AIRFRAME

I. GENERAL DESCRIPTION

The enclosed figures and diagrams of the AH-56A will aid in visualizing the general arrangement and structural characteristics. These figures include the major dimensions, the manufacturing breakdown, the service areas and access provisions.

The AH-56A compound helicopter airframe serves as a functional aerodynamic enclosure or platform, containing various systems which make up the weapons system as a whole. Its major purpose is providing the best possible means of delivery, bringing to bear against an enemy the complete weapons system, and at the same time providing best possible protection for that system as a whole.

Use of common alloys assure that most repairs can be made in the field using available materials and skills. Major structural assemblies are designed to be replaceable. The aft fuselage section is attached by a series of bolts through a bulkhead ring and quick disconnects on cables. Fluid lines allow rapid removal of the entire assembly for repair, replacement, or if necessary, to facilitate air shipment of the vehicle, or for crash retrieval. Wings and stabilizers are replaceable assemblies, utilizing simple and straight-forward mounting provisions and techniques. Absence of movable control surfaces in these assemblies, with their attendant cabling and piping, further simplifies replacement procedures. Major secondary structural fairing components also are readily replaceable, such as: leading and trailing edges of wings and stabilizers; upper fuselage fairings and engine cowling; plus all doors and secondary paneling.

Semimonocoque construction is used throughout the fuselage, consisting primarily of two upper, and two lower longerons, intermediate stringers, and skin. Longerons are aluminum extrusions, and stringers are made from aluminum alclad sheet. Skins are also alclad aluminum sheet.

Functional service centers, for the most part, are housed above the primary structure (WL/116) in secondary fairing structure. Honeycomb construction is used throughout the secondary structure and is designed to be easily replaced if damaged. Multiple structural elements are utilized in strategic locations, such as in the torsion-box main frame forgings, to provide redundancy.

II. COMPONENTS AND LOCATIONS

Name of Component	Number per Aircraft	Location in Aircraft
Forward Fuselage	1	FS/50.9 to manufacturing break at FS/284
Midfuselage	1	FS/284 to disassembly joint at FS/455
Wings, Left & Right	2	Attached to left and right carry-through structure at FS/284 and FS/317, at BL/50.5 L&R
Aft Fuselage & Empennage	1	FS/455 to FS/667, includes ventral, left and right horizontal stabilizers

III. MAJOR COMPONENT DESCRIPTION

A. Forward Fuselage

The forward fuselage extends from the nose (FS/50.9) to the manufacturing break (FS/284), and is semimonocoque construction. Closely spaced stringers, coupled with numerous compartment bulkheads, covered with alclad aluminum skin, imparts a high degree of rigidity to this fuselage section. Rigidity of this assembly is a design necessity to assure gunnery errors are not induced due to structural deflection between the copilot/gunner's sighting station and the two turrets.

The forward portion of each left and right sponson, which is of honeycomb construction, imparts additional torsional stiffness to the primary structure of the forward fuselage. Additionally the sponsons serve as built-in maintenance platforms, house fuel tanks air conditioning package, external power receptacle, and provide aerodynamic forward fairings for the retracted main landing gear.

The forward fuselage is entirely compartmentized into functional service centers with the inherent strength associated with bulkhead construction. In general, compartments are provided with doors containing quick releases and hold-open devices.

Forward fuselage compartments include the XM-51 40 mm grenade launcher in the nose, with quick access provisions to the turret mechanism for maintenance or replacement. The belly turret, between stations 185 and 211, houses the XM-52, 30 mm automatic gun and is easily accessible for maintenance.

The ammo compartment, directly aft of the belly turret, contains two ammo drums. The forward drum supplies the nose turret, while the aft drum contains ammo for the belly turret.

Directly above the ammo compartment on the left side is the electrical load center; opposite, on the right side, is the main avionics compartment.

Immediately forward of the ammo compartment is the battery and flight control linkages compartment.

The canopy, housing the flight crew stations, is constructed of clear acrylic plastic, with windshields of glass. The individual panels are framed in aluminum extrusions. Access to the front copilot/gunner's station, and to the aft pilot's station, is provided by gull-wing type doors on the R.H. side. Mechanical springs provide assistance in opening the doors, and hold-open devices are included. Doors on L.H. side are used for emergency egress or provide access for maintenance.

Crew station boarding provisions include a retractable walk-way on the right side, that can be closed internally by the pilot, or operated externally by ground personnel.

Copilot/gunner's (forward) station and pilot's (aft) station contain conventional type cyclic, collective control sticks, and directional pedals. The cyclic stick requires a control lock when the vehicle is parked. Cyclic and collective control sticks in the forward station are designed to be decoupled on the ground and stowed to provide clearance during swiveling gunner's station operation. They can be quickly coupled in flight.

B. Midfuselage

The midfuselage extends from the manufacturing assembly joint at FS/284 aft to the disassembly tension joint at FS/455. The midfuselage, like the forward fuselage, is semimonocoque construction. The four major longerons are utilized, together with stringers and bulkhead rings. Skins are principally aluminum alclad with fiberglass faced honeycomb panels used adjacent to the fuel tank.

The heart, or foundation, of the primary load carrying structure is the torsion-box section. This section consists of two forged aluminum alloy main frame bulkheads located 33 inches apart at FS/284 and 317. The frames are structurally connected through the longerons and are further rigidized with conventional webs, which also provide compartment separation. Cross-sectionally, the frames conform to the entire circumferential contour of the fuselage. The frames extend to the outboard limits of either sponson, where main landing gear, and wing mounting provisions are provided, while the transmission is mounted on the upper portion of the frame. This assembly forms the torsion-box which provides a stress path carry-through structure for absorbing and spreading wing, rotor, and landing gear loads.

The major sponson sections are assembled to the midfuselage section. They are constructed of ring and stringer stiffeners, skinned conventionally. Sponsons provide a fairing for the retracted main landing gear, the APU, and various other smaller components. Also of vital importance to the mechanic is the fact they provide comfortable work platforms for all top-side service centers. Boarding ladders are incorporated into the trailing edge of each sponson. The left one is manually operated, while the opposite one is electrically actuated automatically by the ground/air safety switch position.

Entrance is gained to the flight control servo package through a door on the underside of the fuselage directly between the two main frame forgings.

Swashplate access is gained through doors at either side of the fuselage below the transmission, and directly between the main frames.

The 300 gallon self-sealing fuel tank is located under the steel deck separating the engine compartment from the fuel tank compartment. Entrance is gained through a removable bottom panel, which exposes the entire tank, and a manhole for internal access.

Just aft of the fuel tank is the aft avionics bay, and doppler antenna with access doors through the lower skin. This area also provides access to the disassembly tension joint, consisting of four major bolts, fluid line disconnects, and drive shaft coupling.

The fairing structure atop the fuselage houses the accessory compartment, transmission, engine air inlet plenum area, and engine. Access doors permit easy entrance into these areas, or the entire fairing structure may be removed to facilitate replacement of major components. Part of this overall fairing consists of the sliding engine cowl, which can be opened or closed by one man from either side.

C. Wings

The wings, which have a total of 195 square feet in area, serve as the prime lifting surfaces during high-speed forward flight, which greatly reduces the load on the main rotor. Major wing components are the main box beam section, leading edge, trailing edge, and wing tip. Each wing has provisions for two pylon stores stations accepting either fuel tanks, or various weapons. Navigation and oscillating beacon lights are installed in the wing tips. In addition, there is an electro-luminescent panel formation light flush mounted on top of each wing.

The wings are each attached to the fuselage main frame bulkheads by 4 principal bolts at FS/284 and FS/317, at BL/50.5 L & R. Inspection panels are provided to facilitate installation of the wing, as well as for maintenance of components associated with external stores.

Various construction materials and manufacturing techniques are incorporated, such as the two layer aluminum bonded one piece leading edge assembly and a fiberglass honeycomb core trailing edge with aluminum skins for antenna. The tip is also of this material and is in two sections. All these components are attached to the box section or to hard structure by screws which assure relative ease of replacement.

D. Aft Fuselage and Empennage

The aft fuselage extends from the disassembly tension joint at FS/455, aft to FS/667. Semimonocoque construction is also used in this assembly, and is visually apparent because few system components are housed within its confines. This assembly attaches to the midfuselage by four principal bolts that tie the four longerons of the two assemblies together. Each of the approximately twenty stringers are joined to the midfuselage by two bolts through angle clips.

All materials are aluminum alloys. Three forged aluminum alloy frame rings are incorporated at FS/612 and FS/634 and 646 incorporating attachment provisions for the three stabilizers, (four principal bolts each), and the three-way integral gear box.

Both horizontal stabilizers are of dual-spar box beam construction. Leading and trailing edges are constructed of easily removable aluminum faced honeycomb sandwich. Tip assemblies are of glass fabric pre-pregnated epoxy material. The left hand stabilizer is slightly heavier, due to an added auxiliary beam which forms a box structure around the tail rotor drive shaft. Also adding to the weight difference is the tail rotor dual servo actuator and drive shaft installed in the inboard leading edge of the left stabilizer. Appropriate access panels are provided.

The vertical stabilizer, or ventral fin, is composed of a two-spar box beam, leading and trailing edges, and bottom fairing. This stabilizer houses the retractable tail wheel strut, flux gate valve, and VHF/FM antennas. The VOR/localizer antennas are attached to either side of the stabilizer.

The easily removable leading edge is made from aluminum faced honeycomb sandwich. The lower portion of the trailing edge houses the VHF/FM antennas under glass fabric pre-pregnated epoxy skins. A small upper portion is of metal skin honeycomb construction.

The lower tip is fiberglass fairing housing the retracted tail wheel. The tail navigation light is mounted in the aft edge.

Appropriate access provisions have been incorporated to facilitate maintenance of internal components, and to assist with removal/installation of the major assemblies.

IV. OPERATIONS

A. Towing

The AH-56A compound helicopter can be towed, or pushed, as the case may be, from the tail landing gear using a standard tow bar attached directly to the tail wheel axle. Offset limits of 45° from vehicle centerline should be observed, in addition to adequate clearances between towing vehicle, tow bar, and protruding aircraft components.

The tail landing gear strut must be properly inflated to provide sufficient fairing clearances, and tail wheel lock must be unlocked. If desired, the rotor brake may be released to enable positioning of the main rotor, tail rotor, or propeller to provide additional clearance of obstructions. Safety pins should be installed as applicable, parking brakes released, and a safety man provided in the pilot's station to apply brakes if necessary, before actually towing the vehicle.

B. Parking

Good judgement is important in parking the AH-56A, as is true with any aircraft, and is dependent to a large degree upon conditions likely to be encountered during the parked period while the vehicle is relatively unattended. Conditions having a detrimental effect on a parked aircraft are numerous and varied, such as: time period vehicle is to be parked, wind velocity and direction, inclement weather, proximity to other operating aircraft, enemy action, and reaction time.

All aspects have been considered in the design of the AH-56A that offer a wide latitude in parking procedures and the amount of protection which appears advisable in any given set of circumstances.

Parking brakes are provided and should be utilized, along with pedal shocks front and rear of each main wheel, especially if anticipating high winds or an extended parked period.

A rotor brake has been included, which when ON, prevents wind or other aircraft from windmilling the rotor. The brake should be utilized whenever the vehicle is parked. To determine if rotor brake is on, attempt to rotate propeller or tail rotor.

Safety pins have been provided where necessary, and should be installed when and where applicable, such as main landing gear, armament components, and external stores.

Visual verification should determine that cyclic stick control locks are engaged. Ensure at the same time that the battery switch is OFF.

Normally, it would be advisable to take advantage of the tail wheel lock as additional protection against high winds rotating the entire aircraft.

Canopy access doors should be positioned open or closed, according to weather conditions and environment. Circumstances will almost always determine to what extent the numerous protective covers and shields will be used.

C. Mooring

Mooring of the AH-56A, as with any aircraft, is usally necessitated by high winds or gust conditions. Properly moored, the aircraft will withstand winds up to 60 knots from any direction, regardless of weight, or center of gravity conditions.

Mooring requirements vary with the ground surface available. If permanent ground mooring facilities are unavailable, a GFE mooring kit is required. The aircraft has five points at which mooring lines can be attached. These points are the three mooring and jacking point fittings, located in the wings and forward fuselage and require installation of jack adapters, plus use of the two rings on the main landing gear struts.

D. Jacking

Two methods of jacking the AH-56A are provided. One method utilizes jack pads located at the lower end of each landing gear strut. The other method uses the wing, and forward fuselage jacking points.

A jack pad is provided at lower end of each gear strut which accommodates a standard low profile hydraulic jack. Landing gears may be jacked individually to facilitate maintenance, observing above caution. All three landing gears may be jacked simultaneously, keeping vehicle level for purposes of weighing, alignment, and symmetry inspections, or other maintenance as desired.

In the course of performing more extensive maintenance, the vehicle can be jacked at the wing and forward fuselage jacking and mooring positions. The fly-away kit adapters and standard hydraulic tripod jacks are required. The adapters screw into clearly marked fittings in aft wing spars and bottom of forward fuselage. The fittings are plugged with a set screw when not in use.

E. Leveling

Maintenance requiring leveling of the AH-56A has been greatly simplified by including plumb bob provisions. A hanger at the top of the fuselage at FS/442.30 is provided to which the plumb bob string is attached. Directly below the hanger on the bottom of the fuselage, is a plate upon which a cross has been inscribed. String length is adjusted so that plumb bob just clears plate. The vehicle is then jacked as required, either from the gear or wings/fuselage until the plumb bob exactly centers the cross. The vehicle is then in a level condition. Access to these plumb bob leveling provisions is provided through the aft avionics compartment door.

V. PRODUCIBILITY COST REDUCTION CONFIGURATION

A. Wing

Changes to the wing are those necessitated or made possible by deletion, or changing of other equipments mounted on the wing, and structural changes for cost reduction purposes. These are:

- The deletion of the AN/ARC-102 radio antenna allows installation of a new all metal trailing edge.
- Provisions for a fuel tank at B.L. 28 and B.L. 117 (R&L) are deleted. Existing structural provisions are retained.
- Provisions at the wing tips for automode antenna and ferry kit tip extensions are deleted.
- The B.L. 28 hard points are deleted.
- The wing structure is revised by changing the ribs to a fore and aft position along with other detail changes. The center section structure is also revised to reduce cost.
- The wing's leading edge is reconfigured to provide droop to obtain higher C_L.

B. Fuselage

The cable cutter provisions are deleted, and the forward ammo bay door attachment has been revised. The hardpoints at B.L. 28 in the mid fuselage are deleted by existing structural provisions in the main frame are retained.

TABLE 1-1. SERVICE COMPARTMENT LOCATIONS AND ACCESS

Compartment or Area	Type and Location of Service Access	Type of Fasteners	Major Equipment
Nose Turret	Removable fairing on top of nose	19 camloc fasteners	Gun feed chutes and mechanism
	Removable fairing on turret	4 camloc fasteners	Gun and turret operating mechanism
Cockpits	Clamshell doors	Bolt latches controlled by handle on lower frame	Copilot/gunner and pilot stations
Swivelling Gunner	Sight head - Rotating bowl shaped cover under fuselage below copilot/gunner station	Solenoid operated latch which may be manually released by pressing with drift punch through hole near top edge to right of sight window opening	Sight head window and vent dessicator unit
	Upper sight elements -- Accessible from copilot/gunner station	------	All elements above the turntable and attachments for sight head
Battery	Connection and disconnection - Access door on left side of fuselage just aft of sight head	2 quick release latches	Battery disconnect and one ICS ground station
Debris bay	Electrically operated doors in bottom of fuselage directly under pilot station	Operated by electrical actuator controlled by switches in ground power receptacle box and in pilot's station	Access to battery, flight control linkage, brake cylinders, weapons control unit

(Sheet 1 of 4)

TABLE 1-1. SERVICE COMPARTMENT LOCATIONS AND ACCESS

Compartment or Area	Type and Location of Service Access	Type of Fasteners	Major Equipment
Boarding Platform	Hinged walkway on right side of fuselage at cockpit area	1 quick release latch	No equipment, used as access walkway to cockpit area
Ground Power Receptacle	Hinged panel in forward end of left sponson	1 quick release latch	Ground power receptacle, debris bay door switch, and ground power monitor switch
Air Conditioning Unit Compartment	Panel on forward end of right sponson	31 semi-quick fasteners	Environmental control air conditioning package
Belly Turret	Removable fairings on and around turret	Camloc fasteners, screws and quick release fasteners	Gun and turret mechanisms
Ammunition Compartments	Hinged access panels	Forward panel - 3 quick release latches. Aft panel - 18 semi-quick fasteners	Nose gun ammunition in forward and belly gun ammunition in aft compartment
Electrical Compartment	Double hinged panel on upper left side of fuselage just aft of cockpit area	5 quick release latches	Electrical system relays, transformer/rectifiers, and supervisory/regulators
Main Avionics Compartment	Double hinged panel on upper right side of fuselage just aft of cockpit	5 quick release latches	Central computer complex, ADF, Inverter and associated items

(Sheet 2 of 4)

TABLE 1-1. SERVICE COMPARTMENT LOCATIONS AND ACCESS

Compartment or Area	Type and Location of Service Access	Type of Fasteners	Major Equipment
Accessory Compartment	Hinged panels on upper fuselage on both sides just aft of electrical and avionics compartments	5 quick release latches on each panel	Hydraulic power packages, alternators, hydraulic fill unit, forward end of main transmission
Swash Plate Compartment	Hinged panel on either side of fuselage just above sponson and centered on rotor mast	30 semi-quick fasteners on each panel	Swash plate and control rod ends
Controls Servo Compartment	Hinged panel in bottom of fuselage directly below rotor mast, also through cutouts in floor of swash plate compartment	14 semi-quick fasteners on bottom panel	Flight control servos
Engine Compartment	Sliding cowl on top of fuselage aft of rotor mast	2 quick release latches with camloc safety locks	Engine and engine accessories
Auxiliary Power Unit Compartment	Panel inboard aft end of left wheel well and hinged panel lower side of left sponson	15 camloc fasteners for both panels	Auxiliary power unit and right angle gear box
Fueling Control Panel	Hinged panel aft end of right sponson	2 quick release camloc fasteners	Pressure fueling connector, drum fueling connector, and fueling controls

(Sheet 3 of 4)

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TABLE 1-1. SERVICE COMPARTMENT LOCATIONS AND ACCESS

Compartment or Area	Type and Location of Service Access	Type of Fasteners	Major Equipment
Left Boarding Ladder	Manually operated ladder stowed in aft end of left sponson	1 quick release latch	No equipment, ladder used for boarding left sponson
Aft Avionics Compartment	Hinged panel in bottom of fuselage just forward of aft fuselage mating joint	25 semi-quick release fasteners	Self contained navigation equipment, radio navigation equipment.

(Sheet 4 of 4)

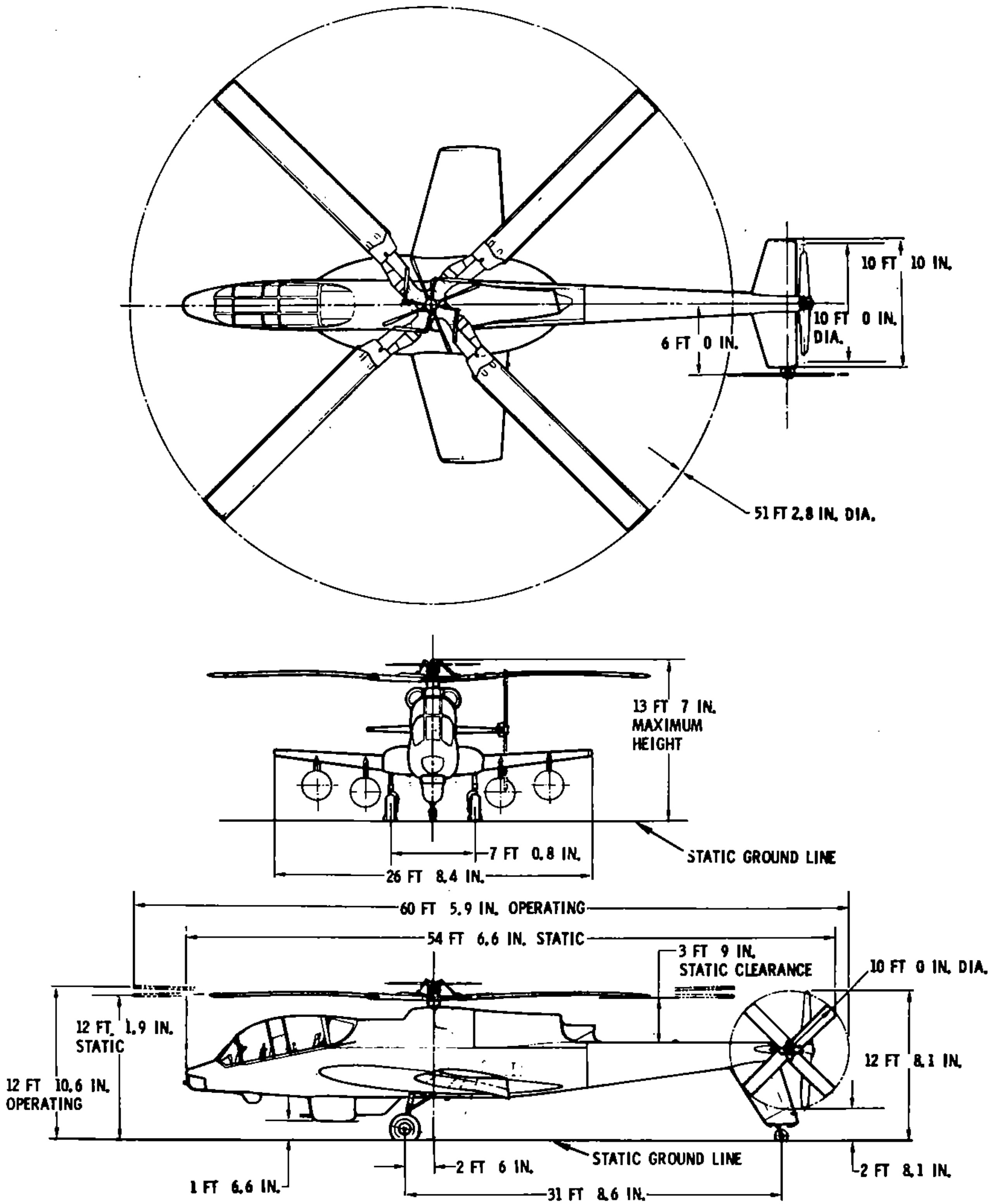


Figure 1-1. Aircraft Dimensions

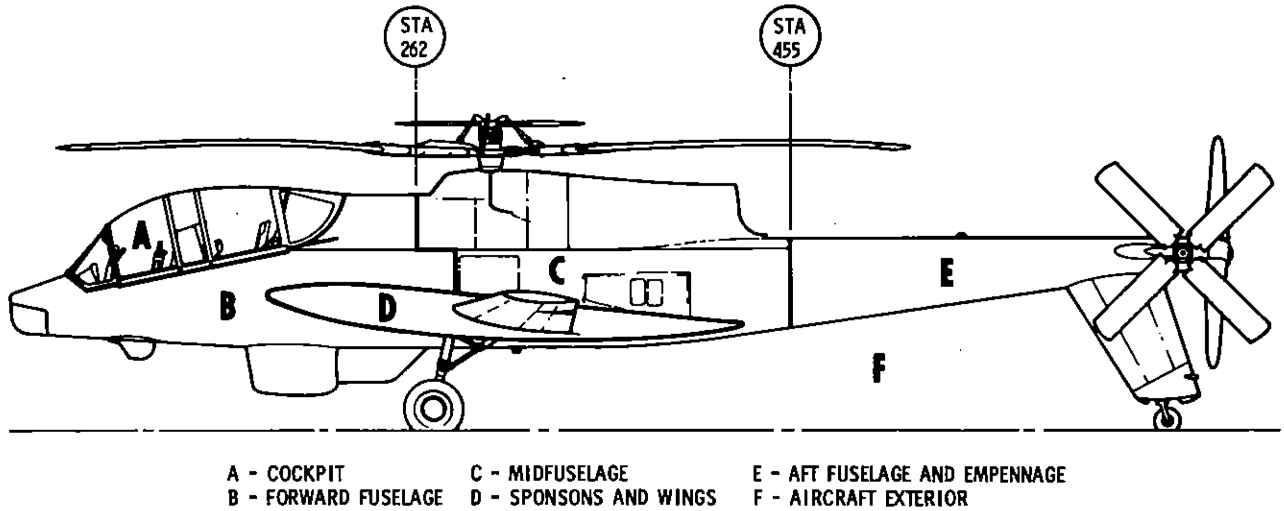


Figure 1-2. General Arrangement (Sheet 1 of 2)

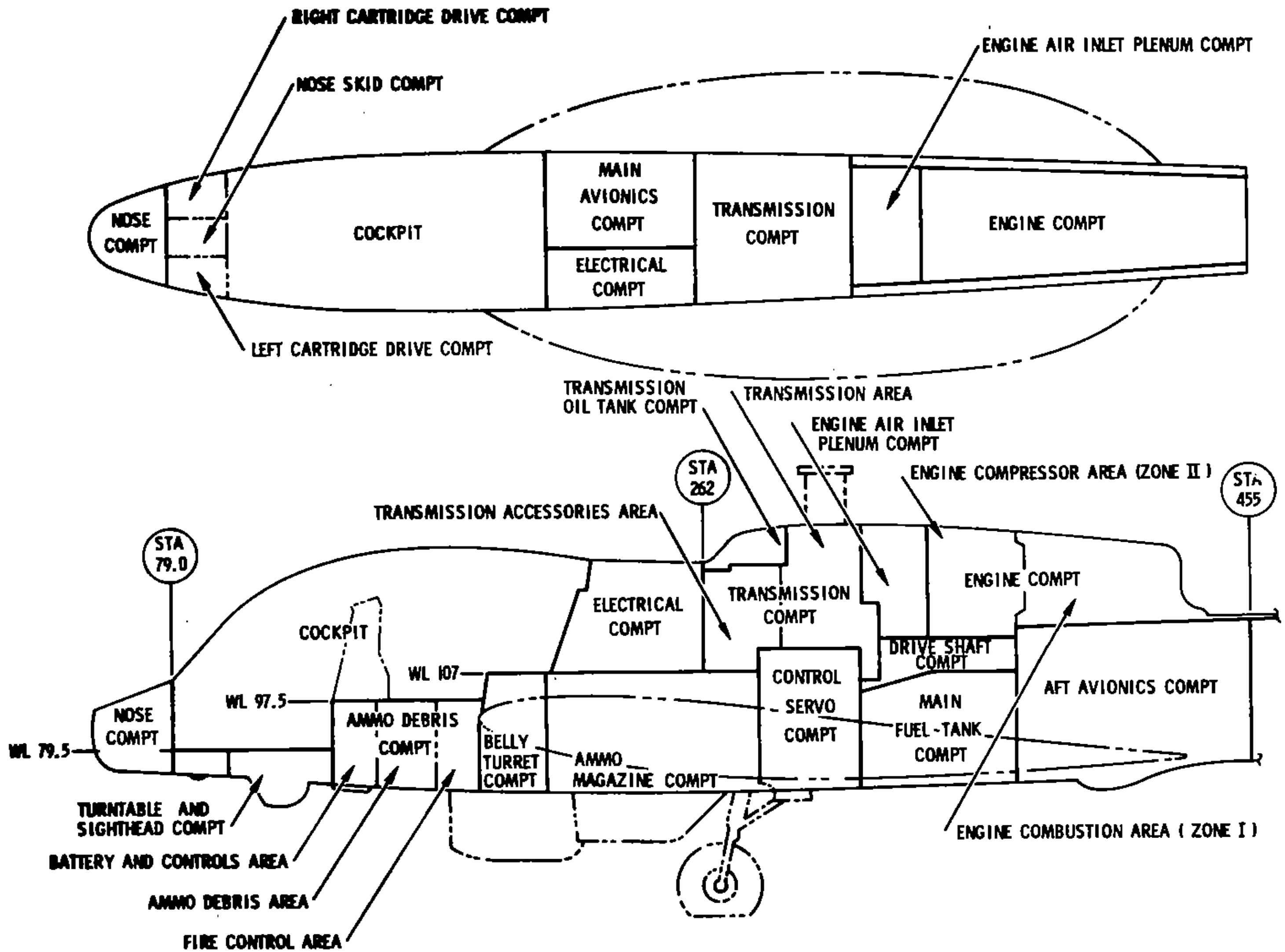
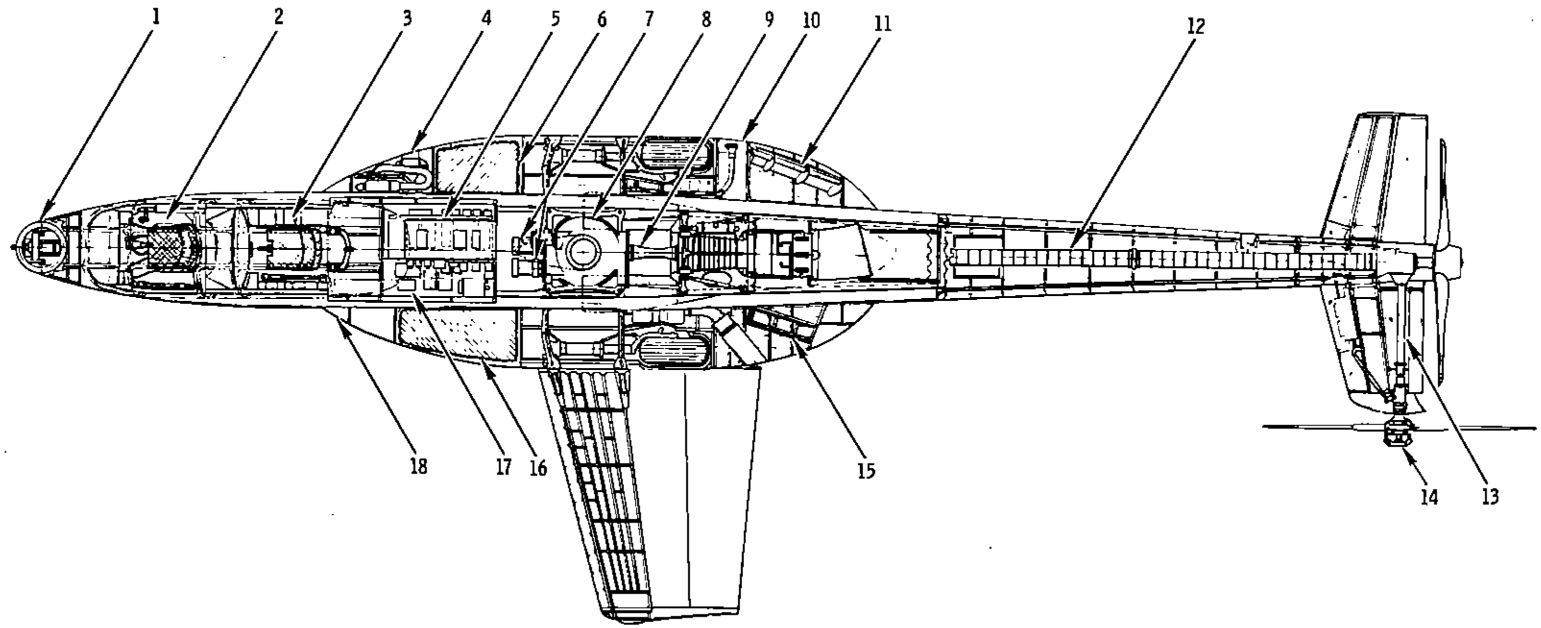
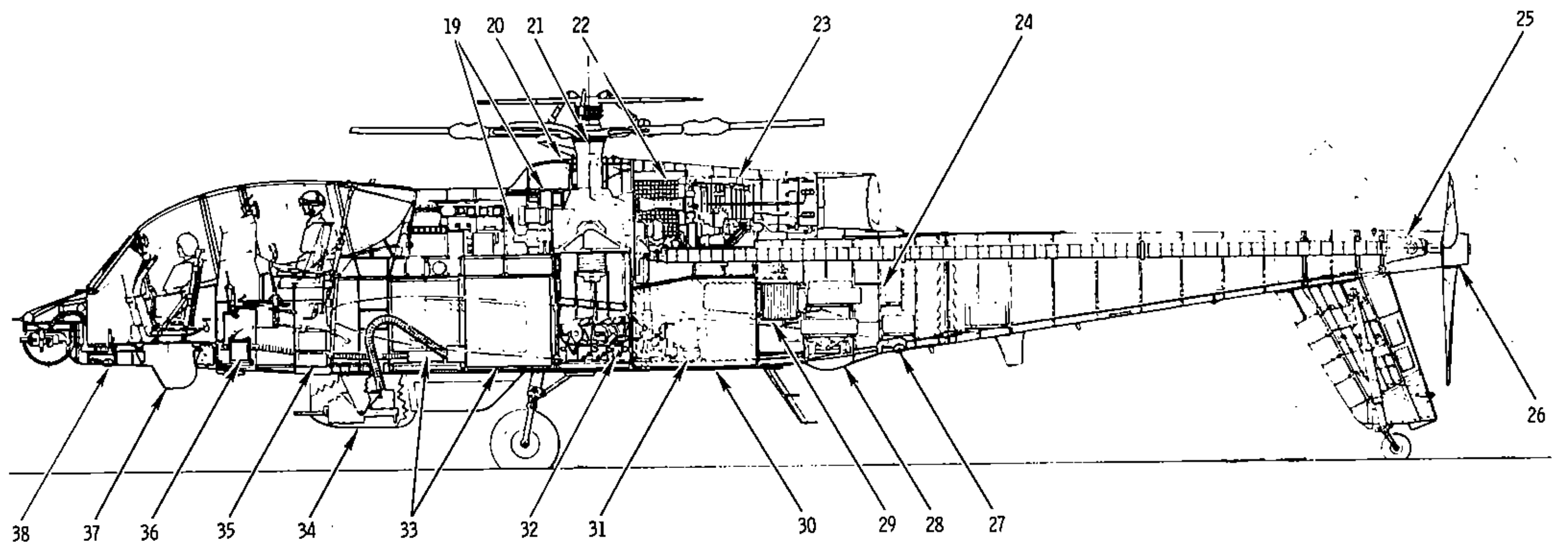


Figure 1-2. General Arrangement (Sheet 2 of 2)



1	Nose turret	10	Fueling station
2	Copilots/gunners station	11	Right boarding ladder
3	Pilots station	12	Propeller-and-tail rotor drive shaft
4	Environmental control unit	13	Tail rotor drive shaft
5	Main avionics compartment	14	Tail rotor
6	Right sponson fuel tank	15	Left boarding ladder
7	AC generators	16	Left sponson fuel tank
8	Main transmission	17	Electrical compartment
9	Torquemeter shaft	18	External power receptacle

Figure 1-3. Inboard Profile (Sheet 1 of 2)



19 Hydraulic power packages
20 Transmission oil tank
21 Main rotor mast
22 Engine air filter
23 Engine
24 Aft avionics compartment
25 Propeller gearbox
26 Propeller
27 Searchlight
28 Doppler radome
29 Oil cooler
30 Main fuel tank
31 APU
32 Main rotor servo actuator package
33 Ammo magazines
34 Belly turret
35 Fire control avionics area
36 Battery
37 SGS sighthead
38 Searchlight

Figure 1-3. Inboard Profile (Sheet 2 of 2)

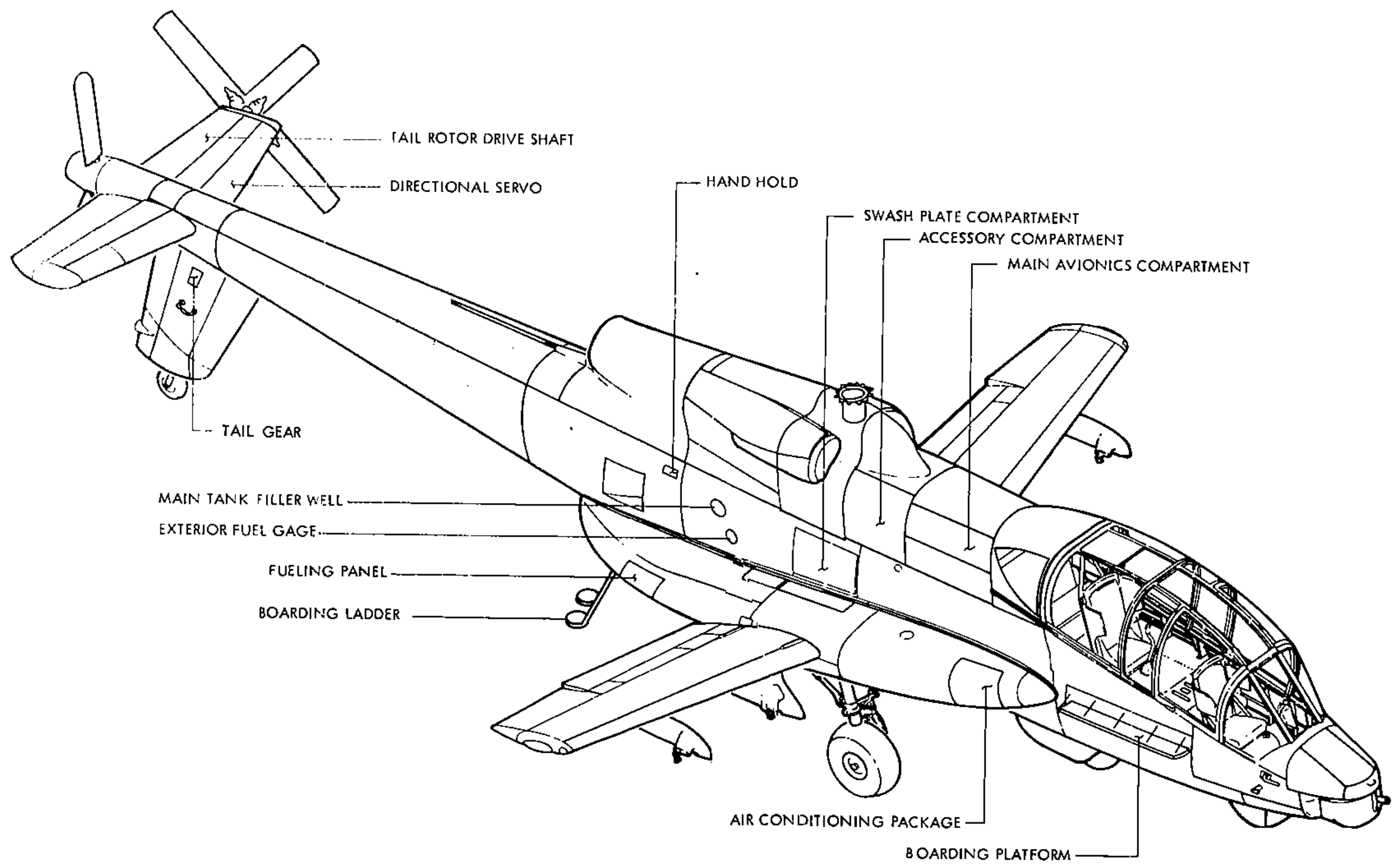


Figure 1-4. Service Areas and Access Provisions (Sheet 1 of 2)

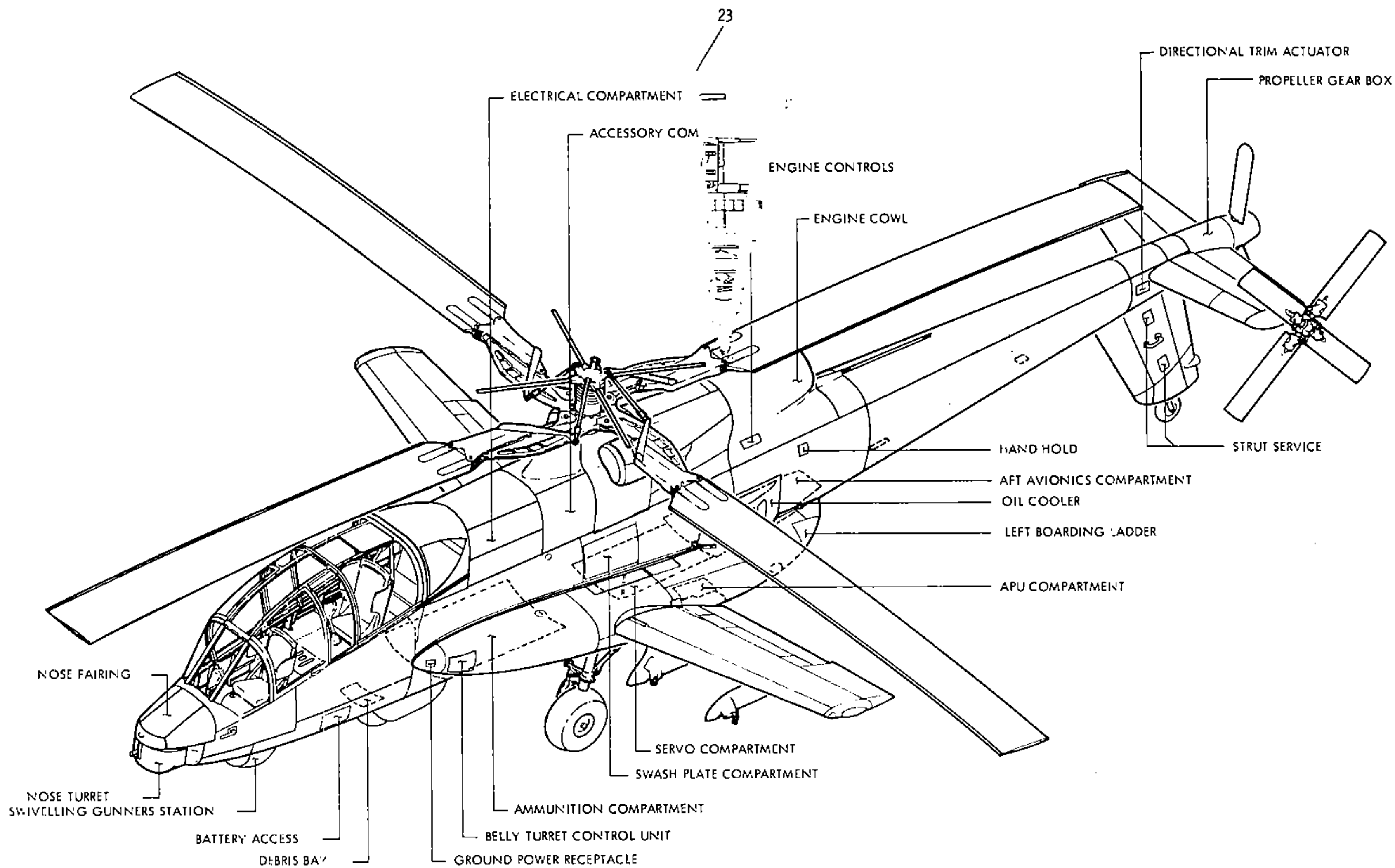


Figure 1-4. Service Areas and Access Provisions (Sheet 2 of 2)

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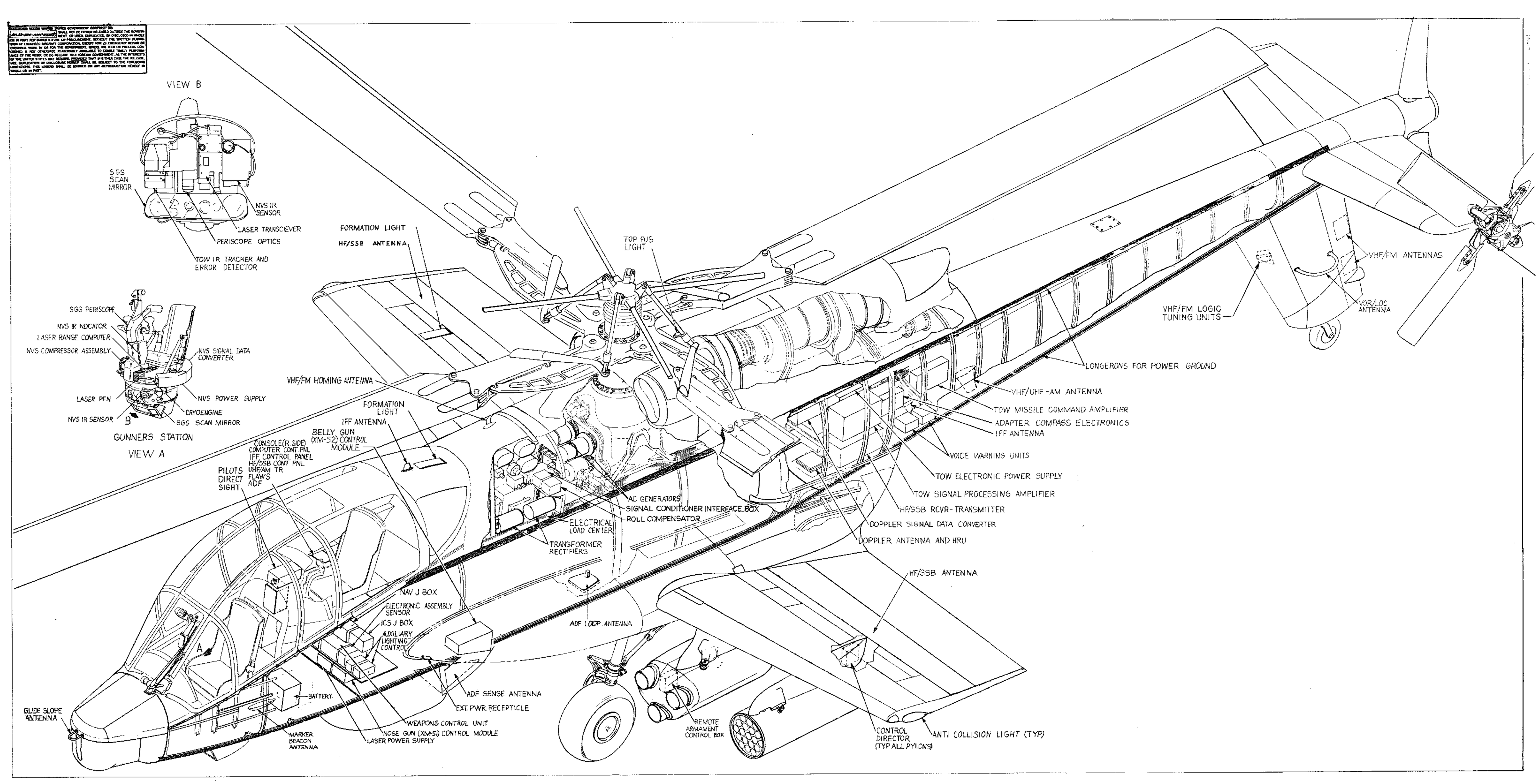

Figure 1-5. Equipment Installation
AH-56A Left Side
(Sheet 1 of 2)

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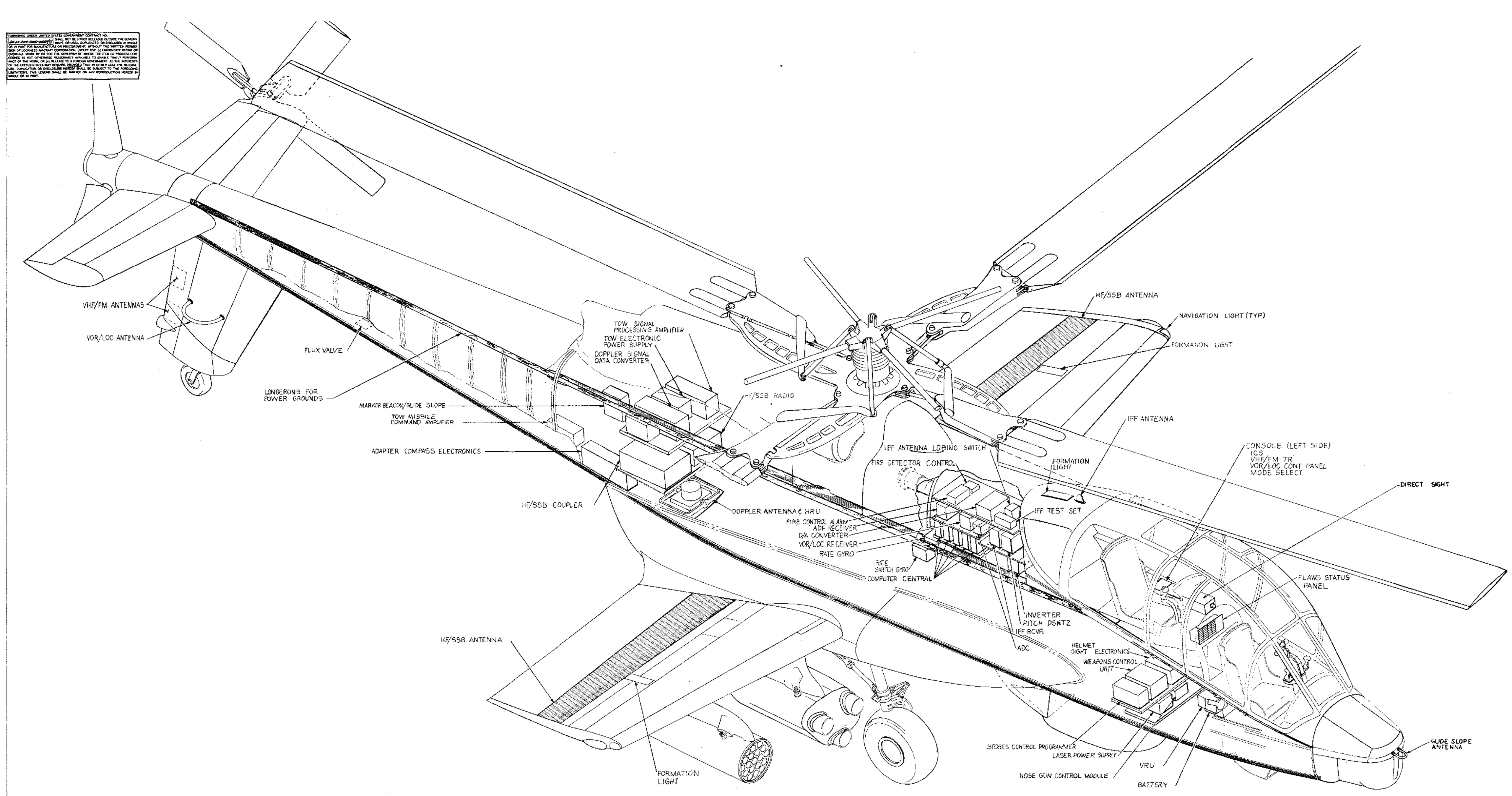


Figure 1-5. Equipment Installation
AH-56A Right Side
(Sheet 2 of 2)

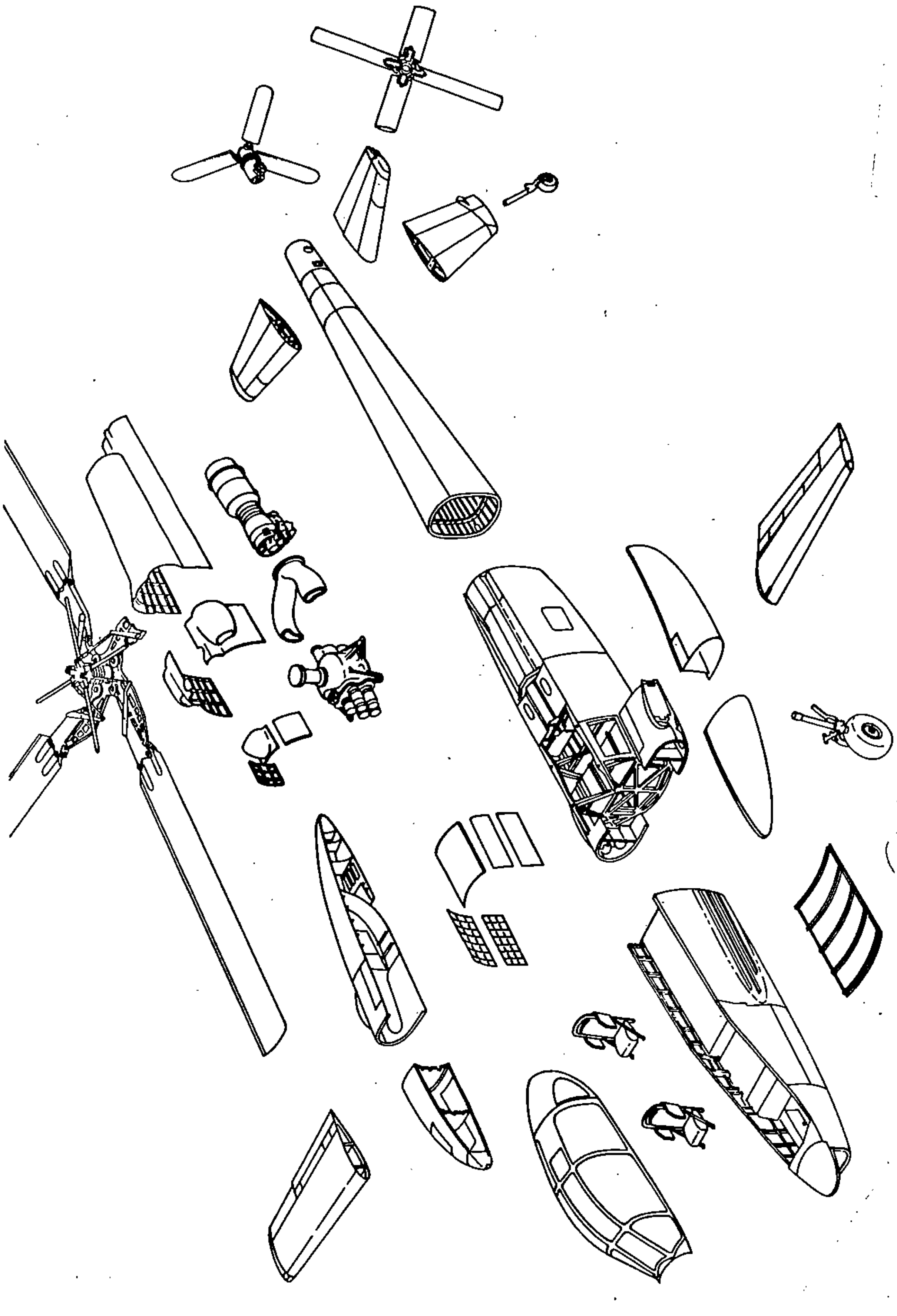


Figure 1-6. Aircraft Structural Breakdown

CREW STATIONS

I. GENERAL DESCRIPTION

Crew accommodations are provided in a tandem arrangement for a pilot in the aft station and a copilot/gunner in the forward station. The aft station floor is raised 18 inches above the forward station floor to provide maximum visibility for the pilot. The cockpit interior is insulated to reduce noise levels and provide thermal protection. Hard plastic trim is installed along the interior sides to maintain a smooth finish.

Flight controls are conventional in location and installation, except that the copilot/gunner's cyclic and collective sticks can be decoupled and stowed out of the way to permit swiveling gunner station operation. Provisions for locking the controls are also provided.

The pilot's collective stick has twist grip controls for complete control of the engine and propeller. The copilot/gunner's collective stick has no provisions for engine starting, idling, or shutoff, but has provisions for inflight control of engine and propeller.

II. COMPONENTS AND LOCATIONS

Name Of Component	Number Per Aircraft	Location In Aircraft
Pilot's Seat Assembly	1	Pilot's station
Copilot/Gunner's Seat	1	Copilot/gunner's station
Rear-View Mirrors	2	Attached to each side of the canopy front frame
Map and Data Cases	2	Right-hand side of each station
Relief Tubes	2	Adjacent to seat in each crew station
Compass Correction Card	2	On glareshield in each crew station

Name Of Component	Number Per Aircraft	Location In Aircraft
First-Aid Kits	2	On aft bulkhead in each crew station
Survival Kits	2	In place of seat cushions
Fire extinguisher	1	Between crew stations
Protective Covers	8	Pitot tubes, sighthead, canopy, turrets and rotor hubs
Protective Shields	4	Engine inlet and exhaust and APU exhaust
Safety Pins	8	One for each main gear and each wing pylon

III. MAJOR COMPONENT DESCRIPTION

A. Cockpit Furnishings

Cockpit furnishings include the rear-view mirrors provided for each crew member. They are installed in brackets mounted on the forward front vertical frame member, just below the hinge line on each entry door. Two map and data cases are provided to accommodate manuals, maps, checklists, and other pertinent data. The map and data case in the pilot station is located in the aft right-hand console. The map and data case in the Copilot/gunner station is located on the cockpit right side. A hand operated fire extinguisher is accessible from either station. Two relief tubes are provided for the convenience of the flight crew. The relief tube in the pilot station is secured in clips located on the floor underneath the stowed position of the collective lever. In the Copilot/gunner station it is stowed in clips mounted on the right side of the vertical bulkhead in back of the seat. The installation consists of rubber tubing above floor lever, and polyethylene tubing below floor lever, terminating at a stainless steel overboard tube at FS 358. A compass correction card is provided for the standby magnetic compass in each cockpit on the glareshield.

