rotor blades.

3. Asymmetric jettison of the external tanks during climb can result in rapid attainment of excessive roll rates and roll attitudes (20^{o} roll in 0.2 seconds).

CENTER OF GRAVITY LIMITATIONS.

It is possible to exceed the CG limits if the helicopter is not properly loaded. To determine placement of load for anticipated missions, refer to the Manual of Weight and Balance Data, T.O. 1-1B-40 and LOAD ADJUSTER, Section IV. The CG limitations will vary according to the gross weights of the helicopter. To determine the most fore-and-aft CG locations for a given gross weight, see figure 5-3. The takeoff and anticipated landing gross weight should be obtained prior to each mission and determined to be within specified limitations. If a locally standardized weight and balance clearance, Form 365-4, showing the helicopter to be within limits is not on file, a Form 365 will be

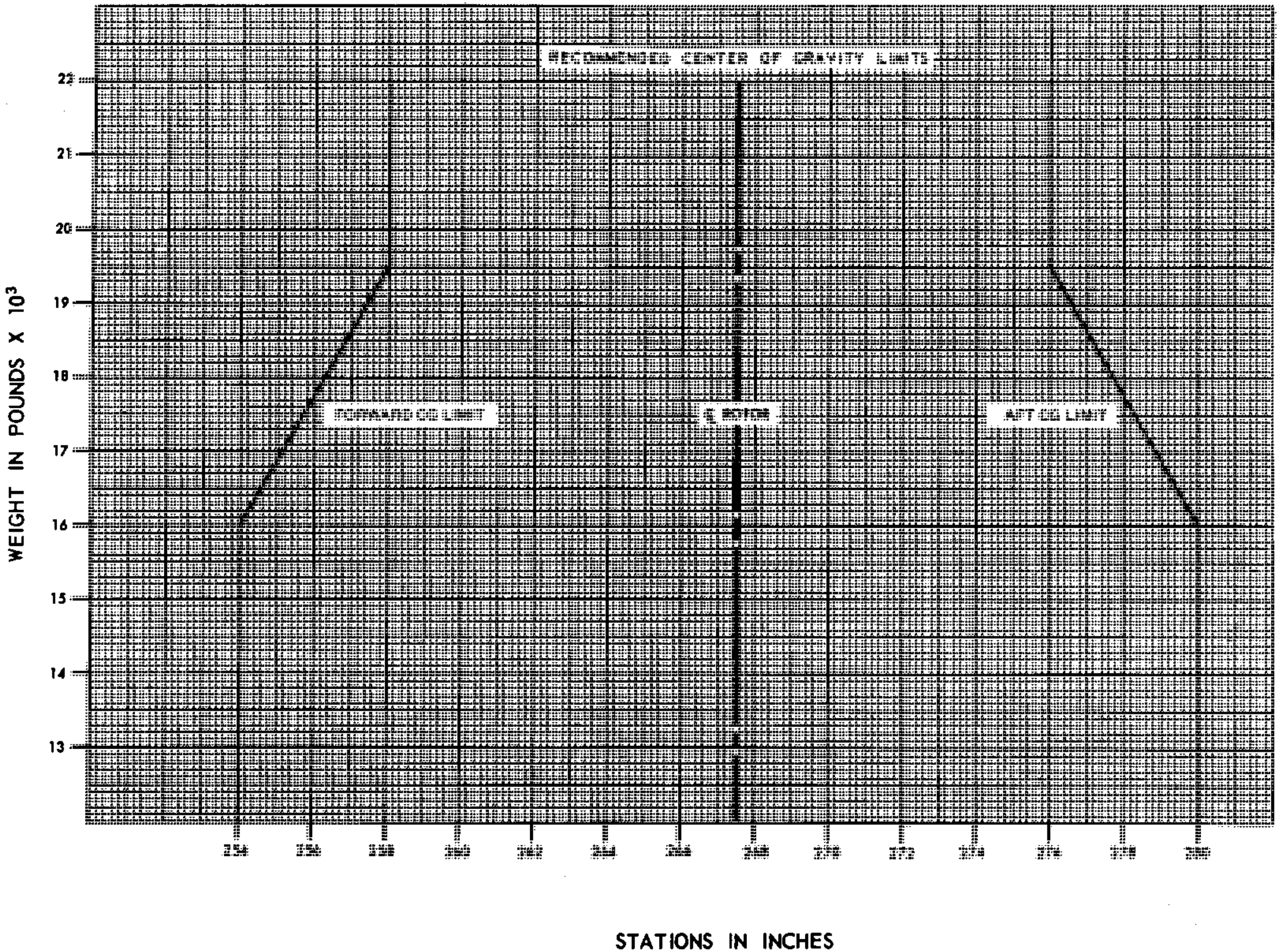


Figure 5-3. Center of Gravity Limitations Chart

completed. The load adjuster may be used to compute the Form 365-4. For additional information, refer to LOAD ADJUSTER in Section IV, WEIGHT LIMITATION, in this section, Manual of Weight and Balance, T.O. 1-1B-40; Basic Weight Checklist and Loading Data, T.O. 1H-3(C)C-5, Cargo Loading Manual, T.O. 1H-3(C)-9, and USAF Aircraft Weight and Balance, T.O. 1-1B-50.