A. Cockpit Furnishings

The seat cushions of each seat can be replaced with survival kits. Two first-aid kits are provided. The Pilot's kit is installed in clips mounted on the centerline of the aft bulkhead behind the seat. The kit in the Copilot/gunner station is installed in clips mounted on the left side of the aft bulkhead, behind the seat. Protective covers are provided for two pitot tubes, the external sighthead assembly, canopy, nose turret, belly turret, main rotor hub and tail rotor hub. Protective shields are provided for the two engine air inlet scoops, engine exhaust duct and the APU exhaust duct. A flagged safety pin is provided for each main landing gear and for each of the wing pylons.

B. Instrument panels consist of a forward main tee-panel, containing flight, navigation, and system monitoring instruments. In addition, left and right lower panels and consoles containing system controls at either side are provided. Instruments are integrally lighted and panels are edge lighted. In addition, flood, utility, and storm lights are provided.

C. Pilot's seat assembly is manually adjustable fore and aft and vertically. Construction is of welded tubular steel with sheet metal back and pan, and a horizontal track assembly. The back and seat pan are assembled as a single unit to the track assembly. The track contains the controls for selective vertical adjustment. Channels function as tracks for the seat pan rollers, providing fore and aft adjustable movement. Fittings at the rear of each left and right channel attach the entire track assembly to the tubular steel support structure. The seat structure is bolted to the floor and overhead cockpit structure. Both seats have removable seat and back cushions so that the seat can accommodate back pack parachutes and survival kits. Both seats are also designed to absorb crash loads to reduce crew injury and are equipped with standard safety belts, crotch straps and shoulder harnesses with inertia reels, and have provisions for mounting armor plate.

D. Copilot/gunner's seat structure is of 6061-T6 aluminum alloy frame with back plate and seat pan attached. The seat pan is removable to permit access to swiveling gunner's station components. The seat is attached to vertical steel supports anchored to the swiveling station. The seat is adjustable only vertically, and is not interchangeable with the pilot's seat.

IV. PCRS CONFIGURATION

The changes in the pilot and copilot crew station reflect the display deletions resulting from avionics and armament system changes, instrument substitutions for cost reduction and reliability improvement, and display, control, and instrument relocation to improve utility (as recommended by the TECOM pilots during an informal P/CR mockup review and by the results of the MOD P-190 EPR's). Deletions are as follows:

Pilot Station

- Range-to-target indication in BDHI
- Accelerometer
- Map plotter and provisions
- AC & DC volt-ammeters
- Power chop arm switch and light
- Ammo debris dump switch
- ARC-102 (HF/SSB) radio control panel
- Retransmit switch
- Plan position display provisions
- Vertical situation display provisions
- Manual external stores release "T" grip
- Distance measuring equipment (DME)

Copilot/Gunner Station

- Radar altimeter indicator
- Marker beacon control
- ARC-114 (VHF-FM)
- Bearing, distance, heading, indicator (subsitute RMI)
- Manual external stores release "T" grip
- Gas generator RPM indicator
- Torquemeter
- Clock

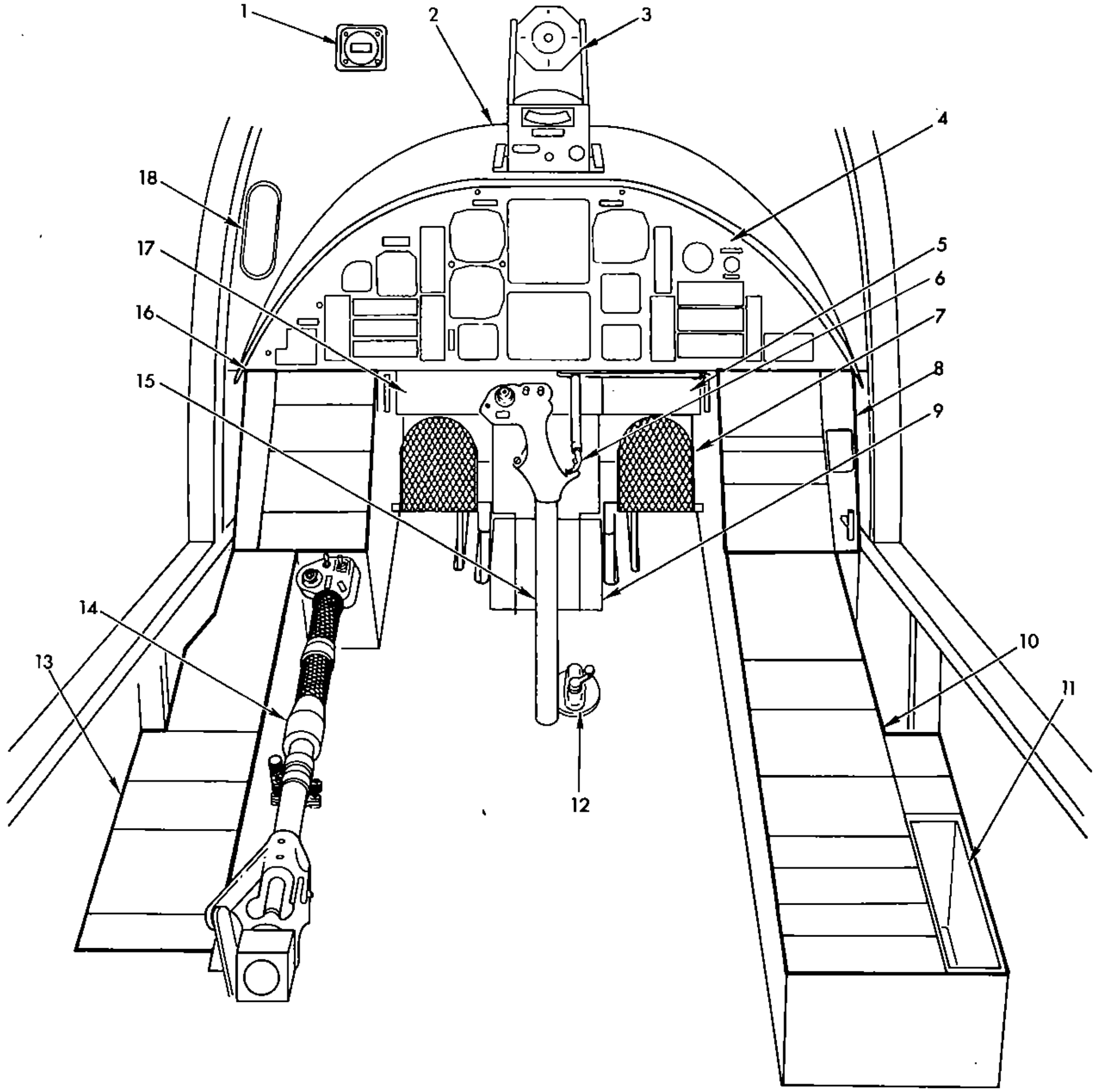
The map plotter is deleted from the pilot's station (but retained in the copilot/gunner's station) because other navigation modes are available to the pilot.

The accelerometer, AC and DC volt-ammeter and power chop arm switch are equipments shown by the MOD P-190 development program to be of marginal utility in an operational environment. The ARC-102 control panel, the ammo debris dump switch, and the vertical situation and plan position displays provisions are no longer required because of the deletion of the ARC-102 radio, the XM-53 gun system and the automodes.

Numerous switches and instruments have been simplified and/or relocated to save costs and improve cockpit effieiency. Only minor functional changes result from these simplifications. The torquemeter in the copilot/gunner's station is replaced by a 100 percent torque indicator light. The power control levers have been relocated at the side consoles.

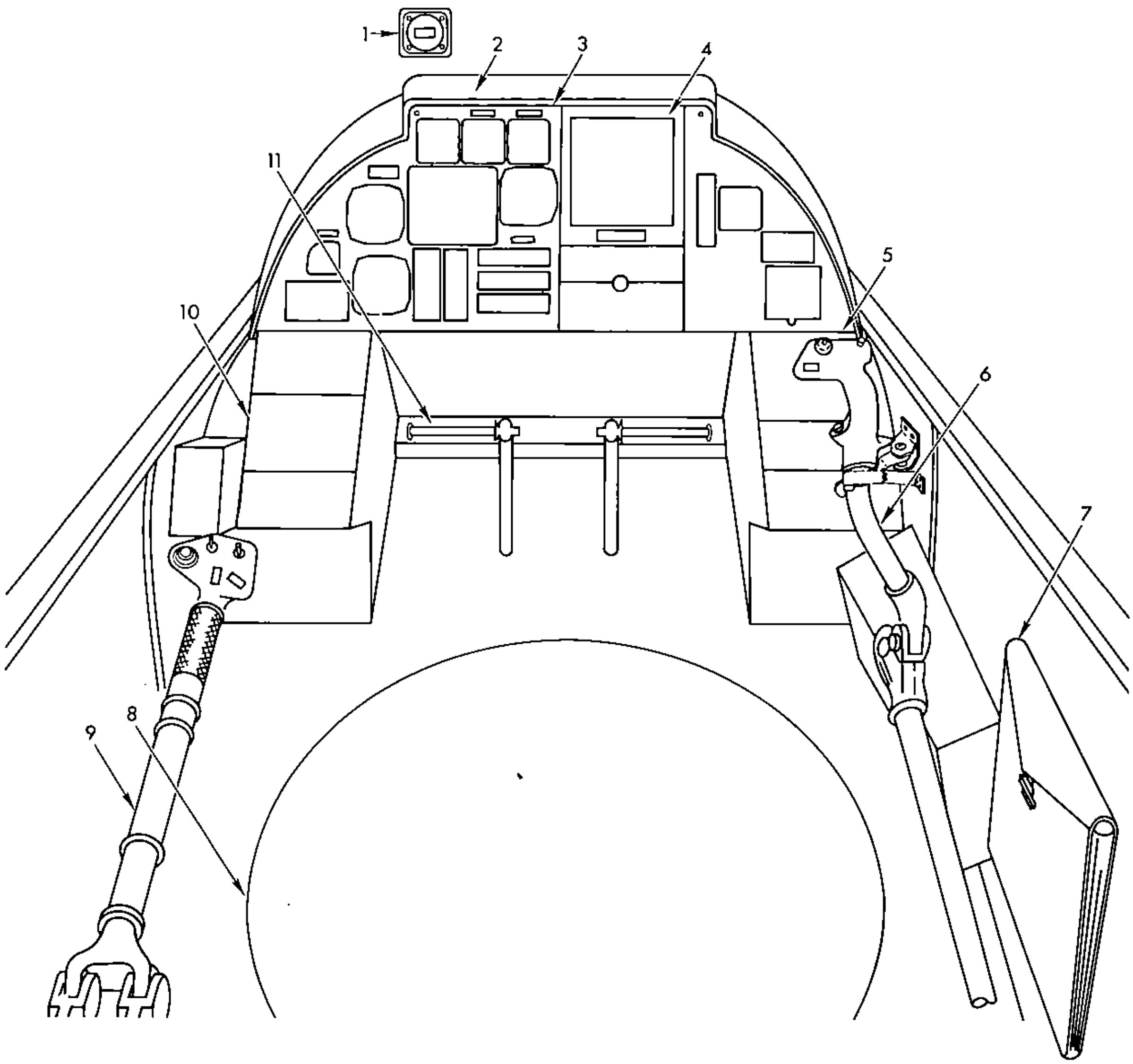
The three-axis trim is reduced to yaw trim only. Pitch and roll trim indications are obtained from the stick position.

The items deleted from the copilot/gunner's station derive from the emergency flight requirements imposed at that station. The flight instruments and navigation displays have been reduced to a minimum necessary for a safe return to base.



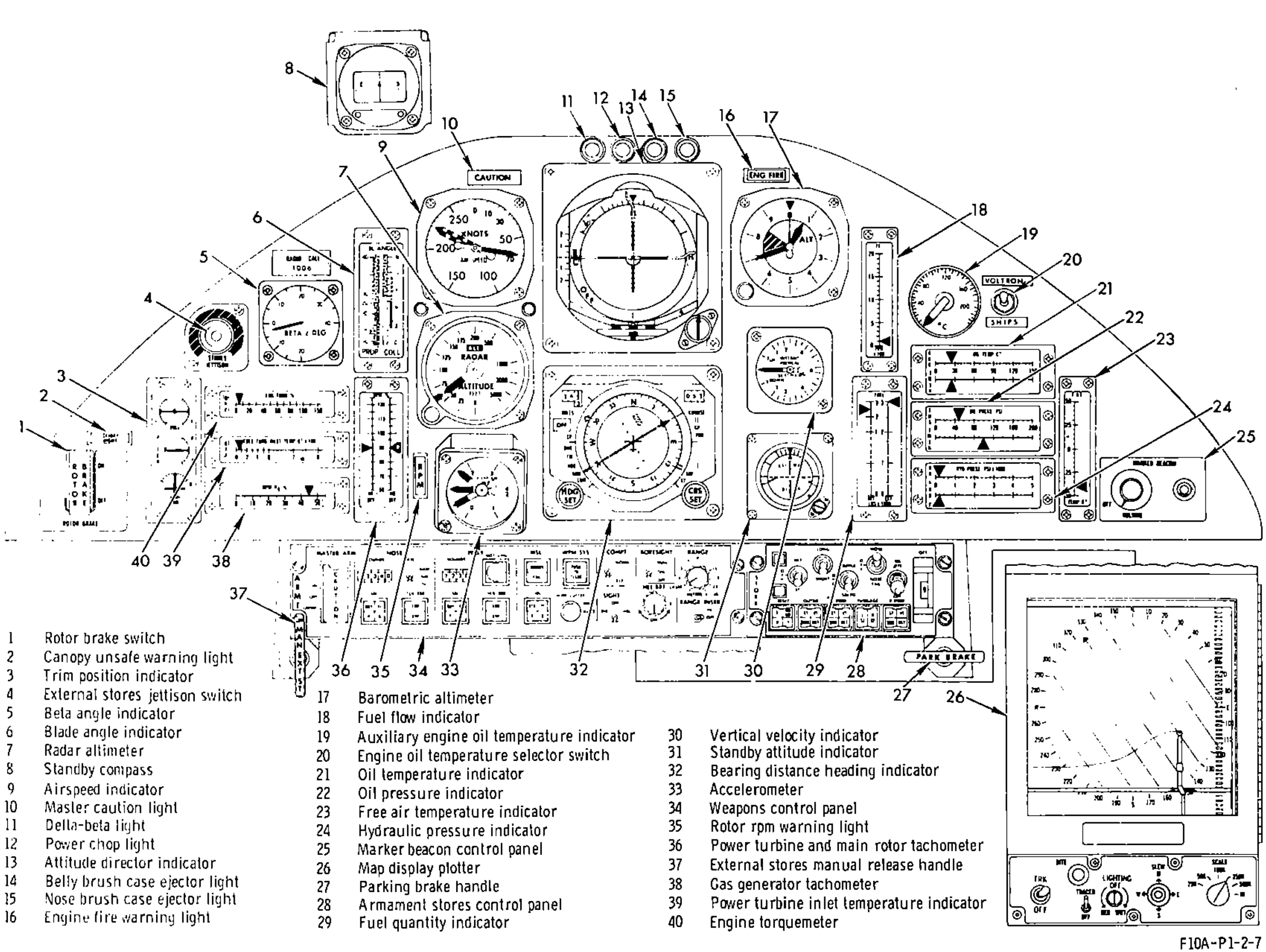
1	Standby compass	10	Right console
2	Glare shield	11	Map case
3	Fixed sight	12	Pedal adjustment crank
4	Instrument panel	13	Left console
5	Stores control panel	14	Collective
6	Cyclic control lock	15	Cyclic
7	Antitorque pedals	16	Left forward panel
8	Right forward panel	17	Weapons control panel
9	Map plotter (stowed)	18	Rear view mirror

Figure 2-1. Pilot Station



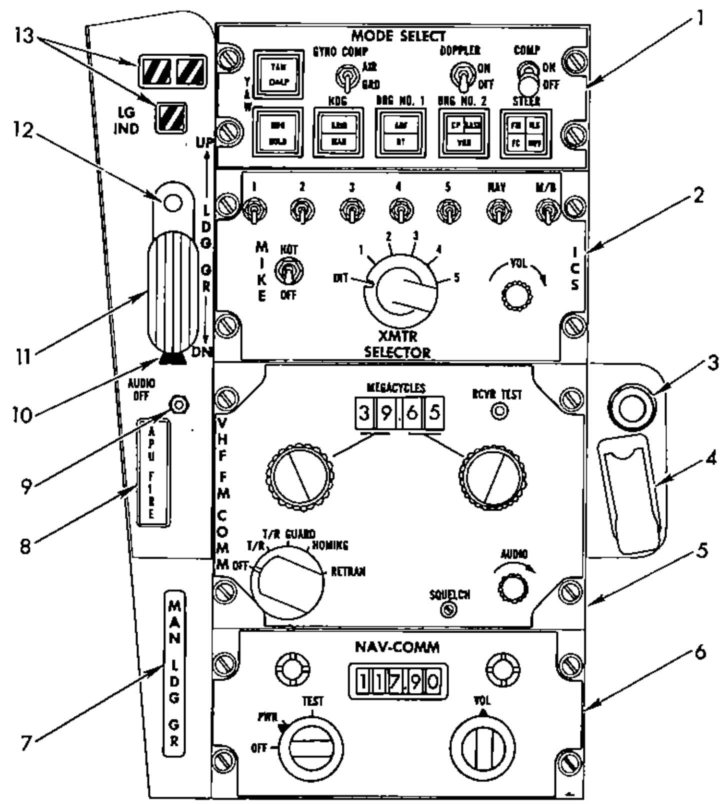
1	Standby compass	7	Map case
2	Glare shield	8	Swiveling gunner station surntable
3	Instrument panel	9	Collective
4	Map plotter	10	Left forward console
5	Right forward console	11	Antitorque pedals
6	Cyclic (stowed)		

Figure 2-2. Copilot/Gunner Station



1	Rotor brake switch	17	Barometric altimeter	30	Vertical velocity indicator
2	Canopy unsafe warning light	18	Fuel flow indicator	31	Standby attitude indicator
3	Trim position indicator	19	Auxiliary engine oil temperature indicator	32	Bearing distance heading indicator
4	External stores jettison switch	20	Engine oil temperature selector switch	33	Accelerometer
5	Beta angle indicator	21	Oil temperature indicator	34	Weapons control panel
6	Blade angle indicator	22	Oil pressure indicator	35	Rotor rpm warning light
7	Radar altimeter	23	Free air temperature indicator	36	Power turbine and main rotor tachometer
8	Standby compass	24	Hydraulic pressure indicator	37	External stores manual release handle
9	Airspeed indicator	25	Marker beacon control panel	38	Gas generator tachometer
10	Master caution light	26	Map display plotter	39	Power turbine inlet temperature indicator
11	Delta-beta light	27	Parking brake handle	40	Engine torquemeter
12	Power chop light	28	Armament stores control panel		
13	Attitude director indicator	29	Fuel quantity indicator		
14	Belly brush case ejector light				
15	Nose brush case ejector light				
16	Engine fire warning light				

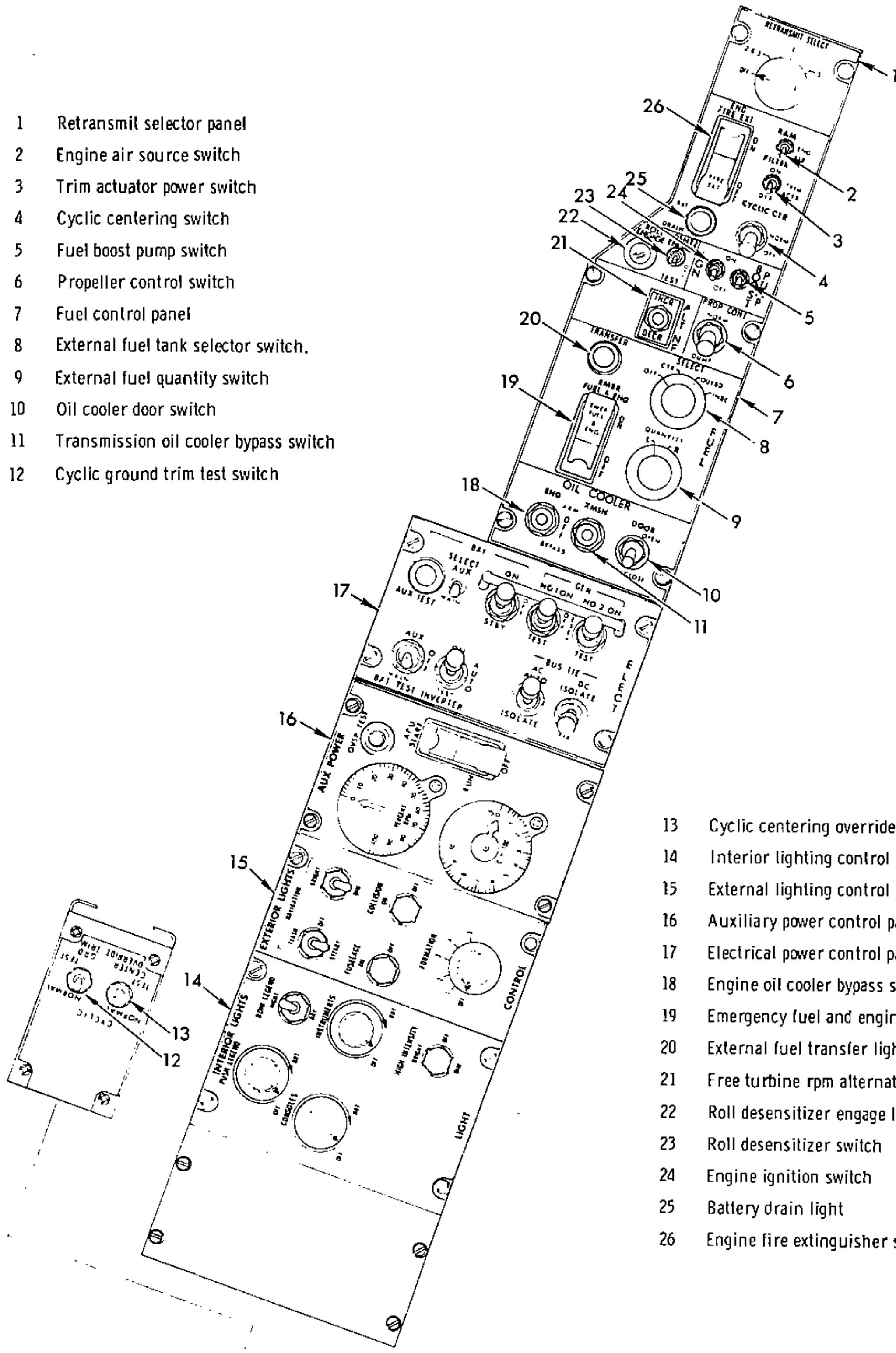
Figure 2-3. Pilot Station Instrument Panel



1	Navigation mode select panel	7	Landing gear manual release handle
2	Intercommunications control panel	8	APU fire warning light
3	Power chop arming light	9	Landing gear audio off button
4	Power chop arming switch	10	Landing gear lever release latch
5	VHF/FM radio control panel	11	Landing gear lever
6	VOR/LOC control panel	12	Landing gear lever override button
		13	Landing gear position indicators

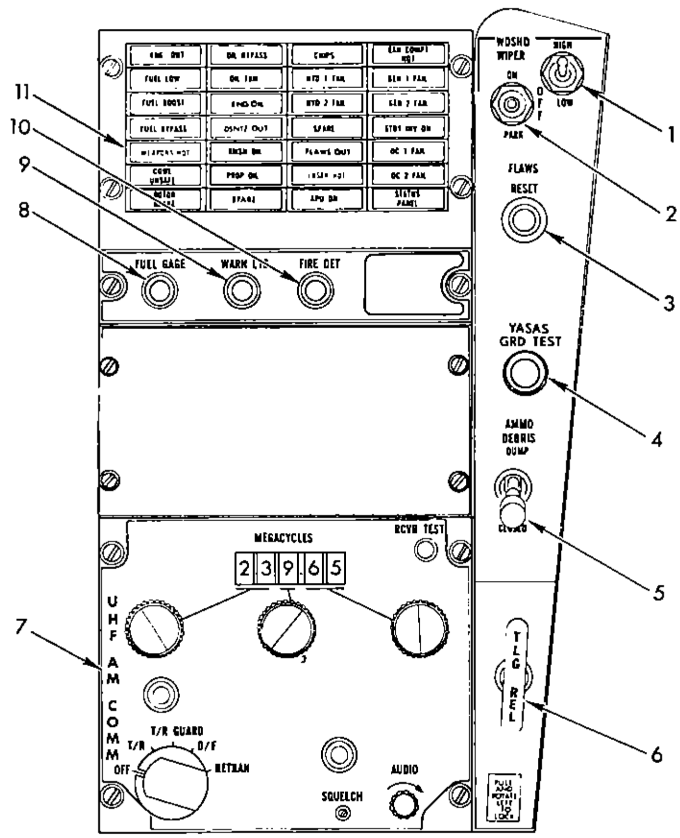
Figure 2-4. Pilot Station Left Forward Panel

1 Retransmit selector panel
2 Engine air source switch
3 Trim actuator power switch
4 Cyclic centering switch
5 Fuel boost pump switch
6 Propeller control switch
7 Fuel control panel
8 External fuel tank selector switch.
9 External fuel quantity switch
10 Oil cooler door switch
11 Transmission oil cooler bypass switch
12 Cyclic ground trim test switch



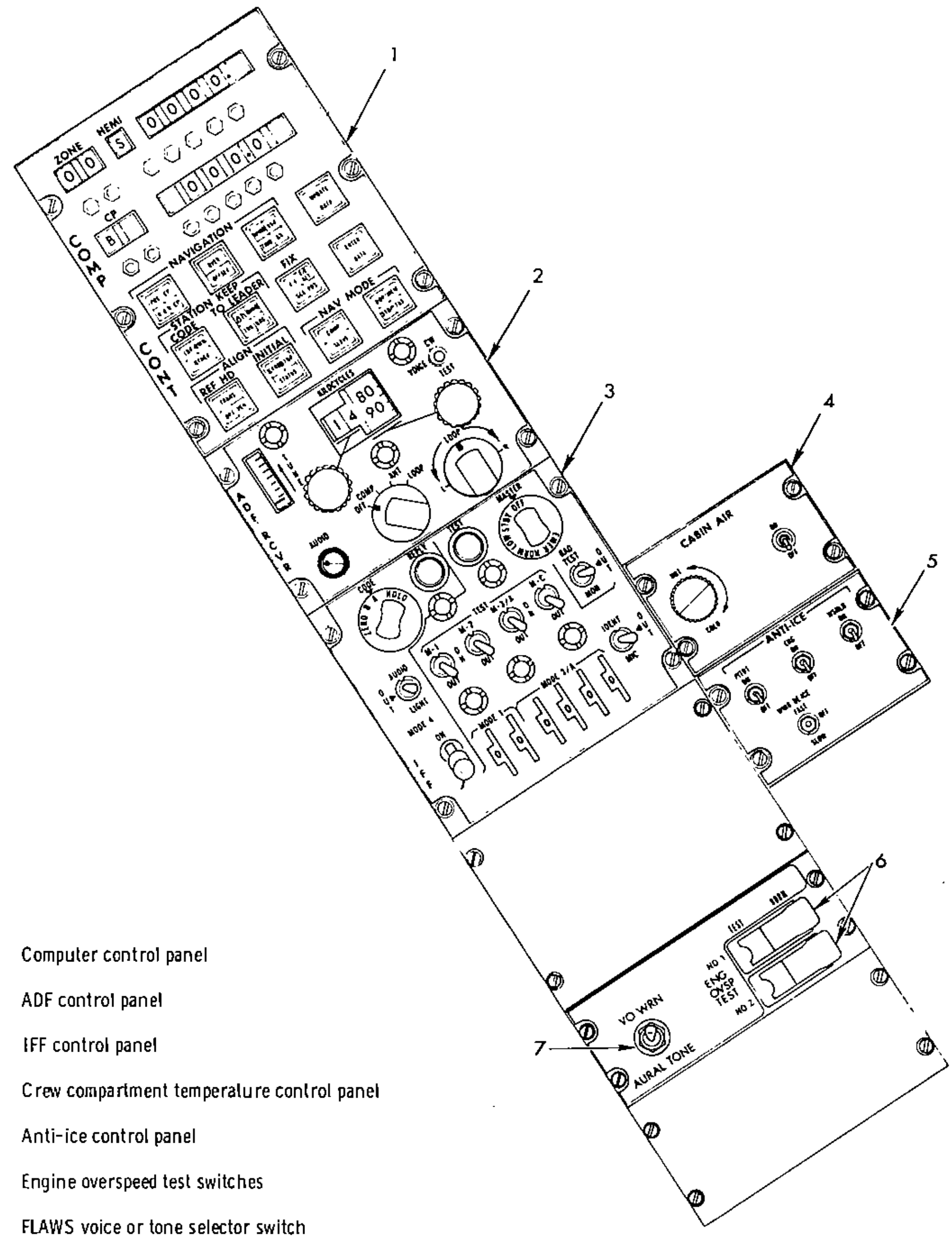
13 Cyclic centering override test switch
14 Interior lighting control panel
15 External lighting control panel
16 Auxiliary power control panel
17 Electrical power control panel
18 Engine oil cooler bypass switch
19 Emergency fuel and engine shutoff switch
20 External fuel transfer light
21 Free turbine rpm alternate set switch
22 Roll desensitizer engage light
23 Roll desensitizer switch
24 Engine ignition switch
25 Battery drain light
26 Engine fire extinguisher switch

Figure 2-5. Pilot Station Left Console



1 Windshield wiper speed switch
2 Windshield wiper motor switch
3 FLAWS voice reset switch
4 YASAS ground test switch
5 Ammunition debris door switch
6 Tail landing gear release handle
7 UHF/AM radio control panel
8 Fuel gage test switch
9 Warning lights test switch
10 Fire detector test switch
11 Annunciator panel

Figure 2-6. Pilot Station Right Forward Panel



1 Computer control panel

2 ADF control panel

3 IFF control panel

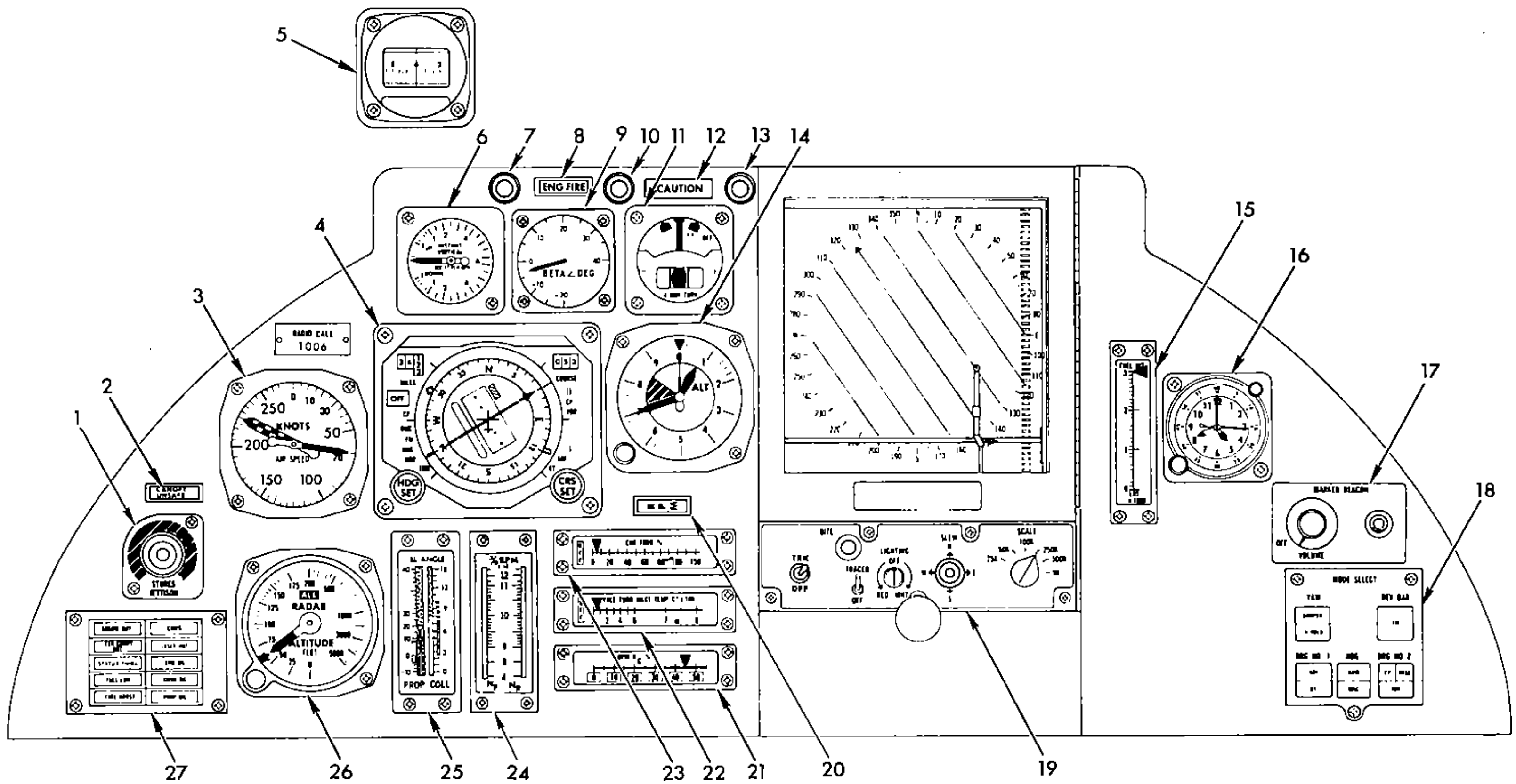
4 Crew compartment temperature control panel

5 Anti-ice control panel

6 Engine overspeed test switches

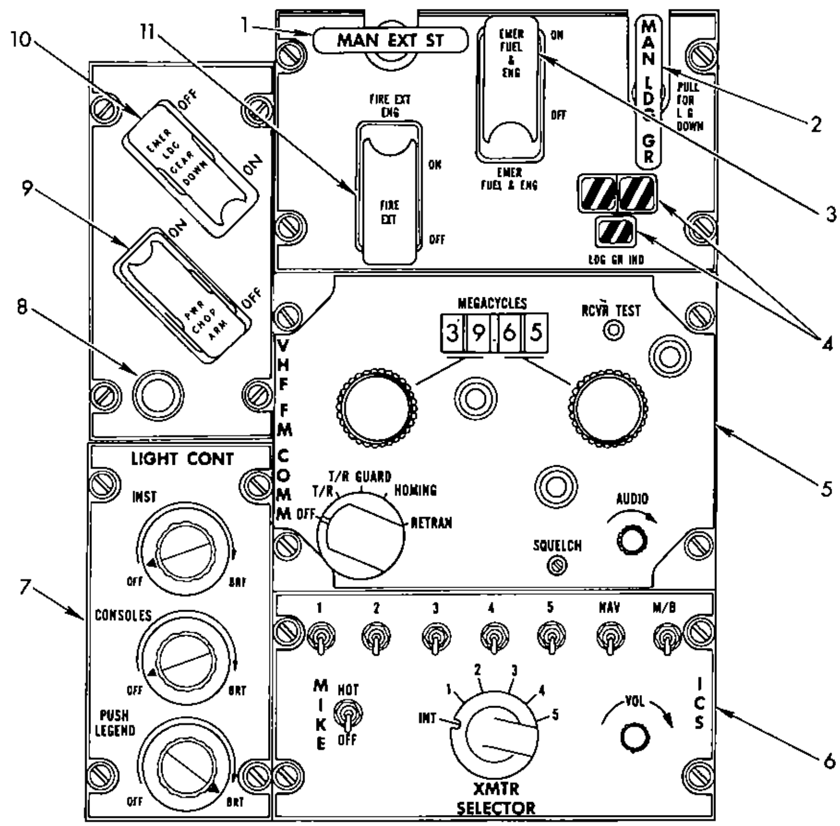
7 FLAWS voice or tone selector switch

Figure 2-7. Pilot Station Right Console



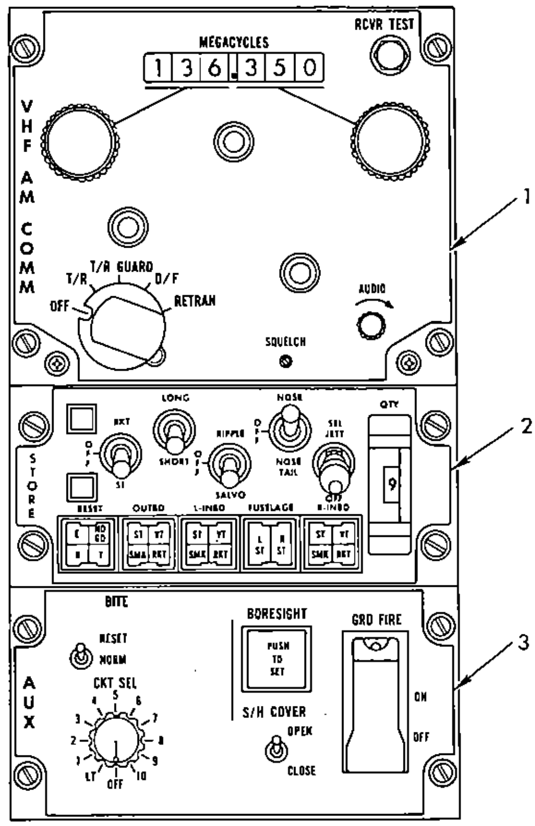
1 External stores jettison switch
2 Canopy unsafe warning light
3 Airspeed indicator
4 Bearing distance heading indicator
5 Standby compass
6 Vertical velocity indicator
7 Oscillograph high speed light
8 Engine fire warning light
9 Beta angle indicator
10 Delta-beta light
11 Turn-and-slip indicator
12 Master caution light
13 Power chop light
14 Barometric altimeter
15 Internal fuel quantity indicator
16 Clock
17 Marker beacon control panel
18 Navigation mode select panel
19 Map display plotter
20 Rotor rpm warning light
21 Gas generator tachometer
22 Power turbine inlet temperature indicator
23 Engine torquemeter
24 Power turbine and main rotor tachometer
25 Blade angle indicator
26 Radar altimeter
27 Annunciator panel

Figure 2-8. Copilot/Gunner Station Instrument Panel



1 External stores manual release handle
2 Landing gear manual release handle
3 Fuel/engine emergency shutoff switch
4 Landing gear position indicators
5 VHF/FM radio control panel
6 Intercommunications control panel
7 Interior lights control panel
8 Power chop arming light
9 Power chop arming switch
10 Landing gear emergency extension switch
11 Engine fire extinguisher switch

Figure 2-9. Copilot/Gunner Station Left Forward Panel



1 VHF/AM radio control panel

2 Armament stores control panel

3 Auxiliary weapons control panel

Figure 2-10. Copilot/Gunner Station Right Forward Panel

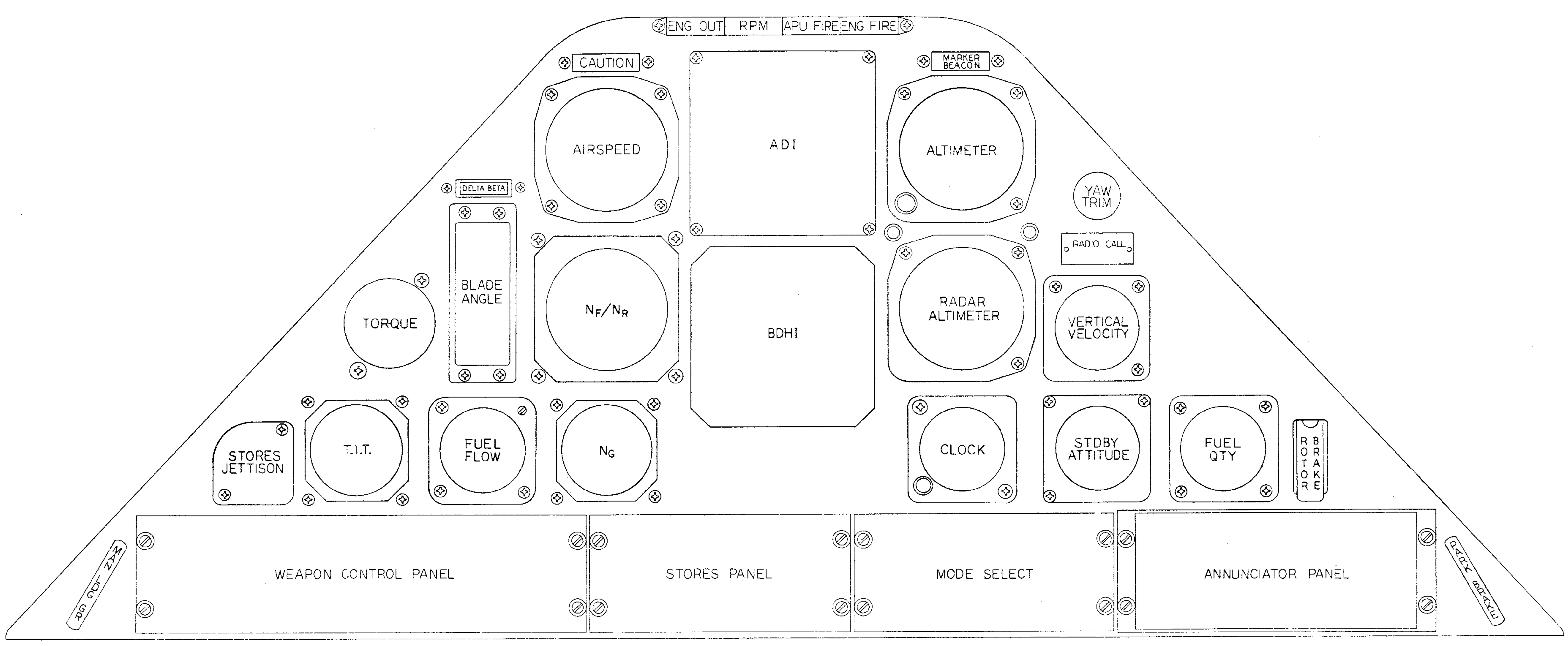


Figure 2-11. Pilot Station Instrument Panel

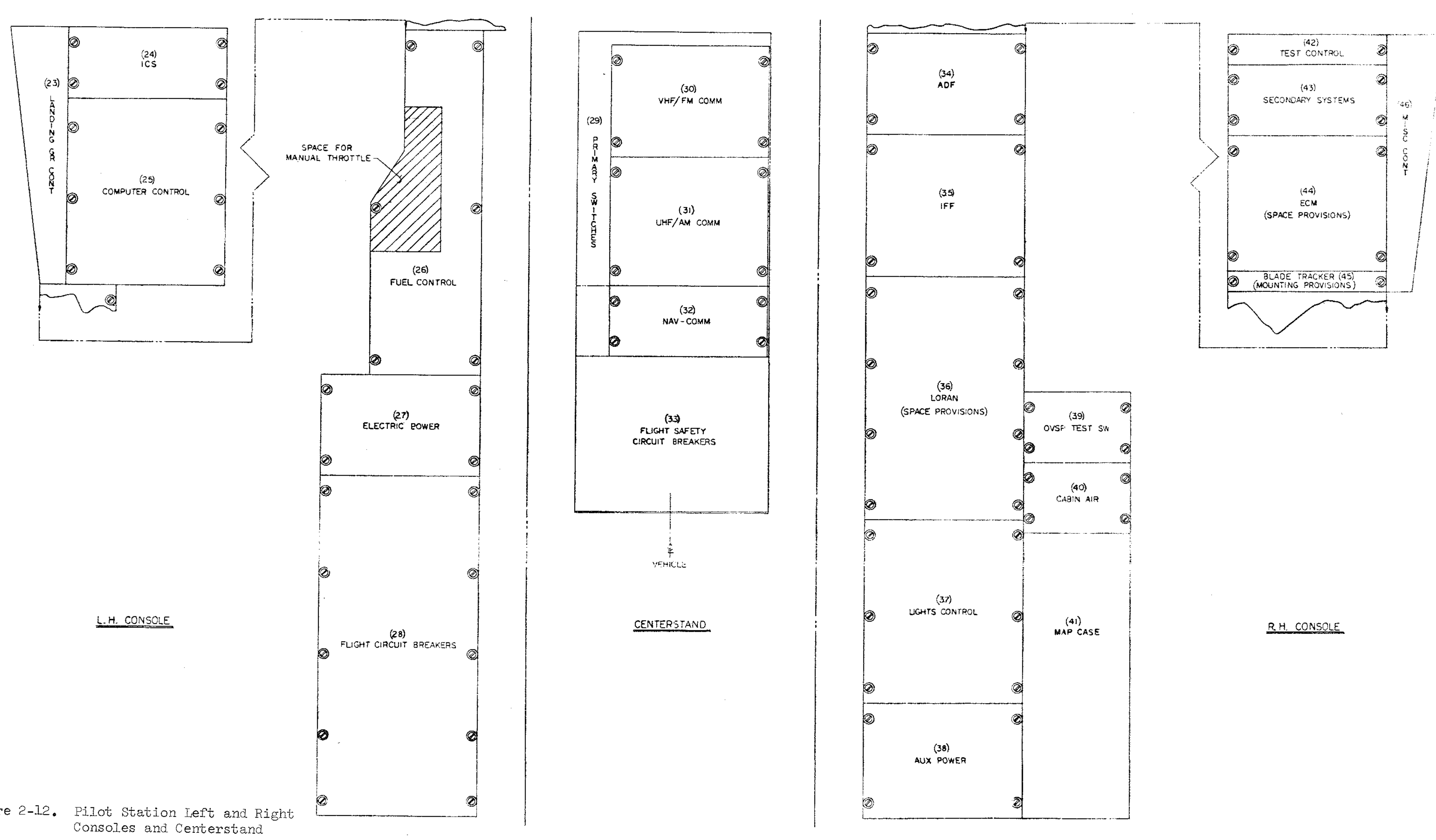


Figure 2-12. Pilot Station Left and Right Consoles and Centerstand

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AUXILIARY POWER UNIT

I. GENERAL DESCRIPTION

The GTP30-106 AiResearch Auxiliary Power Unit with primary power ratings as shown in table 3-1 is installed in a compartment isolated from the rest of the fuselage in the aft end of the left hand sponson. Combustion air is supplied through a screened opening in the left hand main wheel well. An exhaust system ducts the exhaust gases outboard and aft in a nearly horizontal path. This unit uses JP-4 or JP-5 fuel from the normal air vehicle fuel system supply tanks.

Power from the APU is used during ground operation only to drive the transmission oil pump, the No. 2 hydraulic power package, and the No. 1 and No. 2 electrical generators. By the use of power generated in these systems the primary engine (T64-GE-716) may be started and all other power systems may be checked out without operating the primary engine or using any source of ground power. Also, by operting the APU, the electronic equipment may be kept on an alert basis for prolonged periods of time.

The APU is a small turbine engine arranged to provide shaft power and consists of a compressor, combustion chamber, turbine, reduction gearing, self contained oil system, starter motor, fuel control and electronic control.

MIL-L-7808 is used for lubrication of the APU. Lubrication system components include the pump, filter, pressure regulator, pressure switch, sump and jets. The fuel control system provides fuel to the engine to meet the requirements established by sensing elements. During the starting and acceleration mode fuel flow is regulated as a function of compressor discharge pressure and exhaust gas temperature. When the APU reaches its rated speed fuel flow control is shifted from the acceleration limiter to the speed governor. As load changes are applied the speed governor adjusts fuel flow to maintain a speed of 100%.

TABLE 3-1. APU PERFORMANCE

Turbine rpm (continuous operation)	59,000 rpm
Turbine rpm (absolute maximum)	64,900 rpm
Output drive shaft rpm	8,000 ± 240 rpm
Compressor inlet temperature	50°C (122°F)
EGT (absolute maximum)	732°C (1349.6°F)
EGT (continuous)	650°C (1202°F)
Overtemperature thermostat setting	715° - 732 °C (1317° - 1349°F)
Overspeed switch setting	110% rpm
Fuel consumption (maximum APU load)	93 pph
Fuel consumption (minimum APU load)	38 pph
Output shaft horsepower (sea level at 52°C)	76 SHP
Starting time (-7 - 52°C OAT) (-44.6 - 125.6°F)	30 sec
Starting time (-55 - 7°C) (-131.0 - 44.6°F)	60 sec
Fuel inlet pressure	5-21 psi
Weight (APU and right angle drive)	101 lb
Oil pressure	30-40 psi
Oil consumption	0-15 pph
Starter crank limitation	1 minute on (4 minutes off between starts)

Accessibility for service and maintenance is provided through a hinged door on the inboard wall of the left hand wheel well. The oil level sight glass is readily visible through a screened area in this door. Oil servicing and removal of the major accessories may be accomplished through this same door. A removable panel on the lower surface of the sponson provides access for removal of the APU.

II. COMPONENTS AND LOCATIONS

Name of Component	Number per Aircraft	Location in Aircraft
GTP30-106 APU	1	In the aft end of the left hand sponson
Auxiliary Power Control Panel	1	In Pilot's flight station, left console aft
Starter	1	Mounted to drive gear housing
Right Angle Gear Box	1	Mounted on output drive pad
APU Clutch	1	Mounted on output side of right angle drive gear box
Drive Shaft Assembly	1	Connects APU clutch to main transmission
Electronic Control Unit	1	Mounted on top of APU gear housing, aft side.

III. MAJOR COMPONENT DESCRIPTION

A. Auxiliary Power Control Panel

The auxiliary power control panel consists of two instruments and two switches to operate the APU. For APU operation, a tachometer and an exhaust gas temperature indicator are installed. A master start switch with OFF, RUN and START positions, and an overspeed test switch are included.

B. Starter

The starter consists of a direct current electric motor (24 vdc) coupled to a torque limiting and overrunning clutch. The starter is mounted on the gear drive housing in a manner to enable pawls in the clutch assembly to mate with a ratchet in the gear drive assembly. The starter motor will disengage as the APU engine accelerates to approximately 50 percent rpm.

C. Right Angle Gear Box

The right angle gear box is an airframe furnished component that is attached to the six bolt drive pad of the APU. The gear box assembly bends the drive line 90° with an input to output ratio of 1:1.

D. APU Clutch

The APU clutch mounts to the output side of the right angle gearbox between the output drive of the gearbox and the drive shaft that connects the APU to the main transmission. The purpose of the clutch is to effect a smooth connection of power from the APU to the loads applied through the main transmission.

E. Drive Shaft

The drive shaft assembly is approximately 33 inches in length and consists of a tubular shaft and two flexible couplings. The flex couplings absorb minor misalignment which may exist between the right angle gearbox and the transmission.

F. Electronic Control Unit

The electronic control unit receives dynamic speed signals from a magnetic motion-sensing pickup. It initiates and monitors engine starting, acceleration, and shutdown through speed switching functions. Also, this unit controls the starter circuit, fault function (low oil pressure, overspeed, overtemperature) rpm indicator, hourmeter, start counter, and fuel solenoid valve. In the event of engine overspeed, the circuit to the fuel solenoid valve will be opened, thereby, shutting down the APU by eliminating fuel flow to the combustion chamber.

IV. SYSTEM OPERATION

During the operational sequence the electronic control unit senses signals generated during different phases of operation and supplies appropriate commands to provide safe operation. The operational sequence of this unit is shown in the operating sequence diagram. Additional features of this electronic control unit are to transmit a continuous signal from the magnetic motion-sensing pickup to the tachometer indicator on the APU control panel and also, by means of an oscillator, shut down the engine from the overspeed test switch signal. Also, automatic shut down occurs if oil pressure decays to approximately 15 psig or if an overheat condition is sensed in the APU compartment.

V. PCR CONFIGURATION

No change

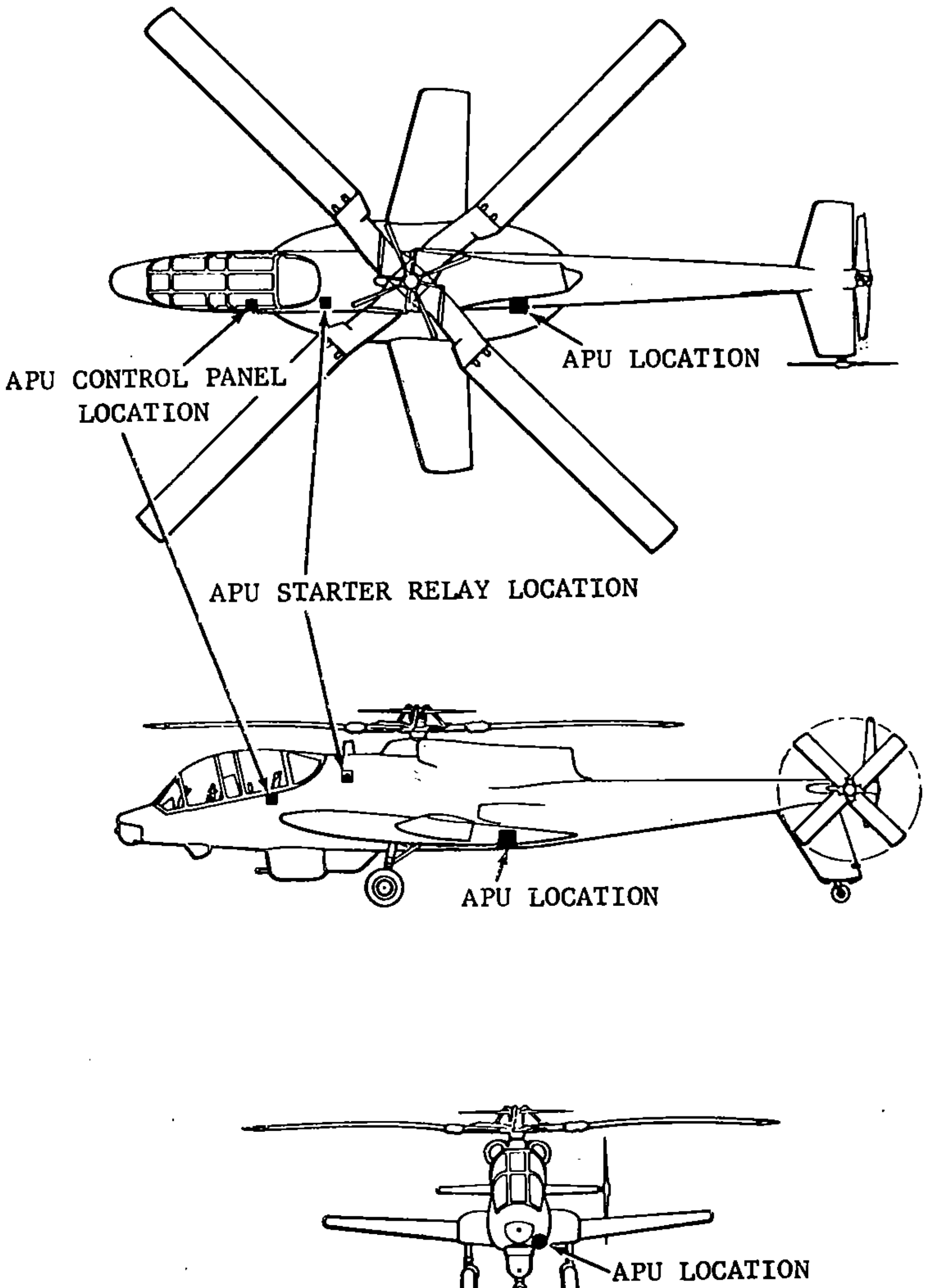

Figure 3-1. APU Location Area

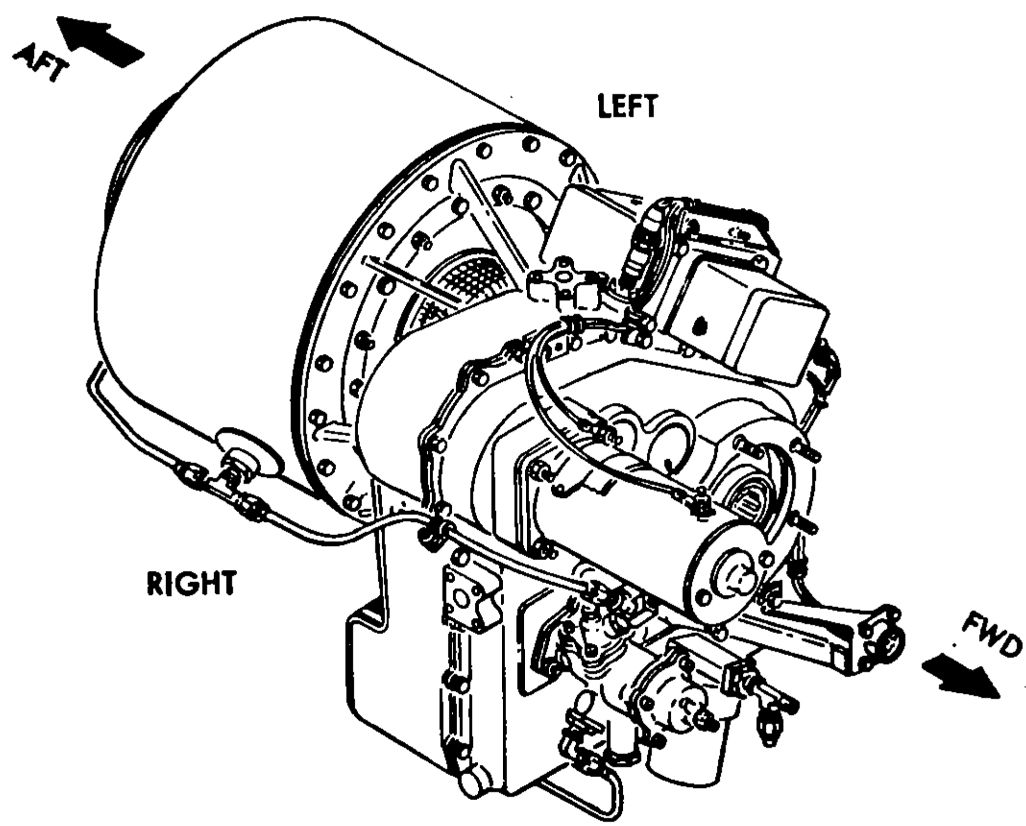

Figure 3-2. APU Orientation

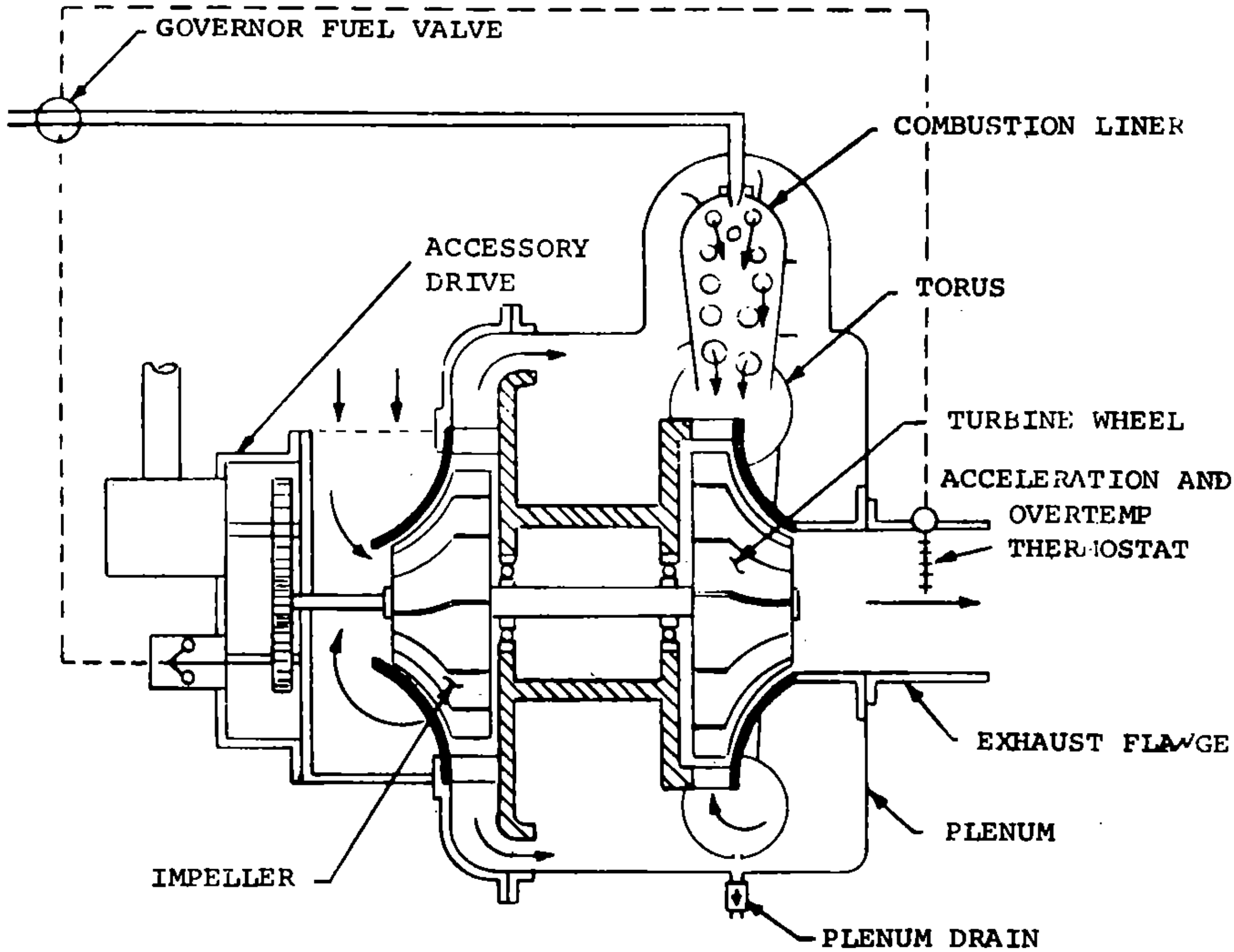

Figure 3-3. Gas Turbine Schematic

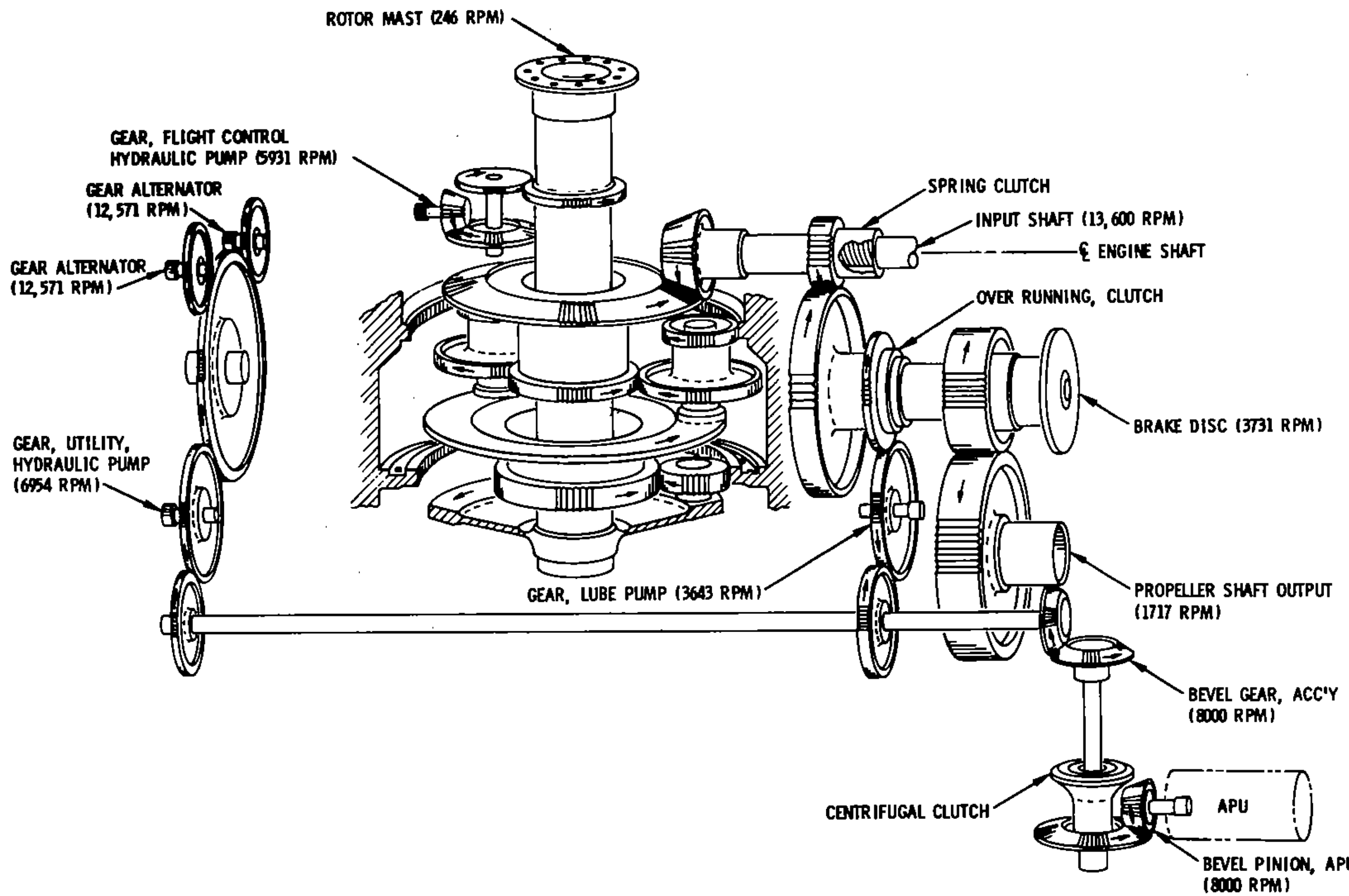

Figure 3-4. Main Transmission and APU Drive Schematic

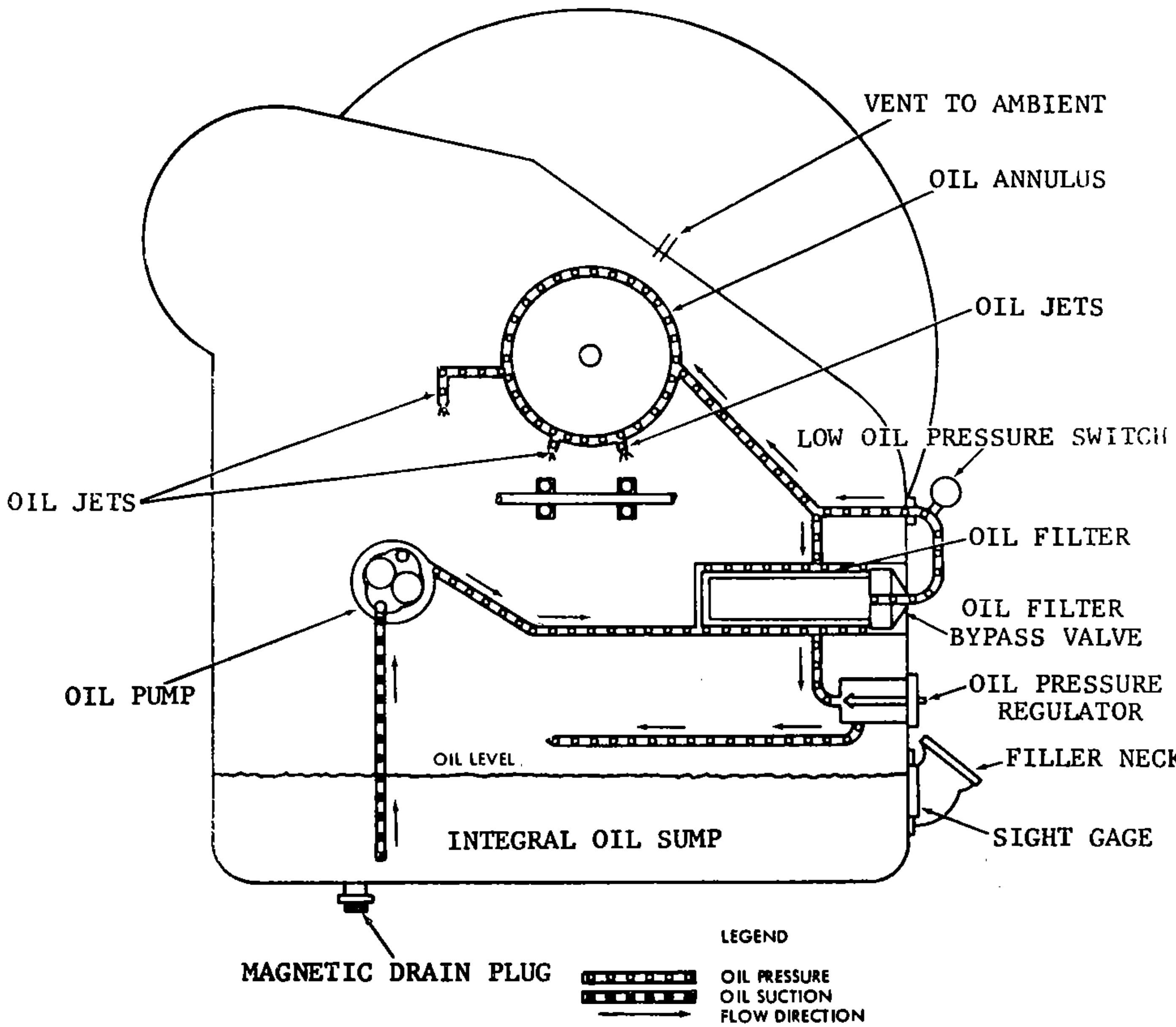

Figure 3-5. Lubrication Schematic

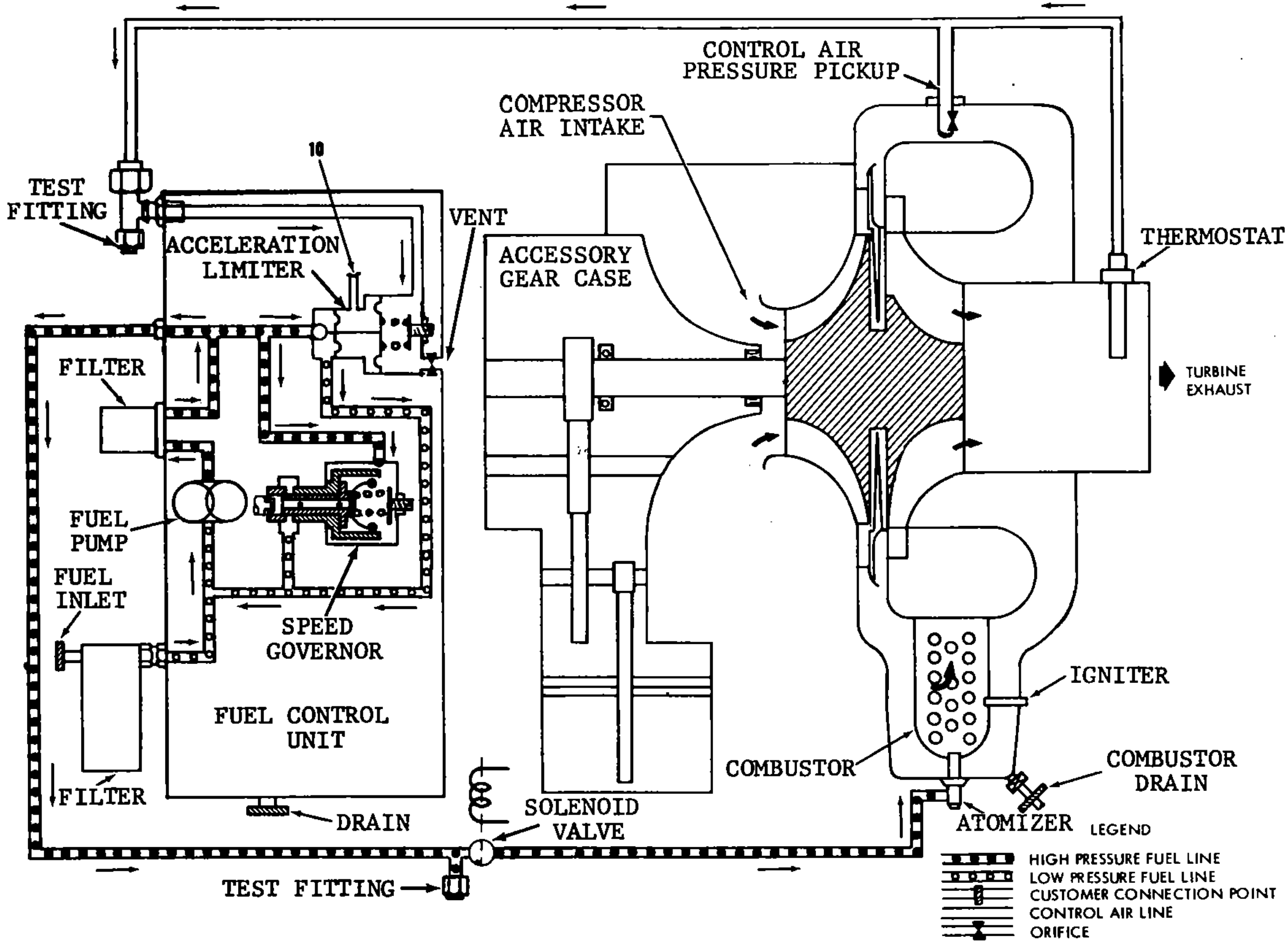

Figure 3-6. Fuel Control System

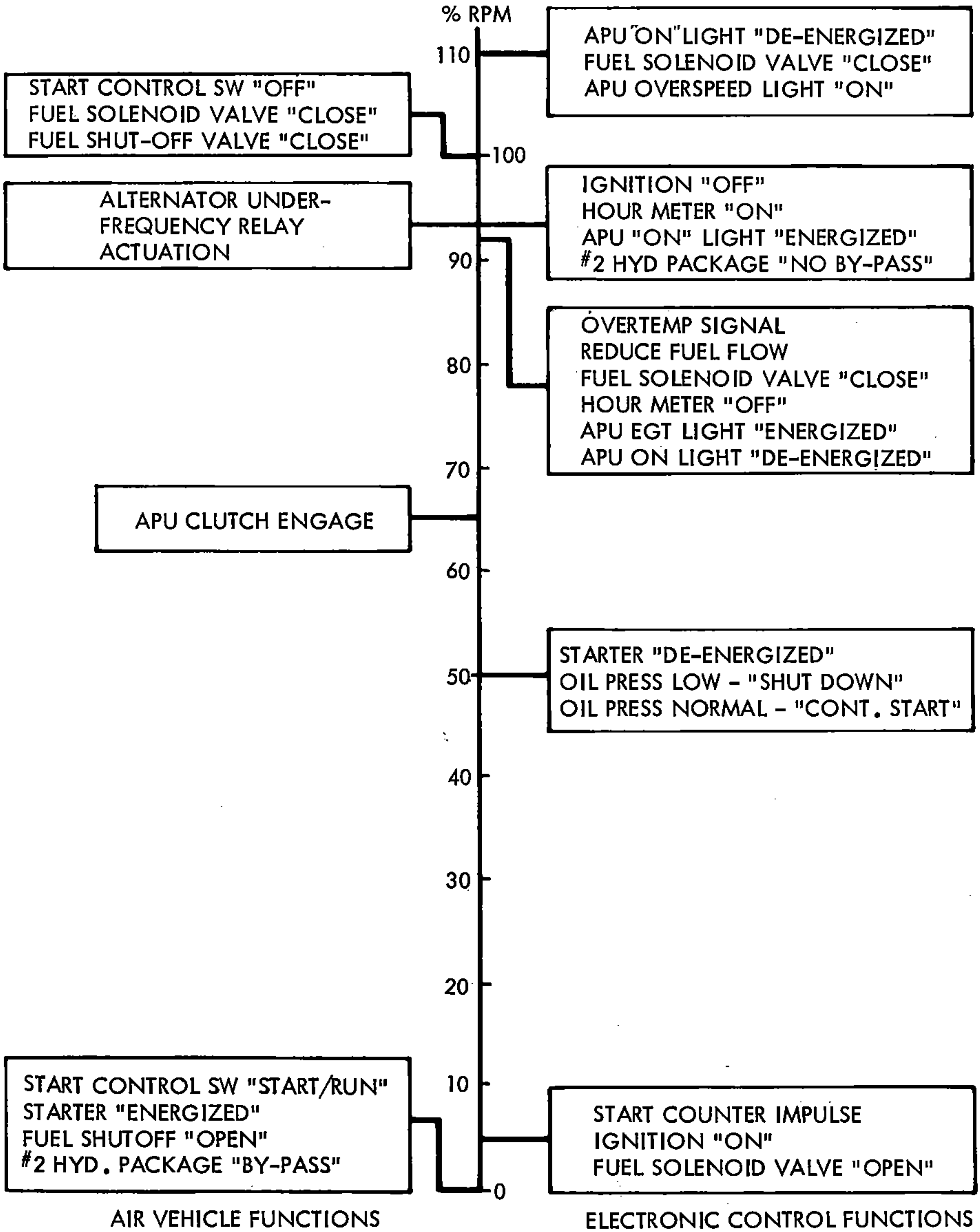

Figure 3-7. APU Operating Sequence

PROPULSION

I GENERAL DESCRIPTION

The AH-56A is powered by a GFM General Electric axial flow, free turbine, turbo shaft engine designated T64-GE-716. This engine in conjunction with supporting aircraft components comprise the powerplant installation.

The T64-GE-716 turbo shaft engine consists of a gas generator driving a power turbine and shaft assembly. The power shaft extends forward coaxially through the gas generator turbine, combustion chamber, compressor, and front frame. The power shaft delivers the driving power to the aircraft main transmission through a torque sensor shaft assembly.

The gas generator incorporates a 14 stage axial-flow compressor with variable inlet flow geometry. The combustion section is a through flow annular type. The gas generator turbine consists of the first two stages of a four-stage axial-flow turbine. These first two stages drive the compressor, the engine power take-off, and the engine accessory gear box with its accessories.

The third and fourth stages of the turbine comprise the power or free turbine. The two power turbine wheels are bolted to a power shaft that extends forward coaxially through the gas generator to deliver the required power to the main transmission. This power turbine is driven by the gases from the gas generator turbine and turns the same direction as the gas generator, but not necessarily the same speed.

A hydromechanical fuel control that senses gas generator and power turbine speeds, inlet air temperature, compressor discharge pressure, and power lever and load control positions, monitors the gas generator speed to keep the power turbine at a speed selected by the pilot regardless of the flight load.

II COMPONENTS AND LOCATIONS

Name of Component	Number per Aircraft	Location in Aircraft
Front Frame	1	Mounted to forward flange of compressor
Speed Control Assembly	1	Driven from forward end of power shaft just forward of front frame - 8 o'clock position
Compressor Section	1	Located between front frame and combustion section
Combustion Section	1	Located between compressor section and turbine section
Turbine Section	1	Located between combustion section and exhaust frame
Exhaust Frame	1	Mounted to aft flange of turbine section
Accessory Gear Box	1	Located at lower forward end of engine. Mounted to front frame
Starter	1	Mounted on aft side of engine accessory gearbox
Power and Load Control Actuator	1	Mounted on engine fuel control
Bellmouth Assembly, Engine Firewall Forward	1	Encircled about compressor front frame inlet

II COMPONENTS AND LOCATIONS (cont'd)

Name of Component	Number per Aircraft	Location in Aircraft
Firewall Assembly, Mid-engine Aft	1	Mounted circumferentially about forward end of engine combustion casing
Gas Generator Speed Transducer	1	Mounted on forward side of accessory drive gearbox
Engine Cowl	1	Mounted on top of fuselage surrounding the engine
Engine Mount	1	Mounted on top of fuselage structure
Inlet Duct	2	On each side of rotor mast
Filter Assembly	2	One in each filter cowl between engine and main transmission.
Debris Scavenge Fans	2	One at debris exit in each filter cowl.
Engine Inlet Air Flaps	2	One centered in each intake duct
Engine Inlet Air Flap Actuator	1	In access hole on top of fuse-lage just aft of main rotor mast
LUBRICATING SYSTEM		
Engine Oil Tank	1	Mounted on engine compressor casing, top left side
Engine Lube/Scavenge Pump	1	Mounted to engine accessory drive, left side of engine

Name of Component	Number per Aircraft	Location in Aircraft
Engine Oil Filter	1	Integral with engine lube scavenge pump
Engine Oil Bypass Valve	1	On engine front frame, left side
Oil Cooler	1	Mounted in mid fuselage, left side
Oil Cooler Bypass Caution Light	1	Pilot's annunciator panel
Oil Pressure Transmitter	1	Installed on bracket on accessory gearbox left side forward
Oil Pressure Warning Switch	1	Installed in tee in oil pressure line 7:00 o'clock
POWER PLANT ELECTRICAL SYSTEM		
Torquemeter	1	(See basic engine section.)
Ignition Exciter	1	Aft flange of compressor casing, right side
T5 Thermocoupler	14	Attached to turbine casing at power turbine inlet
Engine Anti-icing Valve	1	On compressor casing at 11:00 o'clock position
ENGINE FUEL SYSTEM		
Engine Driven Fuel Pump and Filter	1	Mounted on engine fuel control on left side of engine
Fuel Control	1	On accessory gear box - left side of engine

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II COMPONENTS AND LOCATIONS (cont'd)

Name of Component	Number per Aircraft	Location in Aircraft
Fuel Flow Transmitter	1	Mounted on lower left side of engine compressor at 8:00 o'clock position
Flow Divider	1	On compressor casing - 6:00 o'clock position
Fuel Nozzle	12	Installed in 2 manifolds that encircle the engine combustion chamber and frame
Power and Load Control Actuator	1	Mounted on fuel control
Power and Load Control Amplifier	2	In electrical load center
Power Trim Control	1	In electrical load center
Potentiometer, Engine Speed Toroidal	1	In pilot's flight station in aft end of engine condition grip on collective stick
Potentiometer, Propeller Position, Dual	1	In pilot flight station, just aft of collective stick
Potentiometer, Collective Control, Dual	1	Below pilot station floor, operated by collective control mechanism

III MAJOR COMPONENT DESCRIPTION

A. Front Frame

The front frame supports the accessory gearbox, the power takeoff and the speed control. It consists of the outer casing with engine mounting pads, hollow struts and a hub. The No. 1 bearing is contained in the hub. Both struts and hub provide anti-icing air flow.

B. Compressor Section

The compressor section is a 14 stage axial flow unit with a compression ratio of 13 to 1. The inlet guide vanes and the first four stages of stator vanes are variable. These vanes are positioned as a function of airflow to provide efficient operation throughout the entire speed range. The compressor rotor is an assembly consisting of of a shaft, 14 stages of fixed blades, and blade supporting discs.

C. Combustion Section

The combustion section consists of a one piece combustion chamber, an annular combustion liner, the first stage turbine nozzle and an inner casing. The No. 2 and 3 bearings are supported from the inner casing. Engine mounting lugs and compressor air bleed ports are provided at the forward end of the combustion chamber. Twelve fuel nozzles and two igniter plugs are also incorporated in this section.

D. Turbine Section

The turbine section consists of two rotor and two stator casings. Both rotors are driven by hot gases from the combustion chamber. The turbines rotate in the same direction but not necessarily at the same speed. The first rotor consists of two stages and drives the compressor. The second rotor, also of two stages, drives the output shaft which is inside and coaxial with the compressor. The stator casings contain the stator vanes, turbine rotor shrouds and 14 thermocouple probes which sense power turbine inlet temperature.

E. Exhaust Frame

The exhaust frame consists of an outer casing, an inner coaxial casing supported by six radial struts and the aft bearing for the power turbine shaft.

F. Accessory Gearbox

The accessory gearbox is attached to the bottom of the front frame and is driven from a power takeoff from the compressor shaft. The gearbox has drives and pads to accommodate the main fuel control, lubrication pump, a gas generator speed transducer and a starter.

G. Speed Control Assembly

The speed control assembly consists of gear trains and shafting within a housing extending radially in front of the front frame. Engagement is made at the inner end with the power turbine shaft. The outer end, through a flex shaft, drives a governor within the main fuel control.

H. Starter

The starter is a hydraulic motor drawing energy from the No. 2 hydraulic system which can be powered by the APU. The starter is attached to the accessory gearbox by means of an adapter and quick disconnect clamp.

I. Power and Load Control Actuator

The power and load control actuator consists of three electric motors and an actuator which drives the two coaxial shafts of the main fuel control as demanded by pilots input signals.

J. Bellmouth Assembly, Engine Firewall Forward

This is a stainless steel assembly attached to the air inlet flange of the engine front frame to ensure smooth airflow into the engine and to isolate the engine fire zone from other areas of the vehicle.

K. Firewall Assembly, Mid-engine Aft

This stainless steel member separates the accessory section from the hot section of the engine compartment to preclude the progression of fire or flammable fluids from one section to the other. It attaches to the engine and seals against a mating member on the engine cowl.

L. Engine Cowl

The engine cowl, fabricated from stainless steel and aluminum alloy sheet metal, provides a streamlined fairing around the engine. It is attached to the fuselage by rollers and tracks in such a manner that it may be moved aft for access to the engine. The engine exhaust duct and deflector are contained in this assembly. Provision for the attachment of armor plate, to provide protection for vital parts of the engine, are included in this cowl.

M. Engine Mounts

The engine mount provisions consist of two forward pylons, one on each side of the engine, stabilized in the fore and aft direction by a diagonal strut and a stabilizing link under the forward end of the combustion chamber. The forward pylons are bolted to the upper fuselage longerons and attach to the engine by means of ball and socket adapters. Engine loads in all directions are carried by these pylons. The stabilizing link carries only vertical loads and provides freedom for thermal expansion of the engine. Fittings at each end provide attachment to the engine and the fuselage. Provisions are made at each point of attachment to the fuselage for shimming to facilitate alignment of the engine to the transmission. Engine changes may be accomplished without readjustment of the shims.

IV SYSTEM OPERATION

A. Engine Airflow

The engine air induction system consists of inlet ducts, plenum chamber and air cleaners. An inlet duct is provided on each side of the main rotor mast. The leading edge lips of these ducts incorporate provisions for hot air anti-icing. Air from these ducts enters a plenum chamber which contains particle separator type air cleaner units. During normal operation air flows through some of the particle separators. An electrically actuated vane is provided so that, on demand from the pilot during operation in contaminated air, all of the air will pass through the particle separators. Hydraulic motor driven debris scavenge fans discharge the debris overboard. Air leaving the plenum chamber passes through the bellmouth directly into the engine where it is used for combustion, cooling, engine anti-icing, pressurization of bearing seals and air frame bleed requirements.

B. Lubrication

The lubrication system provides a flow of oil, within prescribed temperature limits, to the engine bearings. The combination pressure and scavenge pump unit, in addition to the pump elements, contains the filter, regulating valve, by-pass valve and check valve. This pump unit is furnished with the engine and is mounted on, and driven by, the accessory gearbox. Fuel/oil coolers are incorporated in the scavenge system to dissipate a portion of the heat. The airframe portion of the lubrication system consists of an oil tank, by-pass valve, oil cooler, oil cooler fan and air ducting. The engine oil tank with a storage capacity of 3.25 gallons is installed on the upper right hand side of the compressor section. Contained within the tank are a sight gage, level switch, chip detector and a filler cap. The bypass valve in the scavenge system provides protection against complete loss of oil. At one gallon remaining, the level switch will energize the bypass valve to direct oil flow to the tank rather than through the oil cooler. The three element oil cooler contains the transmission lubricating oil, No. 2 hydraulic system oil and the engine oil. It is of fin and tube construction with pressure and thermostatic bypass features. A hydraulic motor driven fan pulls air through the cooler.

C. Engine Electrical System

An electrical system is provided on the engine for the ignition exciter and igniters, the anti-ice valve, an over-ride for the main fuel control solenoid valve (for check purposes) and the thermocoupler. Additional electrical services are provided for fuel flow transmitter, oil pressure transmitter, oil pressure warning switch, oil by-pass valve, chip detector, oil low level, gas generator speed transducer, engine cowl "safe" indicator, and induction system flap actuator.

D. Engine Fuel System

The engine receives fuel at the inlet to the fuel filter which is integral with the pump and control unit. Operating in response to input signals, the fuel control regulates fuel flow to the engine combustion chamber to provide the power and speed conditions desired. Input signals to the control are: gas generator speed, power turbine speed, compressor discharge pressure, compressor air inlet temperature, overspeed electrical signal, power control shaft position, load control shaft position, and variable stator position. Both power control and load control shafts are positioned by electro-mechanical actuators which are powered through potentiometers and amplifiers in response to movement of the prop pitch grips, the collective levers and the RPM set switches.

Automatic overspeed protection is provided by a motional pickup transducer that senses power turbine speed. Signals from the pickup, through a switch to a solenoid valve on the fuel control, shuts off the fuel flow. When the RPM drops a prescribed amount the signal is removed and the fuel valve is opened. RPM cycling will continue until action is taken by the pilot to reduce the fuel flow thus preventing overspeed and eliminating cycling.

A syncro type fuel flow transmitter is installed downstream of the fuel control. This provides a readout in the cockpit in pounds per hour. The flow divider directs fuel to the dual orifice nozzles through four manifolds. The dual nozzle provides proper fuel atomization over a wide range of fuel flow. During engine starting, and low speed operation, the flow divider supplies fuel to only the primary orifice of the dual nozzle. As fuel flow increases the flow divider allows fuel to flow to both orifices.

V PCRS CONFIGURATION

Changes are limited to the engine air induction system. The air cleaner panels are moved to the outer surface of the plenum chamber and given protection by a set of actuated louvers. This necessitates changes in the cowling contours which extend from the duct inlet to the aft end of the engine cowl. As part of the plenum chamber change the floor is rearranged to improve protection against FOD.

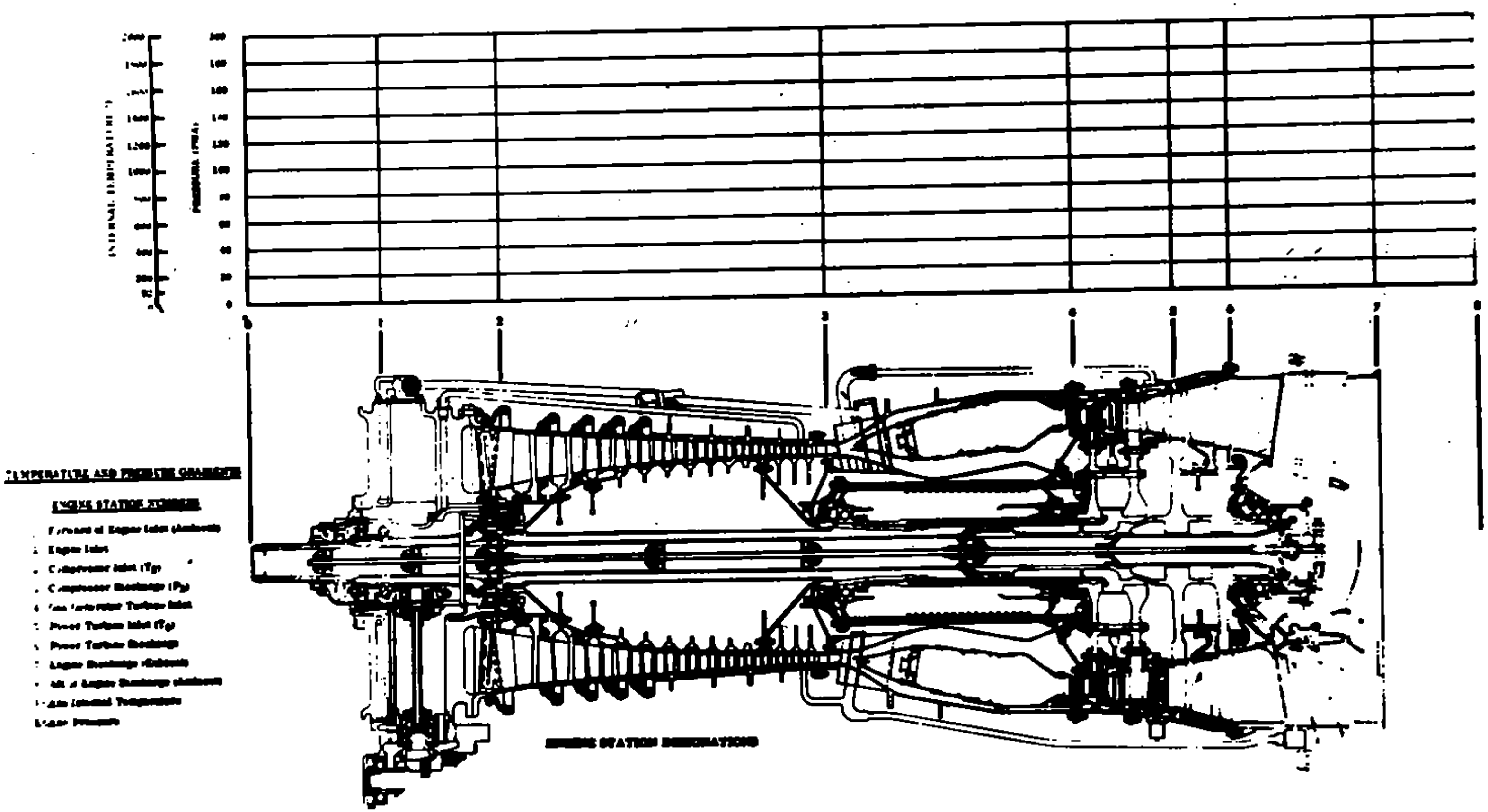


Figure 4-1. Engine Station Designations

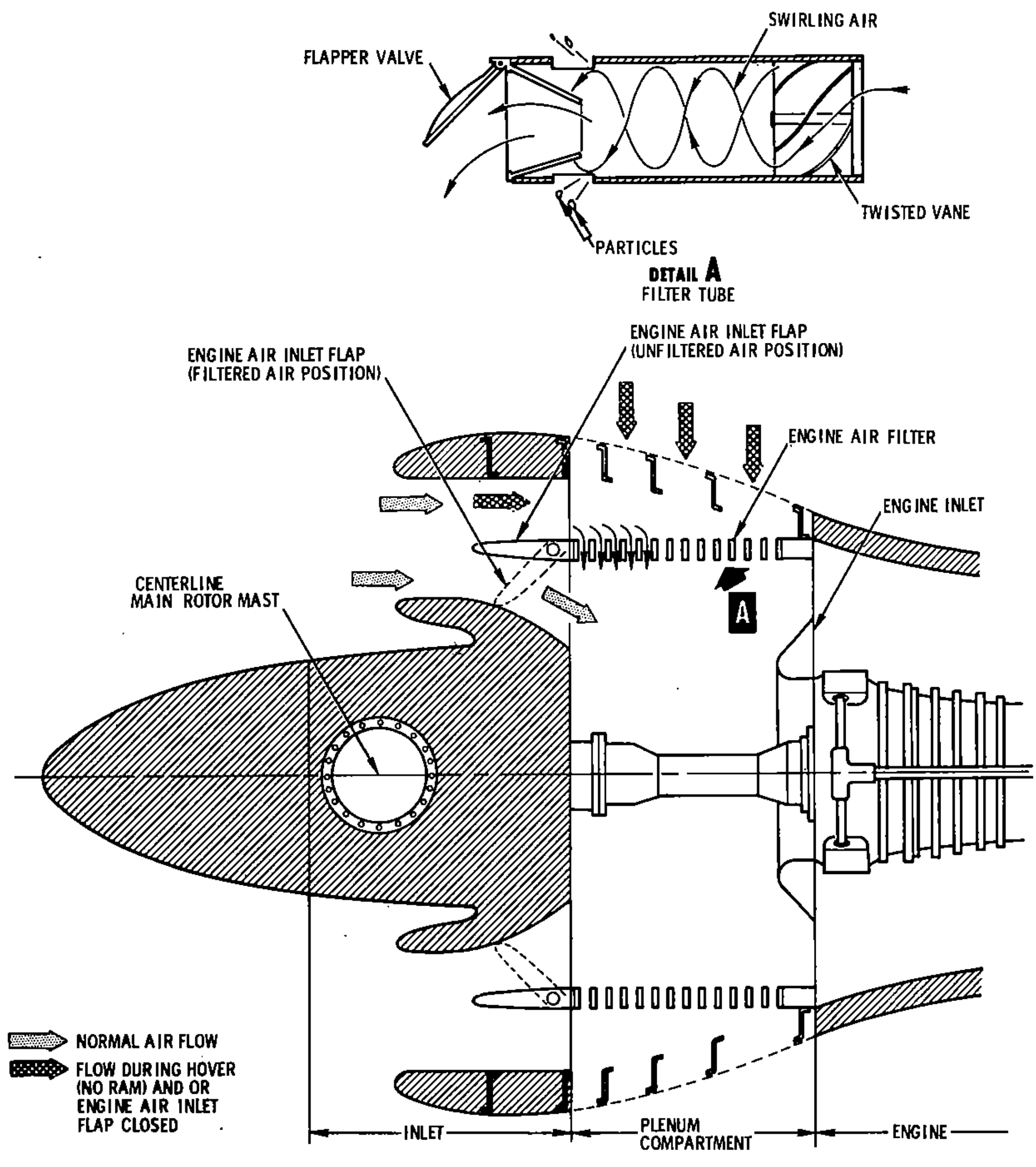


Figure 4-2. Engine Inlet Air Flow Schematic

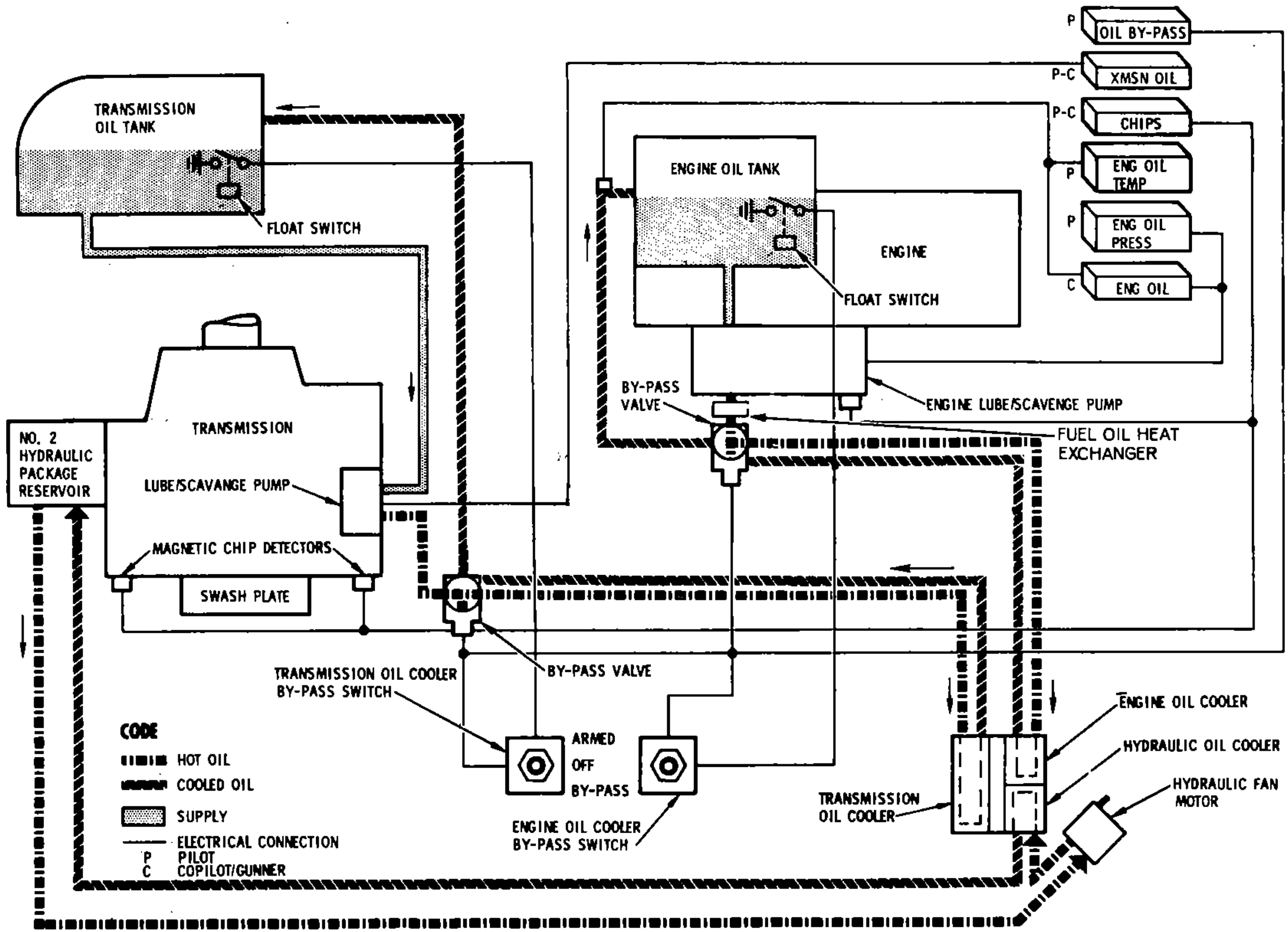


Figure 4-3. Lubrication System Schematic (1 of 2)

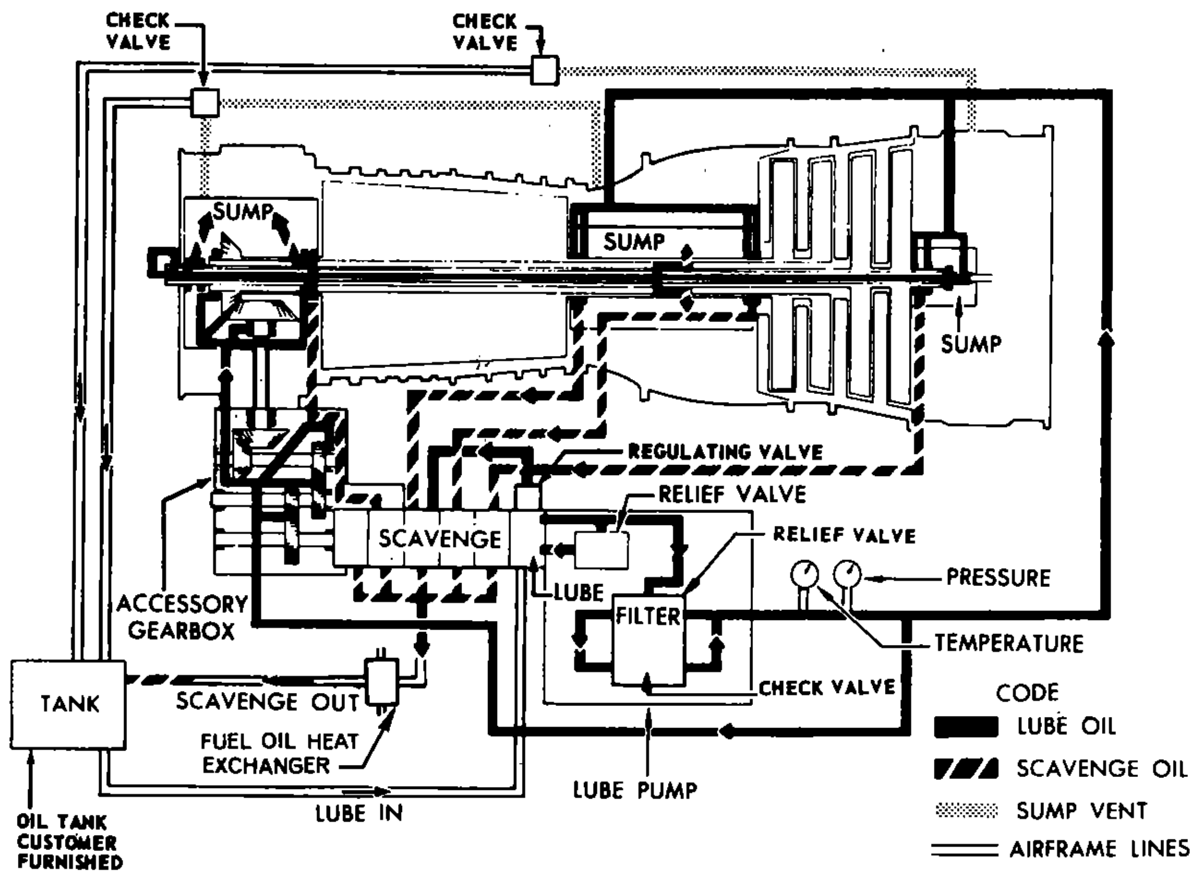


Figure 4-3. Lubrication System Schematic (2 of 2)

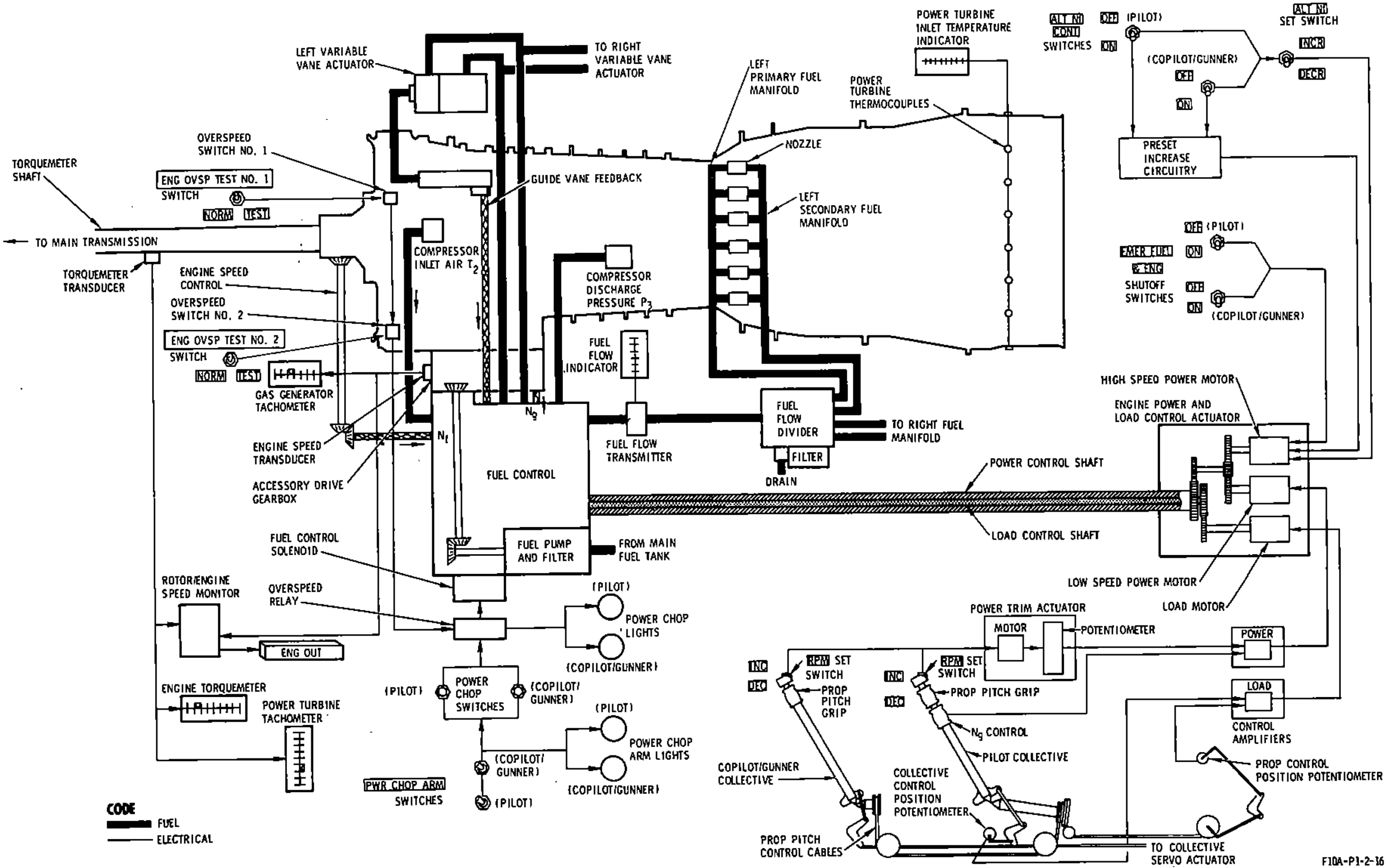


Figure 4-4. Engine Fuel and Control Systems

POWER TRAIN

I. GENERAL DESCRIPTION

The power train consists of those components necessary to transmit power from the engine to the main rotor, tail rotor, propeller and accessories. In addition, an auxiliary power train consisting of those components required to transmit the power from the APU to the main transmission has been included in this design.

II. COMPONENTS AND LOCATIONS

Name of Component	Number per Aircraft	Location in Aircraft
Engine/Torquemeter Shaft Assy.	1	Between engine and main transmission
Engine Speed Control Drive Assy.	1	Forward face of engine
Main Transmission	1	Mounted on the forward and aft major fuselage frames (FS 284 and 317)
Transmission Oil Tank	1	Above transmission accessories
Transmission Oil Cooler Bypass Switch	1	On fuel control panel in the Pilot Station
Tail Rotor Drive Shaft Assy.	1	Between the propeller gearbox and the tail rotor
APU Gearbox Assy.	1	L.H. Sponson aft of the wheel well - mounted on APU
APU Clutch Assy.	1	L.H. Sponson - mounted on APU gearbox assy.
APU Drive Shaft	1	L.H. side of fuselage between APU clutch and main transmission

III. MAJOR COMPONENT DESCRIPTION

A. Engine/Torquemeter Shaft Assembly

The engine/torquemeter shaft assembly provides a load path to transmit 3925 horsepower at 13,600 RPM from the engine to the main transmission and a means for measuring the applied torque. The shaft consists of three sections: 1) a splined coupling to mate the shaft to the engine output shaft, 2) a basic torque/reference shaft with meter pick-up feature, and 3) a splined adapter to mate the shaft with the transmission input coupling. An adapter with an integral spur gear and crowned spline is mounted on the engine output shaft and is locked against axial motion. This spline in conjunction with a similar spline at the transmission end allows the shaft to compensate for any static or dynamic misalignment and/or axial motion between the engine and transmission.

The torquemeter shaft is designed to provide adequate twist for accurate torque readings and has a flange at each end. A reference shaft is mounted on the O.D. of the flanges, fixed at the forward end by a curvic coupling and allowed to float on a bronze bushing at the aft end. Teeth are cut on the O.D. of the reference shaft (aft end) in line with the similar teeth in the O.D. of the aft coupling. A magnetic pick-up mounted in the housing detects any change in the phase relationship between the two rows of teeth, due to twist of the torque shaft as torque is applied, and relates this to the torque indicator in the cockpit.

The torquemeter shaft is enclosed by a two piece cast magnesium housing. The aft section telescopes into the forward section to facilitate removal and, when extended, are bolted together to form a rigid housing.

B. Engine Speed Control Drive Assembly

The engine speed control drive assembly senses the speed of the engine output shaft and transmits this input to the engine fuel control. The gear train in this unit consists of a spur gear and two bevel gear meshes. A flex shaft connects the drive assembly with the fuel control unit. Gears and bearings are lubricated with oil from the main transmission.

C. Main Transmission

The main transmission distributes engine power to the main rotor, propeller/tail rotor and accessories. The transmission mounts on the two major fuselage frames at four points. In addition to being the supporting structure for the gears, it provides a primary structural load path for the external main rotor loads and drive system reactions which are carried through the main housing into the fuselage structure. Power is transmitted from the engine, through an engine/torquemeter shaft, through an overrunning clutch to the input shaft. At this point the power path is divided, one going forward through the main rotor drive section to the main rotor and the other down through the propeller drive section to the propeller/tail rotor. The overrunning clutch automatically disengages the engine from the transmission when the main rotor RPM exceeds equivalent engine speed.