WEIGHT LIMITATIONS.

The basic design or normal gross weight of the helicopter for structural analysis is 19,500 pounds at a limit load factor of 2.5 Gs. The maximum allowable gross weight is 22,050 pounds at a limit load factor of 2.21 Gs. The maximum gross weight for hovering charts in the Appendix give detailed information on the maximum gross weights at whicb the helicopter may be operated under varying conditions of temperature, altitude, wind velocity, and type of takeoff or landing. Gross weight of the helicopter may be determined by referring to the takeoff and landing data (TOLD) card.

MARGIN OF SAFETY AND LOAD FACTORS.

It must be realized that as a structure is loaded to higher weights, its ability to withstand additional loads resulting from maneuvers or gust conditions becomes increasingly less. The margin of safety is the amount of additional load that the structure will sustain before failure occurs. When planning any helicopter mission, consideration must be given to the fact that the maximum permissible weight may depend on the margin of safety desired for the various supporting structures (main rotor, fuselage, landing gear, flooring, etc). If the mission requires excessive maneuvering or flight through turbulent air, it is advisable to maintain a larger margin of safety than if smooth level flight were contemplated. However, the larger the margin of safety, the lower the maximum permissible weight will be. Flight load factors are used as an indication of the margin of safety available for helicopters. Therefore, the structural margin of safety will be equal to the difference between the limit load factor determined for the gross weight and the flight load factor the helicopter is sustaining at any given moment. For example, should the helicopter be loaded so that it is capable of making good a limit load factor of 2.5, and during various phases of flight, flight load factors (G loads) due to maneuvers or gusts of 1.5 and 2.0 are imposed on the helicopter; the margins of safety during these phases would be 1.0 and 0.5 flight load factors, respectively. Therefore, it is important that the maximum flight load factors that will be encountered during a mission be anticipated in order that the helicopter will be loaded in such a manner that the load limit factor it was designed for will never be exceeded during any phase of the flight.

LANDING GEAR LIMITATIONS.

There are no structural limits affecting the extension or retraction of the landing gear in flight. The landing gear is designed for landing at the design normal gross weight of 19,500 pounds, with ground contact at a sinking speed of 480 feet per minute. Caution should be used in taxiing the aircraft in the most critical condition of a braked roll, pivoting and clockwise turning (nose gear only).

CAUTION

A hard landing will be entered in the Form 781, if the limits of figure 5-4 are exceeded, or if any reasonable doubt exists as to the firmness of a landing.

EQUIPMENT LIMITATIONS.

1. Aft ramp shall not be operated in flight at airspeeds above 115 knots. In no case shall the ramp be opened below the horizontal position in flight.

CAUTION

Ramp safety cables must be installed at all times during flight.

2. The personnel door should not be opened or closed during flight at airspeeds above 115 knots.

3. Use of the cargo slings is limited to maximum weights as follows:

   a. External cargo sling (cable suspended) 6000 pounds.

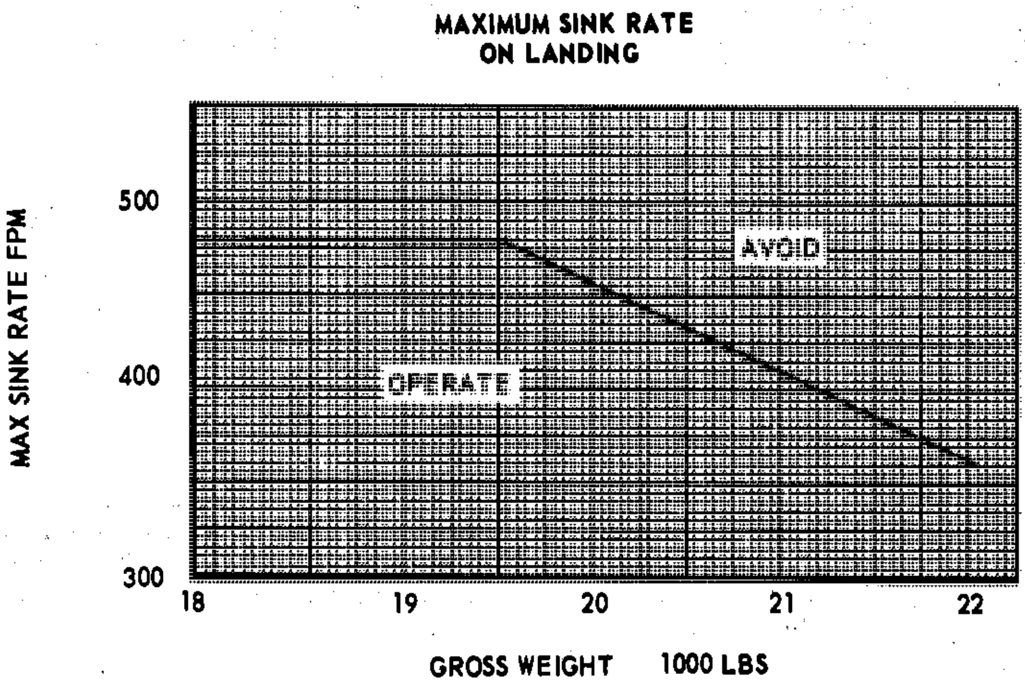


Figure 5-4. Maximum Sink Rate on Landing Chart (Retractable Landing Gear)

b. External cargo sling (low response) 8000 pounds.