The power distribution is:

Engine Input	3925 HP @ 13,600 RPM
Propeller Drive	3880 HP @ 1717 RPM
Main Rotor Drive	3100 HP @ 246 RPM

Two drives are provided for the accessories, one from the rotor mast for the control hydraulic pump and the other from the propeller drive section. An APU input is provided in the aft cover to operate the accessories while on the ground, system check out and/or engine start. All gears and bearings are primarily jet lubricated with MIL-L-7808 or MIL-L-23699 oil.

Power is transmitted from the engine to the main transmission through the input overrunning clutch outer race. The locking action in the clutch is performed by a single row of sprags between the outer and inner race. Concentricity of the inner and outer races is maintained by two bearings, straddling the sprags. The sprags will roll into engagement when the outer ring starts to overtake the inner thus locking the clutch. The power is then transmitted to the input spur pinion shaft which is splined to the clutch inner race.

Power for the main rotor drive system is transmitted from the input spur pinion shaft (propeller drive) through a splined quill shaft, spiral bevel gear mesh, compound planetary, simple planetary and rotor mast to the main rotor. The input speed is reduced at each stage and is tabulated below.

Stage	Gear Mesh	Reduction Ratio	Speed	
			In	Out
1.	Spiral Bevel	4:1	13600 RPM	3400 RPM
2.	Compound Planetary	5:1	3400	680
3.	Simple Planetary	2.763:1	680	246.1

The compound planetary sun gear, located below and splined to the spiral bevel gear, drives the compound planet carrier through five compound planet gears reacting against a ring gear splined to the ring gear support. The splines allows the sun and ring gear freedom for alignment under varying loading conditions and deflections. This will minimize unequal loading between the planet gears. The planet gears are supported by bearings mounted in the upper and lower planet carrier plates. Spacer blocks located between each planet gear separate the upper and lower carrier plates and maintain bearing bore alignment. The spline backlash between the ring gear and ring gear support has been increased and a lever is installed to magnify the ring gear movement during a torque reversal (when the power drive shifts from the engine to the main rotor). This movement is used as an input signal to the Delta Beta System.

The simple planetary sun gear, located below and splined to the compound planet carrier, drives output planet carrier through ten planet gears reacting against a ring gear splined to the ring gear adapter. The planet carrier is a one piece design with cantilevered trunion posts which support the planet gears/bearing assembly. The base of the planet gear is integral with the self-aligning bearing outer race. The self-aligning feature in combination with the alignment freedom of the splined sun and ring gear will minimize unequal leading of the planet gears. The carrier support is splined to the output planet carrier and the rotor mast and contains a shear section. The shear section is designed to fail in the event of a transmission gear train lock up. The failure of the shear section will then allow the main rotor to autorotate freely and independent of the transmission gear train. The rotor mast is a large tube with a flange on the top that interfaces with the main rotor and is made from a titanium forging. It is supported by three bearings, an upper roller bearing and a roller/ball bearing combination at the lower end. The large internal diameter of the rotor mast was designed to provide space to route the controls from the swashplate (located beneath the transmission) to the main rotor.

Propeller Drive Section. The power is transmitted to the propeller drive through the first and second stage spur gear meshes. The input speed is reduced at each stage and is tabulated below:

Stage	Gear Mesh	Reduction Ratio	Speed	
			In	Out
1.	Spur	3.644:1	13600 RPM	3731 RPM
2.	Spur	2.173:1	3731	1717

The power from the input shaft is transmitted through a spur pinion (integral with the shaft)/gear mesh to the intermediate shaft. Included on this shaft are an accessory drive overrunning clutch, second stage spur pinion, and rotor brake disc mounted on the end of the shaft extending outside the aft cover. The second stage spur gear, driven by the second stage pinion, drives a shaft which connects to the propeller/tail rotor drive shaft.

Accessory pads are located on the transmission in three different areas, namely: (1) forward accessory cover, (2) upper forward section of the main housing and (3) aft cover. The power for the accessories can be supplied by either the engine, rotor (i.e., autorotation) or APU as indicated in the following table.

Table I. Transmission Accessory Drives

Accessory	Location	Operated By		APU	SPEED (RPM)	Power Requirements	
		Engine	Rotor			MAX-RATED (HP)	SURGE (HP)
Alternator #1	Accessory Cover	X	X	X	12,571	27.5	55
Alternator #2	Accessory Cover	X	X	X	12,571	27.5	55
Hydraulic Pump-Utility	Accessory Cover	X	X	X	6,954	43.6	45.3
Hydraulic Pump-Control	Main Housing	X	X		5,931	10.5	12.0
Transmission Oil Pump	Aft Cover	X	X	X	3,643	6	10

The power is transmitted through the accessory overrunning clutch through an idler gear (drives the transmission oil pump), and a driven spur gear, mounted on a shaft integral with a spiral bevel gear, to the accessory drive shaft. A spur gear mounted in the forward accessory section receives the power from the drive shaft and directs it to the accessory power take-off drives through an idler and 'ball' gear. The Hydraulic Utility Pump is driven by the idler gear and all the other accessory drive gears are driven by the 'bull' gear.

When on the ground, the above system can be powered by the APU through a spiral bevel gear set mounted in the aft cover. This gear is designed to operate at 100 HP @ 8000 RPM (maximum APU power). The accessory overrunning clutch isolates the accessory drive section from the main rotor and propeller gear train when the accessories are being powered by the APU.

The Control Hydraulic Pump is driven by a spur gear mounted on the rotor mast through a spur pinion and spiral bevel gear set increasing the speed from 246 to 5931 RPM. This drive is on the rotor side of the shear section so that the pump is driven at all times when the rotor is turning thus ensuring hydraulic power to the aircraft flight controls should the shear section fail.

A magnetic pick up mounted in line with the gear teeth of the spur pinion senses the gear RPM and relates this to the cockpit instrument in terms of rotor RPM.

Lubrication System. The Main Transmission Lubrication Pump, is a three element lube and scavenge design and is mounted on the transmission aft cover. The pump contains one lube element and two scavenge elements. One of the scavenge elements has a double inlet functioning as two elements. A drain boss on the pump inlet provides a means for draining the oil tank during servicing. Oil from the tank enters the pump and is pressurized by the lube element, flows past the filter bypass valve, and through the filter. The pump capacity is rated at 25-30 GPM at 100 psi. A differential pressure of 23 to 31 psi will actuate a visual indicator for a clogged filter. The indicator will remain activated until manually reset. The filter contains a disposable 70-micron absolute filter element capable of removing 98 percent of all particles whose two smallest dimensions are greater than 40 microns. The oil flow continues past the pressure relief valve through the lube shut-off valve to the oil discharge ports. The lube shut-off valve is a spring actuated check valve to prevent the oil in the tank draining into the transmission sump. There are two oil discharge ports on the pump, one external port which supplies oil to the lower swashplate and to the speed control drive assembly through external lines, and one port on the pump mounting flange to supply oil to the main transmission. The oil is then routed through internal passages to various jets. The jets direct the oil to gears, bearings and working splines. The oil drains down to the forward and aft transmission sumps and the swashplate sump. The scavenge elements draws the oil from these areas past an electric chip detector, through an inlet screen, over a temperature indicator and a temperature limit switch through the oil cooler into the transmission oil tank. A bypass circuit in the system provides for automatic and manual bypassing of the oil cooler and direct return of scavenged oil to the oil tank. Electrical power is supplied the bypass circuit by the essential dc bus through the XMSN OIL CLR BYPASS circuit breaker.

D. Transmission Oil Tank

The transmission oil tank is self-sealing. The tank is mounted in the cowl above the transmission accessories, just forward of the main rotor mast. An access door, dipstick, cap, and sight gage are located on the right side of the tank. A float switch and drain valve are installed in the tank. Overall capacity of the tank is 8 gallons. When fully serviced to the filler well, the tank holds 6-1/4 gallons. The plumbing from the tank, transmission oil pump, and oil cooler contains approximately 2-1/2 gallons of oil.

E. Transmission Oil Cooler Bypass Switch

The transmission oil cooler bypass switch labeled XMSN OIL COOLER, is a three-position toggle switch located on the fuel control panel in the pilot station. The switch allows scavenged transmission oil to bypass the oil cooler. The three switch positions are marked ARM, OFF, and BYPASS. When the switch is in the OFF position, the bypass valve is set to route scavenged transmission oil to the oil cooler prior to returning it to the tank. When the switch is in the ARM position, scavenged transmission oil is routed to the oil cooler until the oil level in the tank drops to approximately 2 gallons. When the tank quantity reaches 2 gallons, a float switch in the tank closes and positions the bypass valve to route scavenged oil directly to the tank, bypassing the oil cooler. There is a time-delay relay in the circuit to prevent bypassing of oil when the float switch closes, as from sloshing of oil. Also, when the float switch closes, it completes a circuit to the XMSN OIL LEVEL caution light on the FLAWS status panel, causing it to come on. When the switch is in the BYPASS position, the oil cooler is bypassed and scavenged oil is returned directly to the tank; thus, the oil cooler can be bypassed at any time regardless of the quantity of oil in the tank. Normally, the BYPASS switch position is used only during preflight to check out the transmission oil bypass circuit.

F. Tail Rotor Drive Shaft Assembly

A two-part shaft at right angles to the axis of the propeller transmits 600 HP normal, 940 HP maximum and 1200 HP transient at 1238 RPM to the antitorque rotor. It is enclosed in the aft portion of the left-hand stabilizer.

The inboard section of this shaft is a coupling between the right angle spiral bevel gear set in the prop gearbox and the outboard shaft length. Crowned splines compensate for misalignment between the prop gearbox and the tail rotor spindle. The splines are oil lubricated,

G. APU Gearbox/Clutch Assembly

The APU Gearbox/Clutch Assembly transmits 100 HP maximum at 8000 RPM from the APU through an APU Drive Shaft to the main transmission to operate the accessories on the ground for system check out and for engine starting. This assembly consists of a right angle gearbox and a centrifugal clutch.

The gearbox is a self-contained unit with a 1:1 ratio spiral bevel gear set mounted in bearings. An integral pump supplies

MIL-L-7808 oil to lubricate all moving parts and the finned cast magnesium lower cover acts as a sump and oil cooler. A sight glass, filler plug and breathers facilitate servicing of the oil level. The parts are assembled in a cast magnesium housing which bolts to the front face of the APU.

A centrifugally operated, self-energizing, dry-disc type clutch is mounted on the gearbox output shaft spline. It has a torque capacity of 600 inch-pound continuous rating and will start to slip at 1000 inch pounds. A governor contained within the clutch prevents engagement below 5500 RPM so that the APU can become self-sustaining and capable of picking up a load without stalling. The clutch automatically disengages when the driving torque is removed.

H. APU Drive Shaft

The APU clutch-gearbox assembly is connected to the transmission by the APU drive shaft assembly. This assembly is basically two Bendix flexible diaphragm type couplings joined by an aluminum tube.

One coupling is flanged and has a pilot to mate with the flange on the APU clutch housing, while the other coupling has an external spline which mates with an internally splined adapter attached to the APU input gear on the transmission. The spline is grease lubricated and is sealed by an "O" ring on the coupling.

Angular displacements of the system of two degrees continuously and four degrees intermittently are accommodated by flexture of the diaphragms.

IV. SYSTEM OPERATION

A. Rotor Brake

The rotor brake is a single disk, hydraulically actuated mechanism used to decelerate, stop and hold the main rotor, propeller and tail rotor from rotating. In addition it is used to prevent the drive system from rotating during engine start and engine ground idle.

The rotor brake consists of two parts, namely: (1) the brake disk which is mounted on the aft end of the first stage reduction gear shaft, and (2) the brake head assembly which is mounted on the aft cover. The brake is actuated by the aircraft hydraulic system, Figure 18, under a pressure of 3000 psi, forcing the carrier and lining assemblies against the brake disk, that is driven by the rotor gearbox. The braking action resulting from friction slows and finally stops the drive system. Release of the brake hydraulic pressure allows the springs in the adjuster assemblies to retract the carrier and lining assemblies from the disk.

B. Operating Limits

Transmission Oil Temperature Indicator

0-113°C Normal Operating Range	Green Band
113°C Maximum	Red Line
	113°C to 130°C for 30 min. Set PLF* at 90-100 Kts
	130°C to 135°C for 10 min. Total time above 113°C not to exceed 30 min. Set PLF* 90-100 Kts

*PLF = Power for Level Flight

NOTE

With an oil cooler fan failure, maintain 130 KIAS.

LOCKHEED CALIFORNIA COMPANY

Transmission Oil Pressure Indicator

70 psi minimum (100 percent N_R)	Red Line
70 to 110 psi normal operating range (100 percent N_R)	Green Band
110 psi maximum	Red Line

Rotor Brake

Application	Do not apply with engine running
Application	40 percent N_R (with engine off after T.I.T. Below 320°C)

NOTE

Rotor brake may be applied before engine start but must be released at ground idle. Do not attempt to keep rotor brake on beyond ground idle when running up.

D. Failure Detection System

Main Transmission. The main transmission utilizes the oil temperature and pressure switches, electric chip detectors, and the oil temperatures and pressure gages to detect start of failures.

Transmission Oil Caution Light. A caution light, marked XMSN OIL, is located on each annunciator panel. Both lights come on to provide a caution that either the transmission oil temperature is above 120°C (248°F), or transmission oil pressure is below 70 psi. Concurrently, a voice warning message is supplied to the crew. (The copilot/gunner should also check the status of the XMSN OIL LEVEL caution light on the FLAWS status panel to determine if the low-oil-pressure or high-oil-temperature condition is due to low oil quantity.) The lights are activated by an oil temperature and/or pressure switch located in the transmission oil system.

Chip Detector Caution Light. Two chip detectors are installed in the main transmission and one detector in the swashplate. When sufficient chips collect on any of the detectors, an electrical circuit is completed to the FLAWS to cause a CHIPS caution light to illuminate and a voice warning to be heard in the headsets.

Oil Temperature Indicator. Transmission oil temperature is shown on the lower (XMSN) scale of the oil temperature indicator.

Oil Pressure Indicator. Transmission oil pressure is shown on the lower (XMSN) scale of the oil pressure indicator.

V. PCRS CONFIGURATION

The PCR design studies resulted in a number of changes to the Main Transmission. The philosophy of these changes was to bring all life limited parts up to 3600 hours minimum service life and to uprate the time between overhaul (TBO) to 1200 hours minimum. The PCR configuration drawing shows the major changes to the transmission which have been summarized below:

1. One Piece Input Gear - The input spur pinion and spiral level pinion are made integral with a common shaft.

2. Integral Idler Gear Shaft and Gears - The first stage gear and the second stage pinion are made integral with the intermediate shaft. This requires relocation of the accessory drive overrunning clutch to the transmission oil pump drive where the clutch size now becomes compatible with loads.

3. One Piece Compound Planet Carrier - Provides a lightweight rigid carrier with the planet gears and bearings mounted on trunnions so that they will not be affected by carrier deflections.

4. Compound Planetary Ratio Change - To prevent misindexing of the planet gears during assembly.

5. Main Housing - Addition of material in critical areas to facilitate casting production.

6. Bearing Locknut Retention - Change to locknuts with inserts to eliminate extra machining and one time usage of lockwashers.
7. Improve Bearing Key Design.
8. Increase face width of propeller drive gear.
9. Increase bearing capacity on intermediate shaft.
10. Include forward propeller shaft adapter coupling as part of transmission assembly.
11. Other miscellaneous small changes to reduce wear, fretting, etc.

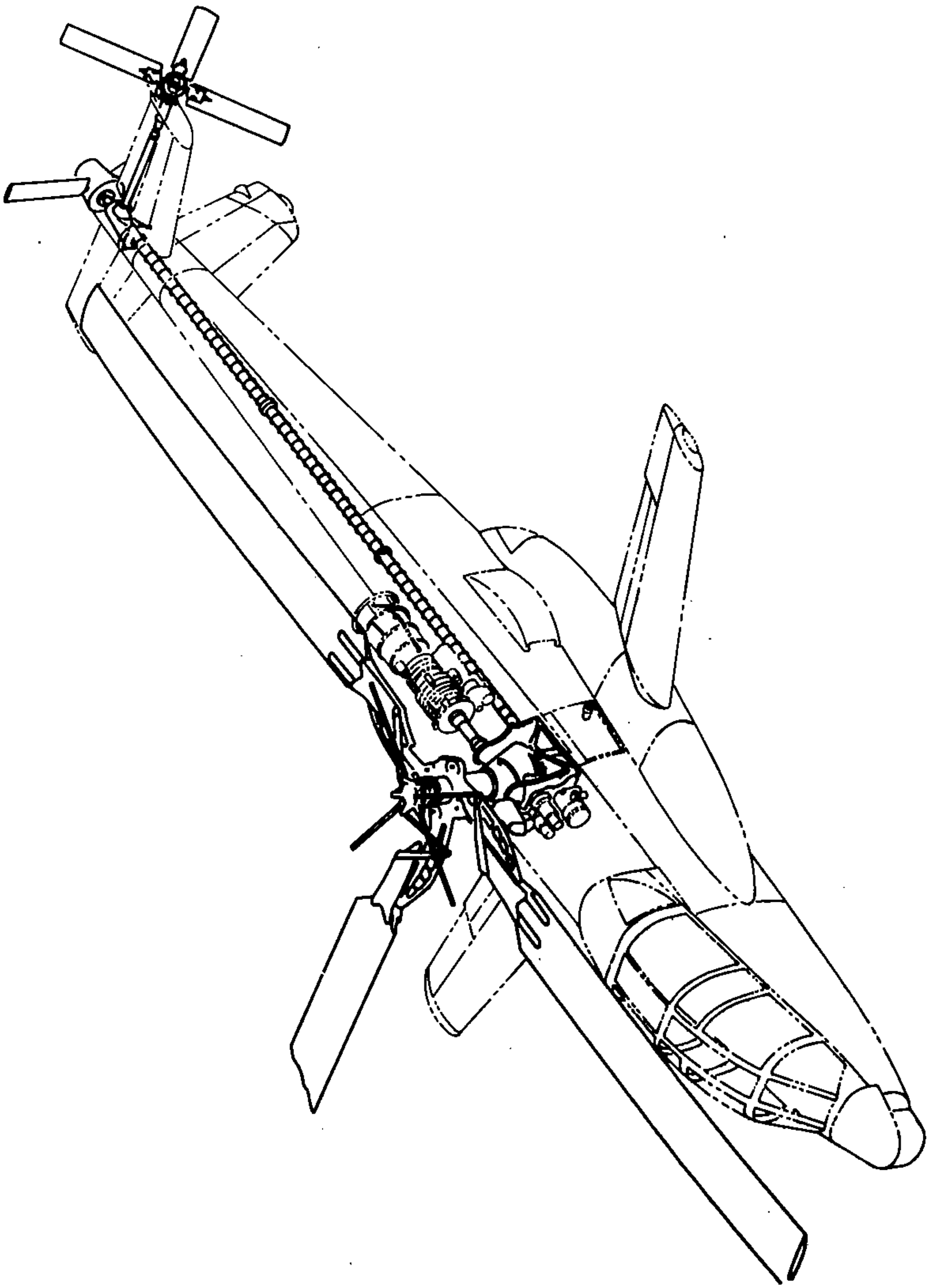
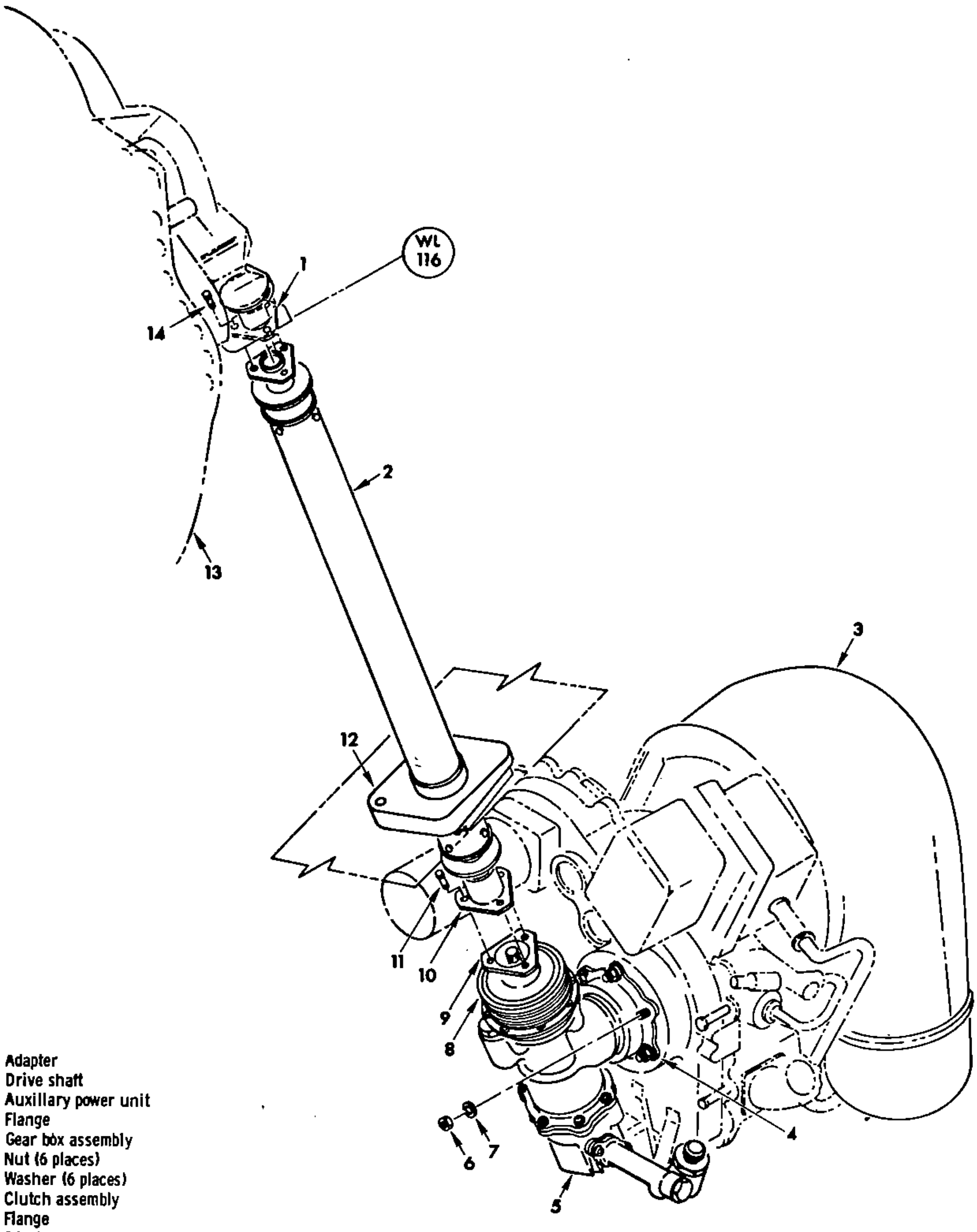


Figure 5-1. Power Train



1 Adapter
2 Drive shaft
3 Auxiliary power unit
4 Flange
5 Gear box assembly
6 Nut (6 places)
7 Washer (6 places)
8 Clutch assembly
9 Flange
10 Adapter
11 Bolt (3 places)
12 Seal block
13 Main transmission
14 Bolt (3 places)

Figure 5-2. APU to Transmission Drive System

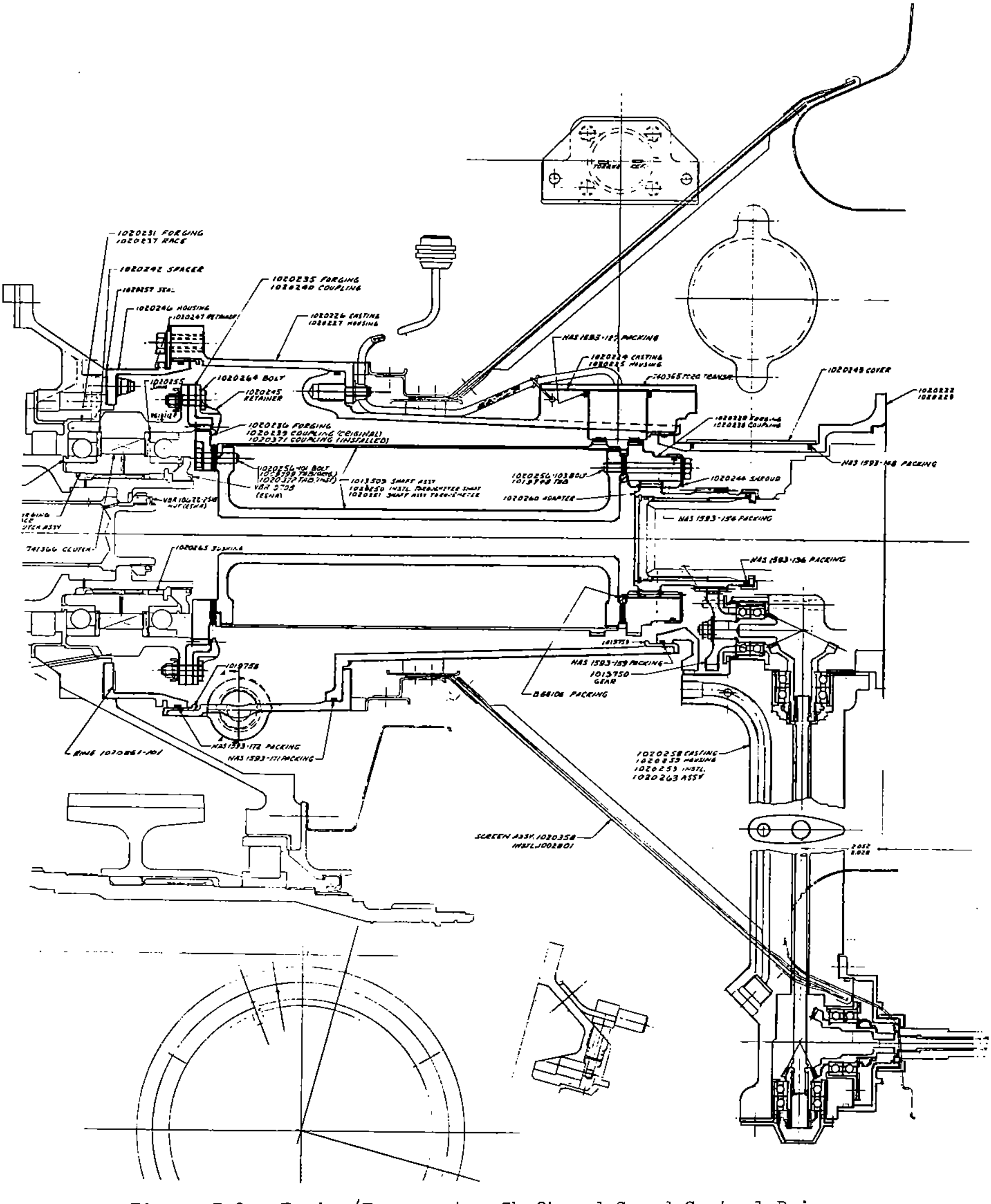


Figure 5-3. Engine/Torquemeter Shaft and Speed Control Drive

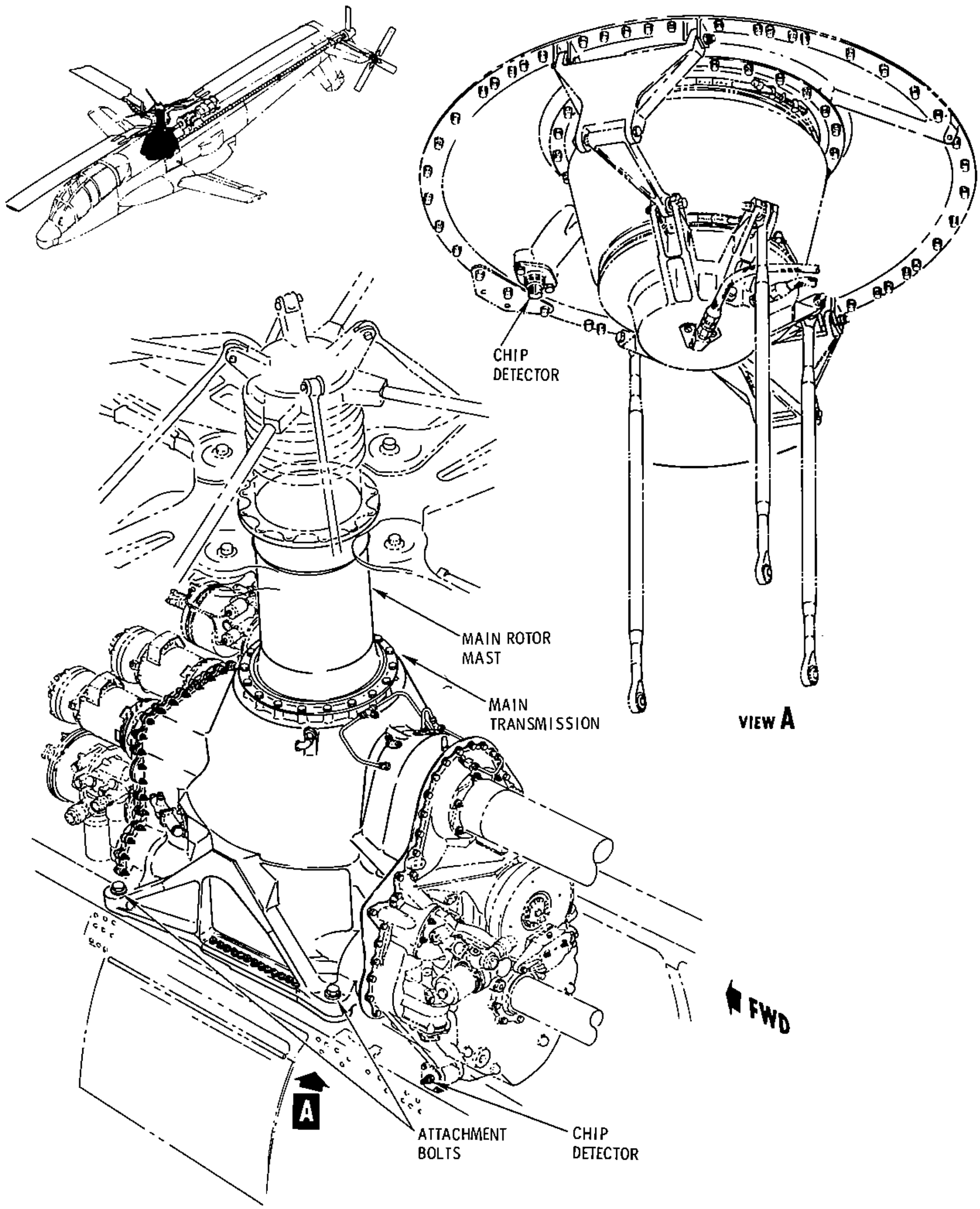


Figure 5-4. Main Transmission

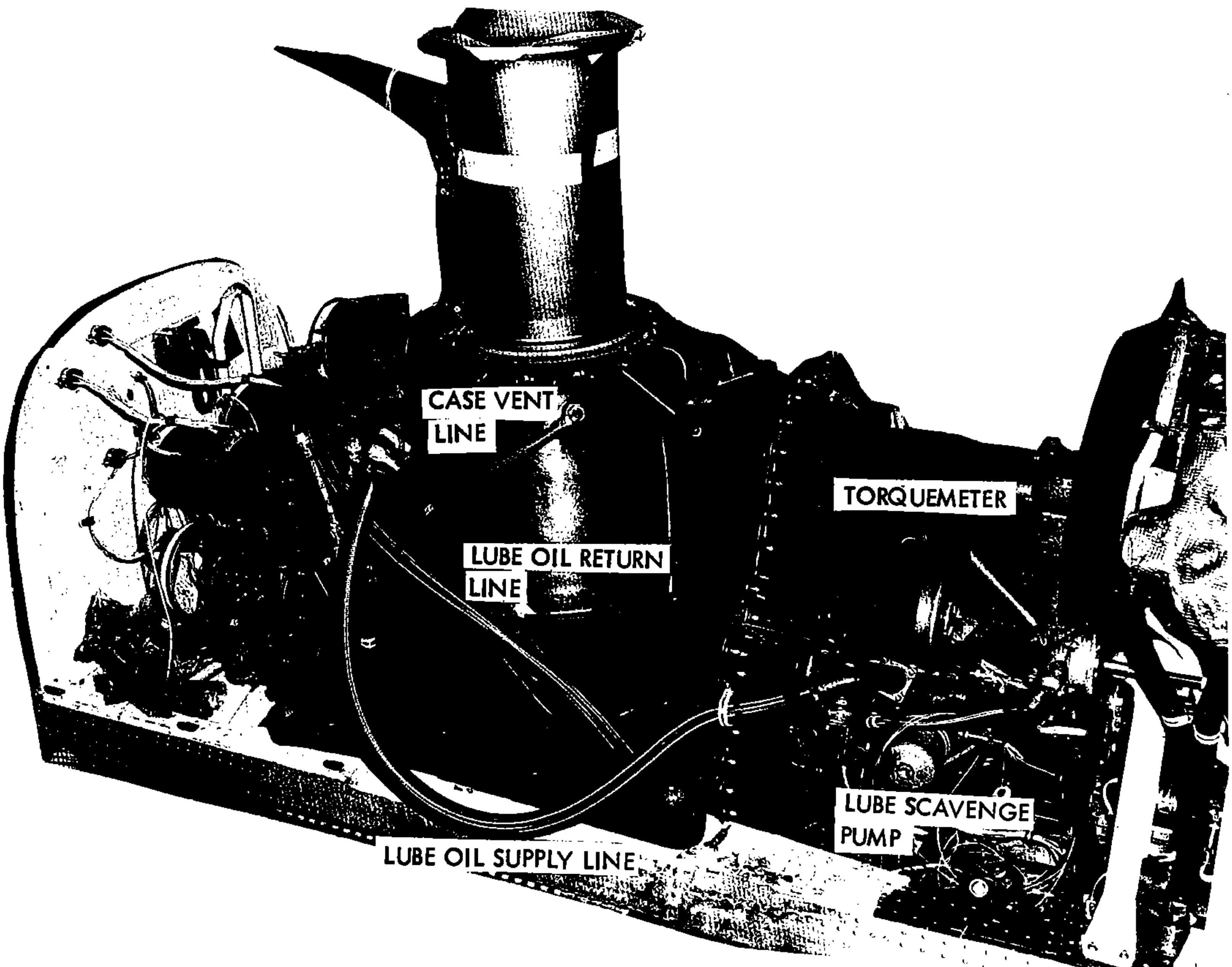


Figure 5-5. Main Transmission Installation - Left Side

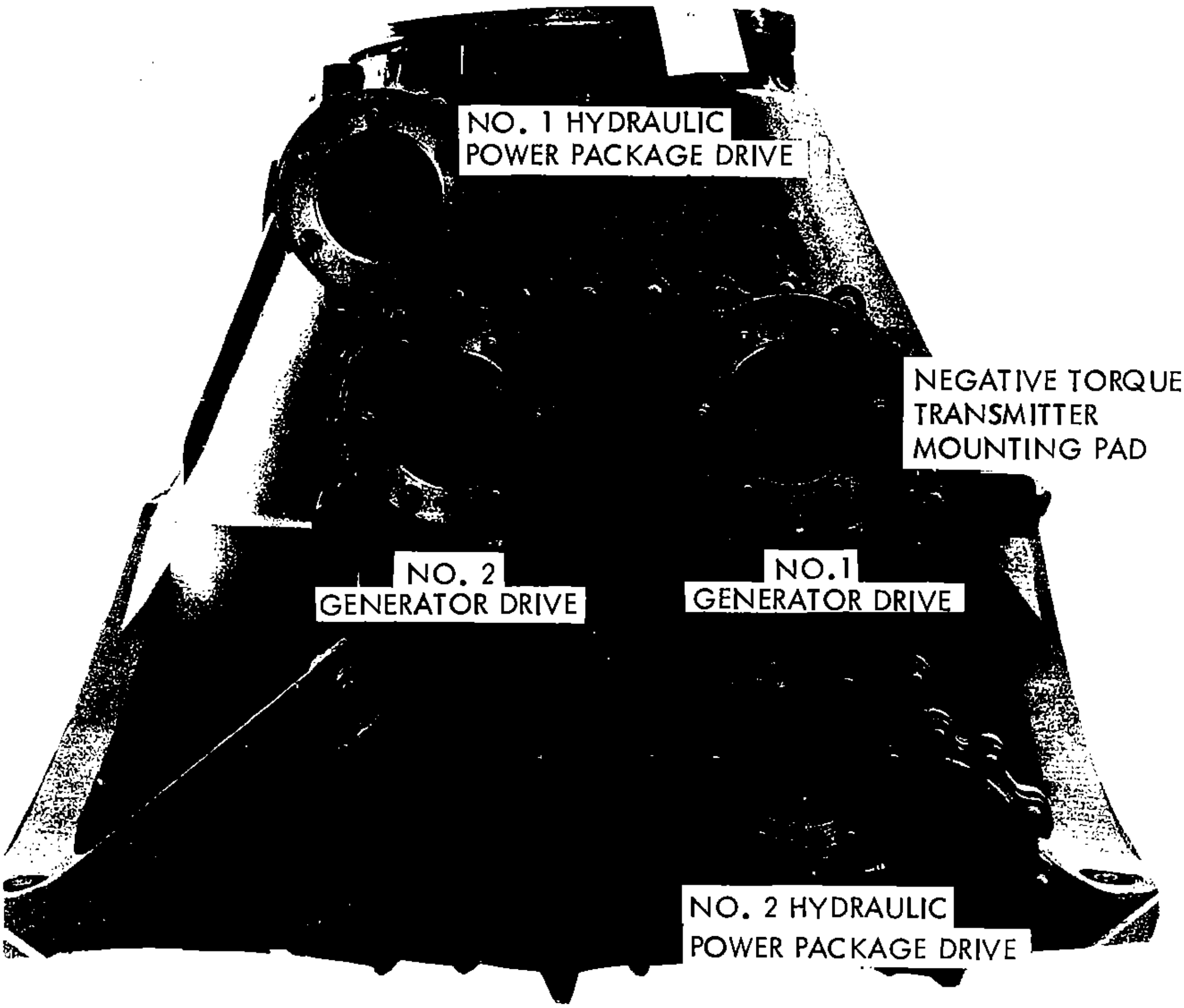


Figure 5-6. Main Transmission Forward Case

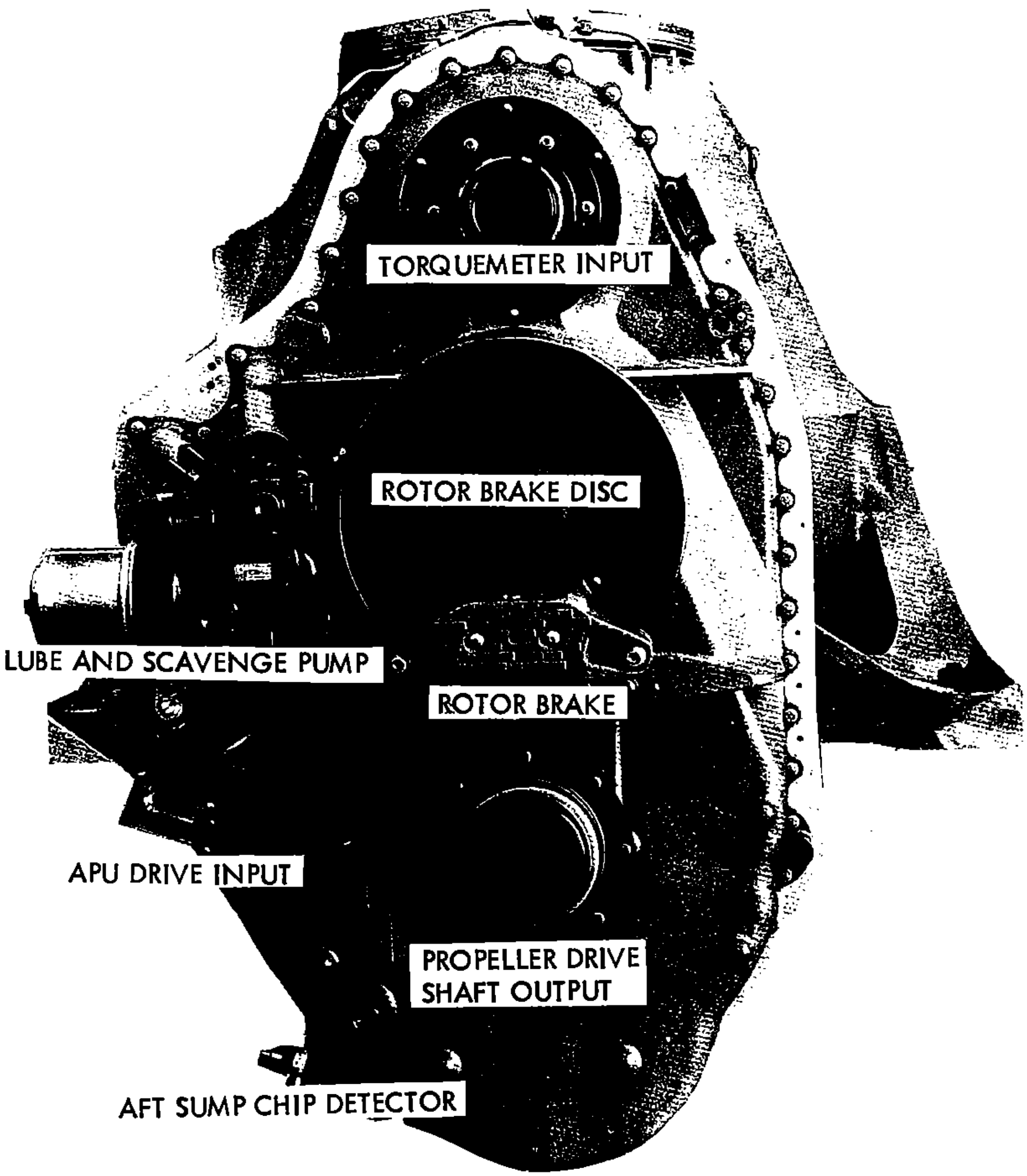


Figure 5-7. Main Transmission Aft Case

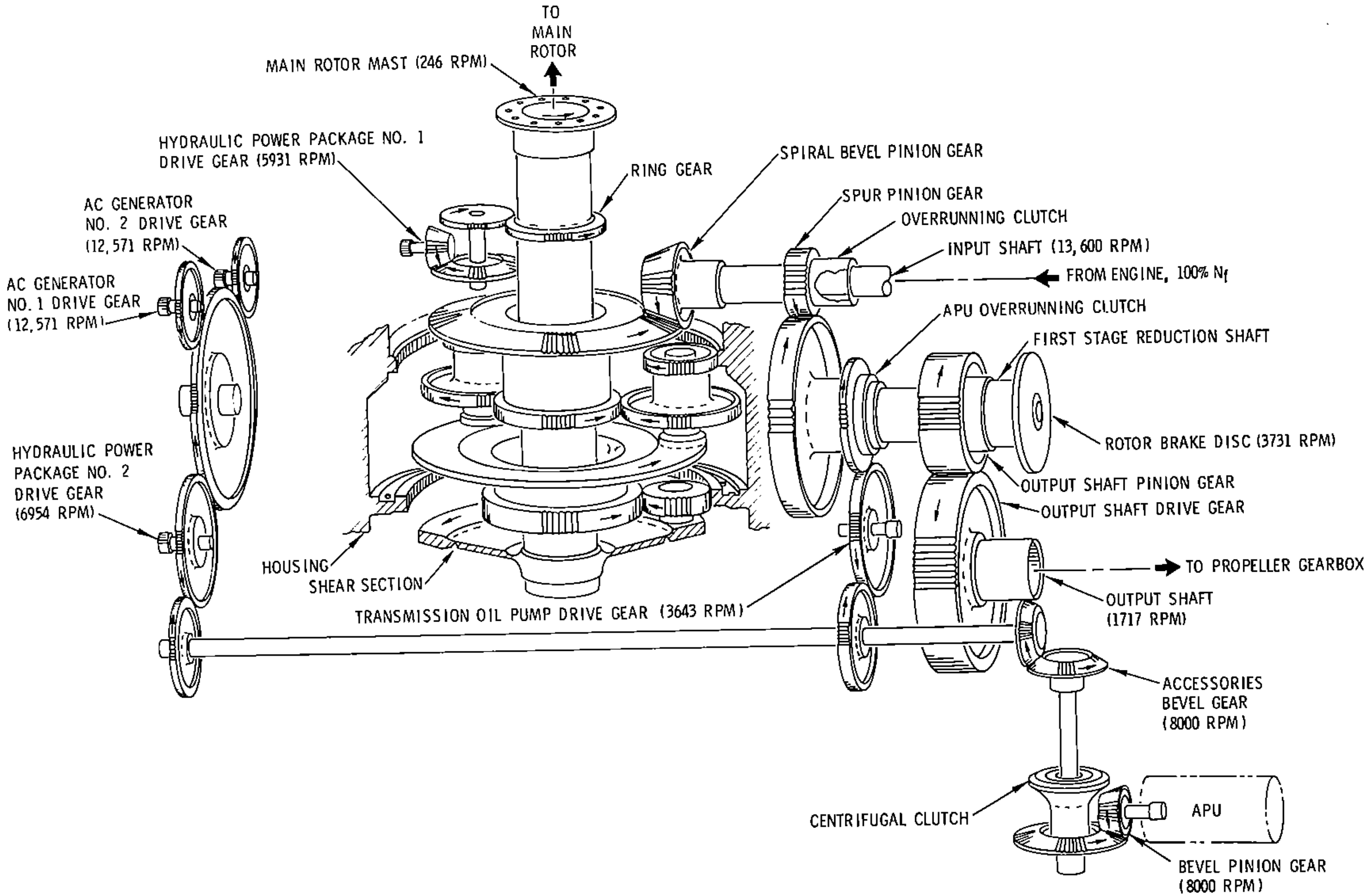


Figure 5-8. Main Transmission Gear Trains

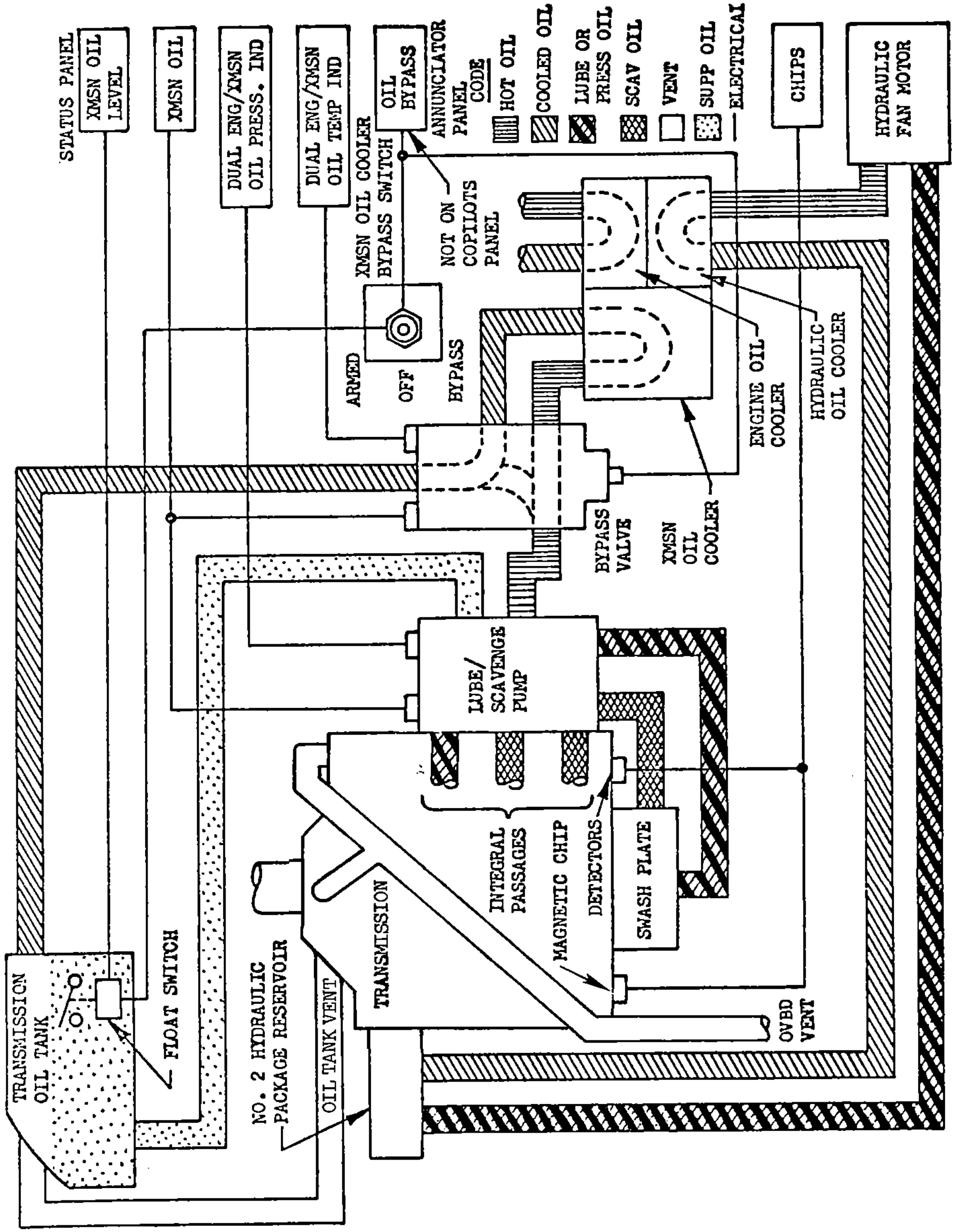


Figure 5-9. Main Transmission Lubrication System Schematic

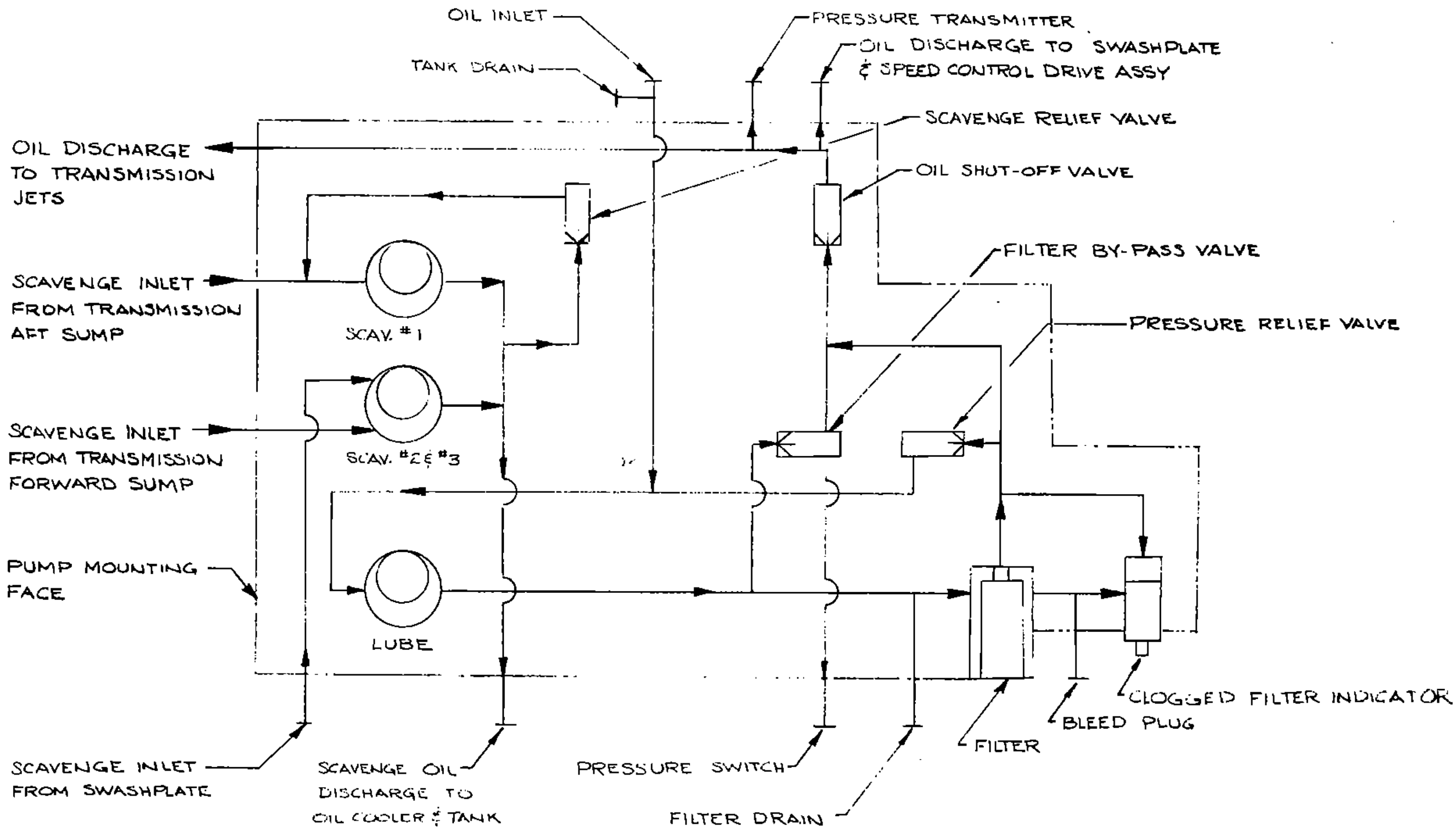


Figure 5-10. Transmission Oil Flow

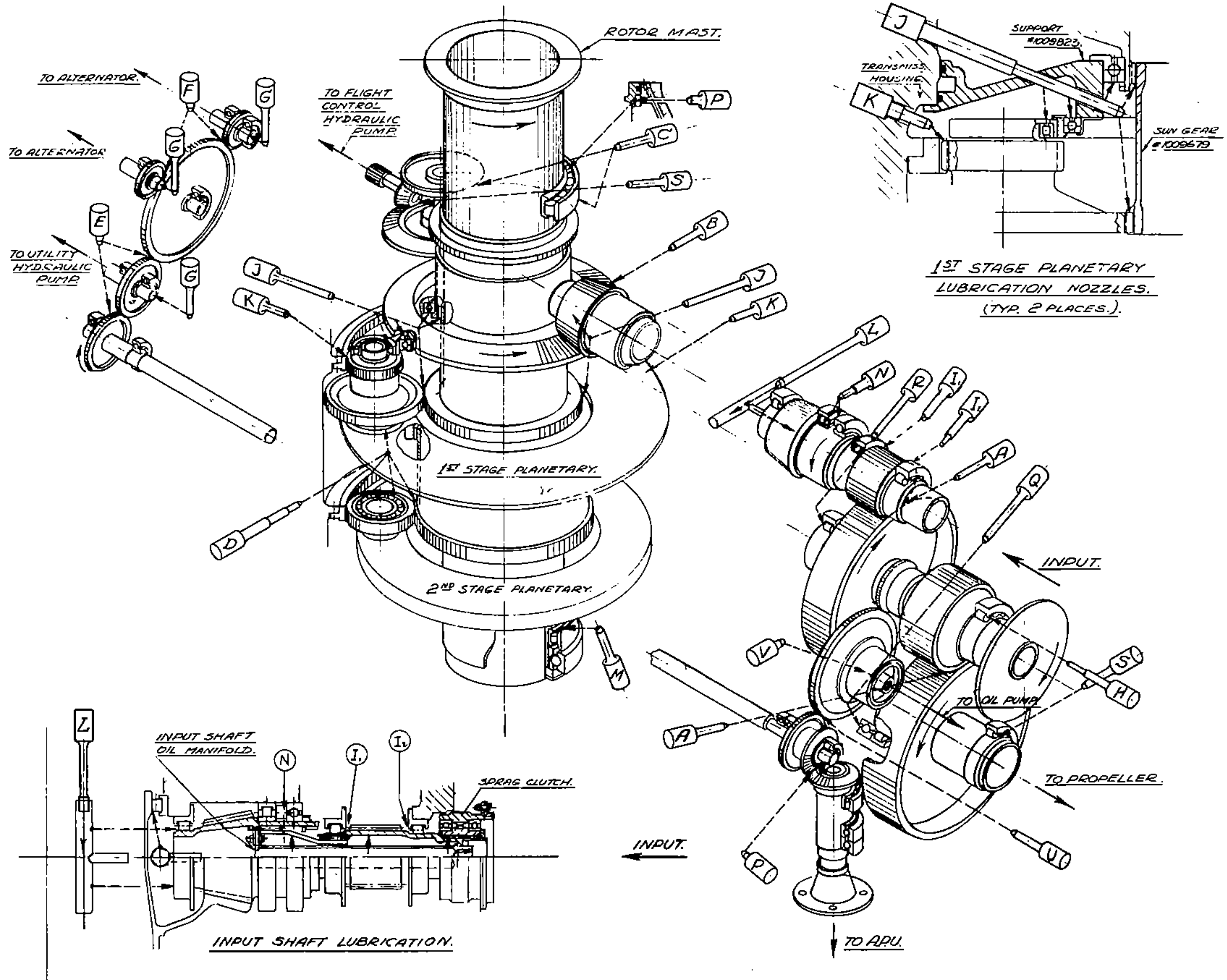


Figure 5-11. Main Transmission Lubrication System

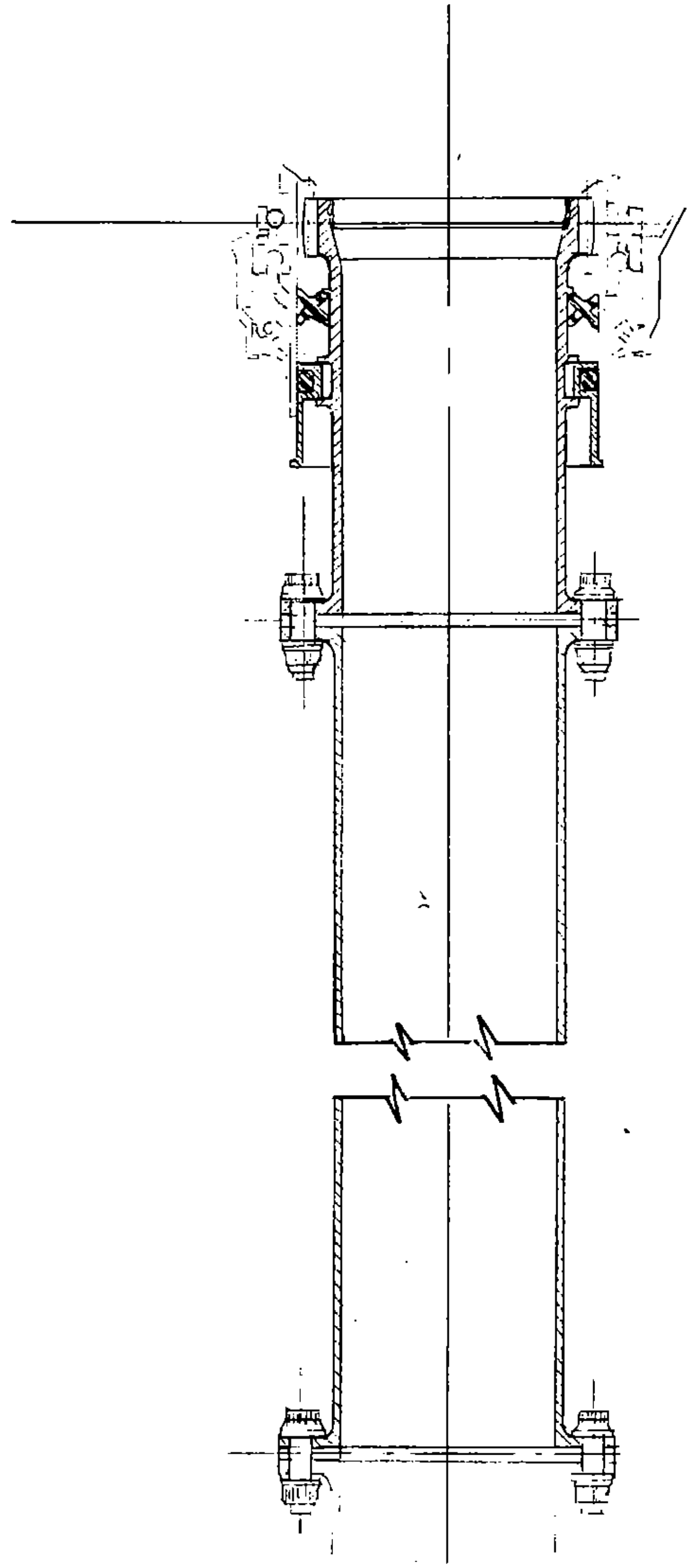


Figure 5-12. Tail Rotor Shaft

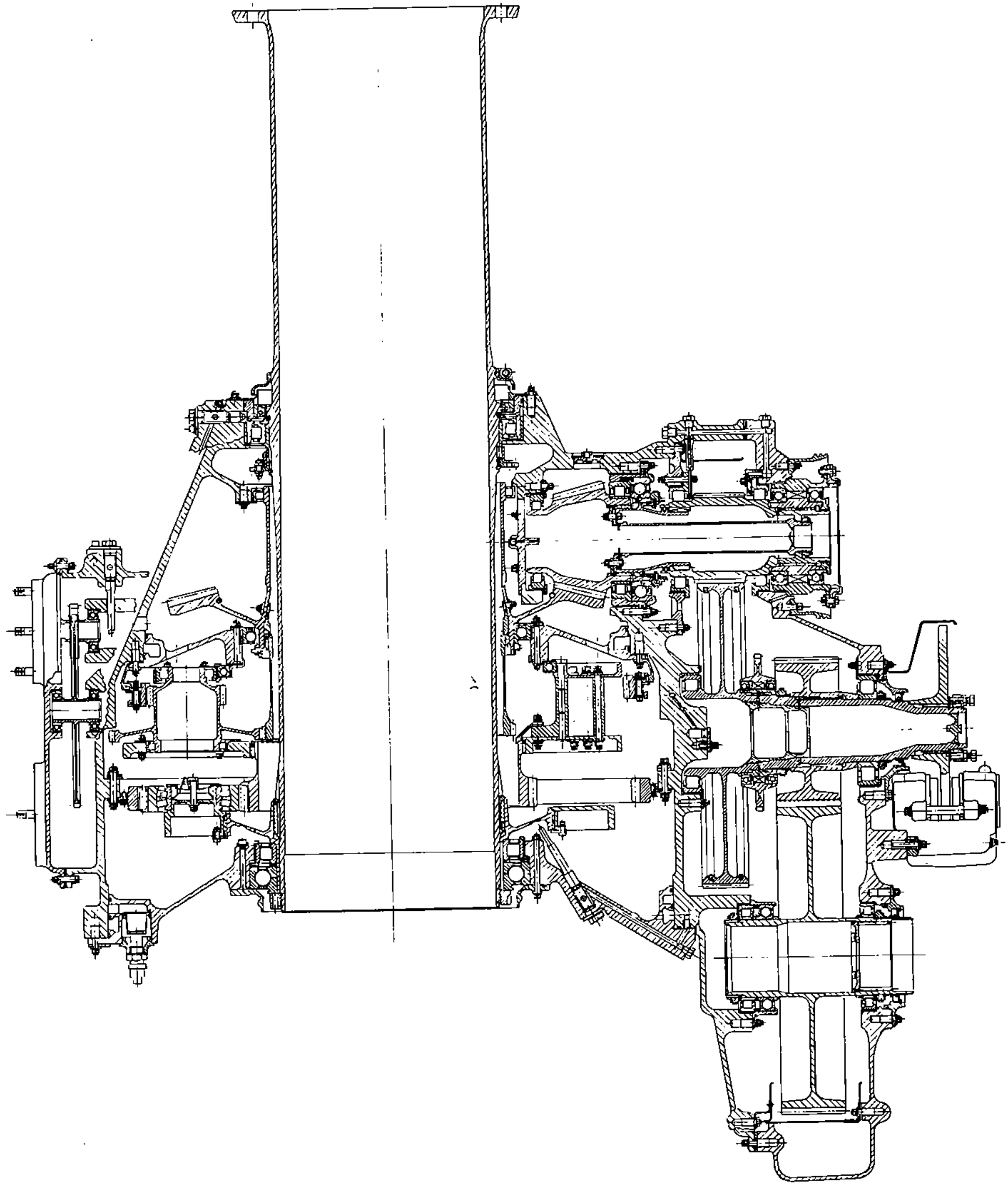


Figure 5-13. Main Transmission Baseline

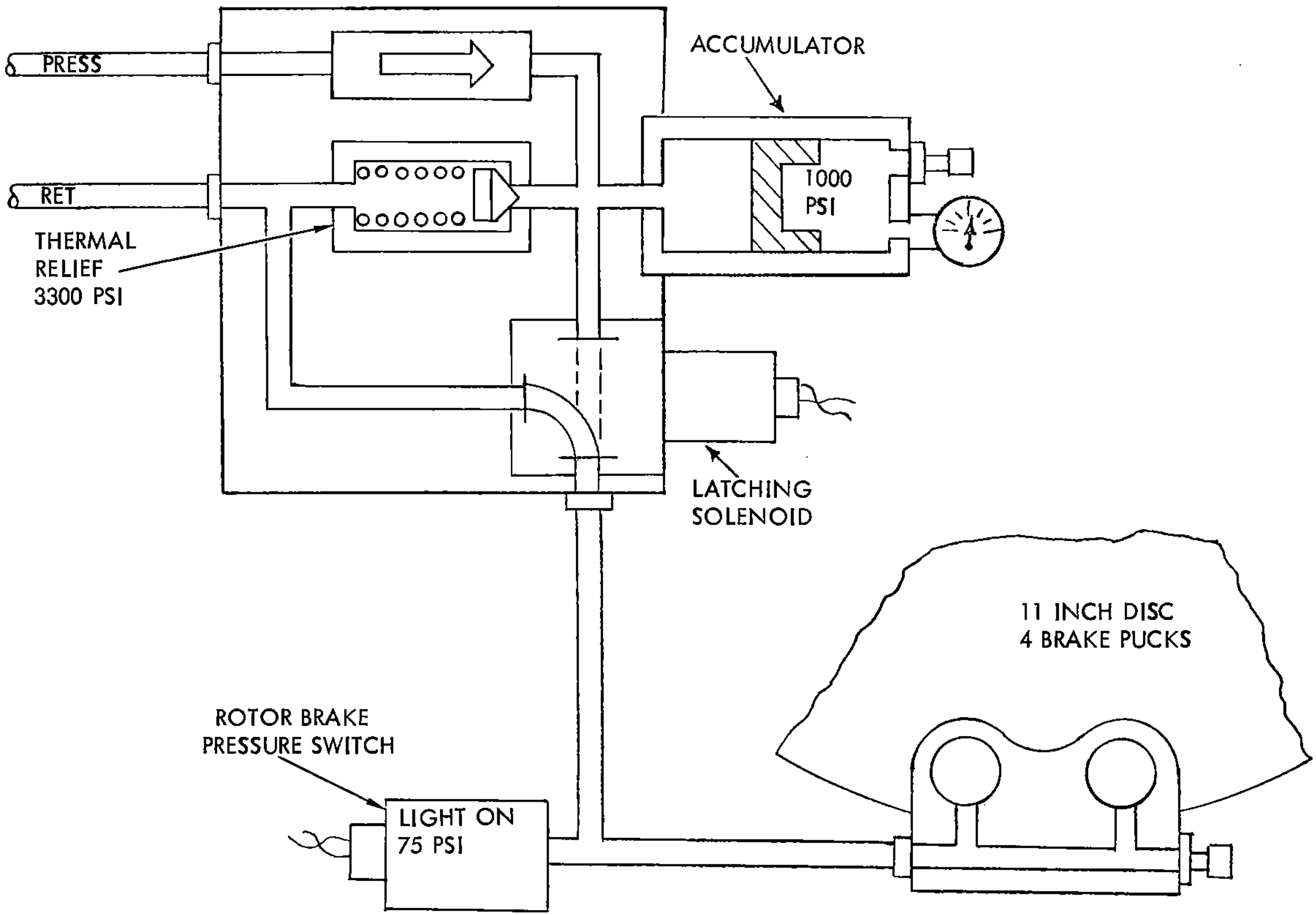


Figure 5-14. Rotor Brake System Schematic

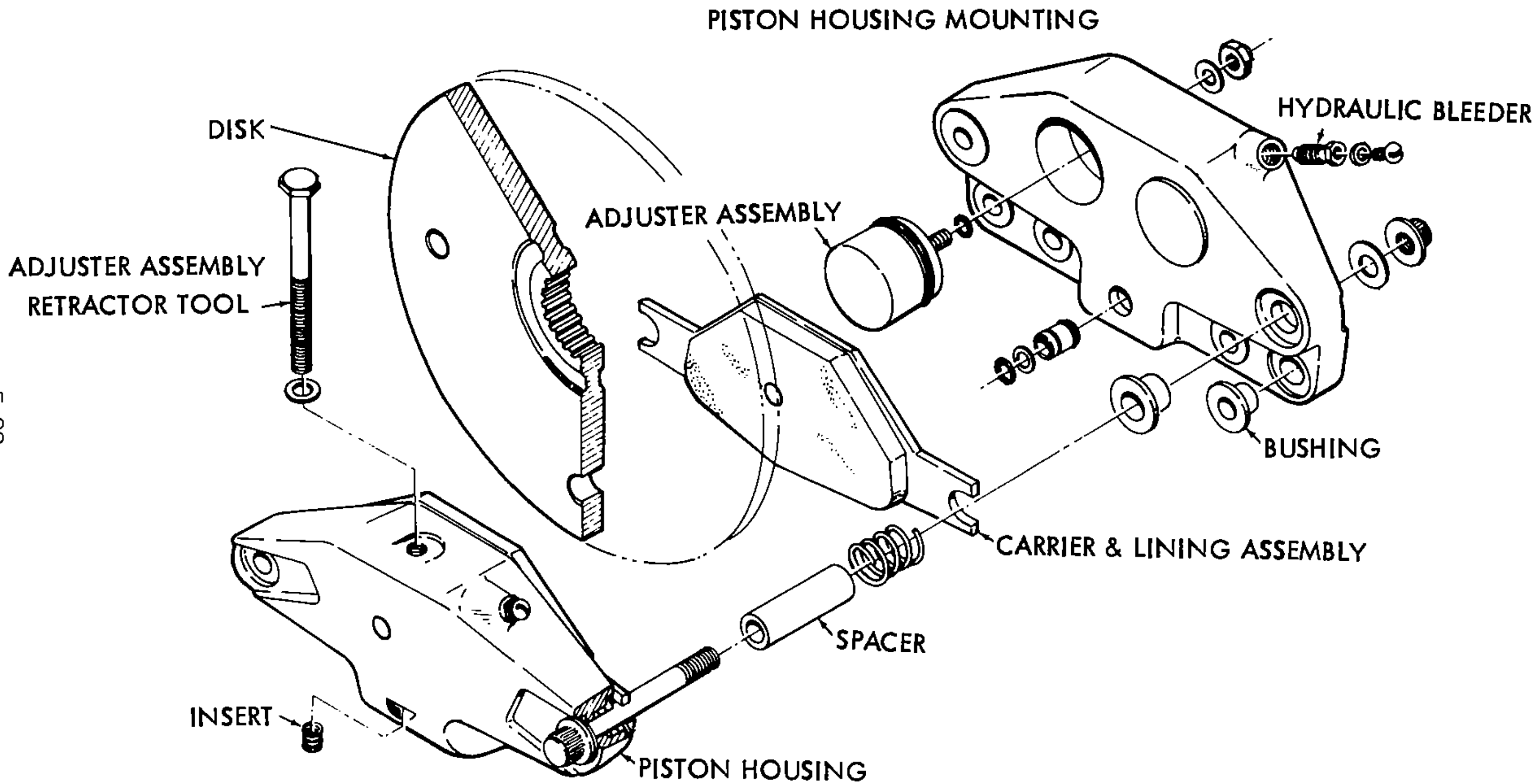


Figure 5-15. Rotor Brake Assembly

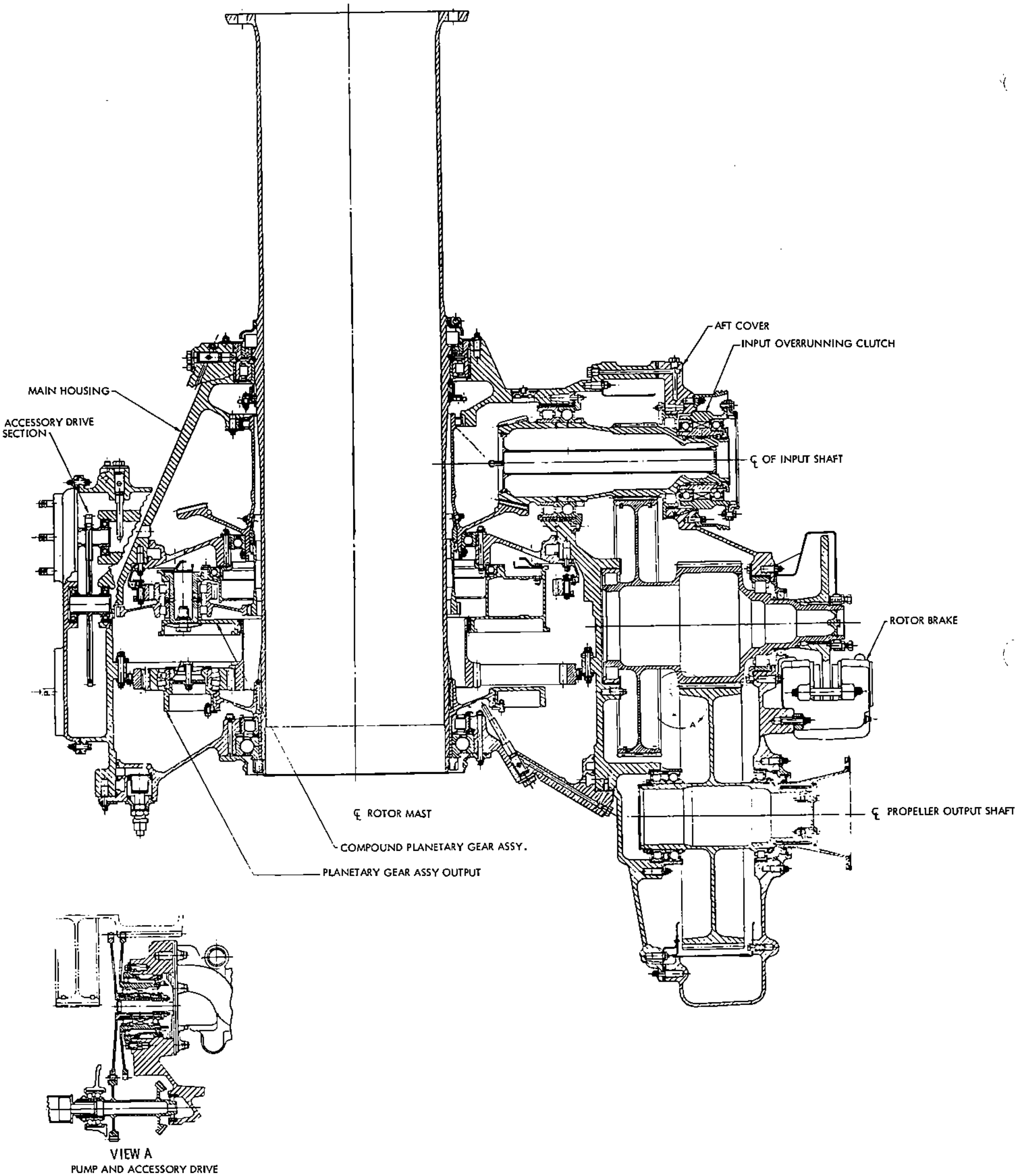


Figure 5-16. PCR Configuration

ACHIEVING HELICOPTER STABILITY
WITH THE LOCKHEED ROTOR SYSTEM

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ABSTRACT

A side-by-side comparison of the stability of a conventional helicopter and of a helicopter with the Lockheed Rotor System is made. It is shown how the use of an attitude gyro and a rotor with cantilevered blades in the Lockheed System provides inherent stability and a short period of oscillation which enhance the matching of the aircraft to the natural capabilities of a pilot.

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ACHIEVING HELICOPTER STABILITY WITH THE LOCKHEED ROTOR SYSTEM

INTRODUCTION

A student pilot reported on his first thirty minute attempt to hover a conventional helicopter: "It's like riding a pogo stick over a floor covered with greasy ball bearings". Another pilot spoke of his first attempt to hover the Lockheed prototype: "For the first thirty seconds, I thought that I would never get the hang of it".

These two comments illustrate an area known as "machine-man matching". If the characteristics of a machine are not matched to the natural capabilities of a man, the machine either cannot be operated or the capabilities of the man must be upgraded by training. One example of the problem is the difference between the training required to successfully ride a tricycle, a bicycle, and a unicycle. In this case, the extra training required is primarily due to the progressive deterioration in stability. Another example is the difference between attempting a docking maneuver with a rowboat, a cabin cruiser, and the Queen Mary. In this example, instability is no problem but the ability to generate accelerations in the required directions becomes more and more degraded until in the case of the Queen Mary the ability to dock without doing damage is completely inadequate and the job must be relinquished to tug boats.

Thus two important elements in machine-man matching can be seen--stability and response to control inputs. The best machine - man matching in the aircraft field involves an aircraft which has high stability and a rapid and positive response to the pilot's control inputs. A conventional helicopter is not only unstable but its response to control inputs is slow, with maximum results appearing several seconds after the pilot starts the control input. These characteristics give the pilot a combination of fear of the instability while making him impatient with the slow response. In many cases, the student pilot, because of his impatience, overcontrols and actually finds himself contributing to the instability rather than damping it.

As in most systems, these poor flying qualities can be improved with electronic black boxes and many helicopters are operational today with black boxes as integral parts of their control systems. Lockheed chose, however, to do the job with a relatively simple mechanical system which involved coupling a gyro to a rotor with cantilevered blades. The replacement of the usual blade flapping hinges with flexible portions of the rotor hub has reduced maintenance and improved reliability of the rotor in addition to improving flying qualities. In the past ten years, three helicopter designs using this concept have been designed and flown with very satisfying results to those involved in the project.

DESCRIPTION OF ROTOR SYSTEMS

Conventional Rotor:

A conventional rotor and its control system is shown in Figure 1a. The rotor blades have individual flapping hinges whose primary function is to relieve blade bending moments. The pilot controls the rotor with a control system whose key element is a swashplate which converts a non-rotating control input from the pilot to a rotating control input at the rotor hub. The rotating control changes the pitch of the blades cyclically each revolution. The response of the helicopter in hovering flight to the pilot's stick input is as follows:

1. The pilot pushes the stick forward.

2. The swashplate is tilted down in front.

3. The pitch of the blade on the right side is reduced and its lift decreases.

4. The blade starts decelerating downward around its flapping hinge. At the same time it is rotating toward the nose. Thus it flaps down over the nose. (This is like the action of a gyro which responds 90° after the control input). In a similar manner, the pitch of the blade on the left side increases and that blade flaps up over the tail.

5. The rotor comes to a stable position with the tip path plane tilted down forward.

6. The thrust of the rotor, which is perpendicular to the tip path plane, is tilted forward and produces a nose down moment about the center of gravity tilting the helicopter nose down which was what the pilot wanted when he pushed the stick forward.

7. The helicopter pitch motion accelerates nose down until the inherent aerodynamic damping of the rotor tilts the tip path plane back perpendicular to the mast at which time a steady rate of pitch is maintained.

Lockheed Rotor:

The Lockheed Rotor System shown in Figure 1b, differs from a conventional system in that the blades do not have flapping hinges but are cantilevered from the rotor mast through a hub with some flexibility in it. In addition, a control gyro is installed between the pilot's control and the blades. The response of the helicopter to the pilot's stick input is as follows:

1. The pilot pushes the stick forward.

2. A spring is compressed in the control system and the resultant force is transmitted up to the left side of the swashplate and then up the rotating control rods to produce a rolling moment to the right on the gyro.

3. The gyro precesses nose down.

4. The pitch of the blade on the right side is reduced and its lift decreases.

5. The blade starts decelerating downward by bending about its root but at the same time it is being rotated over the nose. Thus it bends down over the nose.

6. In a similar manner, the blade on the left side increases its lift and bends up over the tail.

7. The bending of the cantilevered blades into an "S" shape produces a nose-down couple at the top of the mast which tilts the helicopter nose down which was what the pilot wanted when he pushed the stick forward.

8. The helicopter pitch motion accelerates nose downward until the helicopter rate of pitch is the same as the precession rate of the gyro at which time a steady rate is maintained.

   (Note that the input and the final result are identical for both the conventional and the Lockheed helicopters)

RESPONSE TO GUST DISTURBANCES

Conventional Helicopter:

The response to a gust disturbance while the pilot holds the stick fixed is a measure of the stability of a helicopter. Assume that a conventional helicopter is hovering as in Figure 2a and is upset by a gust which tilts the fuselage nose down. The rotor initially stays in its original plane of rotation, but the cyclic pitch generated by the angular difference between the shaft and the tip path plane rapidly causes the rotor to tilt into a position perpendicular to the shaft. The forward tilt of the rotor thrust vector produces a component of force which accelerates the helicopter into forward flight. The increased velocity on the right--or advancing--side of the rotor and the decreased velocity on the left--or retreating--side produces asymetrical lift which makes the rotor flap back around the blade flapping hinges. The resulting rearward tilt of the rotor thrust vector produces a nose up pitching moment about the center of gravity which begins rotating the helicopter back to a level attitude and slowing it down at the same time. This is the so called "speed stability" which rotary wing aircraft have but which is lacking in fixed wing aircraft. As the helicopter develops a pitching rate, the tip path plane lags behind the shaft by an angle which is proportional to the rate. The pitching moment due to this increment of tilt of the thrust vector is thus a damping term. The dynamics of the entire system for a conventional helicopter is such as to produce an unstable oscillation with a period of 4 to 30 seconds depending on the size of the aircraft. Figure 3a shows a time history of such a response following a simulated gust generated by a control pulse. The long natural frequency leads to a time delay between the initiation of a control input used to correct for an upset and the maximum response. This delay can convince the inexperienced pilot that nothing is going to happen and that he should apply even more control only to find several seconds later that he has overcorrected and must quickly apply opposite control.

Lockheed:

The gyro on the Lockheed rotor system provides a horizon reference against which to produce attitude stability. This helicopter is shown hovering in Figure 2b and then upset nose down by a gust. The gyro, however, remains in its original position. Since the pitch of the blades are slaved to the gyro, the blade on the right side receives increased pitch with respect to the shaft which increases its lift causing the blade to bend up over the nose. The pitch of the blade on the left side decreases causing this blade to bend down over the tail, the bending is such as to make the tip path plane stay parallel to the gyro and thus parallel to the horizon. The resulting blade bending moment applied to the rotor hub is a nose up pitching moment which tends to rotate the helicopter back to its original position. Since during this maneuver, the rotor thrust vector remained vertical, no translation of the helicopter was produced and thus the speed stability does not enter into this simplified discussion.

Similarly, the rotor had no pitching velocity with respect to the air so rotor damping is also not an element of the analysis. The use of the gyro as a horizon reference and the use of blade bending moments to produce restoring moments provides a system which is analogous to supporting the helicopter from the horizon with a large spiral spring. This analogy can be used to quickly estimate the natural period of the fuselage oscillation as simply:

$$P_N = 2\pi \sqrt{\frac{\text{Fuselage inertia}}{\text{Rotor stiffness}}} \qquad \text{seconds}$$

For typical Lockheed helicopters the period of this oscillation varies from one half to two seconds depending on the size. The relatively short natural period and the inherent stability produces a helicopter which responds rapidly to corrective control inputs with no danger of overshoot. A typical time history for a pulse control input for the Lockheed Model 286 is shown in Figure 3b. Note that the pitch rate closely follows the control input.

The Lockheed system also incorporates a feature which has the effect of acting as a gust alleviator and as an isolator of the rotor with respect to some of the basic rotor stability effects. This feature consists of sweeping the blade structural axis slightly ahead of the feathering axis as shown in Figure 1b. If the rotor is bent back by a gust, a component of the blade flapping moment will generate a small blade feathering moment. This moment will be applied to the gyro to make it precess nose down which puts in cyclic pitch causing the rotor to return to a position perpendicular to the shaft with no remaining hub moment. This feedback--which is essentially an acceleration feedback for a helicopter in free flight--acts as a gust alleviator by nullifying hub moments produced in rough air which would normally produce a hard ride in a helicopter without blade flapping hinges. In steady flight, the rotor and gyro combination are in equilibrium with no hub moment unless the pilot imposes a steady gyro moment with his control system which must be balanced by the feedback of a corresponding hub moment. For this reason, the natural stability changes caused by blade bending due to changes in speed or angle of attack can be partially, or nearly entirely, eliminated by the action of the hub moment feedback on the gyro depending on the amount of blade sweep forward incorporated.

EQUATIONS OF MOTION FOR A HOVERING HELICOPTER

Conventional Helicopter:

A conventional helicopter is shown hovering in Figure 4a. For this analysis, it will be assumed to have two degrees of freedom: pitch, θ, and speed, V. The rotor tip path plane makes an angle a_{1s} with the plane perpendicular to the shaft. A schematic block diagram of the elements of the conventional helicopter are shown in Figure 5a. The equations of equilibrium without control input are:

Forces parallel to body axis:

$$T a_{1_s} + W\theta = -\frac{W}{g}\dot{V}$$

Moments about the center of gravity:

$$T h a_{1_s} = I\ddot{\theta}$$

For the helicopter in hover, T = W

It may be shown that without a control input a_{1s} is only a function of velocity and pitch rate. The equations of motion in derivative form become:

$$W\frac{\partial a_{1s}}{\partial V}\Delta V + W\frac{\partial a_{1s}}{\partial \dot{\theta}}\Delta\dot{\theta} + W\Delta\theta + \frac{W}{g}\Delta\dot{V} = 0$$

and:

$$W h\frac{\partial a_{1s}}{\partial V}\Delta V + W h\frac{\partial a_{1s}}{\partial \dot{\theta}} - I\Delta\ddot{\theta} = 0$$

The characteristic equation obtained by expanding the determinant of the left hand side of the equations of motion is:

$$\frac{I}{g}S^3 + \left[I\frac{\partial a_{1s}}{\partial V} - \frac{W}{g}h\frac{\partial a_{1s}}{\partial \dot{\theta}}\right]S^2 + W h\frac{\partial a_{1s}}{\partial V} = 0$$

For an example helicopter, let us use the Lockheed Model 286 which has:

W, Gross Weight, = 4700 lbs

I, Pitching Moment of Inertia = 3000 slug ft^2

h, Distance of Rotor Above C.G. = 3 ft

Calculations for the rotor flapping derivatives based on conventional methods which can be found in the helicopter literature give:

$$\frac{\partial a_{1s}}{\partial V} = 0.00045 \ \frac{\text{RADIANS}}{\text{FT/SEC}}$$

and

$$\frac{\partial a_{1s}}{\partial \dot{\theta}} = -0.072 \ \frac{\text{RADIANS}}{\text{RAD/SEC}}$$

when these parameters are substituted into the characteristic equation it becomes:

$$S^3 + 0.354\,S^2 + 0.068 = 0$$

the roots are:

$$S_1 = -0.57$$

$$S_{2,3} = 0.11 \pm 0.33\,i$$

The pair of conjugate complex roots represents a divergent oscillation with a period of 19 seconds which doubles in amplitude every 6 seconds. No wonder helicopter pilots are well paid!

It is of interest to note that if the rotor is moved down to the center of gravity so that h = 0, the characteristic equation reduces to:

$$\frac{I}{g} S^3 + I \frac{\partial a_{1s}}{\partial V} S^2 = 0$$

This equation has two zero roots and a stable root:

$$S_1 = 0$$

$$S_2 = 0$$

$$S_3 = -g \frac{\partial a_{1s}}{\partial V} = -0.0145$$

The stability of such a helicopter has been demonstrated by the DeLackner stand-on helicopter. The effect can be over-done, however. If the rotor is three feet below the center of gravity, a pure divergence results with roots of:

$$S_1 = +0.55$$

$$S_{2,3} = -0.11 \pm 0.33i$$

A helicopter with such a high pure divergence would probably be impossible to fly.

Lockheed Helicopter

Figure 4b shows the elements of a helicopter with a Lockheed rotor system. The equations of motion of the Lockheed rotor system contain two elements which did not appear in the equations for the conventional helicopter: The cantilevered blades which produce a hub moment due to flapping independent of rotor thrust and an attitude gyro controlling cyclic pitch with a feedback proportional to hub moment. A schematic block diagram of the system is shown on Figure 5b and the equations of equilibrium without pilot input are:

Forces parallel to body axis:

$$T a_{1_s} + W\theta = -\frac{W}{g}\dot{V}$$

Moments about the center of gravity:

$$Th\, a_{1_s} + M_H = I\ddot{\theta}$$

Gyro moments:

$$\psi' M_H = -J_G \Omega \dot{\gamma}$$

where ψ' is the blade sweep forward, J_G is the polar moment of inertia of the gyro, Ω is the rotational speed of the gyro, and $\dot{\gamma}$ is the precession rate of the gyro with respect to space axes.

The equations of motion in derivative form are:

$$W\frac{\partial a_{1s}}{\partial V}\Delta V + W\frac{\partial a_{1s}}{\partial \dot\theta}\Delta\dot\theta + W\frac{\partial a_{1s}}{\partial(\gamma-\theta)}\Delta(\gamma-\theta) + W\Delta\theta + \frac{W}{g}\Delta\dot V = 0$$

$$Wh\frac{\partial a_{1s}}{\partial V}\Delta V + Wh\frac{\partial a_{1s}}{\partial \dot\theta}\Delta\dot\theta + Wh\frac{\partial a_{1s}}{\partial(\gamma-\theta)}\Delta(\gamma-\theta)$$

$$+\frac{\partial M_H}{\partial a_{1s}}\frac{\partial a_{1s}}{\partial V}\Delta V + \frac{\partial M_H}{\partial a_{1s}}\frac{\partial a_{1s}}{\partial \dot\theta}\Delta\dot\theta + \frac{\partial M_H}{\partial a_{1s}}\frac{\partial a_{1s}}{\partial(\gamma-\theta)}\Delta(\gamma-\theta) - I\Delta\ddot\theta = 0$$

$$\psi'\frac{\partial M_H}{\partial a_{1s}}\frac{\partial a_{1s}}{\partial V}\Delta V + \psi'\frac{\partial M_H}{\partial a_{1s}}\frac{\partial a_{1s}}{\partial \dot\theta}\Delta\dot\theta + \psi'\frac{\partial M_H}{\partial a_{1s}}\frac{\partial a_{1s}}{\partial(\gamma-\theta)}\Delta(\gamma-\theta) + J_G\Omega\Delta\dot\gamma = 0$$

The quantity $(\gamma-\theta)$ is the cyclic pitch produced by the gyro. For a hinged rotor, the change in blade flapping, a_{1s}, is equal to the change in cyclic pitch. Even though the Lockheed rotor does not have hinges, the blades are flexible enough that the same relationship can be assumed to apply, that is:

$$\frac{\partial a_{1s}}{\partial(\gamma-\theta)} = 1$$

with this identity, the characteristic equation becomes:

$$\frac{I}{g}s^3 + \left[I\frac{\partial a_{1s}}{\partial V} - \frac{1}{g}\left(Wh + \frac{\partial M_H}{\partial a_{1s}}\right)\frac{\partial a_{1s}}{\partial\dot\theta} + \frac{1}{g}\frac{\psi' I}{J_G\Omega}\frac{\partial M_H}{\partial a_{1s}}\right]s^2$$

$$+\frac{1}{g}\left(Wh + \frac{\partial M_H}{\partial a_{1s}}\right)s + \left(Wh + \frac{\partial M_H}{\partial a_{1s}}\right)\frac{\partial a_{1s}}{\partial V} = 0$$

This equation is similar to that for the conventional helicopter with the addition of terms for the rotor stiffness and for rotor feedback to the gyro. Note that the feedback due to blade sweep appears as a damping term in the "S^2" bracket. In addition, an "S" term is obtained which represents a spring effect due to the attitude gyro. For the Lockheed Model 286 the rotor stiffness and gyro parameters are:

$\partial M_H/\partial a_{1s}$ - rotor stiffness = 150,000 ft lbs/rad:
ψ' - blade sweep forward = .01 rad
J_G - gyro polar moment of inertia, 7 slug ft^2
Ω - gyro rotational speed, 37.1 rad/sec

All of the other parameters and derivatives are as listed for the conventional helicopter example. When the characteristic equation is evaluated, it becomes:

$$S^3 + 10.5\,S^2 + 54.7\,S + 0.8 = 0$$

the roots are:

$$S_1 = -0.0145$$
$$S_{2,3} = -5.3 \pm 5.3\,i$$

The roots are all stable. The pair of conjugate complex roots represents an oscillation with a period of 1.2 seconds which damps to half amplitude in .13 seconds--a very desirable combination for machine-man matching. In this case, the terms associated with the rotor rigidity and the gyro overwhelm the inherent speed stability and damping of the rotor. Those derivatives could have been zero with practically no change in the results.

As was pointed out earlier, the natural period could have been approximated from the rotor stiffness and moment of inertia alone:

$$P_N = 2\pi\sqrt{\frac{I}{\partial M_H/\partial a_{1s}}} = 2\pi\sqrt{0.02} = 0.75 \text{ SEC}$$

which is reasonably close to the 1.2 seconds calculated with the full set of equations. Both the rotor rigidity and the gyro are important to the Lockheed concept. Addition of only a gyro to a conventional flapping rotor will produce a stable oscillation but the period will be longer and the damping less than the above example-- a 3 second period with a time to half amplitude of 1.5 sec.

If the rotor rigidity is added to the conventional helicopter without a gyro the characteristic unstable oscillation will not be significantly changed. The period will decrease from 19 seconds to 14 seconds and the time to double amplitude will increase from 6 to 14 seconds. Thus it may be concluded that both the gyro and the rotor rigidity are required to produce the desirable short period and high damping of the Lockheed helicopters.

Although the foregoing discussion has been limited to hover flight, a parallel analysis of stability in forward flight would lead to the same conclusion: that in this flight condition also, the addition to a conventional helicopter of an attitude gyro and of a rotor with cantilevered blades will improve its matching to a pilot's natural capabilities.

CHALLENGES

The control gyro as used in the Lockheed Rotor System makes a definite contribution to good flying qualities but it does represent an additional drag item in forward flight when compared to conventional helicopters. For this reason, there is a requirement to clean up the gyro installation. The minimum value of gyro momentum, $J_G \Omega$, which can be used in a given Lockheed helicopter is governed by friction levels in the control system and must also be high enough to stabilize a potentially unstable rotor-gyro mode associated with aeroelasticity. For this reason the gyro cannot appreciably be reduced in size unless it is changed from a rotor-speed gyro to a high speed gyro. The challenge of this approach is to produce a high speed gyro drive system with the same level of reliability as the existing system since an in-flight failure would be catastrophic. Another suggested approach is to take the gyro off of the rotor hub entirely and replace it with a redundant servo system containing two or more small high speed gyros.down near the swashplate. The challenge this time is to provide a reliable feedback from hub moment to the servo system.

When larger Lockheed helicopters are designed, the ratio of fuselage inertia to rotor stiffness tends to increase thus increasing the natural period of the oscillation and decreasing the papidity of the response. Thus a challenge which applies to large helicopters using this system is to keep the period of the natural oscillation short without taking an excessive penalty in rotor weight.

PROBLEM

Design the simplest possible model which can be used for a classroom demonstration of the equations of motion of the two types of helicopters.