4. Rescue hoists that have a letter Z stamped on the nameplate are limited to a maximum of 550 pounds when raising a load and 300 pounds when lowering a load.

5. The helicopter will not be flown in known icing conditions or visible moisture when temperatures are at or below plus 5°C, without a foreign object deflector installed.

6. One fuel boost pump per engine must be on and functioning during flight to ensure continuous fuel supply to the engine driven fuel pumps while engines are operating under any of the following conditions:

a. Above 5000 feet PA

b. Above 25°C

c. Below 600 lbs of fuel per tank

d. Whenever a fuel filter bypass caution light is illuminated.

e. Whenever a fuel low pressure warning light is illuminated.

WARNING

Engine flameout can be expected in flight if the aircraft is operated without a boost pump functioning under any of the above conditions.

7. Do not change altitude more than 200 feet with BAR ALT engaged without depressing the BAR REL switch or completely disengaging the BAR ALT. Damage to the altitude controller will result if altitude is changed more than 500 feet with BAR ALT engaged.

8. Hoist operations using the winch installation (CH-3E prior to serial No. 66-13285

not modified by T.O. 1H-3(C)C-561) is limited to use in actual emergency rescue or for training purposes using a dummy.

9. The cabin jettisonable windows should not be removed or installed above 70 knots.

SLOPE LANDING LIMITATIONS.

1. To achieve a hovering condition with at least one wheel on a slope, the following limits apply:

CONDITION	SLOPE LIMIT
Noseup Slope	30 degrees
Nosedown Slope	8 degrees
Cross Slope	25 degrees

2. To achieve a landing and rotor shutdown, the following limits apply: (CG listed is the most critical which would apply for each slope condition.)

CONDITION	SLOPE LIMIT	MOST CRITICAL CG
Noseup Slope	8 degrees	280 inches
Nosedown Slope	8 degrees	254 inches
Cross Slope	10 degrees	254 to 280 inches

NOTE

A 20 knot downslope wind condition will reduce noseup and cross slope capability by approximately 1 degree. Effects of a moderate upslope or cross wind slope will be negligible.

SECTION VI

FLIGHT CHARACTERISTICS

TABLE OF CONTENTS

INTRODUCTION.

The helicopter is capable of flight over a speed range extending from approximately 30 knots rearward to approximately 142 knots forward and sideward flight up to 35 knots.

The helicopter has an automatic flight control system (AFCS) which improves its basic flying qualities. The rotor of this helicopter can be safely engaged and stopped in winds up to 60 knots. Cyclic and tail rotor pitch control provide taxi capability in winds from any direction. The tricycle landing gear gives good ground handling characteristics and minimum radius turns are easily accomplished. Water maneuvers in any direction while floating in a level attitude are accomplished with minimum power expenditure. The helicopter lifts into a hover in a generally level attitude; however, this is a function of center of gravity location; aft CG giving a slightly nose high attitude, and forward CG a slight nosedown attitude. Two control system characteristics are incorporated which simplify coordination of controls when lifting off to a hover. As increasing collective pitch results in an increase in main rotor torque, an increase in tail rotor thrust will be required to offset the yawing moment induced. This is accomplished automatically by an integral mechanical coupling within the control system which alters tail rotor pitch as a function of collective pitch. Therefore, pedal applications required by the pilot to hold his heading as power is applied to hover will be very small. The second mode of mechanical coupling, also provided during a collective change, imparts a proportional lateral tilt to the rotor cone to counteract the rolling imbalance and lateral drift induced by the change in tail rotor thrust. Rpm control during all ranges of flight is simplified with the turbine engine installation. Once a rotor speed is selected the engine fuel control will automatically maintain the selected rpm. Only small adjustments to engine trim speeds will be required to compensate for a slight droop which occurs after power changes.

LEVEL FLIGHT CHARACTERISTICS.