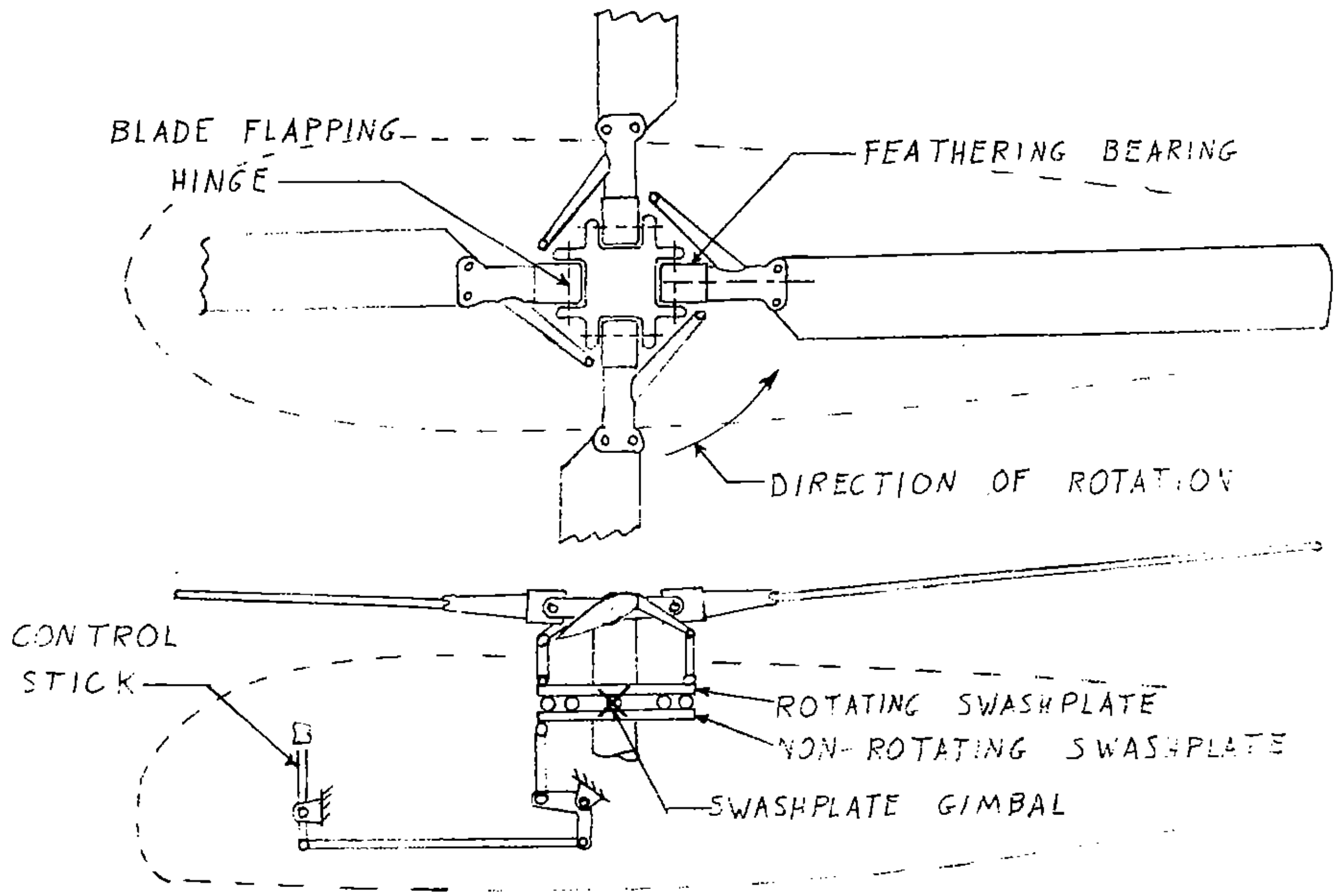

FIGURE 1a. CONTROL SYSTEM OF CONVENTIONAL HELICOPTER

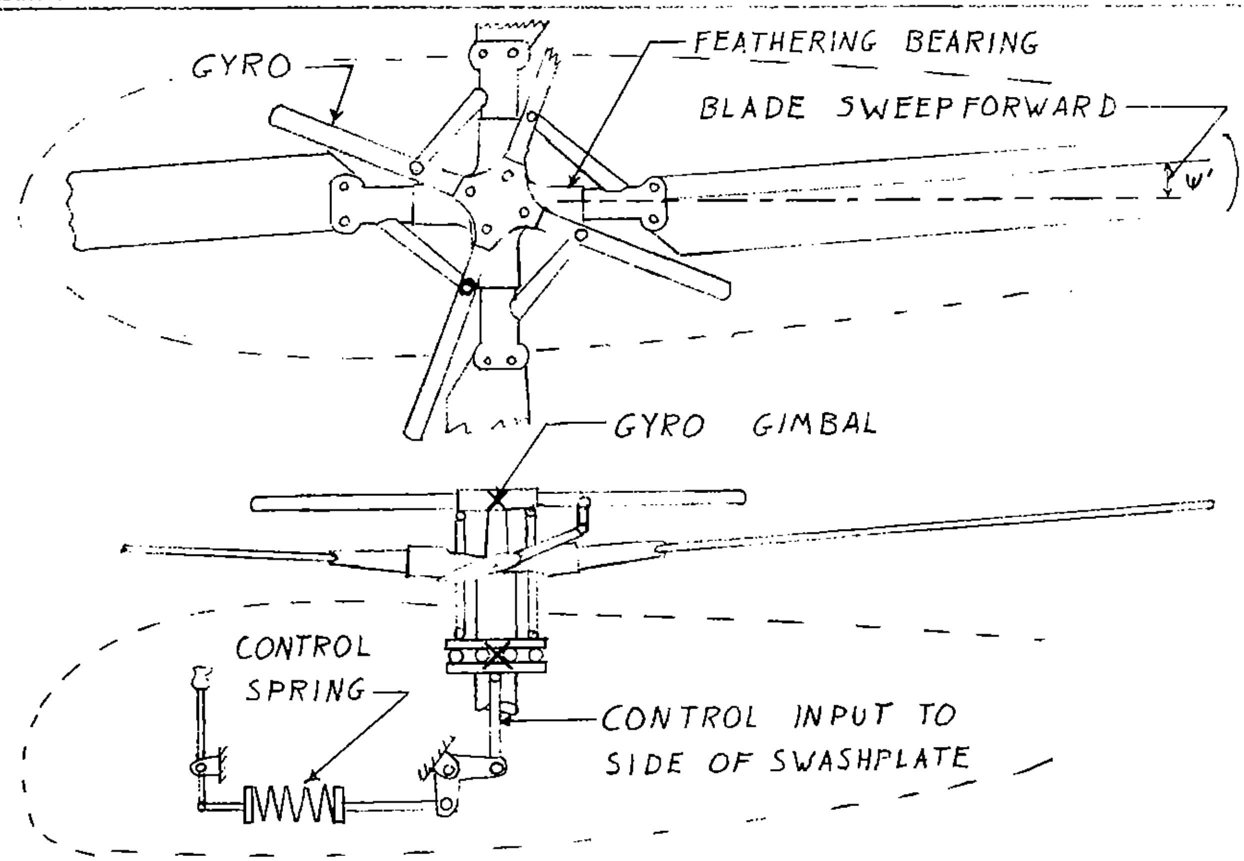

FIGURE 1b. CONTROL SYSTEM OF LOCKHEED HELICOPTER

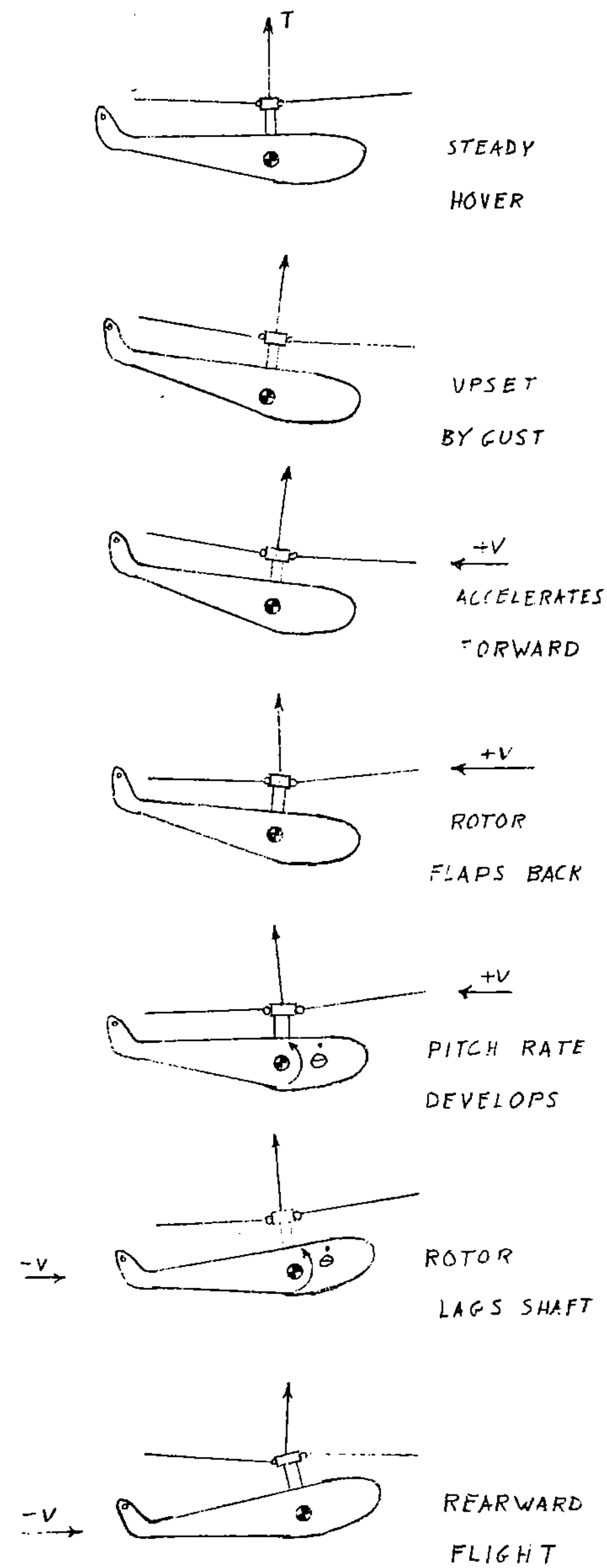

FIGURE 2a. CONVENTIONAL HELICOPTER, RESPONSE TO GUST

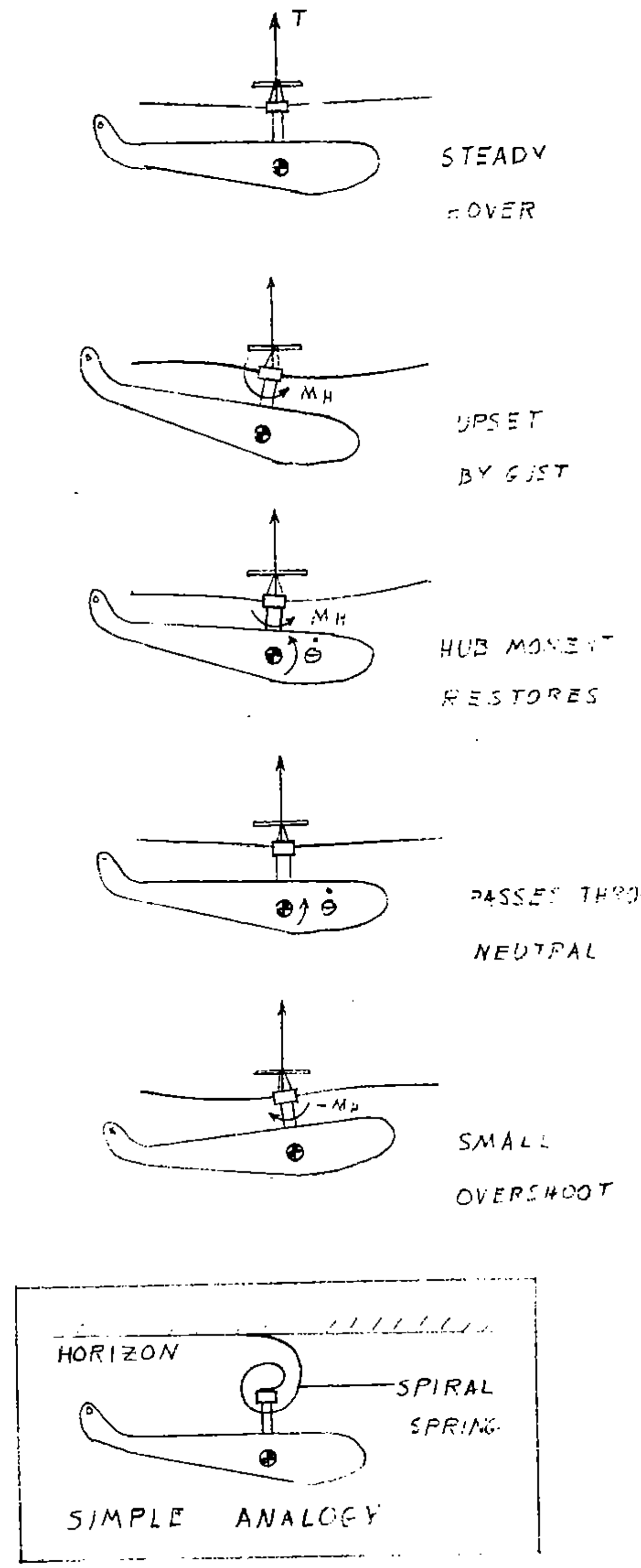

FIGURE 2b. LOCKHEED HELICOPTER, RESPONSE TO GUST

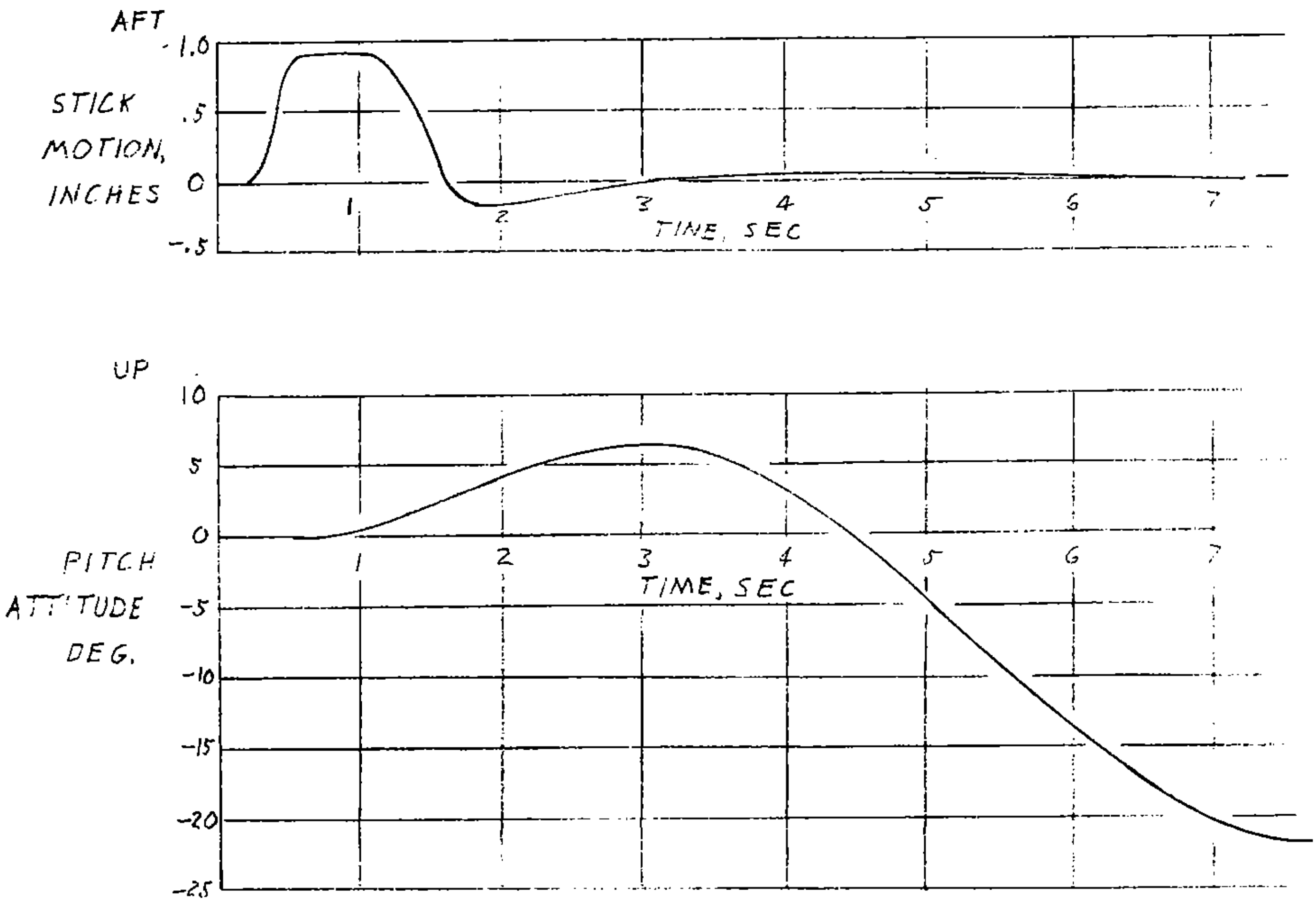

FIGURE 3a. CONVENTIONAL HELICOPTER, RESPONSE TO PULSE, FLIGHT TEST RESULTS

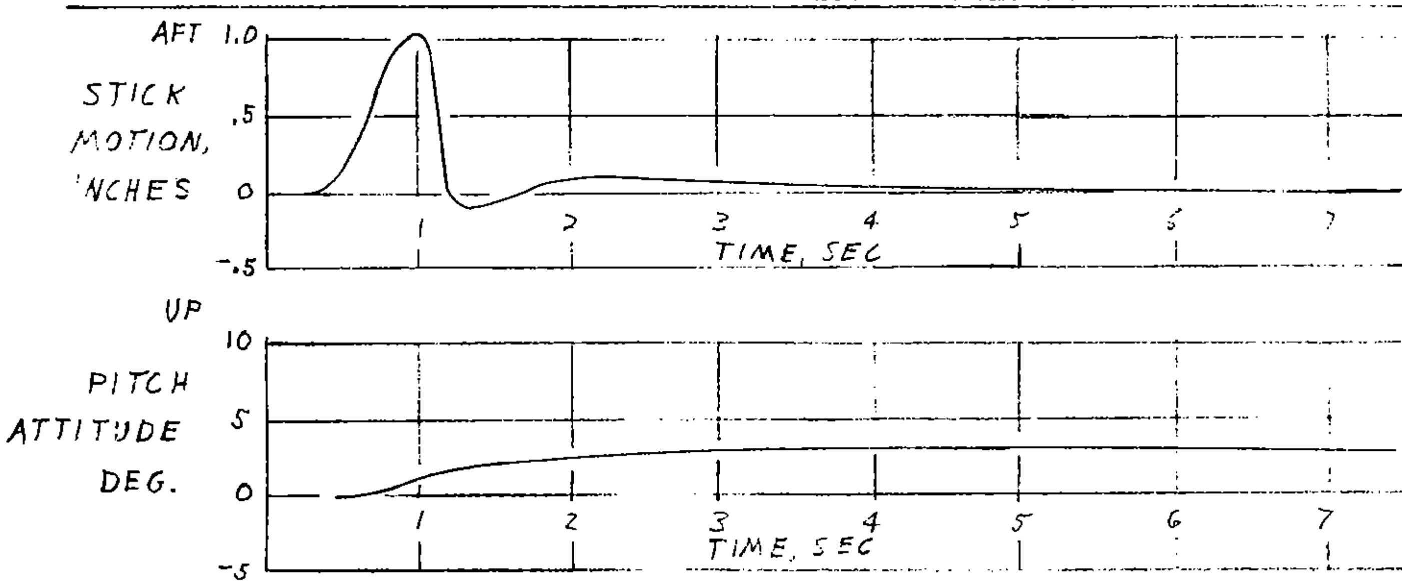

FIGURE 3b. LOCKHEED HELICOPTER, RESPONSE TO PULSE, FLIGHT TEST RESULTS

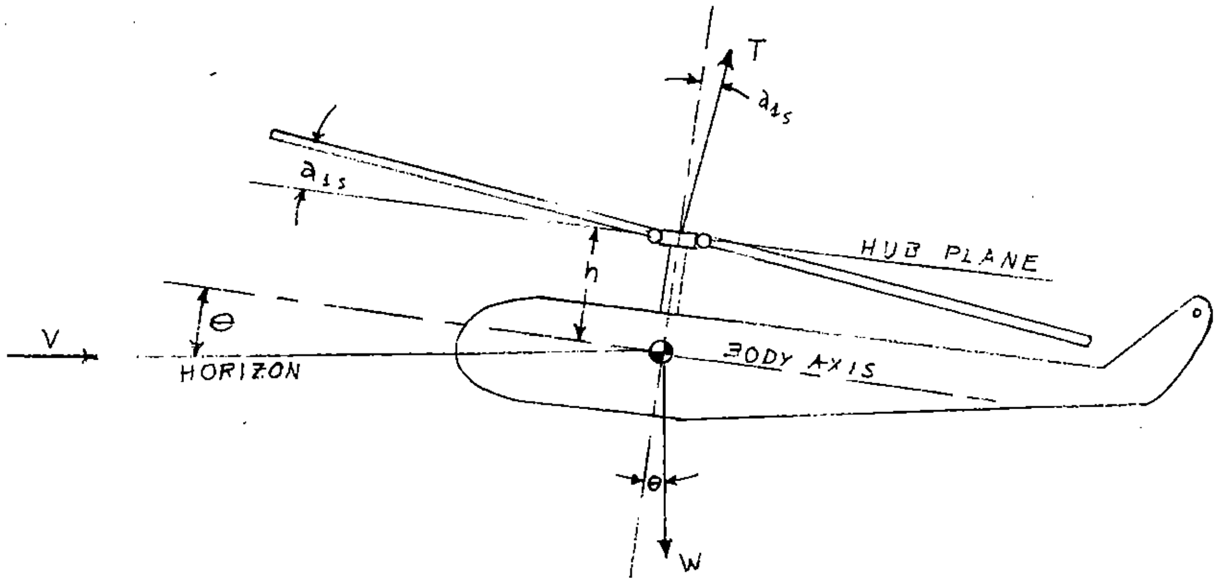

FIGURE 4a. SCHEMATIC OF CONVENTIONAL HELICOPTER

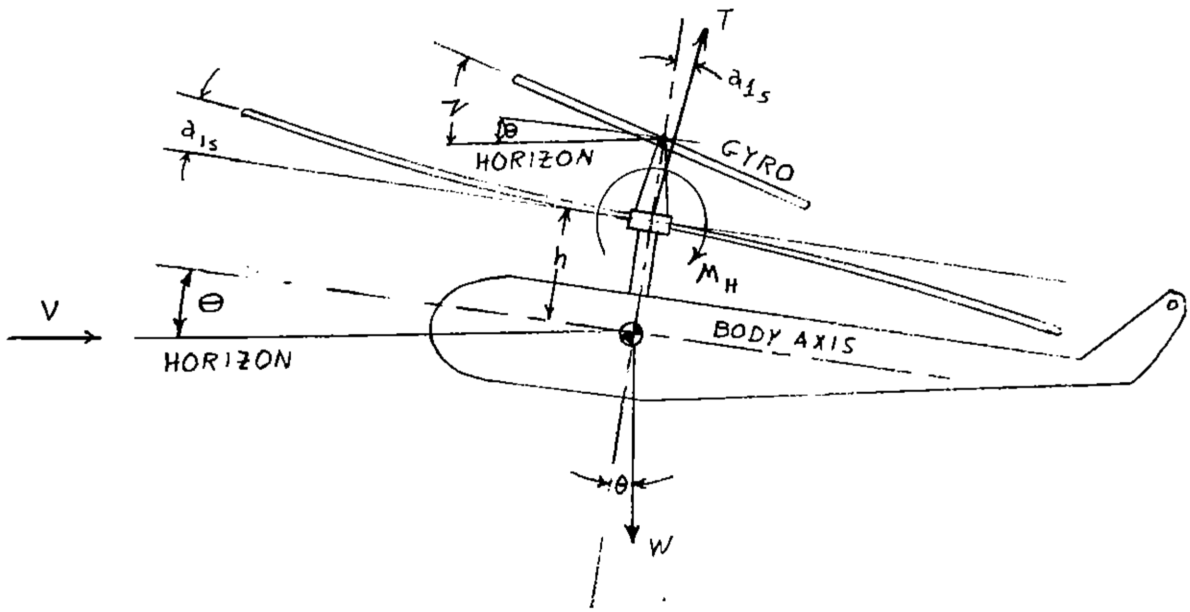

FIGURE 4b. SCHEMATIC OF LOCKHEED HELICOPTER

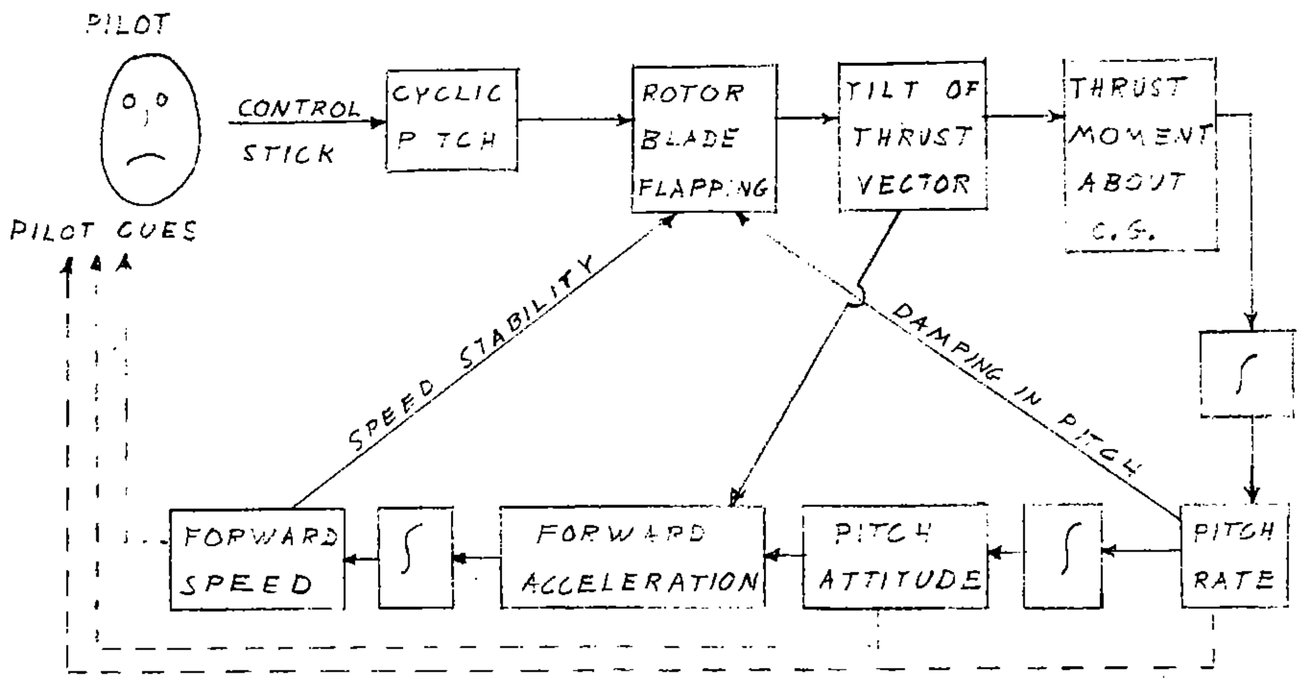

FIGURE 5a. BLOCK DIAGRAM OF CONVENTIONAL HELICOPTER

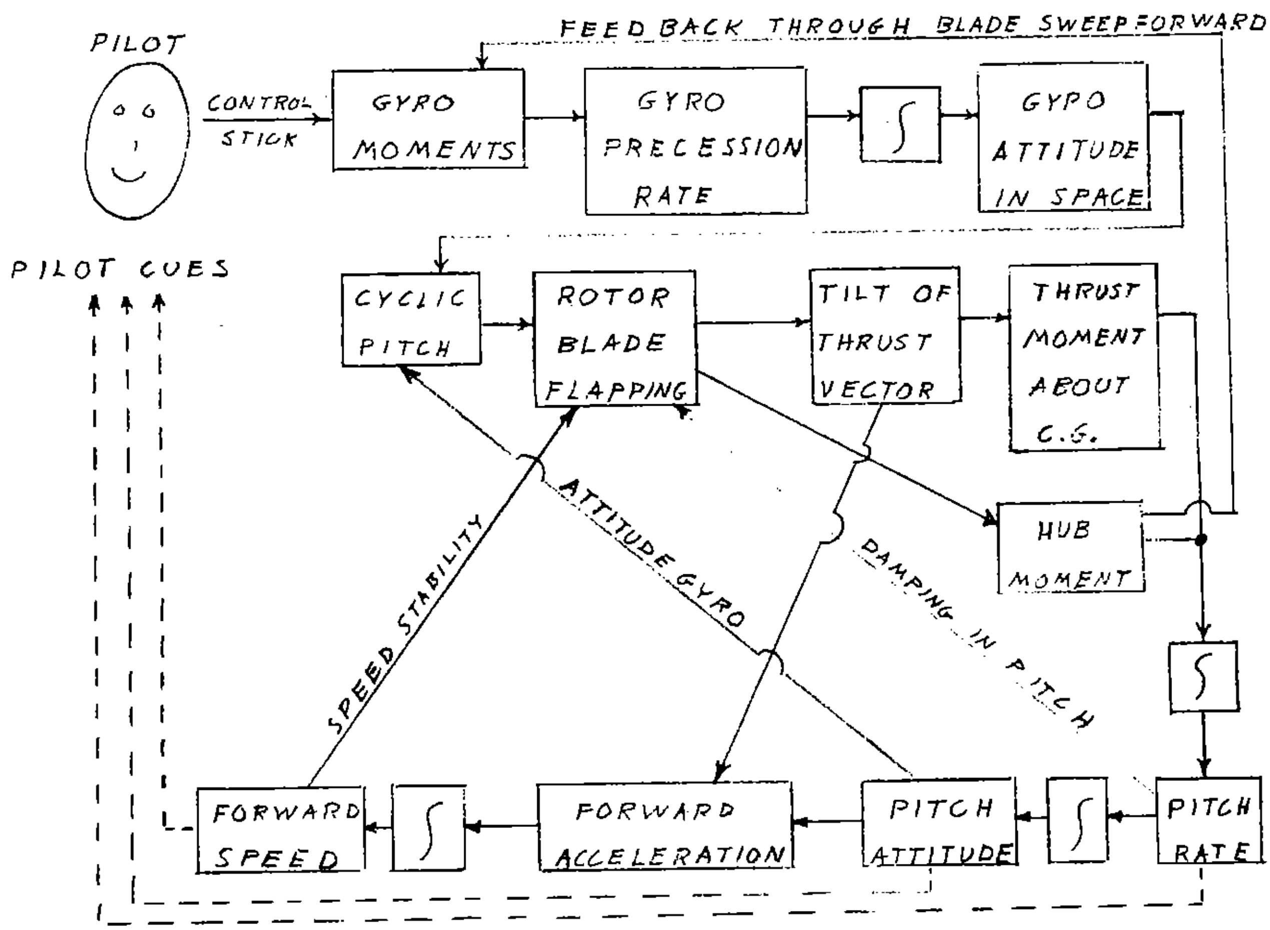

FIGURE 5b. BLOCK DIAGRAM OF LOCKHEED HELICOPTER

FLIGHT CONTROLS

I. GENERAL DESCRIPTION

The primary flight control system for the AH-56A helicopter consists of a longitudinal (pitch) control system, a lateral (roll) control system, a directional (yaw) control system, and a collective (lift) control system. The pitch and roll axes are controlled by a cyclic stick. Inflight pitch control of the aircraft is accomplished in the conventional manner; i.e., pilot fore and aft motion of the cyclic stick changes the pitch attitude of the aircraft. Similarly, pilot right and left motion of the cyclic stick changes the roll attitude of the aircraft. The aircraft directional (yaw) attitude is controlled by changing the pitch angle of the anti-torque tail rotor blades. Conventional directional control pedals are utilized to change this pitch angle.

Aircraft vertical lift is controlled by raising or lowering the collective lever. This action simultaneously changes the pitch on all blades of the main rotor by the same amount.

The four primary flight control systems use dual tandem servo actuators to amplify and transmit pilot or copilot/gunner control inputs to the control surfaces. Power for the dual tandem servo actuators is supplied from two separate hydraulic systems. These systems are powered by individually driven hydraulic pumps; the Number 1 system pump is driven directly off the main rotor shaft, and the Number 2 system pump is driven through the accessory gearbox.

A cyclic centering actuator is installed in both the pitch and roll control systems. The purpose of these actuators is to center the cyclic stick automatically during ground operation when the main rotor is below 80 percent and the aircraft is on the ground. This system can be overpowered by the pilot.

An electromechanical trim system is provided for the pitch, roll, and yaw control axes. The trim systems have 70 percent control authority.

A pitch desensitizer - roll decoupler is provided to reduce pitch axis sensitivity at high speeds and corrects for the vehicle's pitch motion due to roll motion at high speeds. A roll/lift compensator mechanism is included to improve the vehicle's roll damping at low speed and to also reduce cyclic areas coupling due to normal vehicle acceleration changes.

II. COMPONENTS AND LOCATIONS

Name of Component	Number per Aircraft	Location in Aircraft
Servo Assy, Cyclic/ Collective Control (Servo Package)	1	Servo Control Compartment
Directional Servo Actuator	1	LH Stabilizer Forward Beam
Directional Trim/Feel Spring Actuator	1	LH empennage, just fwd of horizontal stabilizer.
Control Gyro	1	FS 300, above Main Rotor Hub
Pitch Link Assembly, Main Rotor	4	Attached to gyro hub and pitch arm (horn)
Swash plate Assembly	1	FS 300, directly under transmission
Coupler Assembly	3	2 in debris bay, and 1 in the belly turret compartment
Copilot/Gunner Cyclic Stick	1	Copilot/Gunner Station, RH side
Pilot Cyclic Stick	1	Pilot Station, center
Copilot/Gunner Collective Lever	1	Adjacent to LH console in in copilot station
Pilot Collective Lever	1	Adjacent to LH console in pilot station
Velocity Gradient Assy	1	In debris bay, upper forward
Maneuver Gradient Platform Assy	1	In debris bay, supporting aircraft battery
Gearbox Assembly, Pedal Adjusting	1	Under floor between pilot's directional pedals

III. MAJOR COMPONENT DESCRIPTION

A. Servo Assembly, Cyclic/Collective Control (Servo Package)

The servo package consists of an assembly of cyclic pitch, roll and collective servo units with associated linkages and equipment and is supported from, and located below, the main rotor transmission case. Major components of the servo package, other than the servos, are: Positive springs (pitch & roll), dampers (pitch & roll). Each servo is a dual assembly - each servo half is operated by an independent hydraulic system.

B. Directional Servo Actuator

The directional servo actuator is a hydraulic power boost for directional (yaw) and stability augmentation system control which is accomplished by the tail rotor. The directional servo is installed near the root end of the front spar of the left hand horizontal stabilizer. This servo is a dual assembly - each servo half is operated by an independent hydraulic system.

C. Directional Trim/Feel Spring Actuator

The directional trim/feel spring actuator functions as a two-speed electro-mechanical actuator with integral double acting spring. The spring provides a feel force at the directional pedals. The actuator has a yaw trim capability at two speeds, the higher controlled by the pilot, and the lower automatically by YASAS heading hold circuitry.

D. Control Gyro

The control gyro consists of four (4) arms extending out from a gimbaled base plate and is located directly above the main rotor. The gyro assembly rotates at the same rpm as the main rotor blades. It serves as a stabilizing gyro within the control and rotor system. The gyro base is connected to the rotor blades by pitch control linkage. Its plane of rotation is displaced when the cyclic and collective pitch inputs are applied through the swash plate. Without pilot inputs, the gyro will remain in its own plane of rotation, even if the rigid rotor/airframe is displaced by an external disturbance.

Such a displacement, if strong enough, will alter the angle between the plane of the rotor and the plane of the gyro. When this displacement occurs, the control gyro immediately changes the rotor blade pitch through the pitch change link to return the rotor disc to its original plane without the necessity of moving the flight controls.

E. Pitch Link Assembly, Main Rotor

There are four (4) pitch link assemblies which connect the four (4) main rotor blades to the control gyro. Each pitch link assembly is adjustable for proper blade pitch to gyro angle relationship. Rod-end and vernier adjustments are both marked.

F. Swash Plate Assembly

The swash plate, located directly under and attached to the transmission, includes one (1) rotating and one (1) non-rotating assembly. Control inputs through static push-pull linkage are transmitted into rotating linkage by the swash plate. Both the swash plate drive and control gyro drive pivot on gimbal joints which are guided axially along the rotor shaft centerline.

A magnetic chip detector is installed in the lower portion of the swash plate assembly to warn the pilot and copilot/gunner of metal chips in swash plate to gyro installation. A voice warning "CHIPS, TRANSMISSION" will be the message heard through the FLAWS system. Also, both annunciator panels will indicate "CHIPS" and both (pilot's and copilot/gunner's) caution lights will illuminate, and the FLAWS status panel will indicate "XMSN CHIPS".

A flexible boot is connected between the swash plate and the transmission mast to prevent oil loss/contamination, and allow swash plate movement. Connected to the non-rotating swash plate arms are the two (2) scissors and the four (4) servo-to-swash plate push-pull rods.

G. Coupler Assembly

The decoupler system is installed in the copilot/gunner's cyclic and collective control systems. It disengages the copilot/gunner's cyclic stick and collective lever from the flight control system. This disengagement permits the copilot/gunner to clear his cockpit area and provide adequate space for swiveling gunner's station operation. When the copilot/gunner's cyclic stick and collective lever are decoupled, they are stowed in clips out of the way of the swiveling gunner's station. During the decoupled mode, the copilot/gunner has limited control of pitch and roll using the cyclic trim switch, and complete control of yaw when the swiveling gunner's station is centered using the directional pedals.

H. Copilot/Gunner Cyclic Stick

Both the pilot's and copilot/gunner's cockpits contain interconnected cyclic sticks. The cyclic stick provides conventional lateral and longitudinal control of the aircraft. Movements of the stick in the fore and aft direction result in pitching motions of the aircraft. Side to side movements result in roll motions of the aircraft. The pilot's cyclic stick has a mechanical control lock that permits locking both cyclic sticks in the centered position.

I. Pilot Cyclic Stick

The pilot cyclic stick performs the same function as that of the copilot/gunner cyclic stick (reference above). Also, it contains a YASAS emergency disengage switch which is mounted just below the pilot's grip. During normal flight operations the copilot/gunner's cyclic stick will be in the stowed position.

Provisions are made in the pilot's feel lever assembly (connects to lower end of cyclic stick in debris bay) for both pitch and roll neutral rig pins, and stick travel adjustments (pitch & roll). To the left of the feel lever assembly the stick moves the roll push-pull rod, and to the right the stick rotates the pitch torque tube assembly. The stick grip wire harness breaks out just below the floor board of the pilot's station.

J. Copilot/Gunner Collective Lever

Both the pilot's and copilot/gunner's stations contain interconnected collective levers. The collective levers provide conventional control of vertical flight. Induced control inputs are transmitted from the pilot, and copilot/gunner when coupled, collective lever through the mechanical linkage to the hydro servo. Boosted collective signals are then relayed to the control gyro to adjust the collective pitch of all four (4) main rotor blades. Pulling the lever up increases rotor blade pitch angle. Lowering the lever decreases rotor blade pitch angle. The copilot/gunner's collective lever contains the propeller pitch grip, engine fuel control power and load demand potentiometers (these potentiometers are located near the pilot's collective lever but can be actuated by the copilot/gunner when his collective lever is coupled to the pilot's), friction release trigger, search light power and swivel control switches, rpm set switch, and yaw trim switch. A mechanical control lock in the pilot's station permits locking both collective levers in the full down position.

K. Pilot Collective Lever

The pilot's collective lever performs the same function as that of the copilot/gunner's collective lever (reference above). In addition it contains the engine condition grip, starter switch, stop-start switch, engine fuel control power and load demand potentiometers, and the hydraulic system selector switch.

L. Velocity Gradient Assembly

A variable feel system provides cyclic stick centering and simulated feel forces as the stick is moved forward and aft, or left and right. These feel forces are induced by the velocity gradient system and the maneuver gradient system. The velocity gradient system provides increasing stick forces as airspeed increases. It adjusts mechanical advantage of cyclic roll and pitch control in keeping with indicated airspeed to simulate airload forces at the cyclic control stick.

M. Maneuver Gradient Spring Cartridge

The maneuver gradient system provides an increasing longitudinal cyclic stick force for a given vertical load factor ("G" force). The system, therefore, produces stabilizing control forces during maneuver conditions resulting in vertical G's. These forces are created by the "G" effect on a bob weight mass composed of a platform, on which is mounted the aircraft battery. The mass is pivoted on a horizontal axis and is connected by mechanical linkage to the velocity gradient assembly. The mass is balanced by a single acting spring so that there is no bob weight input force to the control system under 1 "G" condition. Also, connected in parallel with the single acting spring, is a viscous type rate damper. Both spring and damper are mounted to the forward bulkhead in the debris bay.

N. Directional Pedals

Directional pedals are located in both pilot's and copilot/gunner's stations. The pedals provide control of the heading of the aircraft by varying the pitch of the tail rotor blades. The pilot's and copilot/gunner's pedals are interconnected - no decoupling system. The pilot's pedals contain conventional toe brakes for MLG wheel brake application.

Pilot's pedals can be positioned in and out by cranking the pedal adjustment handle. The copilot/gunner's pedals are foot spar stubs without toe brakes. A segmented quadrant is used to position the copilot/gunner's pedals.

IV. SYSTEM OPERATION

A. Pitch (Longitudinal) Control

Pilot and gunner pitch control inputs are transmitted from their respective cyclic sticks to a common pitch control torque tube assembly.

The pilot's pitch inputs are transmitted from the cyclic stick directly to a pitch control torque tube assembly. Input travel is limited to ±4.5 inches by adjustable stops.

The gunner's pitch inputs are transmitted from his cyclic stick directly to a push-pull rod connected to a coupler mechanism. The output lever on this coupler is attached to a push-pull rod which transmits pitch inputs to a lever attached to the pitch control torque tube assembly. The gunner's cyclic stick is normally decoupled and retained in a position outside its operational locus of travel. To take control, the gunner trips an overcenter latch while grasping the stick grip. Initial movement of the stick triggers the coupler mechanism to provide automatic system coupling when the gunner's stick reaches a position synchronous with the pilot's. The gunner's stick can be decoupled by manual actuation of the coupler mechanism. Access for decoupling is through the debis door with the aircraft on the ground.

A variable feel system is incorporated in the pitch control system to provide simulated feel as the cyclic stick is displaced forward and aft of neutral. In addition, a pitch trim system is provided to relieve the simulated feel forces when the stick is held out of neutral. The variable feel system consists of a double acting feel spring, a trim actuator rod assembly, a velocity gradient actuator, a maneuver gradient system, and a fork lever assembly.

Simulated feel forces and force rates are produced by combining the spring rate of the double acting feel spring with the effective force rate of the maneuver gradient system. Variation of forces and

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force rates with airspeed is accomplished by changing the mechanical advantage between the stick grips and the variable feel system. Trim forces are generated by changing the relationship between the pitch input control and the feel spring.

The variable feel system is connected to the pitch control torque tube assembly. The pitch trim actuator rod assembly connects the inboard lever of the pitch control torque tube assembly to the fork lever assembly. The double acting feel spring is connected to one arm of the fork lever assembly; the maneuver gradient system is connected to another arm. Displacement of the cyclic stick will cause the fork lever assembly to rotate resulting in compression of the feel spring. Change in mechanical advantage between the cyclic stick and the variable feel system is accomplished by changing the orientation of the fork lever assembly. As airspeed is increased, the velocity gradient actuator will rotate the fork lever assembly, thereby decreasing the moment arm of the feel spring force and the moment arm of the pilot input force. This requires the pilot to increase his stick force.

The velocity gradient system consists of an electromechanical actuator and pressure transducer. The pressure transducer measures dynamic and static pressure from the aircraft pitot static system and sends a signal voltage proportional to indicated airspeed to the velocity gradient actuator. A servo amplifier, integral with the actuator, compares this signal voltage with an internal signal voltage proportional to actuator stroke, and drives the actuator to match the internal signal to the external signal. The actuator extension produces the appropriate mechanical advantage at the fork assembly for the airspeed measured by the transducer.

The maneuver gradient system produces stabilizing control system forces during maneuver conditions resulting in vertical g's. These forces are created by the acceleration effect on a bobweight mass composed of a platform and a compensating weight. The bobweight platform is pivoted about a lateral axis and is connected through intermediate torque tubes to the fork lever assembly in the variable feel system. The bobweight mass is balanced by a single acting spring so that there are no force inputs to the control system during 1 g flight. Damping of bobweight motion is accomplished by two viscous dampers connected in parallel with the balance spring. Changes in mass forces resulting from conditions other than 1 g provide the maneuver force gradient. These forces increase with increasing airspeed by the change in mechanical advantage in the variable feel system.

Cyclic pitch control inputs are transmitted from the right outboard lever of the pitch control torque tube assembly through push-pull rods interconnected by bellcranks to the input arm of the dual tandem pitch control power servo. The output ram of the servo is connected to a C type torsion spring. Compression of the C type spring results in a force that is transmitted through a series of rods and cranks to two of the four nonrotating pushrods that are attached to the swashplate assembly.

The swashplate and control gyro are interconnected by three symmetrically spaced push-pull rods routed inside the rotor drive shaft. The control gyro and swashplate are individually driven at rotor speed by constant speed linkages. Both the gyro and swashplate pivot on gimbal joints which are guided axially along the rotor shaft on ball splines. Control gyro tilt is limited to 15° by a circumferential stop on the drive assembly.

Servo amplified control input forces are transmitted to the gyro by the three rotating push-pull rods. The gyro is connected to the blade feathering arms through pitch links. When a control input is made, the torqued gyro precesses, changing the angle of attack of the main rotor blades about their feathering axis, which establishes the control moment on the air vehicle.

Damping of the cyclic motion of the main rotor control gyro and swashplate is accomplished by viscous dampers linked to the nonrotating swashplate. The damper configuration consists of four linear dampers symmetrically located about the swashplate. Damping rate for each linear damper is 44 lbs./in./sec. at 8.2 cps and ±0.020 inch amplitude. Shear rivets within each damper rod assembly are provided to insure that a failed damper will not restrict cyclic motion.

Longitudinal system stick force for the pilot and gunner is a function of stick position at hover (0 to 25 knots) and also at speed (225 knots and higher). When the cyclic stick is trimmed from center, the force gradient will shift so that servo stick force at the grip occurs at the trimmed position of the stick.

The pitch trim actuator has 70 percent control authority. Limitation of control authority is always in the opposite direction from a runaway trim actuator. Limitation in control is due to the feel spring bottoming out, and is a function of airspeed (i.e., velocity gradient actuator position). There is no limitation of pitch control with a runaway trim actuator for speeds under 150 knots.

A pitch desensitizer - roll decoupler mechanism reduces the pitch axis sensitivity at high speeds and also corrects for the vehicle's pitch motion due to roll motion at high speeds.

B. Roll (Longitudinal) Control

Pilot and gunner roll control inputs are transmitted from their respective cyclic sticks to a common roll bellcrank assembly.

Pilot control inputs are transmitted from the cyclic stick to the roll bellcrank assembly through a push-pull rod. The pilot input travel is limited to ±3 inches by adjustable stops.

The gunner's cyclic stick is attached to a longitudinally mounted torque tube and crank assembly. The output end of this assembly is attached to a push-pull rod. This rod is connected to a coupler mechanism that has its output end attached to another push-pull rod. This control rod is attached to the common roll bellcrank assembly. The coupler mechanism is identical to the pitch system coupler and functions in the same manner.

A feel system is incorporated in the roll control system to provide simulated feel as the cyclic stick is displaced left or right of neutral. In addition, a roll trim system is provided to relieve the simulated feel forces when the stick is held out of neutral. The simulated feel force rate is produced by the spring rate of a double acting feel spring. Trim forces are generated by changing the relationship between the roll input control and the feel spring.

The feel system is connected to the roll bellcrank assembly through the roll trim actuator rod assembly. The trim actuator rod assembly is connected to one arm of a bellcrank; the other bellcrank arm is connected to the double acting feel spring. Roll trim inputs are produced by extending or retracting the actuator rod assembly causing the cyclic stick to displace from the neutral position. A rotary damper is attached to one arm of the common bellcrank assembly to provide stick damping.

Cyclic roll control inputs are transmitted from the roll bellcrank assembly through push-pull rods interconnected by bellcranks to a summing lever at the input arm of the dual tandem roll control power servo. The output ram of the power servo is connected through a series of bellcranks and control rods to the roll positive torsion spring assembly. Torquing of the torsion spring results in a force being transmitted through two roll crank assemblies to two of the four nonrotating push-pull control rods. Power boosted roll control inputs are transmitted from the nonrotating push-pull rods to the three rotating push-pull rods through the swashplate bearings. Operation of this portion of the system is the same as the longitudinal control system with the exception of the feedback linkage system. Feedback linkages are connected from one of the roll crank assemblies back to the summing lever at the servo actuator input arm. This system is tailored so that the roll servo actuator backs off with change in pitch gyro precession at a rate such that the gyro feels equal forces from the roll and pitch force springs. Control stick position is not affected by this feedback.

Lateral system stick force for the pilot is a function of stick position. The lateral stick force gradient does not vary with airspeed. When the cyclic stick is trimmed from center, the force gradient will shift so that zero stick force at the grip occurs at the trimmed position of the stick.

The roll trim actuator has 70 percent control authority. A runaway trim actuator can be overpowered by the pilot, and full roll control can be retained.

The lift/roll compensator mechanism improves the vehicle's roll damping at low speed and also reduces cyclic areas coupling due to normal vehicle acceleration changes.

C. Cyclic Trim and Stick Centering

The cyclic trim system consists of pitch and roll trim actuators, three omni trim switches, and associated control relays. The pitch and roll trim actuators are part of the trim actuator rod assemblies located in the pitch and roll control systems. These actuators are controlled by any one of three trim switches which are located on the pilot's cyclic stick, the gunner's cyclic stick, and the left-hand grip of the gunner's swiveling gunner's station (SGS). Power to the cyclic trim system comes directly from the main battery through the CYCLIC CTR. switch.

The trim actuators are provided with trim centering switches which automatically null the trim actuators when the trim actuator centering relay is energized. This relay is controlled by the landing gear GRD/AIR safety switch. The trim actuators will automatically return to a null trim position in response to a signal from both the right-hand and the left-hand main landing gear ground/air safety switch. A cyclic trim ground test switch is provided so that the trim centering system can be checked out on preflight.

An automatic stick centering system is provided to center the cyclic controls during startup and during shutdown should the pilot forget to engage the manual cyclic stick lock. The stick centering system automatically centers the cyclic stick in both the pitch and roll axes when the main rotor rpm is less than 80 percent and the aircraft is on the ground.

This system consists of a hydraulic actuator and cam assembly attached to the input bellcranks of both the pitch and roll power servos. Hydraulic pressure to the stick centering actuators is controlled by two independently operated solenoid valves in series connection. These valves receive pressure from the Number 2 hydraulic system and are attached to the housing of the pitch servo actuator. In the event of loss of system pressure, an accumulator and check valve will assure that hydraulic pressure is present to actuate the stick centering system. When both solenoid valves are energized the hydraulic actuator and cam assemblies position the servo input bellcranks to neutral. The forces at the cyclic stick to overcome the force produced by the centering actuator are 13 lbs. in the pitch axis and 11 lbs. in the roll axis.

The centering solenoid valves are independently controlled: one valve is controlled by the rotor speed monitor, and the other valve is controlled by the main landing gear GRD/AIR safety switches. Power to the stick centering control circuit by-passes the main battery switch and comes directly from the main battery through the CYCLIC CTR. switch. This feature permits the cyclic stick centering system to remain enabled should the pilot elect to disconnect main electrical power to nonessential systems in the event a crash landing is anticipated. A CYCLIC CENTER OVERRIDE switch is provided so that the stick centering system can be checked out on preflight.

D. Collective Control

Pilot and gunner collective control inputs are transmitted from their respective collective sticks to a common torque tube assembly.

Pilot control inputs are transmitted from the collective stick hub through a push-pull rod to the inboard horn of the common torque tube assembly. The pilot's stick is counter-balanced with a mass balance connected to the torque tube assembly.

The gunner's collective inputs are transmitted from the collective stick hub through a push-pull rod to the input arms of a coupler mechanism assembly. The output arm of the bellcrank transmits control inputs through a push-pull rod to the center horn of the common torque tube assembly. The gunner's stick is counterbalanced with a 1 g mass balance. The gunner's stick is normally decoupled and retained in a stowed position. To gain control, the gunner pulls the stick along its line of normal actuation; disengaging it from its snap-type retainer. Initial movement of the stick triggers the coupler to provide automatic system coupling when the gunner's stick reaches a position synchronous with the pilot's. System decoupling is by manual actuation of the coupler in the debris area.

The pilot to servo control input system is provided with a manually adjustable stick friction device and an electrically actuated friction brake. The manually adjustable friction device has an adjustable range of 6 ±1 lbs. minimum to 9.4 ±1 lbs. maximum. The electrically actuated brake has a fixed value of 11 ±1 lb. A friction device at the servo input lever permits servo input position retention at any point in its stroke in the event that system continuity between servo and collective stick is lost. All friction values are related to the pilot's collective stick reference point and when combined provide a maximum force of 25.5 lbs. with electric brake energized and a minimum force of 6.5 lbs. with the electric brake off. The electric brake can be controlled by either pilot or copilot by actuation of a trigger switch.

A collective stick down stop is positioned by a solenoid actuator in combination with a spring to limit collective blade angle to a zero lift position when the aircraft is on the ground. The down stop is spring loaded and either pilot or copilot can override the down stop with a maximum force of 33.5 lbs at the stick reference point.

Collective control inputs are transmitted from the left outboard horn of the common torque tube assembly through push-pull rods interconnected by bellcranks to the input arm of the dual tandem collective power servo. The output arm of the power servo is connected through a link assembly to a bellcrank which transmits servo amplified collective inputs through a push-pull rod to the collective yoke. Rotation of the collective yoke results in a vertical movement of the pitch and roll bellcrank assemblies which are connected to the four nonrotating pitch and roll push-pull rods.

The collective control inputs are transmitted from the nonrotating push-pull rods through the swashplate bearings to the three rotating push-pull rods interconnecting the swashplate with the gyro. Pitch links interconnecting the gyro with the blade feathering arms transmit collective control inputs to the blades proportionate to the mechanical advantage of each blade to the gyro.

E. Yaw Control

Pilot and gunner yaw control inputs are transmitted through conventional rudder pedals to a common yaw control bellcrank assembly.

The pilot's left and right pedal assemblies are pivoted on a single laterally fixed shaft. Each pedal assembly is connected to a push-pull rod. Yaw control inputs are transmitted through the push-pull rods to the lower arms of the common yaw control bellcrank assembly. Pedal travel is limited to 3.25 inches right pedal and 2.61 inches left pedal by adjustable stops. Fore and aft pedal adjustment of ±3 inches is provided by positioning the pedal pivot shaft about the lateral axis. A lever at the end of the pedal pivot shaft is attached to an irreversible screw jack controlled by a pedal adjust crank.

The gunner's pedals are pivoted on a lateral fixed shaft and connected to a push-pull rod which transmits yaw inputs to a bellcrank assembly. The output lever of this bellcrank transmits control inputs through push-pull rods interconnected with bellcranks to a lower arm of the common yaw control bellcrank assembly. The gunner's pedal travel is the same as the pilots and is limited by the same adjustable stops. The gunner's pedals can be adjusted with a lever in increments of 0.86 inches. The upper arms of the common yaw control bellcrank assembly are connected to cables that extend aft through conventional pulleys and fairleads. Yaw control inputs are transmitted through the cables to the lower arms of a bellcrank assembly located just forward of the tail rotor servo. The center arm of this bellcrank assembly is attached to the feel and trim actuator assembly. The upper arm of the bellcrank assembly attaches to a push-pull rod which connects to the input arm of the dual tandem yaw control power servo.

The output ram of the servo is connected to a ratio arm which transmits boosted yaw control inputs through a push-pull rod to a yoke assembly attached to a sleeve co-axial with the tail rotor driveshaft. The outboard end of the sleeve is attached to a four armed spider assembly. Servo amplified yaw control inputs are transmitted to the tail rotor through pitch links which connect the spider assembly to the blade pitch arms.

Displacement of the foot pedals produce simulated feel forces by compression of the spring in the trim/feel spring actuator assembly. Pedal centering forces are provided by spring preload in the double acting cartridge. An electromechanical actuator is combined with the spring to provide trim and heading-hold function. Control of the yaw trim actuator is by means of a switch on the collective stick. Trim forces are generated by shifting the feel spring ground to balance the input deflection. Pedal force is a function of pedal position when the trim actuator is nulled. The tail rotor blade angle is a function of trim actuator position.

Trim capability is 100 percent left pedal travel and 90 percent right pedal travel. A spring overtravel of .71 inches beyond maximum pedal travel deflection provides 71 percent right pedal control capability and 88 percent left pedal control capability in the event of a runaway trim motor.

The yaw trim control system consists of two yaw trim switches, an actuator and related control relays. The trim switches are located one each on the pilot's collective lever and the gunner's collective lever. Power to the yaw trim system comes from the essential dc Bus through the Trim Control Yaw circuit breaker.

The yaw trim switches control two relays: the yaw actuator extend high speed relay and the yaw actuator retract high speed relay. When either relay is energized, the trim actuator will extend or retract, causing the tail rotor to change blade angle.

F. Hydraulic Power Servo Actuators

The four power servo actuators used in the flight control system are similar in construction and operation and differ only where necessary to accommodate mounting, plumbing, output thrust, or rate requirements. Each is a multiple input, irreversible positional servo, powered by dual hydraulic systems and configured in such a way as to separate these systems completely so that a mechanical rupture of the housing containing one hydraulic system cannot propagate into the housing containing the other.

The pitch, roll, and yaw control servo actuators are designed such that each half of the tandem output actuators can independently provide 50 percent of the maximum output servo power. Thus, with loss of either the No. 1 or No. 2 hydraulic system, there is sufficient control power left in the remaining system.

The collective actuator is the same as other actuators except that the effective area of the collective actuating pistons is increased to provide additional single system output. With only one hydraulic system operating, one half of the tandem output actuator can independently provide 60 percent of the maximum output servo power. When both systems are operating normally, the two cylinders combined could exceed the output design loads. A load limiter, attached to the servo housings, acts in such a way that fluid is by-passed from one side of the output pistons to the other whenever the ΔP (with both systems operating) across the pistons exceeds 2000 psi. With only one system operating, the load limiter is locked out of operation.

Each servo unit is divided into two housing halves; each housing contains one unit of a tandem output actuator (ram), a servo valve assembly, interconnecting levers and cranks, and a micronic filter. The No. 2 side of the pitch and roll servo actuators each contain a series servo (mod piston), a transfer valve, an engage valve, and a feedback transducer. These components are used for pitch desensitizing and roll damping, respectively.

V. PCRS CONFIGURATION

The provisions for the negative loaders removed in MOD P-190 are deleted, and the provisions for automodes are deleted from the collective and cyclic servo package. However, items are retained for the roll compensation and pitch desensitizing capability. The YASAS system is deleted from the directional actuator. The pilot/copilot collective control coupler is simplified, and the collective brake and friction mechanisms are combined to simplify their installation.

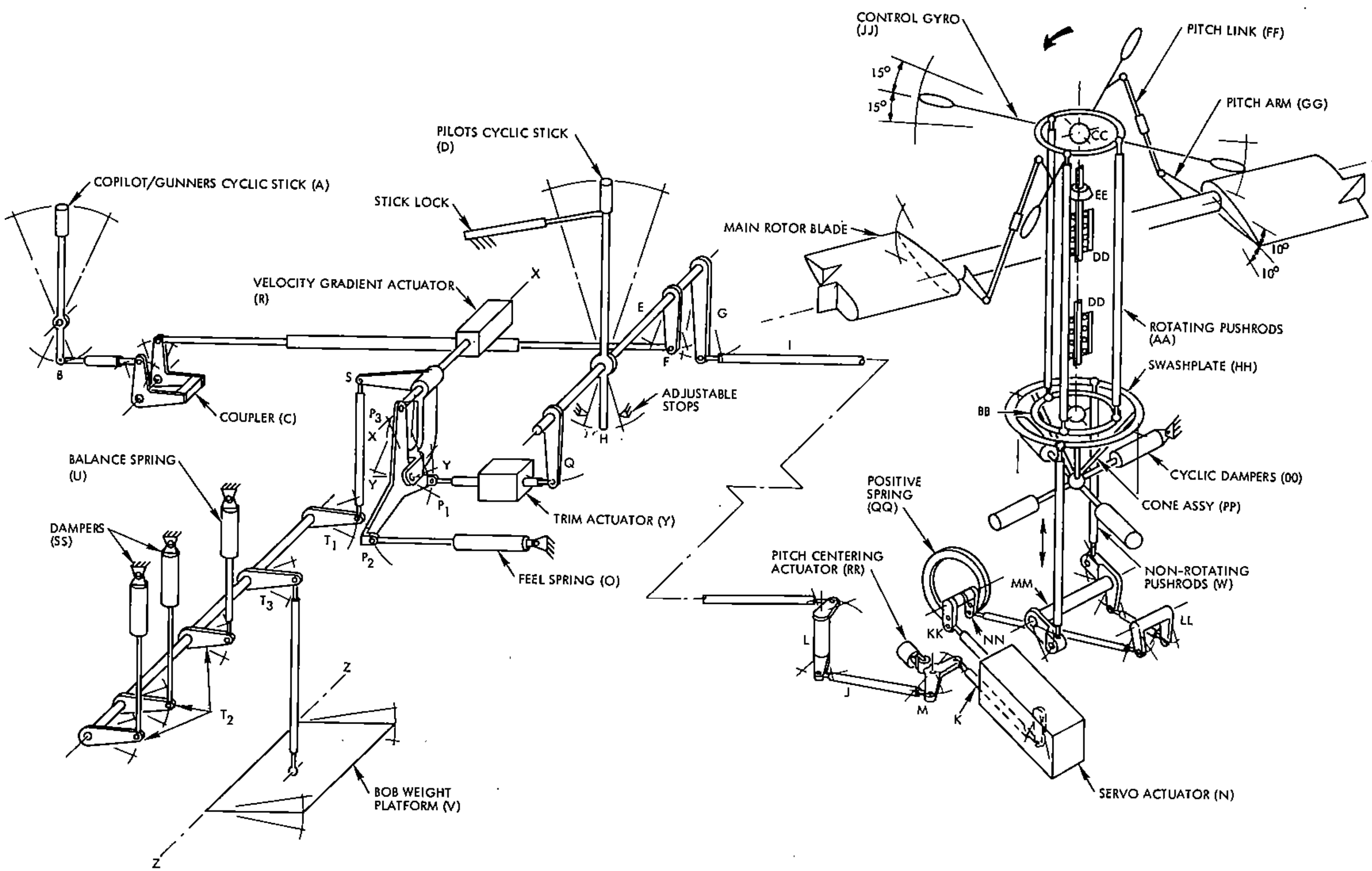
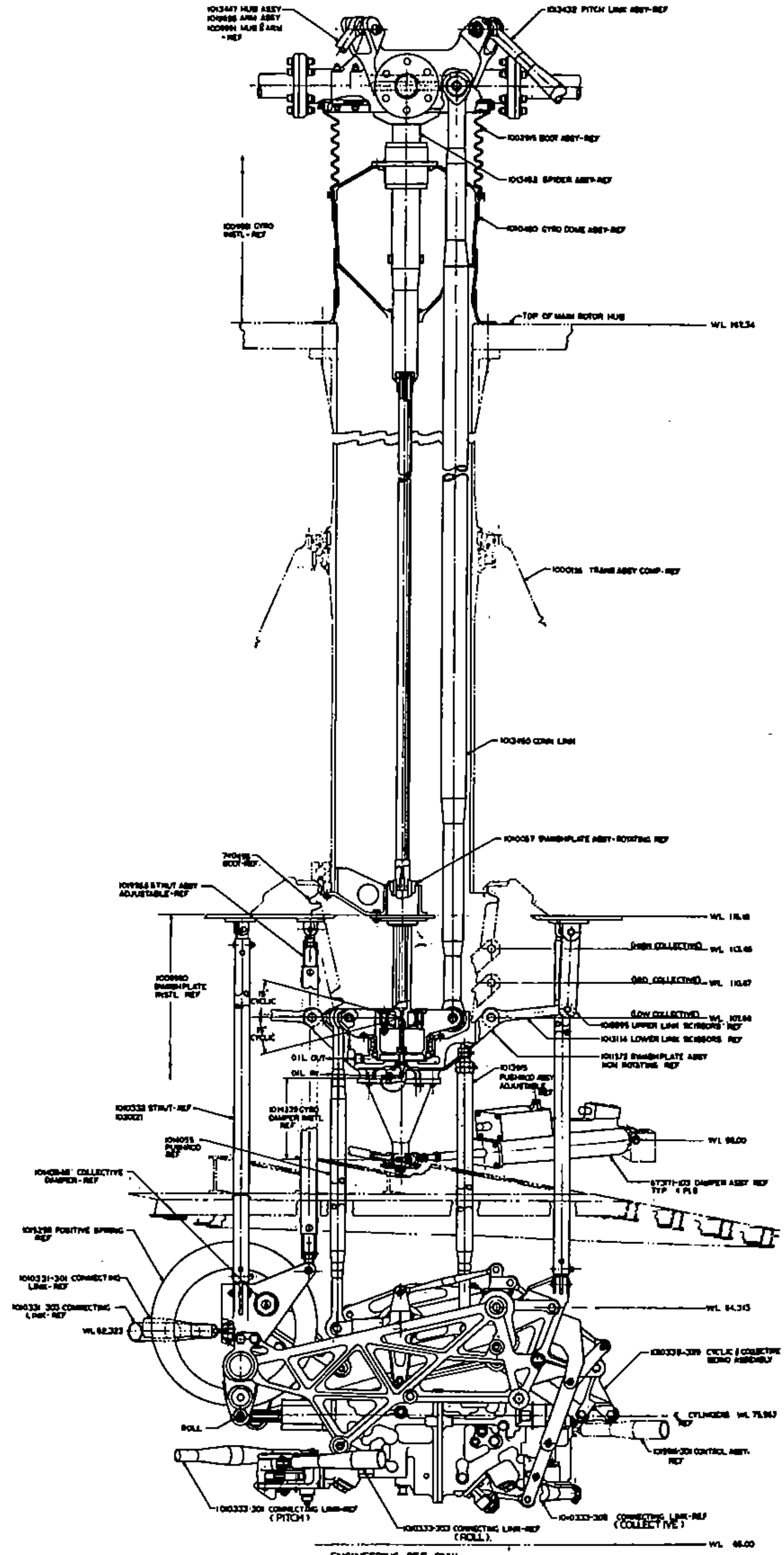


Figure 7A-1. Longitudinal Control System Schematic



TOP OF MAIN ROTOR HUB
OIL OUT
OIL IN
ROLL
ENGINEERING REF ONLY

Figure 7A-2.