Control displacement while moving away from a hover is positive in all directions; however, initial response to a cyclic input will be slightly greater with the AFCS operating. Any oscillation induced by an aerodynamic disturbance, while hovering, will be dampened within one cycle with the AFCS operating. With the AFCS off, the oscillations are mildly divergent but are easily controlled by the pilot. While transitioning from hovering into forward flight, without AFCS at high weights, momentary application of aft cyclic to maintain desired pitch attitude after the basic applications of forward cyclic may be noted, but control is easily maintained. During climb, control is positive about all axes and no difficulty is encountered in maintaining best rate-of-climb speed. The heading will be maintained at all flight conditions with yaw channel of the AFCS operating, and the helicopter will return to original heading without oscillation if disturbed by a gust. Without the yaw channel operating, any disturbance in yaw will be dampened in a few cycles. When entering sideslips, control positions will always be in the proper direction, (right lateral cyclic with left pedal) and no control reversals will be encountered throughout the range of sideslip angles. There is adequate tail rotor power available to accomplish any desired directional maneuvering. In the speed range from 50 knots to Vmax, a forward cyclic increment will be required to increase airspeeds. The helicopter will always exhibit a tendency to return to trim following a disturbance in pitch or roll with AFCS engaged. With the AFCS off, any oscillations in pitch and roll will be dampened in a few cycles. At high rotor speeds, disturbances in pitch and roll will be dampened best. At higher rotor speeds, control margins are greater. No violent helicopter motions are encountered when entering autorotation in forward flight; however, some trim change will be noticed. The amount of cyclic trim change required is a function of airspeed at the time of autorotational entry. At high forward speeds, some right lateral and aft longitudinal trim changes are required. The amount of trim change on autorotational entry will decrease with airspeed. Rpm decay rate after a sudden reduction of power is high if collective pitch is not lowered rapidly; however, rotor rpm builds up again on reduction of collective pitch. Adequate tail rotor power is available during autorotation to accomplish all maneuvering and directional control is positive in sideslip. Flare is effective in reducing airspeed and rate-of-descent, if recommended rotor speed is maintained in autorotation and there is adequate rotor inertia for an effective flare. Power off touchdown is made in a slightly nose high attitude, but the helicopter is capable of low autorotation touchdown speeds when desired. Power on approach and vertical landing is accomplished using normal technique. Run-on landings are easily accomplished on properly prepared surfaces with final approaches and touchdowns made in a near level attitude. The tricycle landing gear affords good control after touchdown.

LEVEL FLIGHT CHARACTERISTICS UNDER VARIOUS SPEED CONDITIONS.

For hovering or low speed flight, high rotor rpm is required because of the high power and control necessary. When hovering or flying at low speed and increased forward speed is desired, the cyclic stick is moved forward. A momentary settling may occur with rapid acceleration, and then the helicopter will begin to climb because the main rotor blades encounter an increased flow of air due to the forward movement of the helicopter. As the helicopter accelerates to approximately 50 KIAS, collective pitch should be steadily decreased to maintain a constant altitude. To maintain the same altitude above approximately 60 KIAS, an increase in collective pitch is necessary until maximum speed is reached. At maximum speed, a higher collective pitch setting is required than for hovering and power turbine speed should be between 100 and 103%, depending where it is smoothest. As forward speed is increased, the helicopter will assume an increasing nosedown attitude. This is caused by the rotor blade flapping hinges that are located at a distance from the center of the rotor hub. When the main rotor blade tip-path plane is tilted forward to increase forward speed, the centrifugal force of the blades will tend to align the plane of the rotor hub, and consequently the fuselage, with the forward tilted tip-path plane. The automatic stabilization equipment introduces fore-and-aft cyclic control corrections to maintain a given fuselage pitch attitude thus providing automatic cruising speed control. As the helicopter is decelerated from cruise condition, power required decreases until 50 - 80 knots is attained. Below this airspeed, power required increases as airspeed decreases. (Figure 6-1 portrays only unaccelerated level flight conditions.) During conditions of light gross weights, low altitudes and temperatures, power required will normally be less than power available allowing a margin for maneuver. At heavy gross weights, high altitudes and temperatures, the power required may exceed the power available; thereby; preventing level flight at slower speeds. Figure 6-1 illustrates conditions where power required exceeds power available resulting in descending flight until increased forward airspeed is attained. When the helicopter is decelerating, descending or rotor rpm has decayed and the condition is to be reversed, power required will be even greater. Even if there is sufficient power available to reverse the rate of descent or deceleration, the engines may not fully accelerate before the speed or rpm has decayed below the point where level

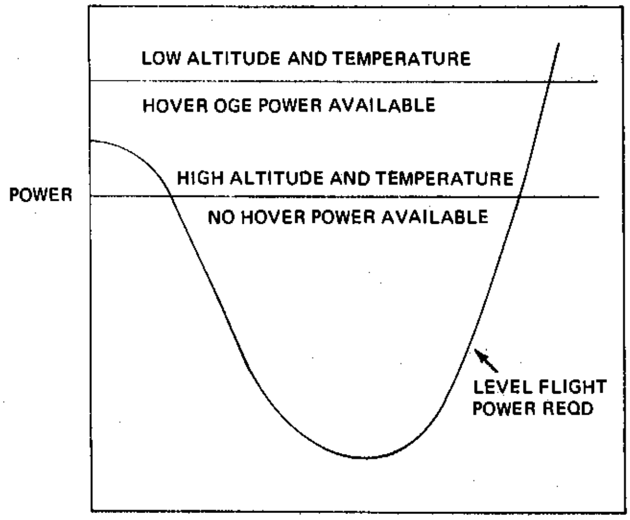


Figure 6-1. Forward Airspeed Unaccelerated Level Flight

flight is possible. When operating in conditions where OGE hover is not possible, level flight below 50 KIAS and less than 103% N_r should be avoided.

STALLS.

Stalls, as applied to a fixed-wing aircraft, will not occur in a helicopter. However, the helicopter may encounter a stall condition referred to as blade stall described in this section.

BLADE STALL.