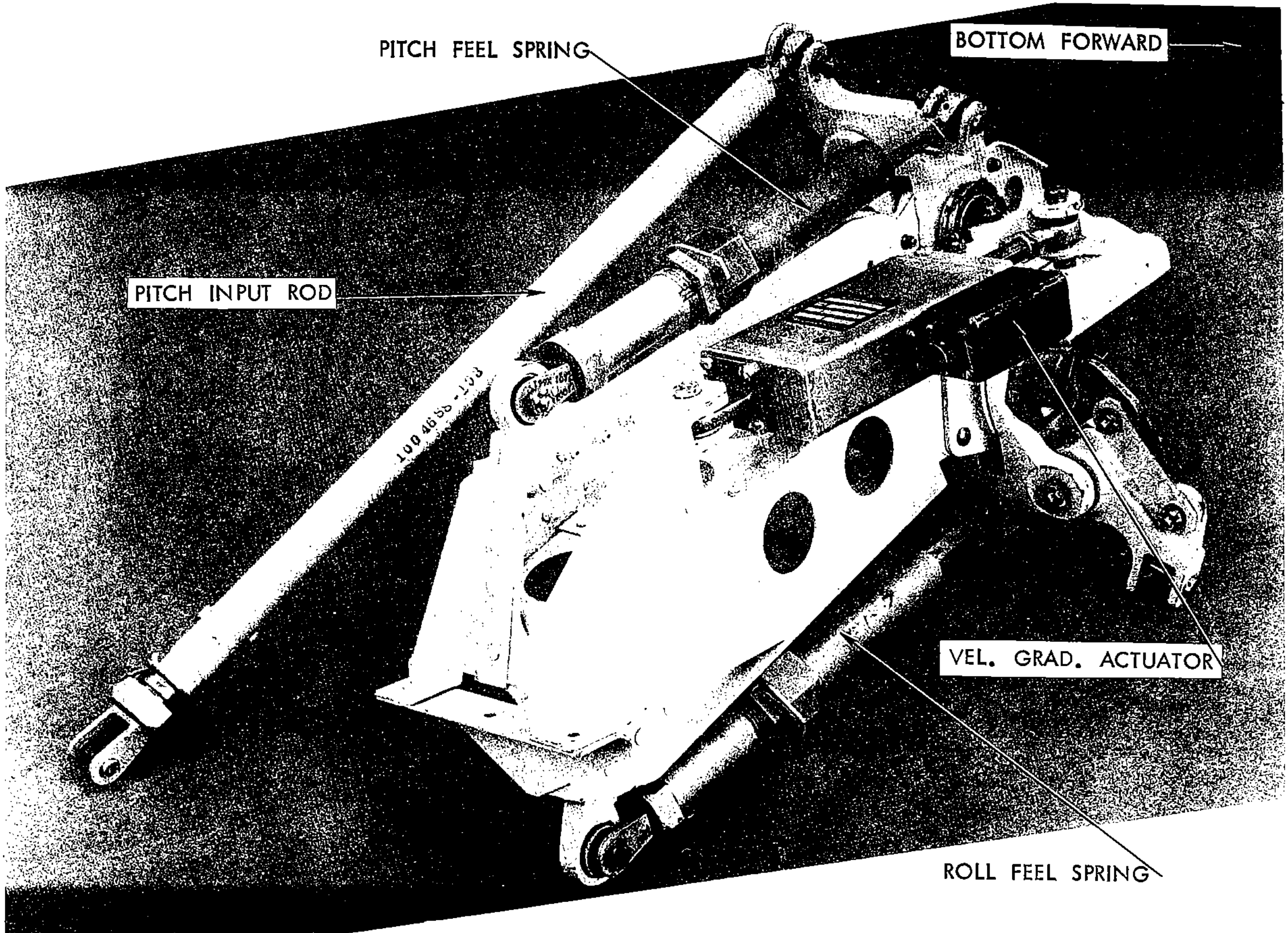


Figure 7A-3. Velocity Gradient Assembly

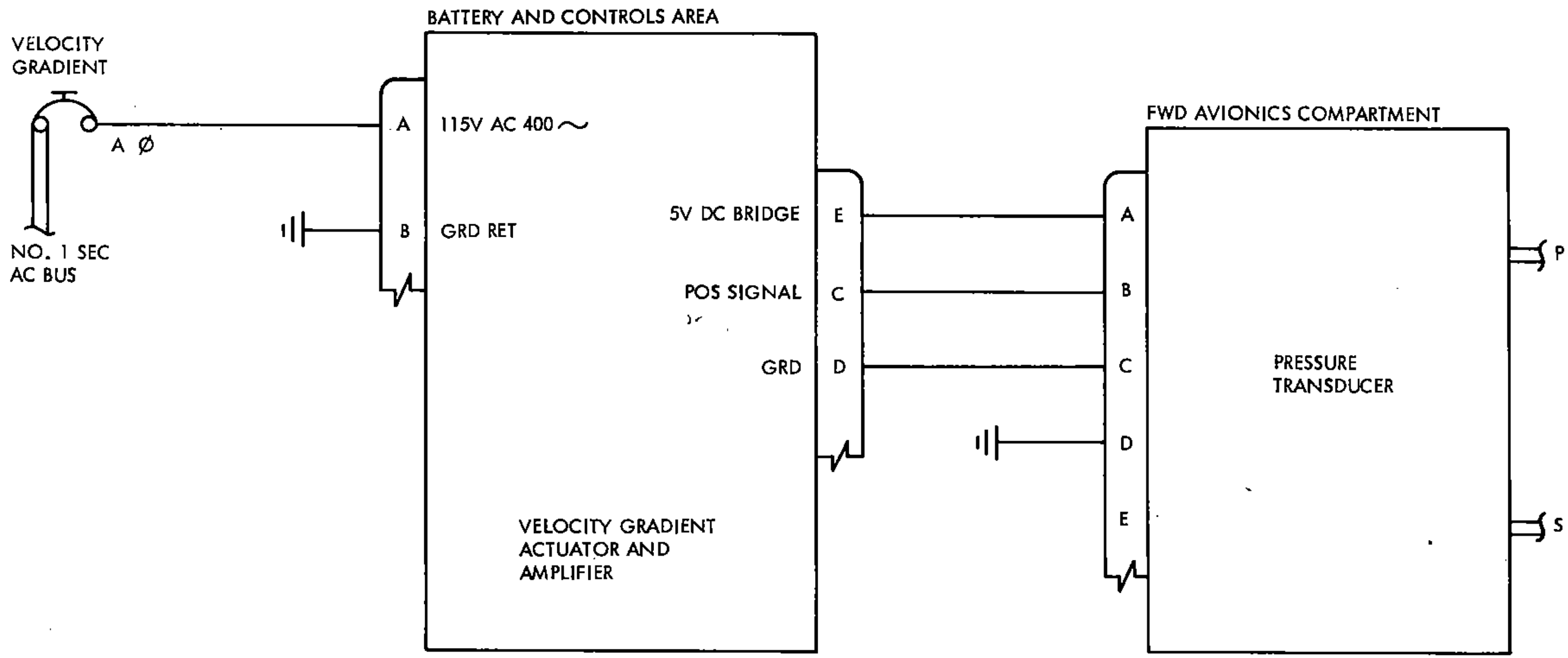


Figure 7A-4. Velocity Gradient Circuit

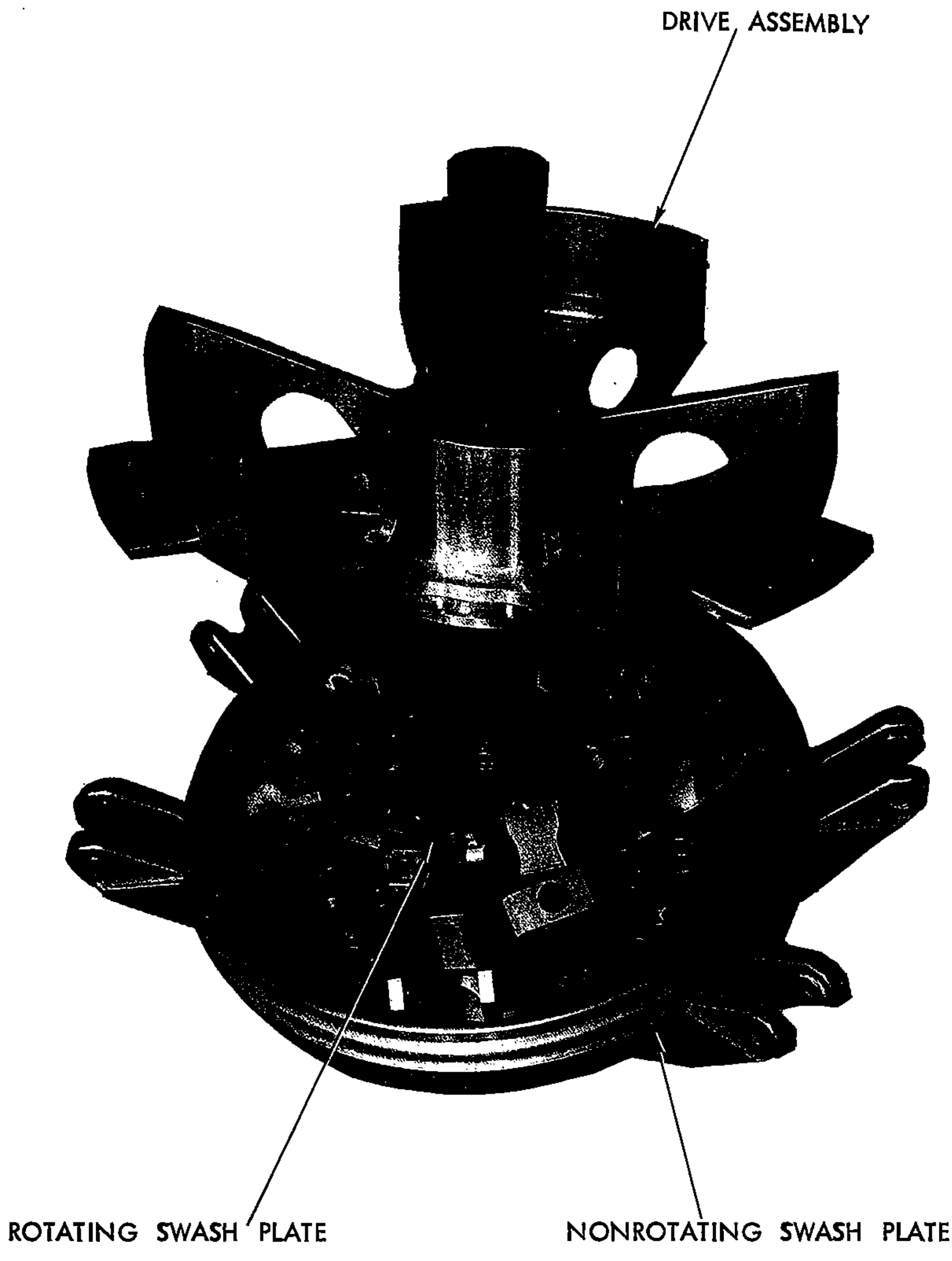


Figure 7A-5. Swash Plate Assembly & Drive Housing

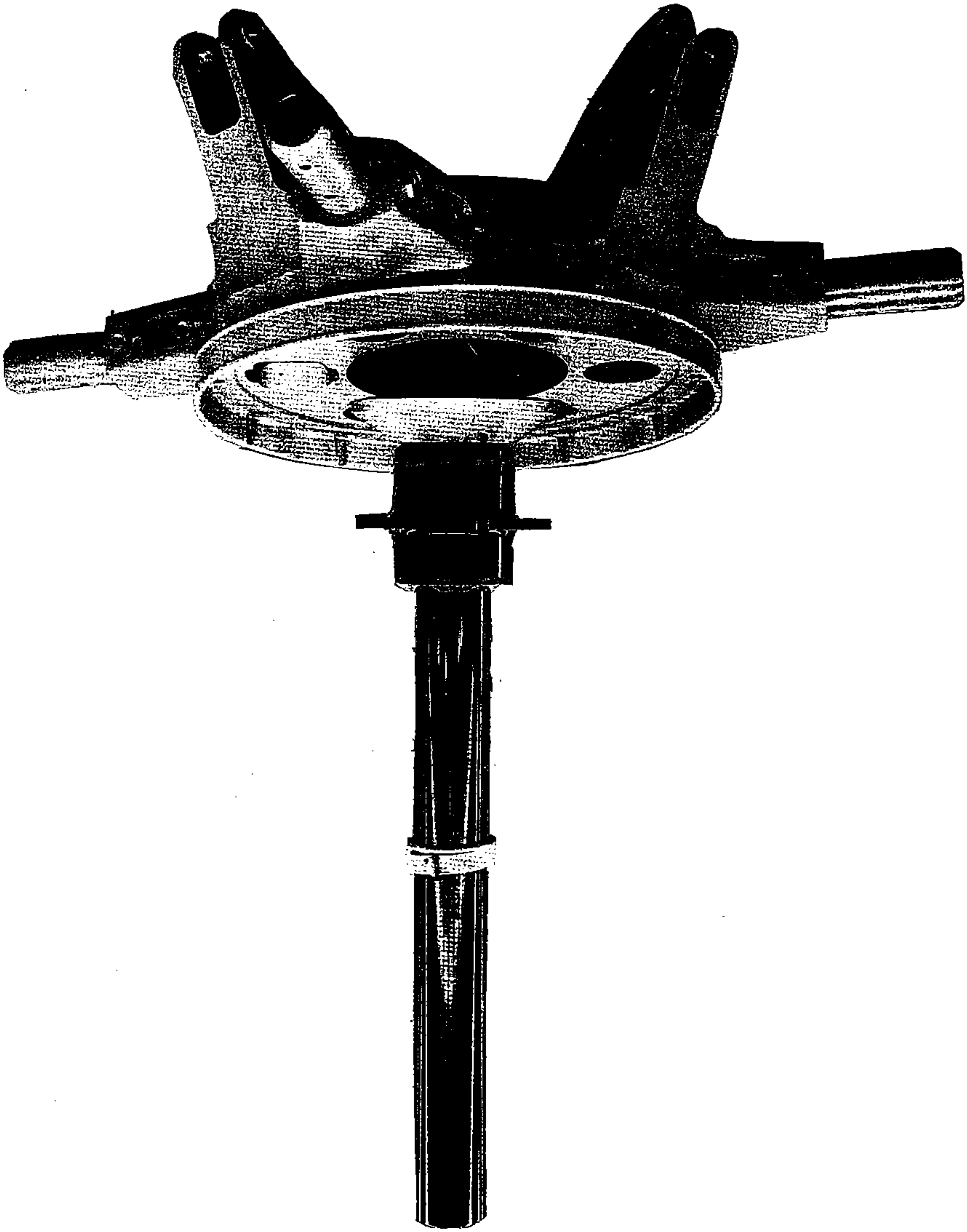


Figure 7A-6. Control Gyro

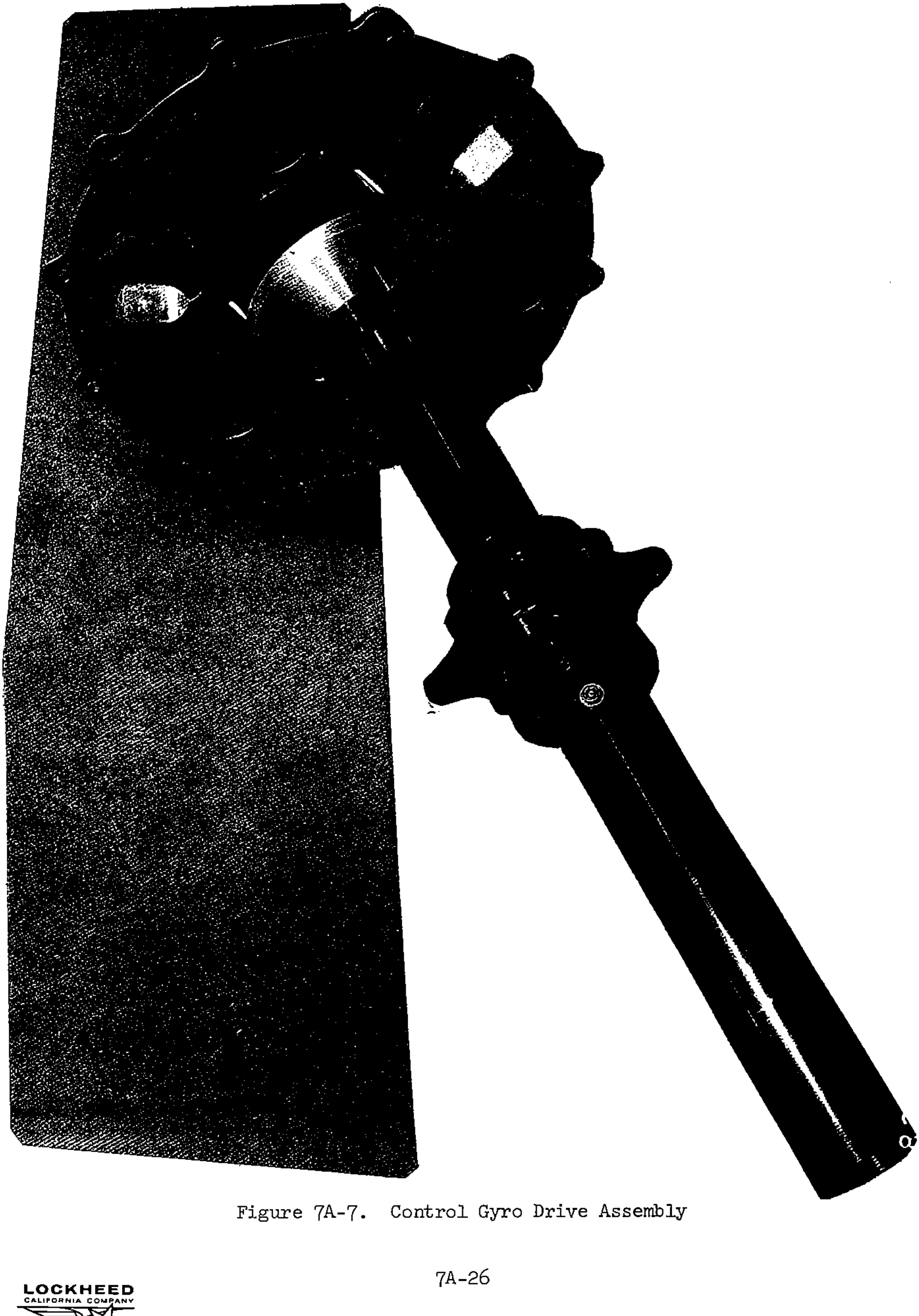


Figure 7A-7. Control Gyro Drive Assembly

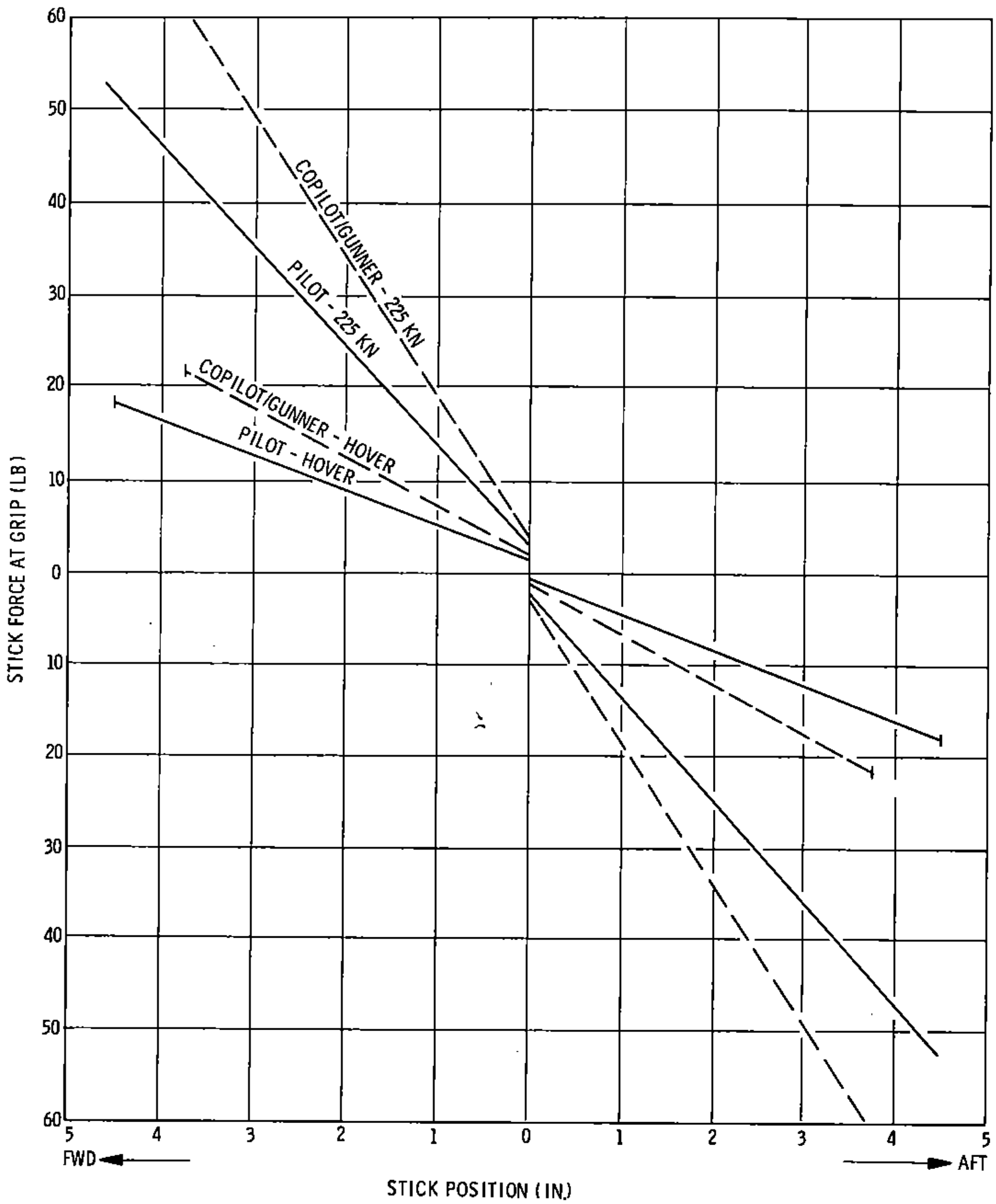


Figure 7A-8. Longitudinal Control System Cyclic Stick Force Versus Stick Position

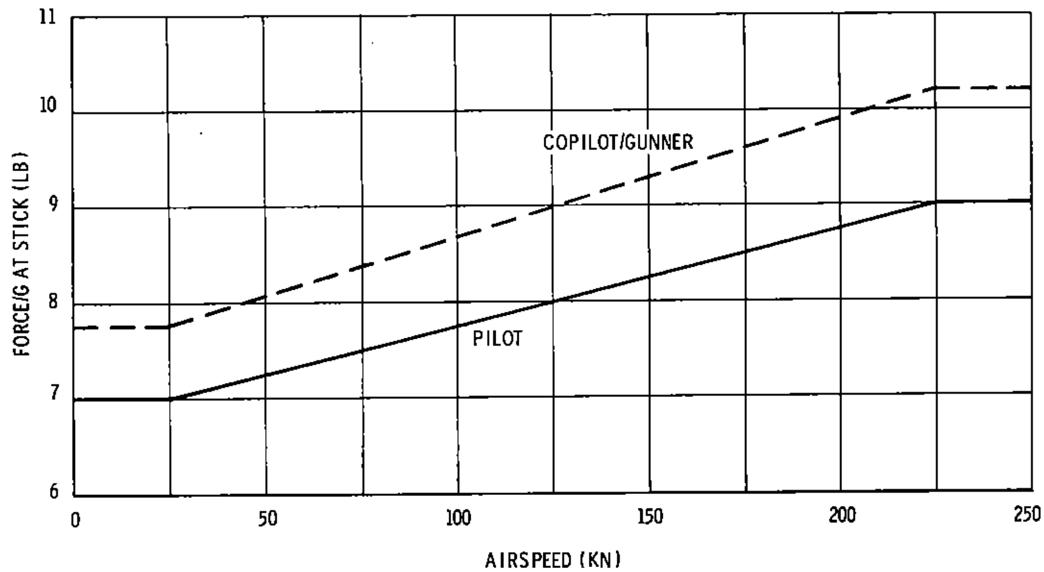


Figure 7A-9. Longitudinal Control System Cyclic Force Per G Versus Aircraft Velocity

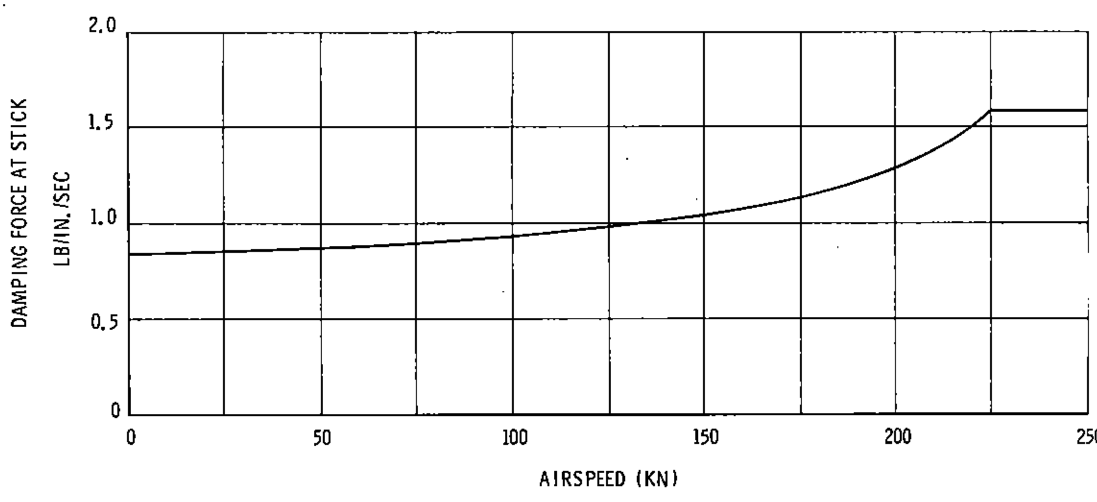


Figure 7A-10. Longitudinal Control System Cyclic Damping Rate Versus Aircraft Velocity

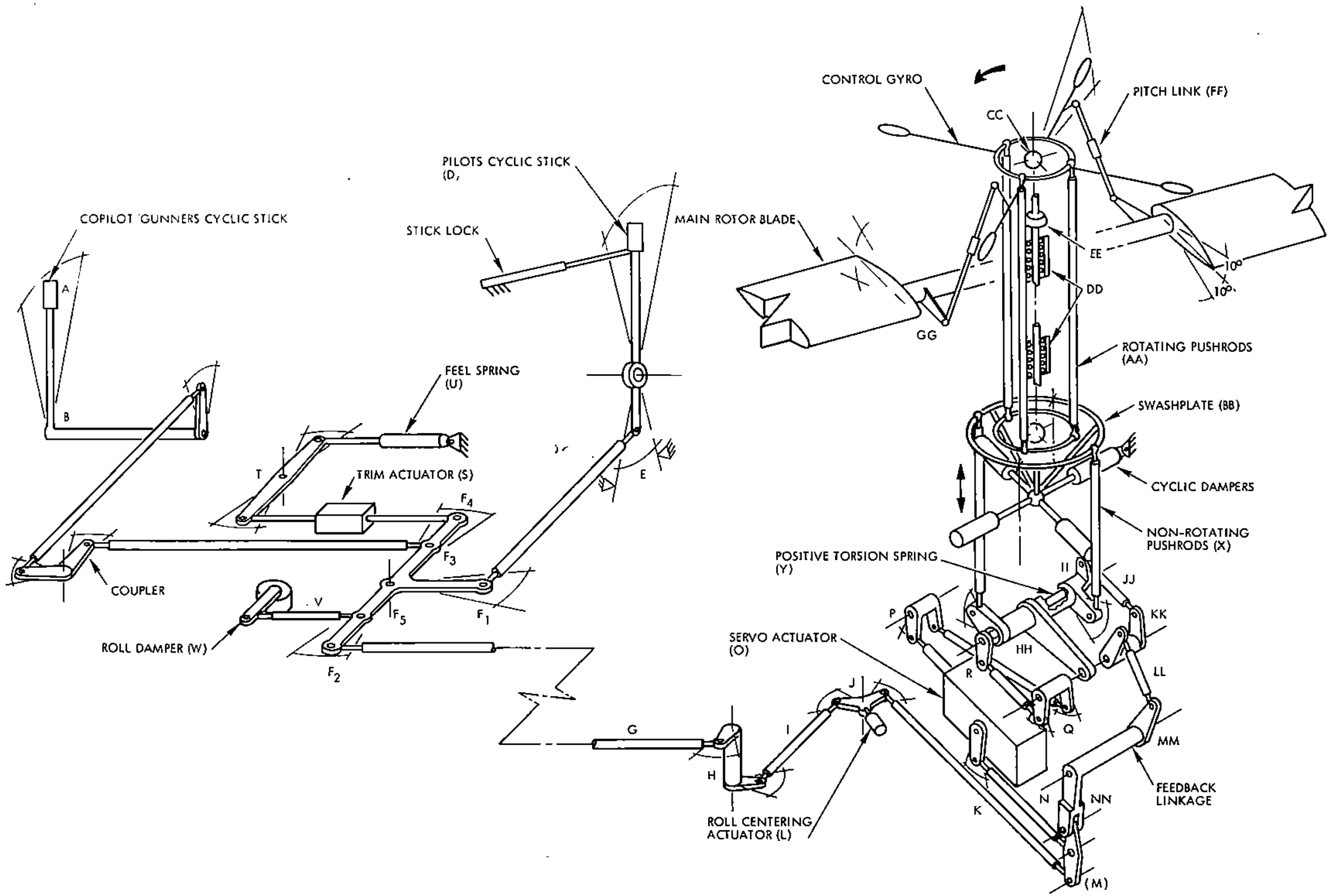


Figure 7A-11. Lateral Control System Schematic

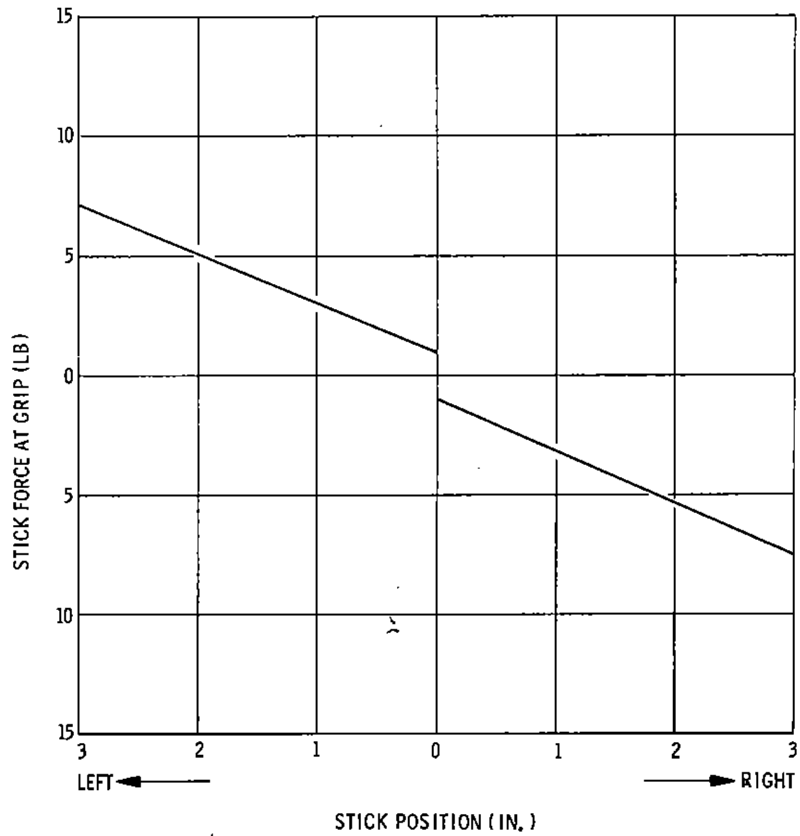
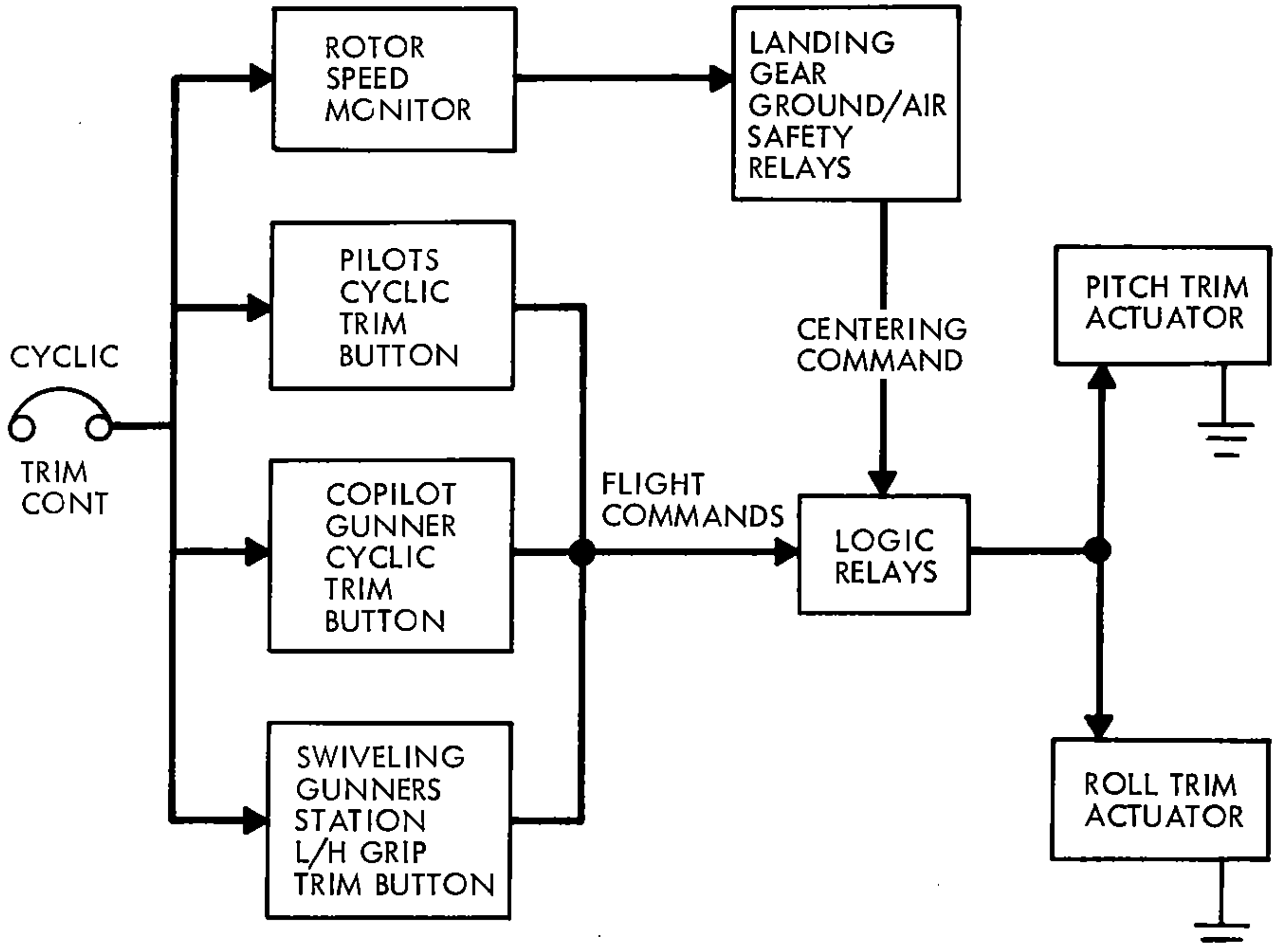


Figure 7A-12. Lateral Control System Cyclic Force Versus Cyclic Position



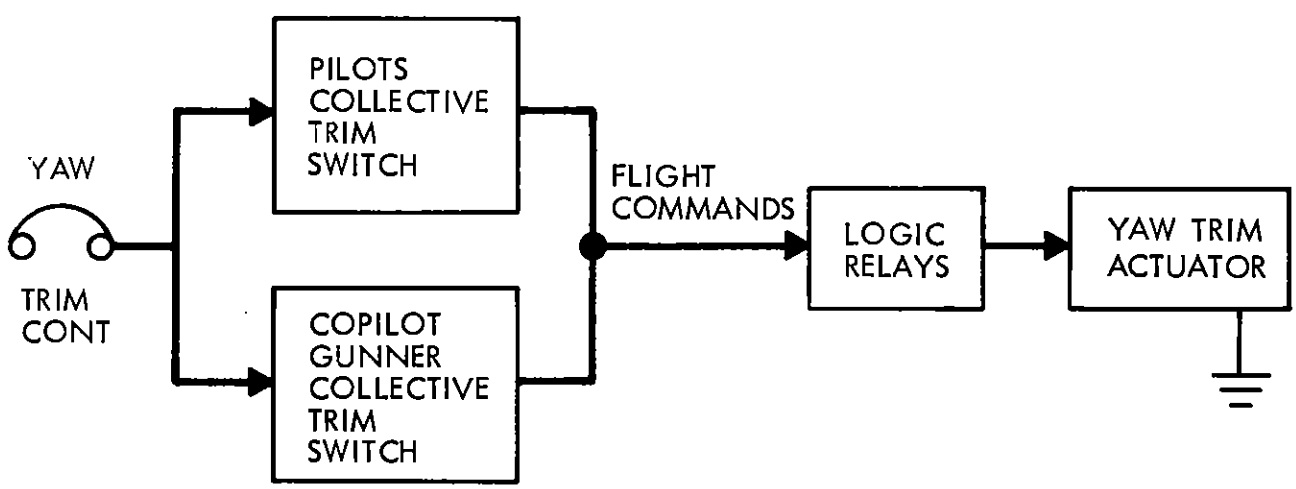
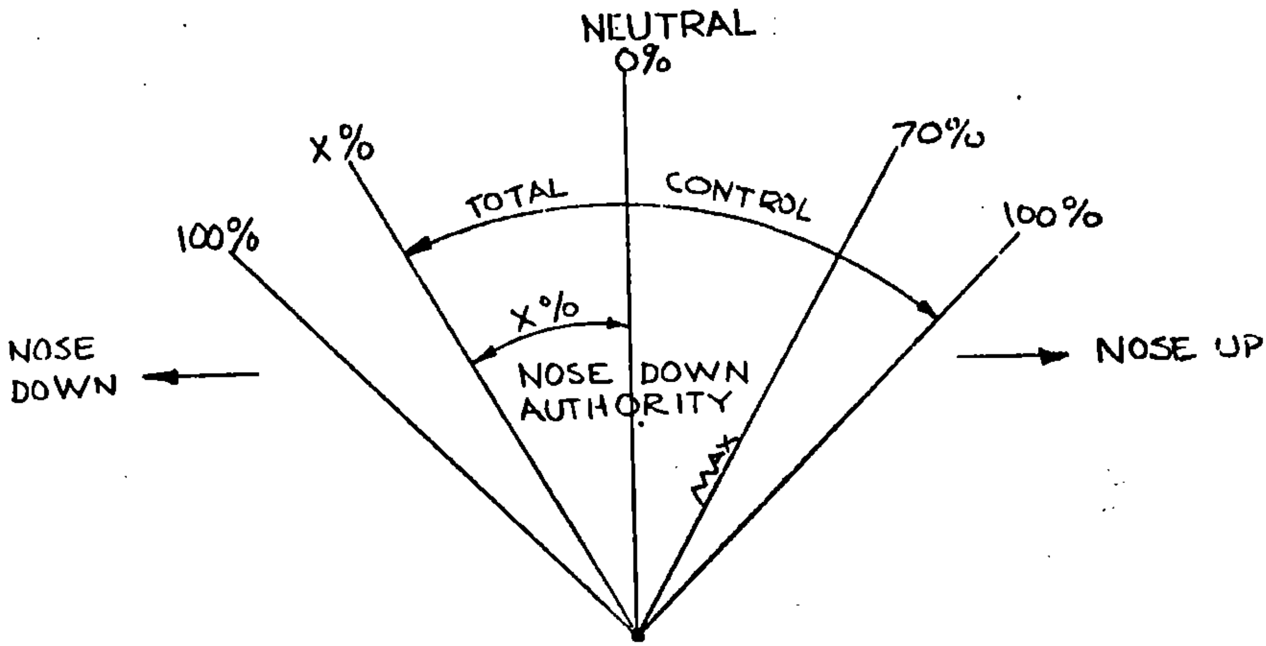


Figure 7A-13. Flight Control Trim System



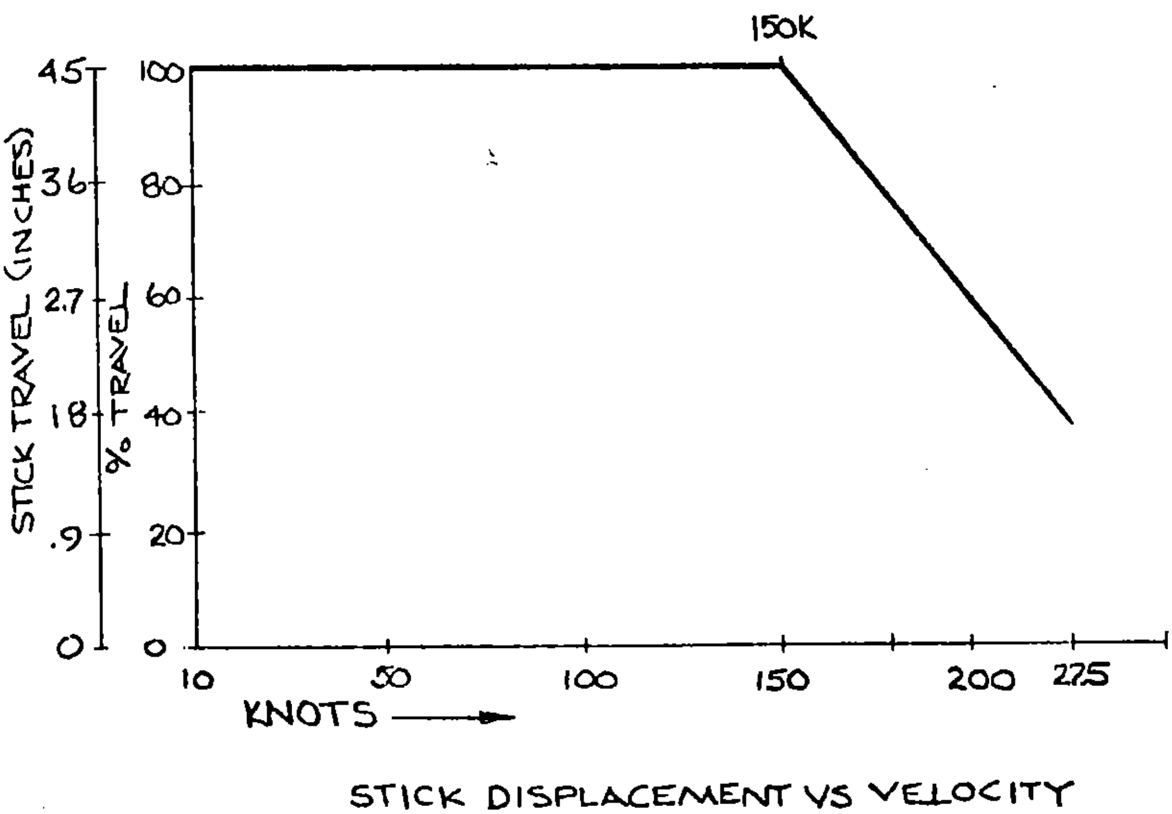


Figure 7A-14. Pitch Control Limitation for a Runaway Trim Actuator

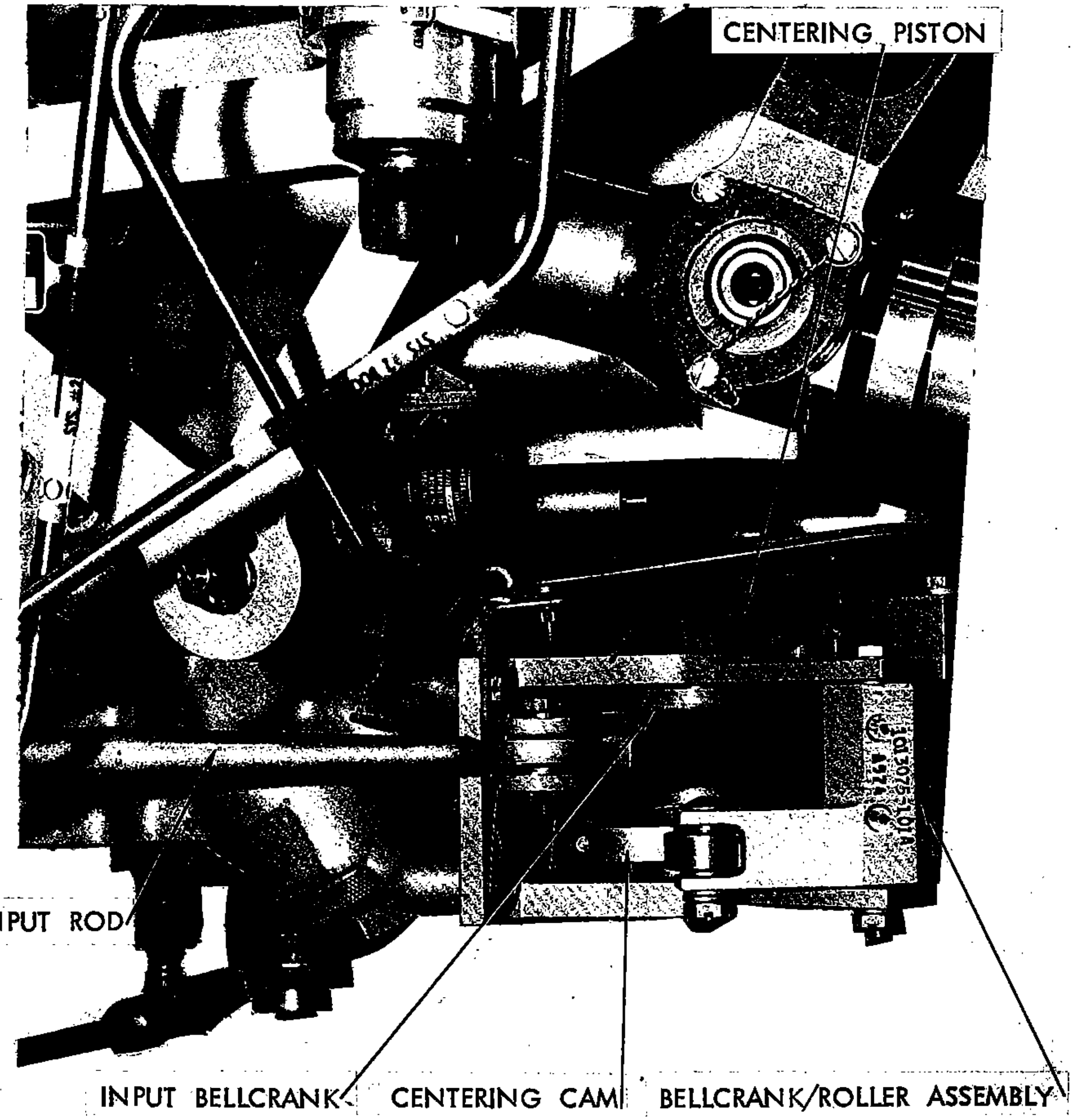


Figure 7A-15. Cyclic Stick Centering Assembly

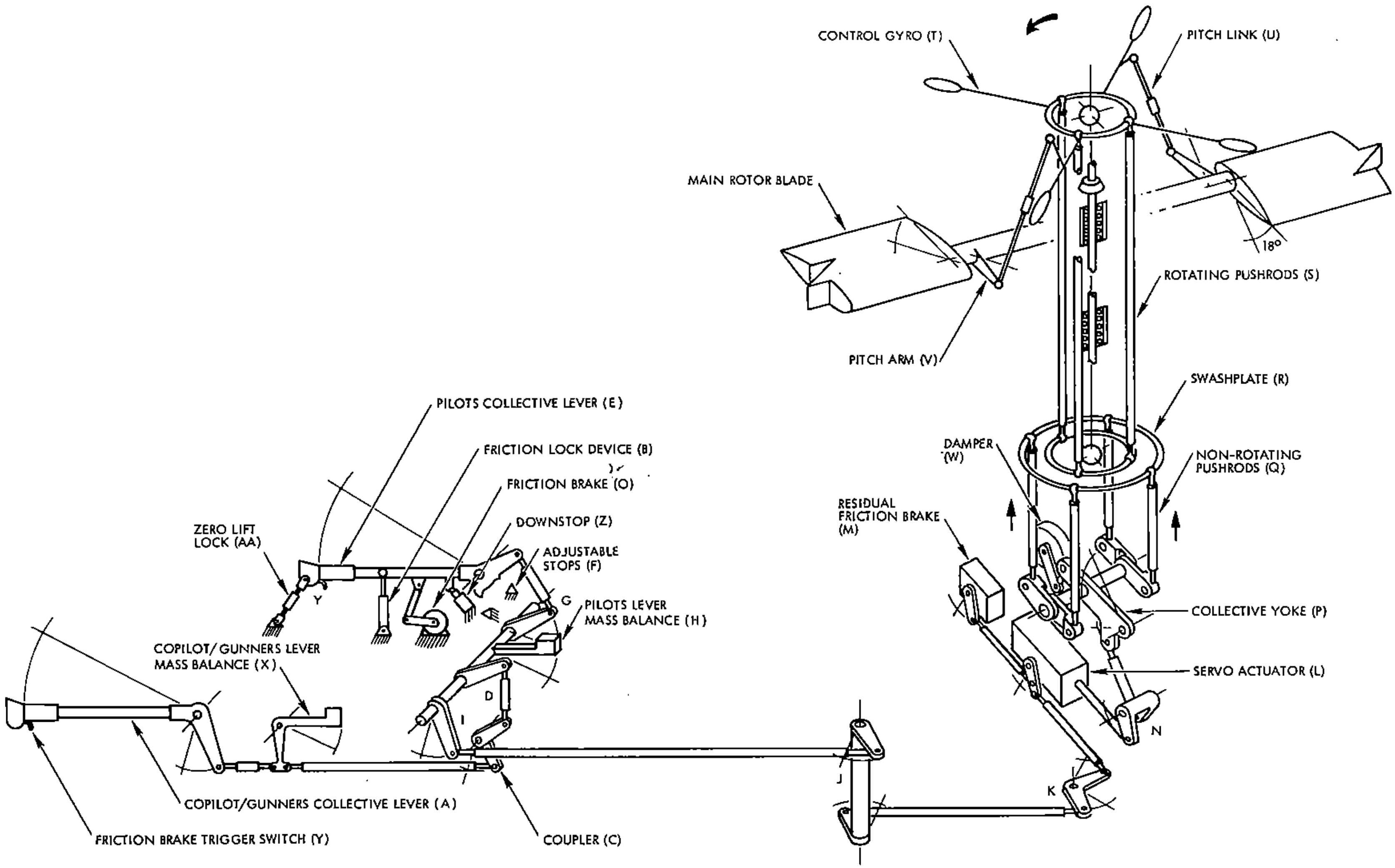


Figure 7A-16. Collective Control System Schematic

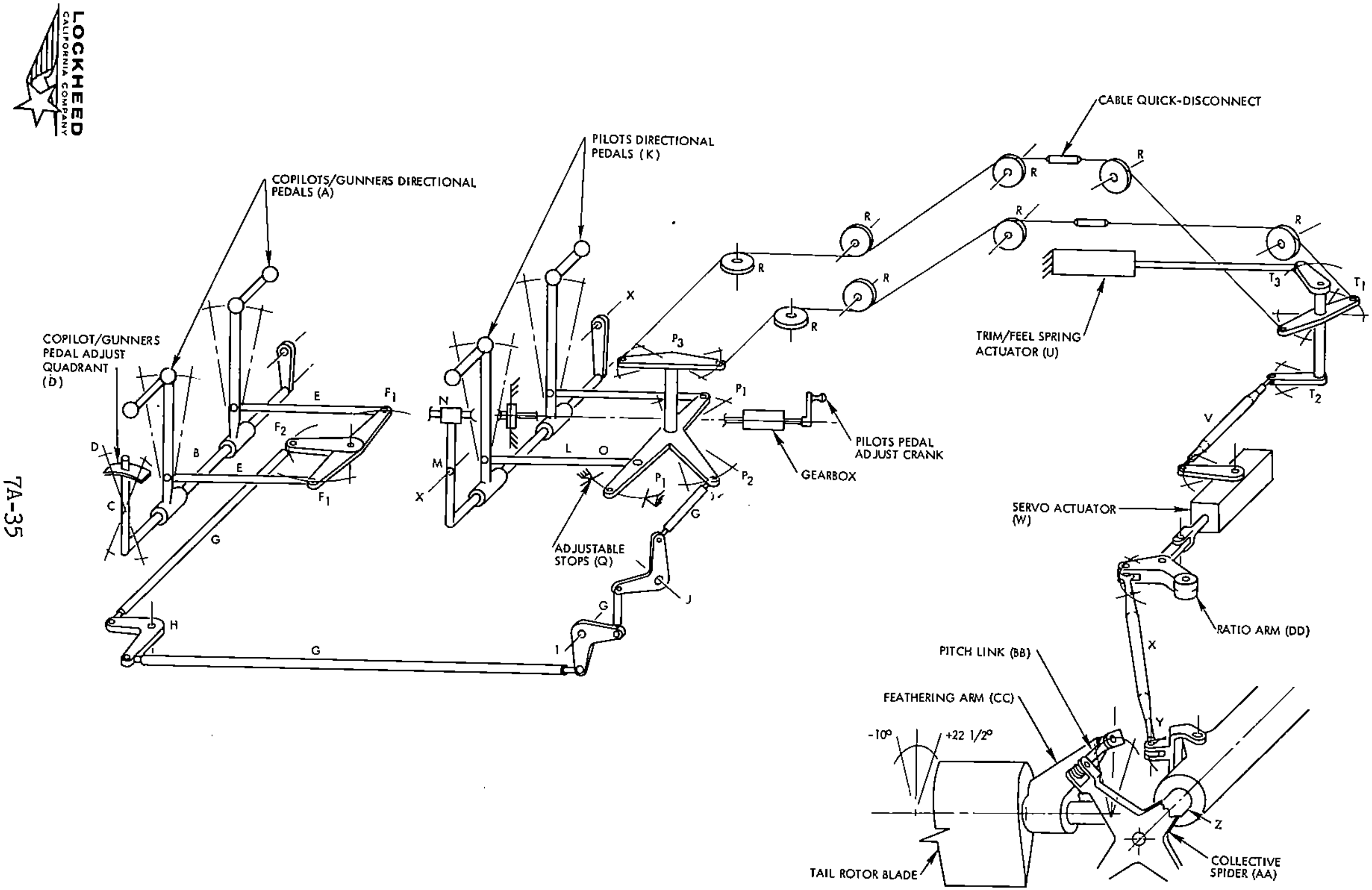


Figure 7A-17. Directional Control System Schematic

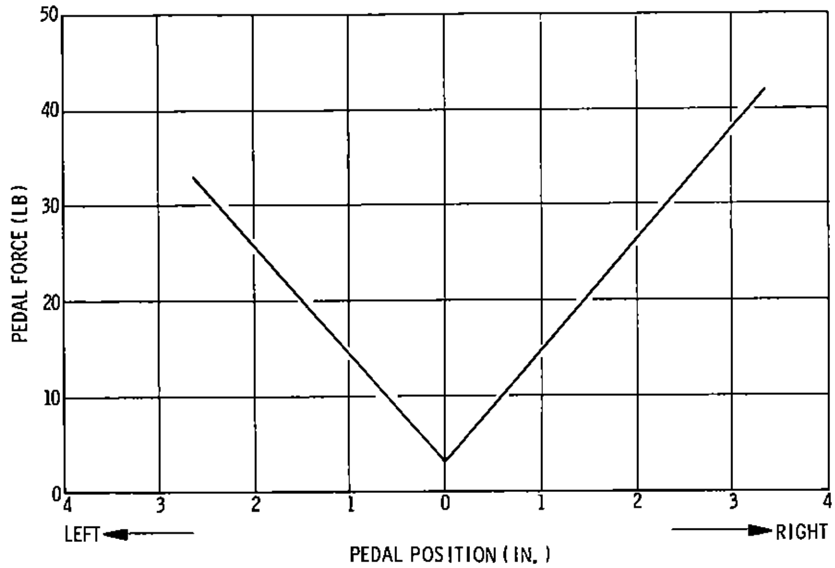


Figure 7A-18. Directional Control System Pedal Force Versus Pedal Position

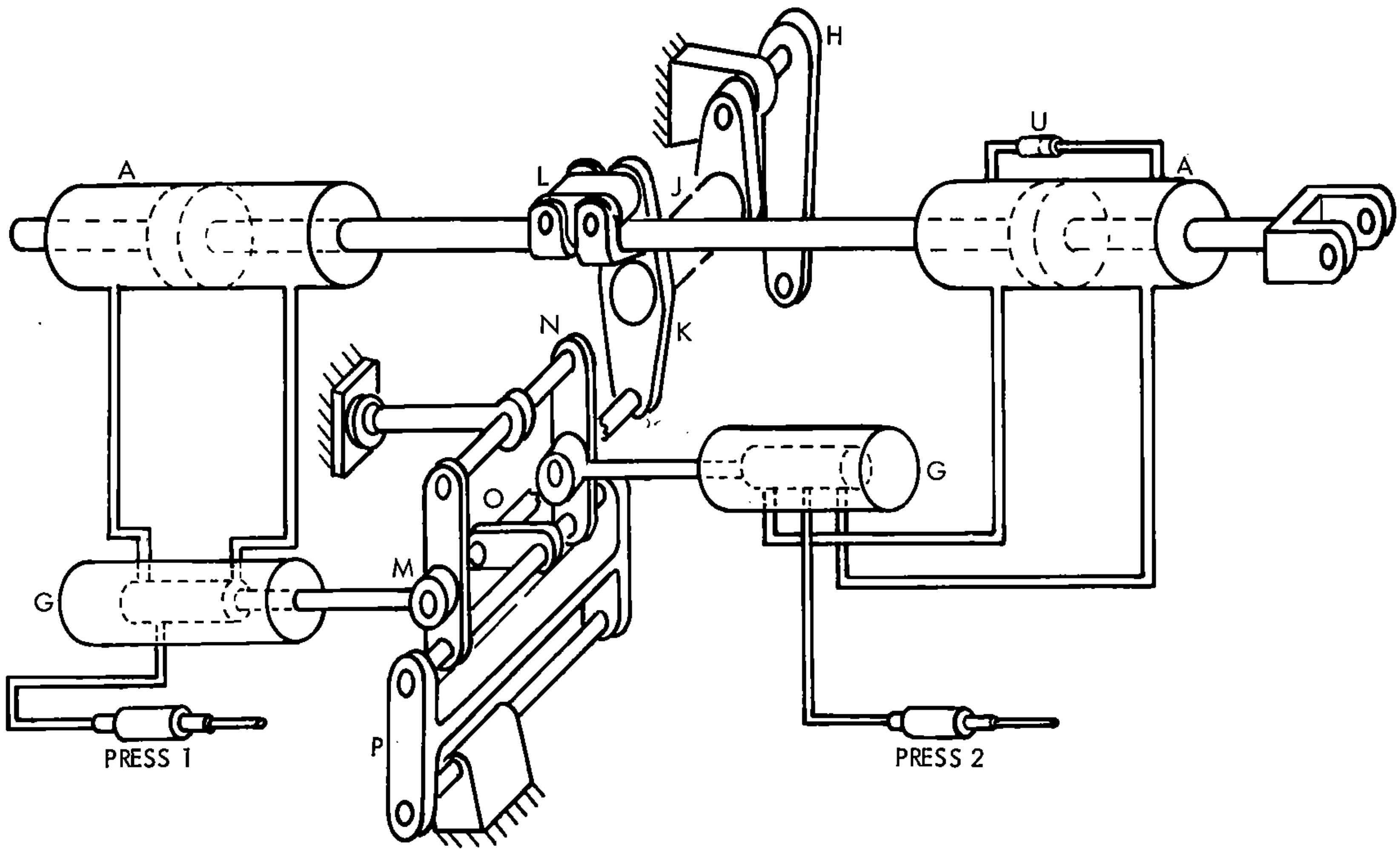


Figure 7A-19. Power Servo Schematic Directional (Yaw)

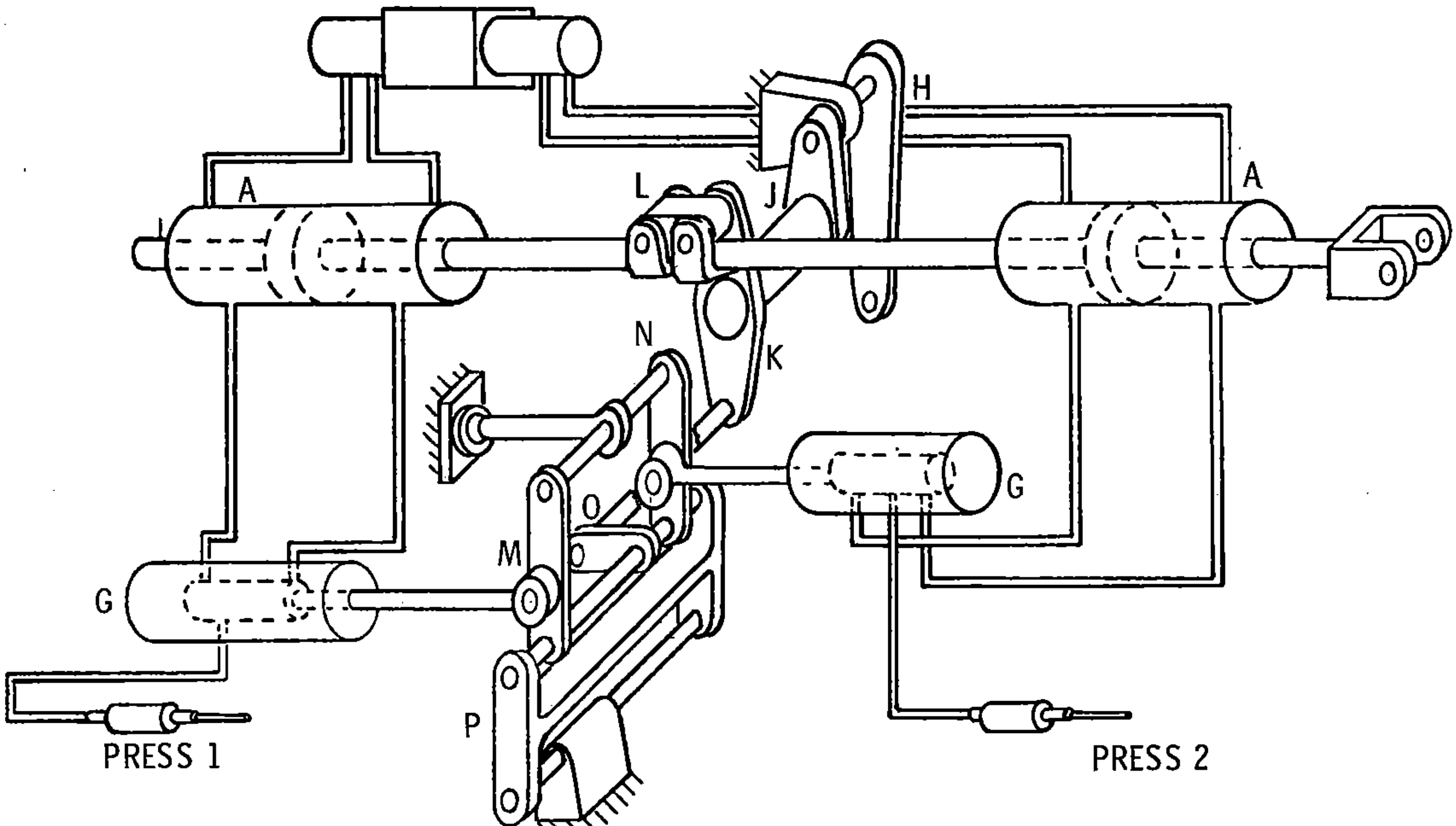


Figure 7A-20. Power Servo Schematic - Collective

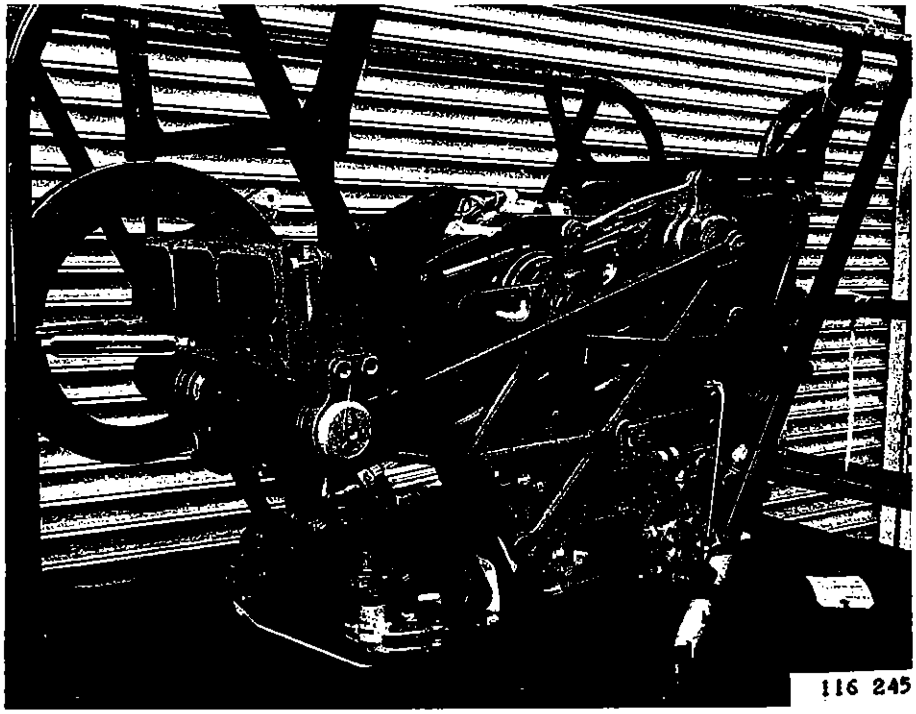


Figure 7A-21.