Blade stall, the tendency of the retreating blade to stall in forward flight, limits the high speed potential of the helicopter, increases stress, and decreases component life. The retreating blade (the blade moving away from the direction of flight) has a tendency to stall because the blade tip is traveling at the rotational velocity minus the forward speed of the helicopter. As the velocity of the retreating blade decreases, the blade angle of attack must be increased to equalize lift to provide stabilized flight. As the angle of attack increases, with the highest blade angles being at the tip, the blade will stall (lose lift and increase drag). The increased drag will cause loss of rotor speed unless power is increased. The advancing blade (the blade moving into the direction of flight) on the other hand is traveling at a substantially higher speed and has relatively uniform low angles of attack and is not subject to blade stall. Blade stall will first occur at the blade tip and is most likely to occur when operating at high values of airspeed, gross weight, density altitude, and power and especially with low rotor rpm. Maneuvers, acceleration, or turbulent air, all of which increase g load factors, will induce blade stall by reducing the airspeed at which blade stall will occur. The blade stall chart as presented in figure A-36 portrays the airspeeds at various pressure altitudes, temperatures, gross weights, rotor speeds, and load factors (angle of bank) as limited by blade stall. This blade stall chart establishes maximum recommended airspeeds to allow for turbulence, mild maneuvers, and necessary control inputs to maintain the desired flight attitude. At these speeds roughness, encountered by reasonable maneuvers or mild turbulence, can be tolerated. Severe turbulence or abrupt control maneuvers at this point will increase the severity of the stall and the helicopter will become more difficult to control. In the blade stall condition, each main rotor blade will stall as it passes through the stall region and create vibrations per revolution equal to the number of blades. If stall is allowed to fully develop (speeds in excess of those shown in figure A-36), loss of control will be experienced and the helicopter will pitch upward and to the left. The use of forward cyclic stick to control this pitch up is ineffective and may aggravate the stall as it increases the blade angle of attack of the retreating blade.

METHODS OF ELIMINATING ROUGHNESS CAUSED BY BLADE STALL.

If blade stall is causing roughness in the helicopter during high speed flight, or when maneuvering at lower speeds, the roughness may be eliminated by accomplishing one or any combination of the following:

1. Decrease collective pitch.
2. Decrease the severity of the maneuver.
3. Gradually decrease airspeed.
4. Increase rotor rpm.

SETTLING WITH POWER.

At high altitudes, at high gross weights, or when operating with reduced power, it may not be possible to maintain level flight due to a lack of power that will cause settling to occur. Settling with power can occur at any airspeed or altitude combination whenever power required exceeds power available, thus preventing level flight at slower speeds. This settling is of minor consequence except at certain rates of descent and low forward airspeed where it is extremely critical since vortex ring state may develop.

VORTEX RING STATE.

Sometimes referred to as power settling, this extremely critical situation may develop at any altitude or gross weight when the airspeed is below translational lift and rate of descent is high. Under certain power and rate of descent combinations, the down wash from the rotor begins to recirculate up, around, and back down through the effective outer portion of the rotor disc. The helicopter sinks into the air mass it has just displaced in trying to obtain lift and the main rotor blades work continually in their own turbulent airstream. The velocity of the recirculating air mass may become so high that full up collective pitch lever may not produce sufficient lift to control rate of descent. Increasing collective normally has an adverse effect since this antagonizes the turbulent airflow. The possibility of entering vortex ring state is further increased during conditions of low airspeed and high rates of descent when compounded with tailwinds or large or rapid applications of aft cyclic or collective. When vortex ring state develops, roughness and partial loss of control may occur, indicated by ineffectiveness of controls. To recover from the condition, increase forward airspeed, decrease collective pitch, or enter autorotation if altitude permits. A considerable loss of altitude may occur before the condition is recognized and recovery is completed. During approach for landing, the conditions causing vortex ring state should be avoided.

CAUTION

When operating below translational lift, avoid rates of descent in excess of 500 feet per minute for normal operations, and 300 feet per minute when operating in conditions of high density altitude/ high gross weights.

DYNAMIC ROLLOVER CHARACTERISTICS.

WARNING

The helicopter may roll over on its side if the bank angle (roll angle) reaches 15 degrees when one landing gear is on the ground and the thrust of the rotor system is approximately equal to the weight of the helicopter. Reduce collective to stop the roll and correct the bank angle to a wings-level attitude.

Landings and takeoffs in a skid or side drift can cause the helicopter to roll, pivoting about one landing gear which is in contact with the ground. If the angle of roll (or bank) exceeds 15 degrees with the landing gear on the ground, then the helicopter may continue to the rolling movement which cannot be stopped with full cyclic, opposite to the direction of roll. The degrees of roll angle, beyond which recovery is not possible, will become less than stated above if any of the following conditions exist: there is a crosswind; the right landing gear is the pivot point; the helicopter's c.g. is displaced laterally; left pedal input is increased. When performing maneuvers with one wheel on the ground (e.g., cross-slope landings and takeoffs and one wheel landing on uneven ground), maintaining a trimmed, wings-level attitude will preclude this condition from developing. When the helicopter is in a banked attitude with one wheel on the ground, lateral cyclic control response becomes sluggish and less effective. Collective pitch is more effective than lateral cyclic in controlling the rolling. Increasing collective will increase and accelerate the rolling movement. When cross-slope landings and takeoffs are performed, the "one wheel on-the-ground" maneuver is unavoidable because the up-slope wheel touchdown first and lift off last. If the helicopter starts to roll toward the up slope side (5° - 8° bank angle) reduce the collective pitch to correct this situation. Return to a wings-level attitude before continuing the take-off or landing. Do not increase collective pitch suddenly to get airborne in attempting to recover from the banked condition. The angle of roll will increase and the roll rate will accelerate before the wheel will lift off the ground. The angle of roll may increase beyond the critical roll angle and the helicopter will roll over on its side.

WARNING

Use smooth, moderate collective movements when recovering from a banked attitude with one wheel on the ground. Rapid reduction of collective may cause fuselage-rotor contact or a high rate of roll in the opposite direction when on a slope.

FLIGHT CONTROLS.

FLIGHT CONTROL SERVOS.

The primary servos at the rotor assembly and the auxiliary servos at the mixing unit are both in operation at all times. Because of the servo units, the control forces are virtually eliminated and are constant throughout their full range. This may cause a tendency to over-control at first, because there is very little feel in operating the cyclic stick unless the cyclic stick trim system is in operation. If either servo system should fail or malfunction, it may be turned off, provided there is hydraulic pressure in the other system. Both servo systems may be turned off by the pilot; however, the switching prevents both servos being turned off at the same time. If the primary boost, which physically controls the lower swashplate is turned off, movement of the lower controls and swashplate is accomplished through the auxiliary boost which is located near the cockpit controls. In this instance, feel of the pilot's control remains almost unchanged except for small differences due to friction and lost motion in the control system. If the auxiliary boost is turned off, the pilot physically moves the push-pull rods and bellcranks up to the primary boost. In this instance, a friction force of several pounds is noticeable and stability augmentation through the AFCS is lost. In addition, if the auxiliary servos are inoperative, the tail rotor control forces will be transmitted to the tail rotor pedals since the tail rotor does not have a separate servo.

COORDINATION OF FLIGHT CONTROLS.

The climb and descent of the helicopter is controlled primarily by raising or lowering the collective pitch lever; however, coordinated movements of the tail rotor pedals and cyclic stick are necessary to maintain a constant heading. When collective pitch is increased to ascend, additional torque is developed by the main rotor. This torque can be compensated for by use of the tail rotor pedals. However, minor changes are accomplished by the yaw compensator, which is a mechanical coupling within the flight control system that changes tail rotor pitch proportionately with a change in the main rotor collective pitch. The torque-compensating tail rotor pitch changes are accomplished automatically when the automatic flight control system is engaged. Sideward flight from hovering is accomplished primarily by lateral displacement of the cyclic stick; however, it is necessary to use tail rotor control to prevent the nose from swinging toward the direction of flight. When flying sidewards to the right, the cyclic stick is displaced to the right and the left tail rotor pedal is used to keep the nose of the helicopter in the original direction. For sideward flight to the left, the cyclic stick is displaced to the left and the right tail rotor pedal is used. In hovering with no wind no appreciable movement of the cyclic stick is necessary; however, with a wind condition the cyclic stick should be held into the wind to maintain the same relative position above the ground. Turns while hovering are accomplished primarily by depressing the right pedal for a right turn and the left pedal for a left turn. During forward flight at low speeds, coordinated movements of the cyclic stick and tail rotor pedals are necessary to accomplish turns. In high speed flights, less pedal displacement is necessary to accomplish turns.

HELICOPTER VIBRATION.

The inherent vibrations in any helicopter are those created by the mechanical functions of the engines and transmission systems, dynamic action of the main and tail rotors, and aerodynamic effects on the fuselage. The overall vibration level is influenced by the many individual frequencies of vibration and combinations thereof. Many multiples of a basic frequency are felt, and often two or more different superimposed frequencies create beats. The overall magnitude is the resultant of the amplitudes of all the frequencies. It would be difficult for the pilot to completely separate all the types of vibrations encountered. Generally, these are divided into three categories; namely, (1) low (2) medium, and (3) high frequencies. Varying magnitudes of all three types of vibrations are often present in an individual helicopter. Only through experience will the pilot be able to judge what is normal to the model and what is abnormal and correctable.

LOW FREQUENCY VIBRATIONS.

One Times Main Rotor Speed (One Per Revolution).

This vibration emanates from the main rotor system and is generally caused by main rotor head or blade imbalances. This vibration produces a rotary excitation of the fuselage which feels like a lateral oscillatory roll or wallow to the pilot. If this vibration is present in all regimes of flight, it should be noted in Form 781. The most probable causes are:

1. Main rotor blades out of track. A blade track adjustment is not warranted even though the blades appear to be slightly out of track, if a one per revolution vibration is not present. Out of track condition could be caused by:

   a. Damaged main rotor blade trailing edges.

   b. Main rotor blade static balance beyond tolerances.

2. Worn or loose control rod end bearings.

   If the vibration is present in a hover only, the cause could be the same as item 1. as well as:

   a. Main rotor blade dynamic balance beyond tolerances.

Ground Roll.

Ground roll is a one per revolution lateral roll of the helicopter which often occurs during rotor engagement, and is due to the in-plane misalignment of the main rotor blades, causing an out-of-balance condition in the main rotor system. When the rotor attains flying speed, centrifugal force normally aligns the blades and the vibration disappears. If the vibration continues with the rotor up to speed at flat pitch, but disappears when the helicopter is lifted into a hover, then the cause could be as follows:

1. Static balance of main rotor blades.

2. The landing gear struts need servicing.

2/3 Times Main Rotor Speed (2/3 Per Revolution).

In flight conditions that result in high main rotor blade flapping angles, a condition of negative pitch lag coupling can occur in which the capability of rotor system damping is exceeded. This condition, called pitch lag instability, is felt as a heavy lateral rotary oscillation which can become increasingly violent if airspeed is allowed to build up or N_r is further decreased. It is not desirable to remain in this condition. Immediate corrective action is to lower the collective, increase rotor speed, and reduce airspeed and/or the severity of the maneuver.

Flight conditions under which pitch lag instability may be encountered are as follows:

1. At forward CG loadings.
   a. High forward speeds.
   b. Right sideward flight or hovering in a right crosswind.
2. At any CG loading.
   a. Exceeding allowable forward speed.
   b. High gross weight.
   c. Low rotor speed.
   d. Steep turns, level or climbing.
   e. Gusty wind conditions.
   f. Abrupt pull up from a dive.

If 2/3 per revolution vibrations should be experienced during the above conditions, it should be noted in Form 781.

Tail Shake.

Tail shake, sometimes erroneously referred to as two per revolution vibration, is an aerodynamic effect of the tail rotor passing through the disturbed air of the main rotor system in certain flight regimes. This vibration will be felt as a random impulse around the yaw axis. The trailing position of the tail rotor relative to the main rotor head, resulting from flying the helicopter in a right slip, can induce this vibration in the speed range of 50-80 knots, especially with an aft center of gravity loading.

TAIL ROTOR BUZZ.

Under certain flight conditions, a medium frequency tail rotor vibration may occur. The pilot can identify this condition, commonly referred to as tail rotor buzz, by a loud buzzing sound and simultaneous airframe vibration originating in the tail rotor area. Buzz has not been encountered in forward flight. Buzz can occur in hovering flight regimes in which the relative wind is from the right. Susceptibility to buzz increases with increases in tail rotor blade pitch, relative wind velocity, helicopter gross weight, main rotor rpm, and relative wind direction (increasing from 020 to 060, diminishing after 100 degrees).

Critical relative wind directions result from several flight regimes. Some examples are listed:

a. Crosswind hover, wind from right, front quarter.

b. Right sideward flight while headed into the wind, the combined velocities producing a resultant wind from the right front quarter.

Recovery from buzz may be achieved by reducing the tail rotor blade pitch or changing the relative wind direction. Applying right tail rotor pedal produces the most rapid recovery, as it simultaneously reduces tail rotor pitch and brings the relative wind direction toward the nose of the helicopter. Lowering the collective pitch decreases the buzz vibration and might be employed when external clearance precludes turning right and when altitude permits a descent and/or landing.

MEDIUM FREQUENCY VIBRATIONS.

Five Times Main Rotor Speed (Five Per Revolution).

The five per revolution vibration is caused by the dynamic response of the main rotor blades to asymmetrical aerodynamic blade loading. Its intensity is greatest at high forward speeds, at low gross weights, and during transition to a hover at high gross weights. It is felt in transition to a hover as a steady vertical shake caused by the main rotor blades traversing the downwash of preceding blades. This is normal to the helicopter when felt at the point where the collective pitch is increased to sustain the hover, or when hover taxiing the helicopter into and out of translational lift. The effect can be reduced in transition to a hover by leveling the helicopter just prior to applying collective pitch, and by planning the approach so that final pitch application at a slow rate will be sufficient to attain the hover. At high speeds, the difference in the lift distribution between the advancing and retreating main rotor blades results in heavy vibratory loads on the rotor head as the spanwise center of lift of each blade moves in and out. It is felt as a combination of vertical and lateral shake at the same frequency. The bifilar absorber assembly is designed to counteract the four per revolution inplane force (in the plane of rotation of the rotor head). The five per revolution vibration levels in the cockpit and cabin have been substantially reduced in the vertical as well as lateral levels by the reduction of the airframe torsional response obtained by reducing the four per revolution in plane force. Once properly tuned the bifilar absorber assembly will remain in a tuned condition over a wide range of rotational speeds. The lateral portion of the vibration is often reflected in the left tail rotor pedal or the copilot's collective pitch stick. Lateral vibration will usually decrease as N_r is increased. If five per revolution vibration is excessive at high forward speeds, an attempt should be made to distinguish between vertical and lateral vibration. Fly the helicopter to maximum speed to see if the visual cues of lateral vibration are noticed. A possible cause could be improper torque of the main gear box tiedown bolts. The type of vibration felt should be noted in Form 781. If visual cues of lateral vibration are noticed, beginning at some airspeed above 100 KIAS, and worsen as airspeed increases, the ramp may be at fault. Reduce airspeed below 100 KIAS, lower the aft ramp to the extent of the safety cables, and then fly the helicopter to maximum speed. If the vibration is noticeably reduced, then the cause is the ramp interfering with normal fuselage motion. Note the type and cause of vibrations in Form 781. If lowering the aft ramp in flight does not reduce the vibration sufficiently, and the vibration is encountered randomly in various flight regimes, the tail rotor may be the cause.

One Times Tail Rotor Speed.

Vibration (caused at 1243 cycles per minute at 100% N_r) is usually due to tail rotor blade pattern dissymmetries and is not easily identifiable by the pilot because of its proximity to five times main rotor frequency. It is evidenced by an increase in overall helicopter vibration. Since this frequency is close to five per revolution (1015 cycles per minute) the two frequencies sometimes modulate (beat) at a frequency of 228 cycles per minute, which is felt as a shudder throughout the helicopter and is hard to distinguish from one per revolution (203 cycles per minute). When excessive vibration is suspected in all regimes of flight, it should be noted in Form 781.

HIGH FREQUENCY VIBRATIONS.

High frequency vibrations may be felt as a tingling sensation in the soles of the feet or a tickling in the nose. In extreme cases, the instrument pointers will appear to be fuzzy. High frequency vibrations will normally emanate from the engine, main gear box input section, or tail rotor drive system, and are often equally apparent in a ground run as in flight. The most important cue, by far, to high frequency vibration will be the associated sounds. If a crewmember is available in the cabin area, he can assist in locating and defining vibrations as well as visually monitoring the tail rotor tip path plane.

Tail Rotor Drive Shaft Vibrations.

Generally, these vibrations are caused by an imbalanced drive shaft or bad bearings. These vibrations can be identified during ground run by feeling the fuselage.

Main Gear Box Vibrations.

The main gear box contains many possible sources of high frequency vibrations such as the various gear box mounted accessories, the accessory gear train, the APU and APU clutch, oil cooler blower, and the input bevel gear and freewheeling units. These vibrations are generally heard rather than felt in the airframe. Combinations of these high frequency vibrations in extreme cases could result in the pilot sensing low or medium frequency vibrations. These would be detected as vibrations which are affected only by variation in main rotor speed, and may be just as apparent in a ground run as in flight. There are also numerous gear clash sounds that occur under various conditions, the acceptability of which can only be determined by experience or measurements by instrumentation.

Engine Vibrations.

The engine gas generator and power turbine will normally beat together at various N_g and N_f combinations, or with N_f split off from N_r. To the pilot, the only obvious evidence of excess vibrations will be greatly increased high pitch noise levels. If the magnitude appears abnormal, it is well to check alignment of the power turbine or high speed main gear box input shaft and conditions of engine mounts. It is often possible to reduce this vibration by rotating the engine main drive shaft in relation to the main gear box input shaft in 90° increments until in a ground run the vibration is diminished. If the noise level of one engine seems excessive compared to the other engine at the same power condition, and if the excessive noise varies, with N_g or N_f changes and is perhaps accompanied by a tingling vibration in the engine control levers, then a bad engine bearing or rubbing compressor blades may be indicated. Listen carefully to the engine during normal shutdown. Any unusual noises during coastdown after the speed control has been shut off might require an engine change. Note any engine vibrations in Form 781.

MANEUVERING FLIGHT.

The high degree of maneuverability makes it possible to execute many maneuvers such as hovering, vertical takeoffs, and vertical landings which permit operations from extremely small areas.

DIVING.

Blade stall limitations are especially critical during recovery when G load factors are increased. The recovery from a dive should be made very shallow to prevent the occurrence of blade stall.

FLIGHT WITH EXTERNAL LOADS.

The helicopter has no unusual characteristics when carrying an external load, except in strong or gusty winds when the cargo may tend to swing.

WARNING

External loads which have aerodynamic characteristics may cause oscillations to the extent that the load may oscillate into the rotor blades and/or fuselage. Oscillations can usually be controlled by slowing the forward speed of the helicopter or entering a coordinated turn so that centrifugal force will aid in re-centering the external sling load.

TAIL ROTOR EFFECTIVENESS.

Exceptional circumstances may sometimes lead to power-on operation between 98 and 100% N_r. Below 98% N_r, a number of factors may work in combination to cause a reduction in yaw control authority available to the pilot. Yaw control authority (Tail Rotor Thrust Available) is directly influenced by main rotor rpm, cyclic, collective and rudder pedal inputs, power settings, flight path, wind, and density altitude. Generally, the most important factor affecting yaw control authority will be main rotor rpm. Consequently, when at or near maximum engine power settings, at high density altitudes, yaw control authority will be lost at some point as main rotor rpm is progressively decreased below 98% and the aircraft will yaw to the right. (The point at which yaw control authority is lost is variable and dependent upon the factors listed previously. Loss of yaw control authority is insidious in nature because of rudder pedal inputs made by the AFCS.)

When the engine is operating at or near topping, the application of any control requiring an increase in torque will cause the main rotor rpm to decrease. Control applications that cause this effect are increasing collective, left tail rotor pedal and rapid aft cyclic.