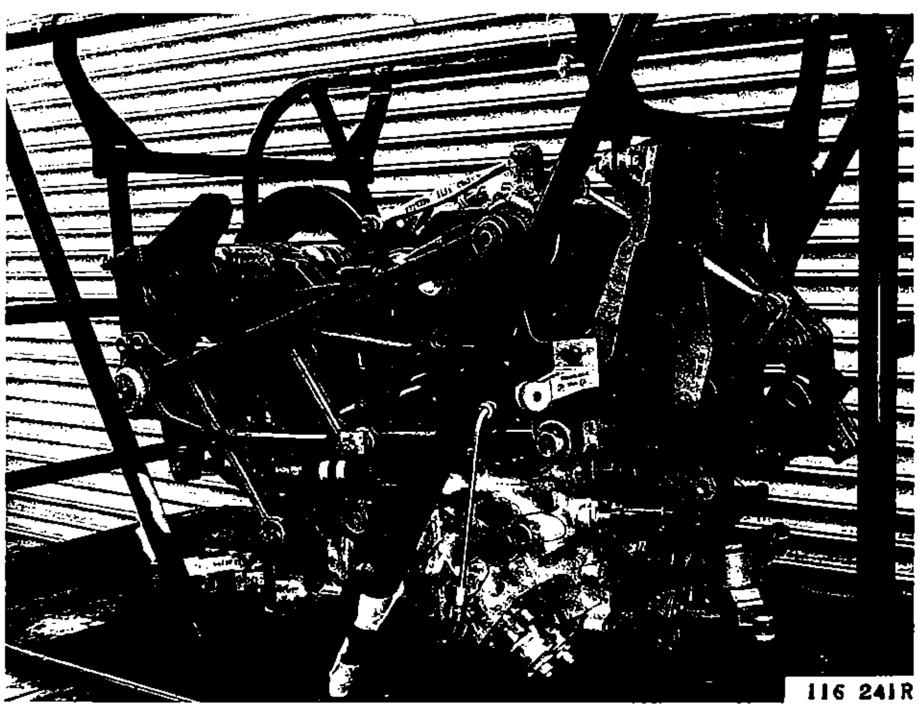


Figure 7A-22.

AMCS DESCRIPTION

The AMCS (Advanced Mechanical Control System) is an improved version of the basic, gyroscopically controlled and stabilized, flapping feedback, rigid rotor control system introduced by Lockheed on the Model 475 helicopter and since incorporated in the Lockheed XH-51, Model 286, XH-51 compound, and AH-56A helicopters. It retains the principles of the basic system but, by a different implementation and arrangement of components, provides certain advantages described below. The key element of the AMCS is the feedback system from the rotor to the gyro. The arrangement of this mechanism is completely different from the original system. The component arrangement of the remainder of the AMCS is largely dictated by this new feedback system arrangement.

For clarity and ease of discussion, the original system, as finally implemented for the AH-56A, is identified herein as the ICS (Improved Control System). The improvements associated with this system are in detail areas; the basic system is identical in concept and arrangement to the original system.

Figure 1 is a block diagram schematic of the ICS. Referring to Figure 1, it is seen that the gyro is in the cyclic feathering system and is connected directly to the rotor blades. The pilot commands cyclic blade feathering by impressing a force, through a power actuator and spring system, on the gyro. The gyro responds by precessing in space. Resultant tilting of the gyro with respect to the spin plane of the rotor feathers the blades cyclically. The feedback from the rotor to the gyro arises from flapping bending of the rotor blades and hub arms through the mechanism of forward sweep of each rotor blade with respect to its feathering axis. This is shown schematically in Figure 2, where the blade sweep angle is exaggerated to illustrate the cranking moment of the blade about the feathering axis resulting from flapping bending resolved through the sweep angle. When opposite blades flap asymmetrically, causing a disturbing or maneuvering moment on the rotor shaft and vehicle, the resultant feathering moment on each blade is transmitted through its pitch arm and pitch link to the gyro, producing a moment on the gyro. The gyro responds to this moment by precessing. The phasing of the feedback moment to the gyro is such as to cause the gyro to tilt with respect to the rotor in a direction which feathers the blades cyclically to oppose and remove the flapping moment which generated the feedback in the first place. Any external disturbance, such as a gust, or steady angular or oscillatory activity of the rotor and vehicle with respect to the gyro, which is a stable body in space, generates feedback and blade cyclic feathering to damp out this activity and thereby stabilize the system. When pilot commands and resultant feedback are impressed on the gyro, the level and direction of imbalance of these moments on the gyro determine the acceleration, rate and direction of a desired maneuver.

The AMCS utilizes a direct flapping feedback system which eliminates certain undesirable noise in the feedback loop of the basic system. As a result of this purification of the feedback signal, the AMCS provides improvements in the following characteristics:

a. Stick force and stick position gradients in the retreating blade stall regime are essentially constant over the entire operating envelope, including the retreating blade stall regime (see Figure 3).

b. Lift/roll coupling is significantly reduced (see Figure 4).

Because of the criteria established for system stiffness, free play and control power at the start of design of the AMCS, improved capabilities, beyond AH-56A contractual requirements, are also obtained:

a. Higher gross weight capability.

b. Higher speed capability.

c. Higher load factor capability.

As noted previously, the AMCS utilizes the same kinds of components as the ICS but arranged in a different sequence. This is illustrated in the block diagram of Figure 5. Here the gyro is placed outside of the blade cyclic feathering system; the feedback to the gyro does not go through the feathering system; and the gyro feathering commands to the rotor are amplified by the cyclic power actuators. This disassociation of the gyro from the feathering system derives from the implementation of a feedback system which senses rotor blade flapping only and eliminates noise in the feedback loop. This implementation is illustrated in Figure 6, which also includes a picture of the ICS feedback system for comparison. A feedback arm, which is simply a stiff cantilever beam, is attached rigidly to each fixed hub arm (a nonfeathering element of the rotor) at a point radially outward from the center of the hub where there is adequate flapping activity of the fixed hub generated by blade flapping. The feedback arm measures and amplifies linearly the flapping slope and deflection changes of the fixed hub arm. The hub arm slope and deflection changes are directly proportional to the rotor flapping moments. No other motions are transmitted. The inboard end of each feedback arm is connected to a push rod which extends vertically downward inside the rotor mast. The lower end of each feedback rod operates a crank which compresses or extends a feedback spring. Each feedback spring is link connected to the rotating mass of the small gyro, which is positioned below the transmission, inside the fuselage. The feedback springs convert the flapping motions of the rotor arms into proportionate moments on the gyro. All of the above elements are rotating at rotor speed. When the gyro precesses under the action of these moments, its change in tilt with respect to the mast is transmitted as a cyclic blade feathering position command to the cyclic power actuators, which feather the blades.

The pilot command system for vehicle control is shown in Figure 7. The pilot's cyclic stick position is transmitted through the normal control mechanism from the cockpit to the valve on a small servo-actuator. When the actuator ram changes position in accordance with this command, it compresses or extends the positive spring which, identically to the feedback springs, produces a moment on the gyro. Each positive spring (roll and pitch) is connected to the outer, nonrotating gimbal ring which supports the rotating gyro mass. As in the case of the feedback, the resultant gyro tilt is transmitted as a cyclic blade feathering position command to the cyclic power actuators, which feather the blades. All of the above components are in the nonrotating system.

The hardware implementation of the AMCS in the AH-56A is determined principally by the following considerations:

1. Stiffness criteria

2. Free play/wear criteria

Initally, general configuration studies included conventional swashplate and push rod arrangements with both "inside-the-mast" and "outside-the-mast" mechanizations. The selected configuration, described below, satisfied the above conditions with the least weight and greatest life potential of all configurations investigated. It provides a minimum number of joints (bearings) in the feathering system, located to provide deamplification of bearing free play as seen at the rotor blades, and permitting the use of large diameter, low pressure loaded bearings to minimize wear and achieve long life. It provides the required ratio of collective to cyclic feathering system stiffnesses (approximately 2 to 1) in a single, rigid structural/mechanical element which replaces the normal assemblage of swashplate, bearings, and push rods in a conventional system. This element, designated the sliding spatial lever, is connected directly to both the cyclic and collective power actuators without intermediate, highly loaded linkages.

The weight of the prototype AMCS is 61 pounds heavier than the ICS. In production, where interface elements such as the main rotor fixed hub forgings, could be changed, the AMCS weight is currently predicted to the 42 pounds lighter than the ICS. No quantitative comparisons are available at this time on the Maintenance Man Hours per Flight Hour. However, the ICS portion of the total vehicle MMH/FH is very small, so that any changes in maintenance time for AMCS will have a negligible effect on vehicle availability and operational costs.

The AMCS configuration is illustrated schematically in Figure 8. For clarity, the rotor mast, transmission, and structural supporting elements of the control system itself are omitted. The entire system is supported by the rotor mast. The upper SSL bearing fits into an antifretting ring bonded to the inside diameter of the mast about a foot below the top of the mast. Vertical location and rotational driving of the upper SSL bearing is provided by an aluminum tube, bolted to the top of the mast and extending downward, just inside the mast, to the upper SSL bearing. The upper SSL bearing is keyed to the SSL for rotational driving; vertical slots in the SSL engage the driving keys so that the SSL is free to slide up and down inside the bearing for collective control. The bearing itself contains a spherical element which accommodates tilting of the SSL for cyclic control. Four sets of ferrules, one set for each feedback rod, are bonded to the inside of the mast below the SSL bearing to support the long, slender feedback rods.

Indexed and bolted to the bottom of the mast is a downward extending, steel, tubular member, approximately 12 inches in diameter, which is the main structural supporting element for the gyroscope, feedback springs and the servo actuators. A large diameter, double row ball bearing is installed at the lower end of this rotating mast extension tube for connections of the static structural elements to which the actuators are mounted directly. The first such element, immediately below the mast extension, is an assembled pair of cast aluminum alloy housings which support and enclose the cyclic power actuators and their associated linkages. The cyclic actuators' valves, the pilot input actuators, and the positive spring and gyro damper assemblies are bolted to the outsides of these castings.

Extending downward from the cyclic actuators' housing is a welded, steel tubular framework, similar to an engine mount trusswork tilted up vertically, which is attached to the housing with four bolts. At the bottom of this framework is a cast, steel, X-plan form beam, to which is mounted the base of the vertically disposed collective actuator.

An enclosure with access doors extends from the top of the cyclic actuators' housing upward to the base of the transmission. This enclosure surrounds the gyro and reacts any torsion in the static structural members caused by friction in the bearings which are between the nonrotating elements and the rotating members in the control package. The enclosure is isolated structurally from all other loads by means of flexible elements at its upper and lower edges.

Two sealing boots are installed, one at the top of the mast, between the rotor hub and the sliding spatial lever, and the other at the lower section of the system extending downward from the bottom of the cyclic actuators' housing to the top of the collective actuator body. The entire feathering system, with the single exception of the collective actuator lower support bearing, is contained within the lubricated, environmentally protected area bounded by these boots. The control system is lubricated by spraying oil through nozzles mounted in this enclosed area. This lubrication system is selfcontained, independent of the transmission and engine lube systems. It includes a small, electrically driven pump submerged in a spherical reservoir and appropriate plumbing for circulation.

The sliding spatial lever is a tapered, steel tubular member, extending vertically through the rotor mast with four, mutually perpendicular, radially directed, horizontal arms at its top. The arms are joined to the vertical stem by a single electron beam weld in the uppermost tubular section. Each of the four arms is connected at its outboard extremity to the upper end of a pitch link. The lower end of the pitch link is attached to an arm on the movable (feathering) portion of the hub. Vertical motions of the sliding spatial lever produce collective blade angle changes. Tilting motions of the sliding spatial lever produce cyclic blade angle changes. Accordingly, the collective actuator is mounted vertically and attached to the bottom of the SSL so as to slide it up and down. The cyclic actuators are mounted horizontally at 90° to each other and slightly above the bottom of the SSL. Motions of these actuators tilt the SSL about the upper SSL support bearing, which serves as a fulcrum for the lever-like action of this member under these conditions. Tilting of the SSL may be in any azimuthal direction to achieve pure pitch, pure roll, or any combination of the two.

The attachment of the collective actuator to the SSL is through two bearings: (1) a spherical bearing at the top of the actuator ram, which in turn is attached to a clevis fitting with an upper tubular body containing (2) a double row, preloaded ball bearing whose inner race is mounted to the SSL. This arrangement provides rotational isolation between the rotating SSL and the nonrotating collective actuator through the ball bearing while the spherical bearing accommodates the "knee-break" action which occurs when the SSL is tilted by the cyclic actuators.

The cyclic actuators are connected to the SSL through a set of concentric bearings within a single housing. The innermost bearing is a cylindrical sleeve which rotates with the SSL but provides freedom for the vertical motion of the SSL associated with collective feathering. Surrounding this cylindrical bearing is a spherical, roller bearing. This bearing isolates the rotational motion of the SSL from the actuators and accommodates the misalignment between the tilting SSL and the horizontally supported actuators. One of the cyclic actuators is connected rigidly to the outer race of this bearing. Surrounding this bearing is a cylindrical bearing to which the second cyclic actuator is rigidly attached. The purpose of this bearing is to accommodate the relative angular motions between the cyclic actuators which occur in a horizontal plane when either one or both of these actuators move to produce cyclic blade angle changes.

From the mounting of the actuators and the motions of the SSL, it is apparent that tilting of the SSL for cyclic feathering would induce collective feathering also if the collective actuator did not sense this motion and change ram position accordingly. Similarly, roll and pitch would be cross-coupled if each cyclic actuator did not sense the motion of the other and compensate accordingly. Sensing linkages are incorporated for both these purposes which provide very accurate compensation and virtually eliminate any cross-couplings from these basic kinematic motions.

All of the servo actuators are fail-safe and fail-operational members. Each contains dual valving and pistons operating on dual hydraulic systems driven by separate pumps and completely noninterconnected. The collective and pilot input actuators are conventional dual tandem units. The cyclic actuators each have three cylinders arranged side-by-side rather than in tandem. The center cylinder works on one hydraulic system and has a piston area twice that of each of the outer cylinders. The outer cylinders are operated by the second hydraulic system. The cyclic servo valves are located remotely from the actuators themselves but are hydraulically connected through rigid plumbing to maintain stiffness and resolution requirements. The actuators' load capabilities, even with one hydraulic system inactive, are adequate to handle maximum predicted blade feathering loads.

Virtually every component in the AMCS is designed by stiffness or wear criteria so that operating stresses are extremely low. The system life requirement is 3600 hours. Bearings and seals must achieve a minimum 1200 hour life. All other components have a 3600-hour life requirement.

ICS
CONCEPT
BLOCK DIAGRAM

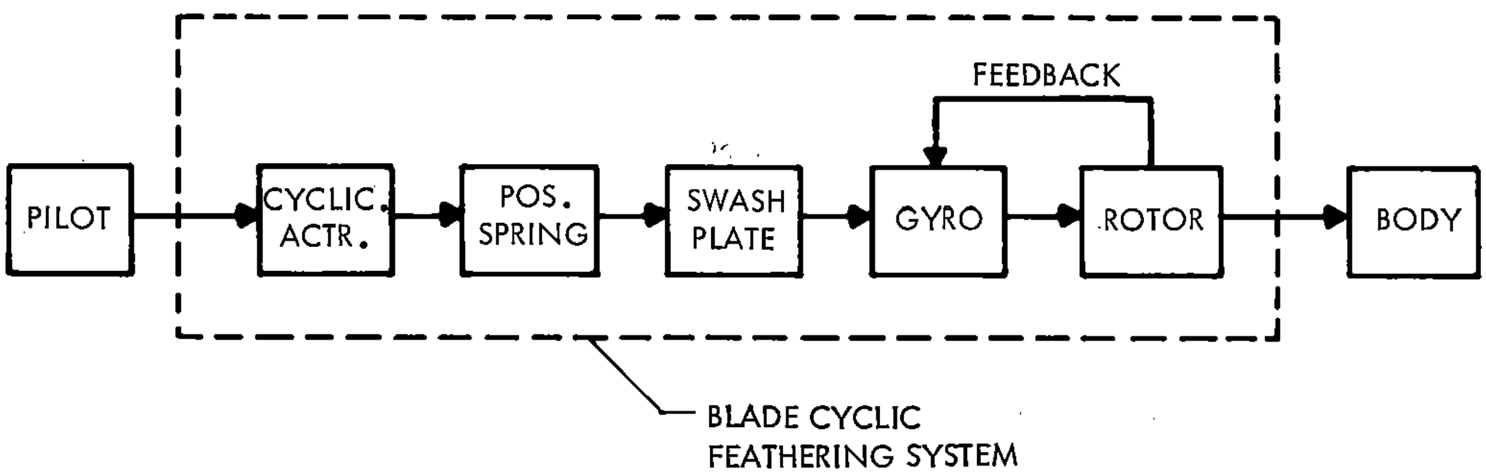
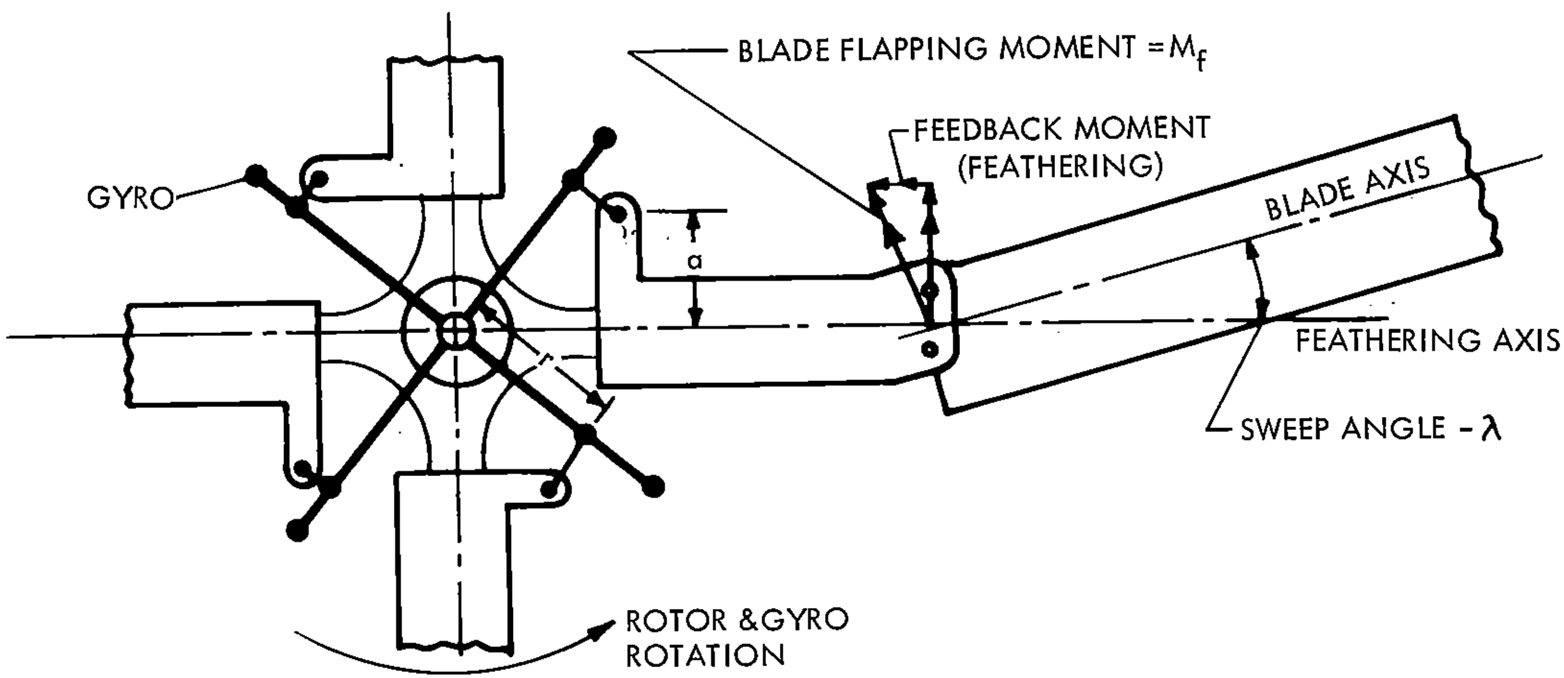


FIGURE 1

ICS
CONCEPT
BLADE SWEEP FEEDBACK



FEEDBACK MOMENT TO GYRO = $\lambda M_f \frac{r}{a}$

FIGURE 2

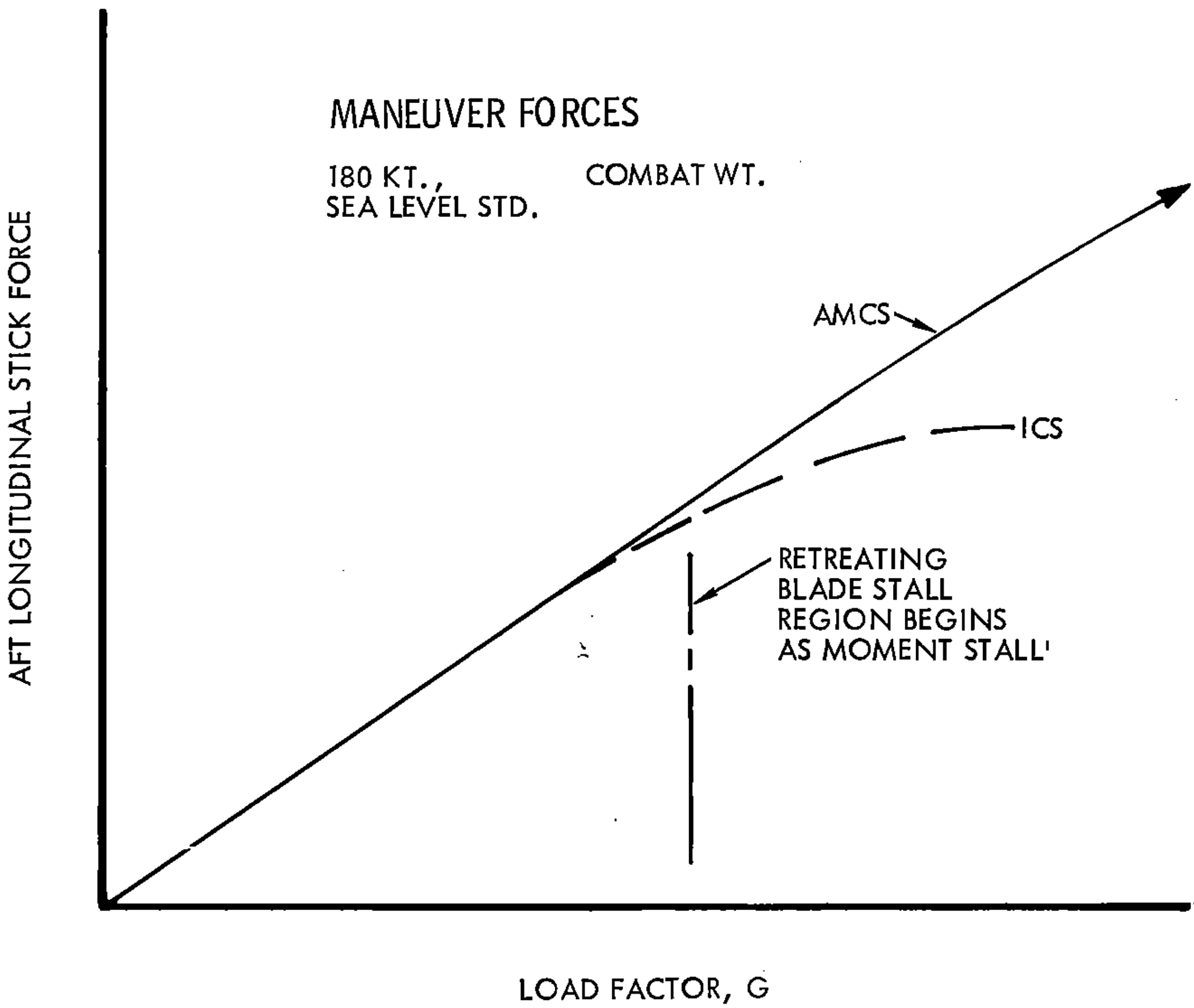

MANEUVER FORCES
180 KT.,
SEA LEVEL STD.
COMBAT WT.
AMCS
ICS
RETREATING
BLADE STALL
REGION BEGINS
AS MOMENT STALL
AFT LONGITUDINAL STICK FORCE
LOAD FACTOR, G

FIGURE 3

MANEUVER CROSS - COUPLING

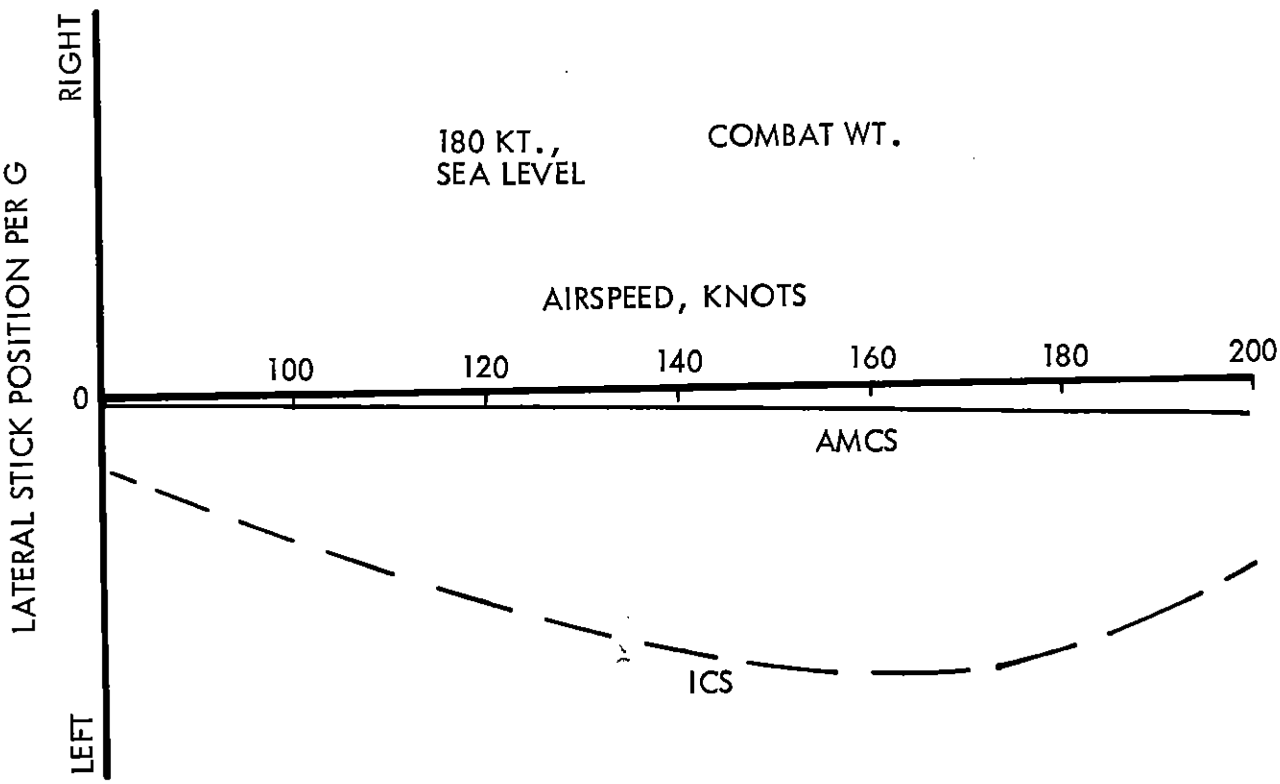


FIGURE 4

AMCS

CONCEPT
BLOCK DIAGRAM

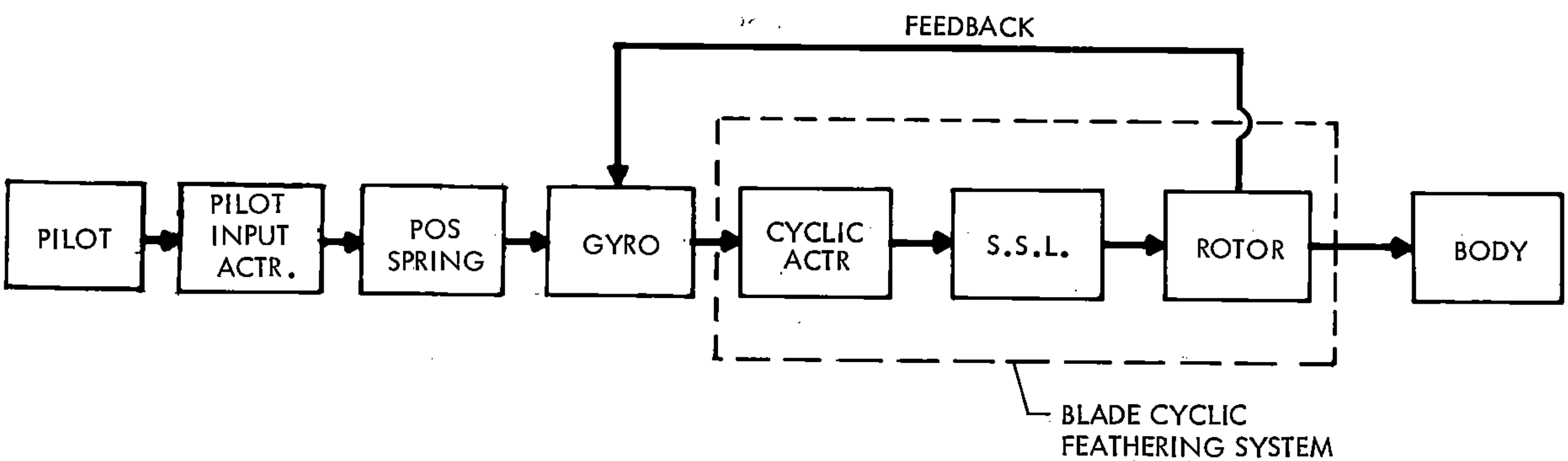


FIGURE 5

MOMENT FEEDBACK

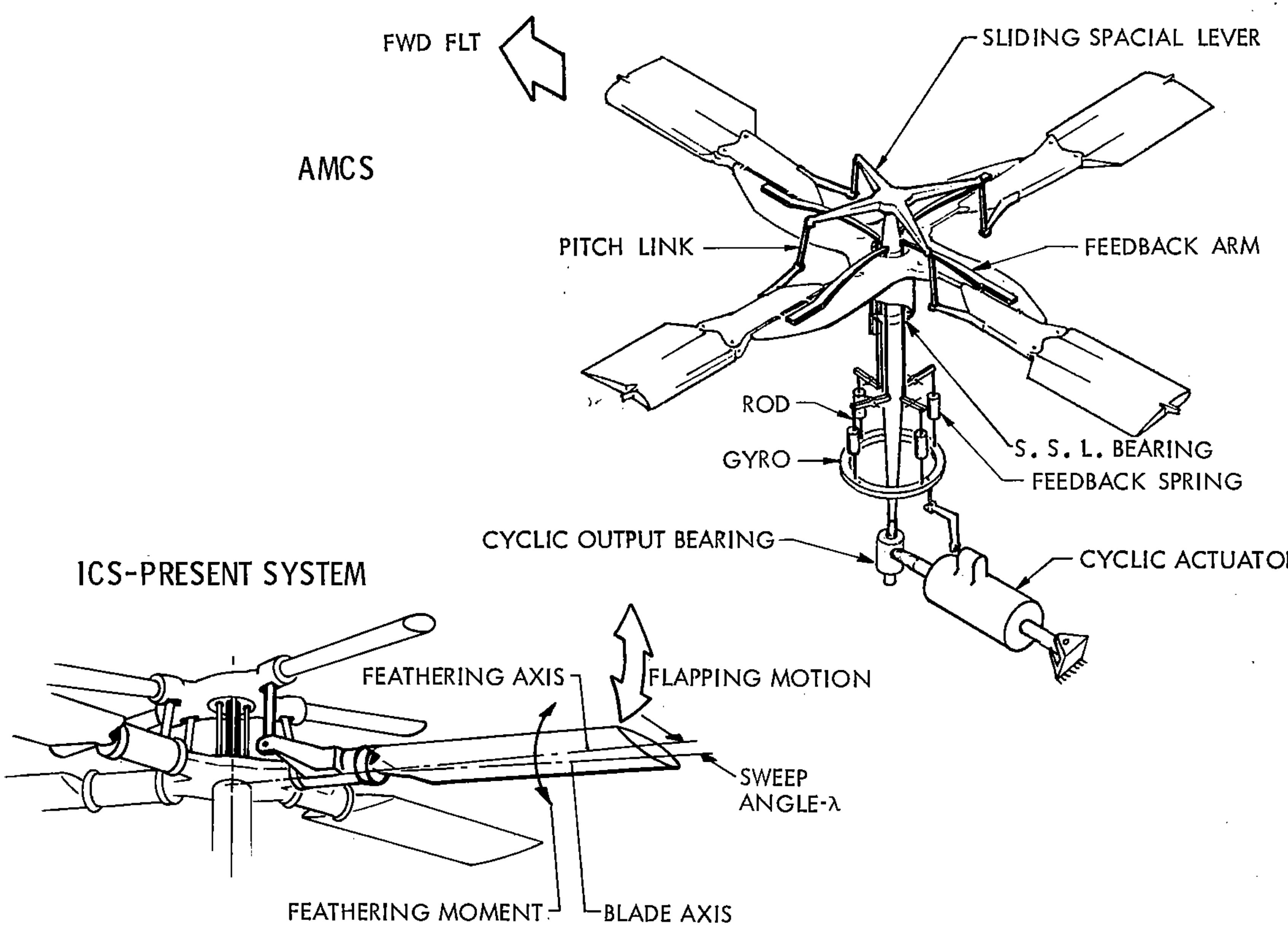


FIGURE 6

AMCS

PILOT INPUT SYSTEM

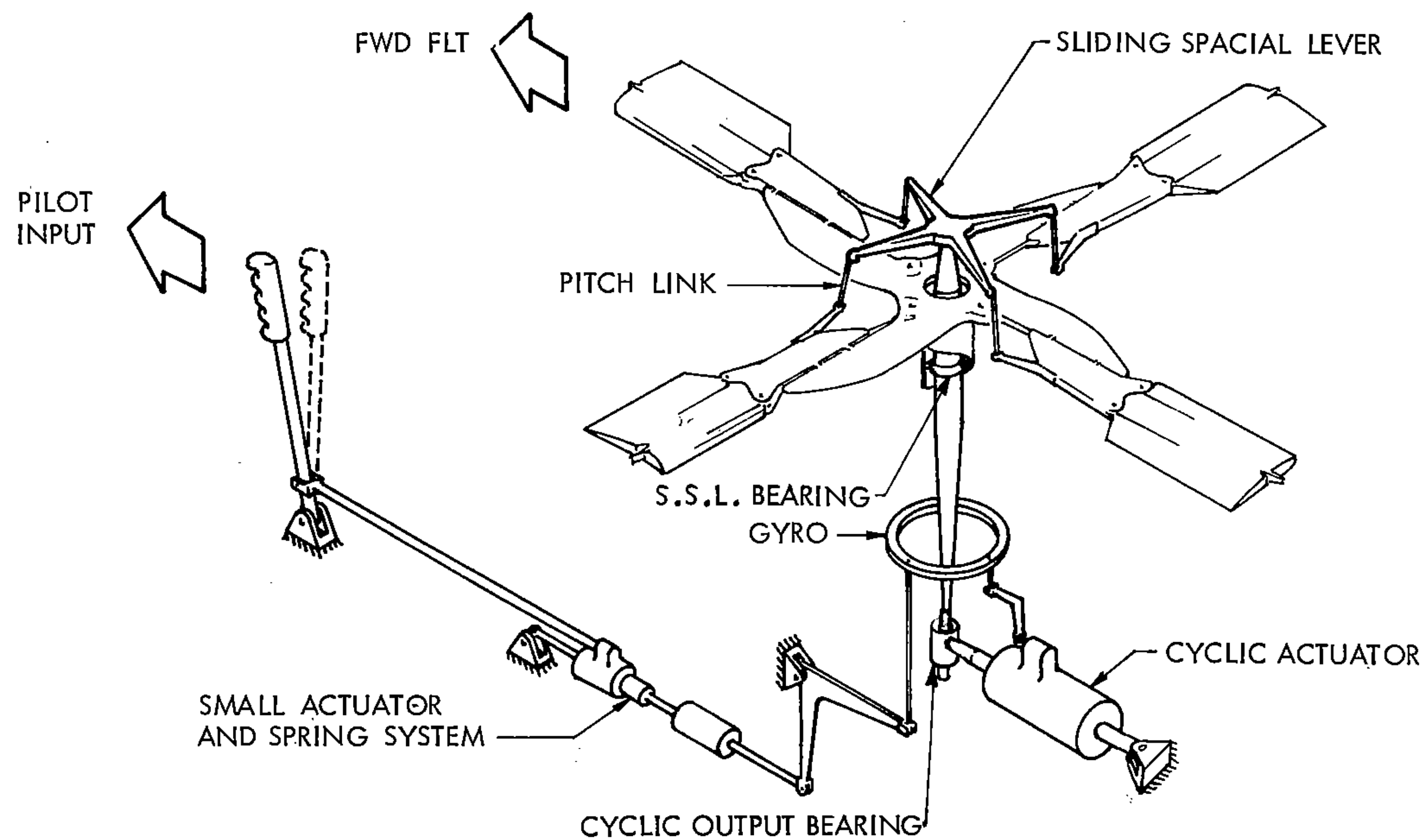


FIGURE 7

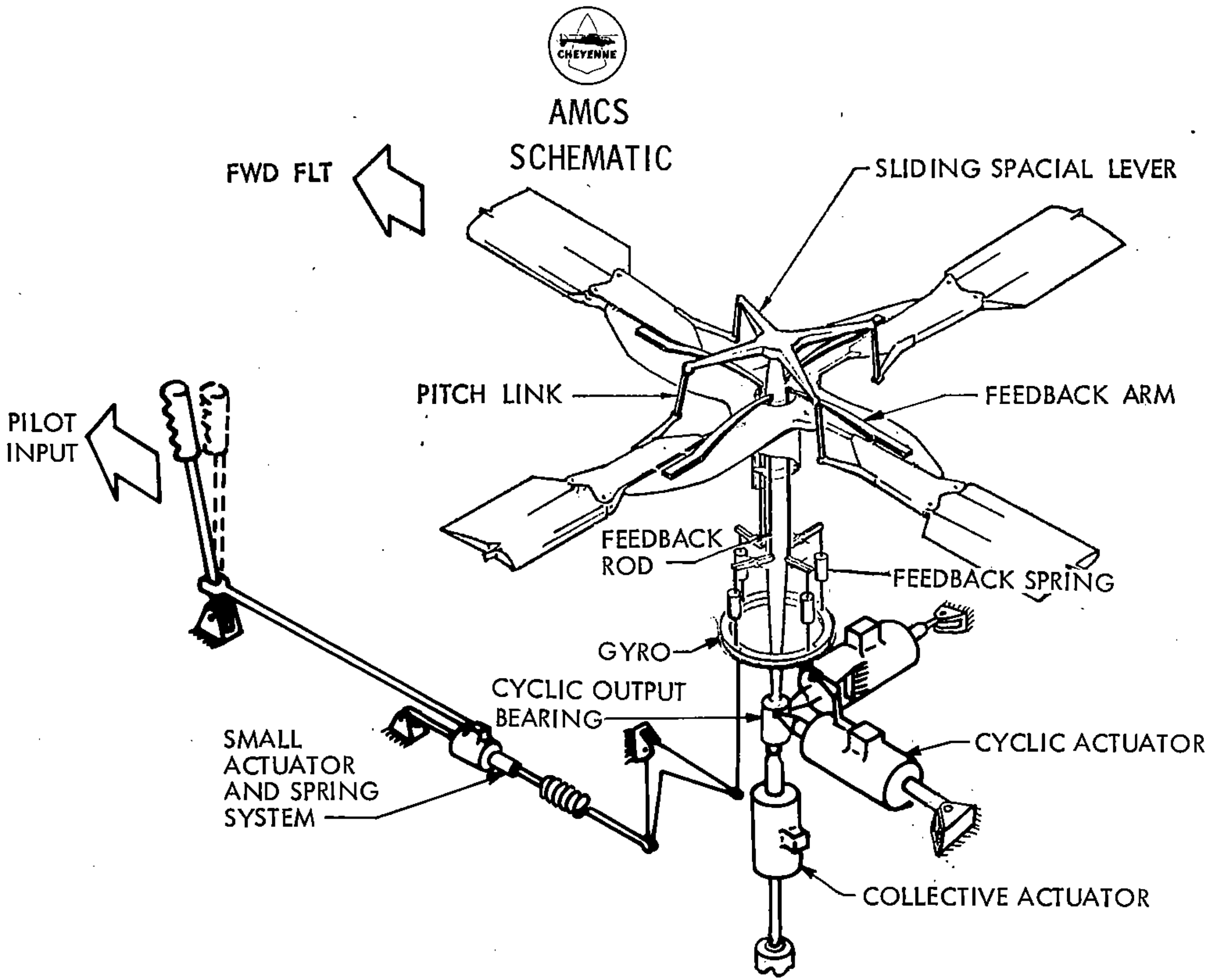

CHEYENNE
AMCS
SCHEMATIC
FWD FLT
PILOT
INPUT
SLIDING SPACIAL LEVER
PITCH LINK
FEEDBACK ARM
FEEDBACK
ROD
FEEDBACK SPRING
GYRO
CYCLIC OUTPUT
BEARING
CYCLIC ACTUATOR
SMALL
ACTUATOR
AND SPRING
SYSTEM
COLLECTIVE ACTUATOR

FIGURE 8

MAIN ROTOR ASSEMBLY

I. GENERAL DESCRIPTION

The main rotor is mounted rigidly to the aircraft. A control gyro is used to transmit control motions through pitch links, pitch arms and movable hubs to the blades.

The main rotor is relatively simple as there are no flapping hinges, lead-lag hinges, dampers and NO GREASE FITTINGS. Each of the four blades are attached to a movable hub section by two bolts. One of these bolts is a special expansion type that can be easily removed for blade folding. The movable hub is attached to an arm of the fixed hub by two fully exposed, self lubricated, replaceable hinge feathering bearings and a tension-torsion pack.

Blades are prebalanced and pretracked. A vernier adjustment device in the pitch link permits accurate indexing. The three cyclic and collective control rods are located inside the 12 inch diameter main rotor mast where they operate in an oil mist environment and are not vulnerable to battle damage. The mast is a hollow, machined, titanium forging which is an integral part of the transmission.

II. COMPONENTS AND LOCATIONS

Name of Component	Number per Aircraft	Location in Aircraft
Basic Hub (Fixed Hub)	1	Attached to top of rotor mast
Movable Hubs	4	Attached to each arm of fixed hub
Tension-Torsion Packs	4	Attached between each movable and fixed hub arm
Rotor Blades	4	Attached to movable hubs

Name of Component	Number per Aircraft	Location in Aircraft
Rotor rpm Indicators	2	Pilot and copilot/gunner's instrument panels
Rotor rpm Warning Lights	2	Pilot and copilot/gunner's instrument panels

III. MAJOR COMPONENT DESCRIPTION

A. Basic (Fixed) Hub

The hub consists of four machined titanium forgings welded together by electron beam welding. The hub is indexed to the gyro dome and mast by protruding inserts, which engage over-sized holes in the mast and gyro dome.

B. Movable Hubs

Four machined titanium movable hubs, including the pitch arms, are installed on the basic hub fixed stubs by two self-lubricating bearing hinge points. The pitch link arms are rigidly attached to the top inboard end of each movable hub by four bolts.

C. Tension-Torsion Packs

The tension-torsion packs are connected to the root end of each movable hub and the fixed hub. These units carry all centrifugal loads. The packs are fabricated of a continuous strand of steel wire looped around end eyelets. The area between eyelets is seized with wire and embedded in epoxy.

D. Rotor Blades

The rotor blades are attached to the movable hub by two attach bolts. The forward section of the all metal blades consist of a stainless steel strip extending back nine inches on both sides of the blade and a balance weight in the leading edge to provide a forward CG.

The aft blade section consists of thin sheet stainless steel skin and trailing edge. The entire blade aft of the forward cavity is filled with aluminum honeycomb for structural strength. A series of seven stainless steel doublers extending out from the blade root 102 inches are bonded to the blade. The blade is completely sealed to guard against the possibility of moisture entering the blade.

The constant chord measurement is 28 inches, rotor diameter is 51.2 feet. The pitch links are adjustable to compensate for the pretrack figure which is stenciled on the blade. Also, trailing edge tabs are provided on the blade for tracking.

The blade airfoils taper in thickness from 12 percent of the chord at the root to 6 percent at the tips to provide good high subsonic Mach number characteristics for the advancing blade at high forward speeds. Camber in the leading edge of the airfoil varies from maximum near the root of the blade to minimum camber at the tip to achieve a high maximum lift on the blade without again compromising high Mach number characteristics. A nominal 5° of twist is incorporated in the blades to improve hover performance.

E. Rotor RPM Indicators

The pilot's rotor tachometer is a dual linear type instrument combining power turbine and rotor rpm in percent, labeled N_f and N_r. The instrument has an electrical power off flag. The unit is calibrated from 0 to 65 percent for starting and 60 to 125 percent for normal operating range. The rotor rpm scale is electrically connected to a magnetic pickoff mounted on the transmission. The transmitter senses rpm from a gear on the number one hydraulic pump drive geartrain. Electrical power is supplied from the essential dc bus through a circuit breaker labeled ROTOR TACH and two circuit breakers labeled ROTOR TACH and SPEED MONITOR. The copilot/gunner has a single scale indicator for rotor rpm, which operates the same as the pilot's.

F. Rotor RPM Warning Lights

A rotor rpm warning light is located on both the pilot's and copilot/gunner's instrument panels. The lights are powered by the essential dc bus through a circuit breaker labeled FLAWS FLT SAFETY. The light will flash on and off to indicate that the main motor is below the low rpm limit (90 percent) and glow steady if the rotor exceeds its maximum rpm limit (110 percent). These signals are received from the engine/rotor speed monitor located in the forward right side of the transmission accessory compartment.

IV. SYSTEM OPERATION

A. Blade Folding

The main rotor blades may be folded manually by using a jackscrew device (GSE).

Two rotor blades are aligned longitudinally with the aircraft. The two transverse blades are folded by removing the trailing blade attaching bolt and installing the jackscrew in the lugs provided on both the movable hub and blade. By using a speed handle the jack-screw is extended, pivoting the blade about the forward blade attaching bolt until it is aligned with the preceding blade. When the rotor is folded two blades are positioned forward and two blades positioned aft, aligned parallel with the fuselage. The jackscrews remain installed to prevent the blades from rotating about the forward attaching bolt and coming in contact with foreign objects.

V. PCRS CONFIGURATION

The main rotor hub is modified to improve fatigue life and time between overhaul to 3600 and 1200 hours, respectively, as well as to improve producibility. The fixed hub feathering hinge lugs are increased in strength; the basic hub section has increased fatigue strength; and the tension-torsion pack installation is revised for increased life. The movable hub life is increased through higher-strength feathering and fold joint lugs, and revised T-T pack installation. The feathering hinge axis is relocated with respect to the hub centerline to reduce loads in the hinge joint. The feathering hub clearance is increased. The dynamic characteristics of the rotor system are unchanged.

The main rotor blade life is improved to 3600 hours by increasing the fold joint lug fatigue life. The hub fairings are reduced in size and the aerodynamic seal on the main rotor shaft is deleted. These two items cause a reduction in maximum speed of 1.2 knots.

The effective droop of the main rotor is the same as that of the baseline. However, since the fixed and movable hubs are revised as defined above, wherein the angularity of the feathering hinge line with respect to the movable hub centerline is different from baseline, the geometric droop of the blade with respect to the hinge line is altered accordingly. The sweep is the same as baseline.

TAIL ROTOR ASSEMBLY

I. GENERAL DESCRIPTION

The tail rotor is driven by the transmission, via the propeller and tail rotor drive shaft, through a right angle drive in the propeller gearbox. The rotor rotates clockwise, when viewed from the left of the aircraft. The collective pitch control consists of a rotating and non-rotating section, which move as a unit in and out along the spindle drive shaft. Pitch links connect the rotating section to pitch change arms fixed to the root of each blade. The non-rotating collective input yoke (from the servo) midpoint attaches to the stabilizer. The hub assembly attaches to the gimbal ring and contains four spindles to which the blades are attached.

The 10 foot diameter tail rotor is a four bladed, fully gimbaled, delta hinged type. It rotates at 1240 RPM and is mounted on the outboard end of the left horizontal stabilizer. The tail rotor assembly consists of a drive shaft, spindle, hub, four blades, collective control, collective input yoke, and gimbal ring.

II. COMPONENTS AND LOCATIONS

Name of Component	Number per Aircraft	Location in Aircraft
Tail Rotor Drive Shaft	1	Aft area of left horizontal stabilizer
Collective Control	1	Left horizontal stabilizer
Collective Input Yoke	1	Left horizontal stabilizer
Gimbal Ring	1	Left horizontal stabilizer
Rotor Blades	4	Left horizontal stabilizer
Spindle	1	Left horizontal stabilizer
Hub	1	Left horizontal stabilizer
Tail Rotor Blades	4	Left horizontal stabilizer

III. MAJOR COMPONENT DESCRIPTION

A. Tail Rotor Drive Shaft

The tail rotor drive shaft is 38.355 inches long and drives the tail rotor from the propeller gearbox right angle drive output at 1240 rpm. The shaft connects to the gearbox drive with a crown spline, an oil dam type seal, and retainer ring. The tail rotor end of the shaft is connected to the drive adapter by a curvic coupling and clamp.

B. Collective Control

Collective control is a sleeve assembly made up of rotating and non-rotating sections. The two sections move as a unit, in and out, along the drive shaft. Pitch links connect the rotating section to pitch arms on the root of each rotor blade. An oil reservoir, with sight gage, bolts to the non-rotating section and provides oil to lubricate all bearings in the drive shaft and collective control assembly. All items requiring lubrication are in the oil reservoir cavity. The reservoir is serviced with approximately one quart of MIL-L-7808 or MIL-L-23699 oil.

C. Collective Input Yoke

The collective input yoke attaches to the stabilizer. One end of the yoke is connected by a pushrod to a dual hydraulic servo, while the other end attaches to the non-rotating section of the collective control.

D. Spindle Mechanism

The spindle assembly provides a method of supporting the tail rotor drive mechanism. It also contains the pitch control, which transfers linear motion through the collective rotating mechanism. The components within the assembly are lubricated by centrifugal force with oil supplied from an external tank. The oil tank is mounted on the non-rotating portion of the collective assembly and has a integral passage for return oil. Oil is supplied by gravity feed, to the spindle through a hose connected to a port on the inboard portion of the spindle.

E. Hub Assembly

The hub assembly attaches to the gimbal ring and has four stubs. A spindle and sleeve assembly consisting of two feathering bearings, and a tension-torsion pack, with provisions for blade installation, is mounted on each stub. The spindle and sleeve assembly is retained to the hub stub by the tension-torsion pack. The blades are bolted to the sleeves. Each spindle and sleeve assembly contains an oil reservoir, to lubricate the feathering bearings.

F. Tail Rotor Blades

The tail rotor blades are aluminum honeycomb bonded to an "H" shaped titanium spar and covered with titanium. The leading and trailing edge cavities are filled with a foam plastic material. All blades are interchangeable.

IV. SYSTEM OPERATION

The tail rotor drive shaft rotates at a constant 1240 rpm at 100 percent N_f, driving the spindle which carries the gimbal ring, hub assembly, and blades.

In order to change rotor blade pitch angle, the hydraulic control servo actuating cylinder moves the collective input arm. Movement of the input arm carries the collective control assembly in or out, along the drive shaft. When the collective control is moved, it actuates the pitch links, which rotate the blades about their feathering axis.

V. PCRS CONFIGURATION

The aerodynamic fairing on the hub is deleted.

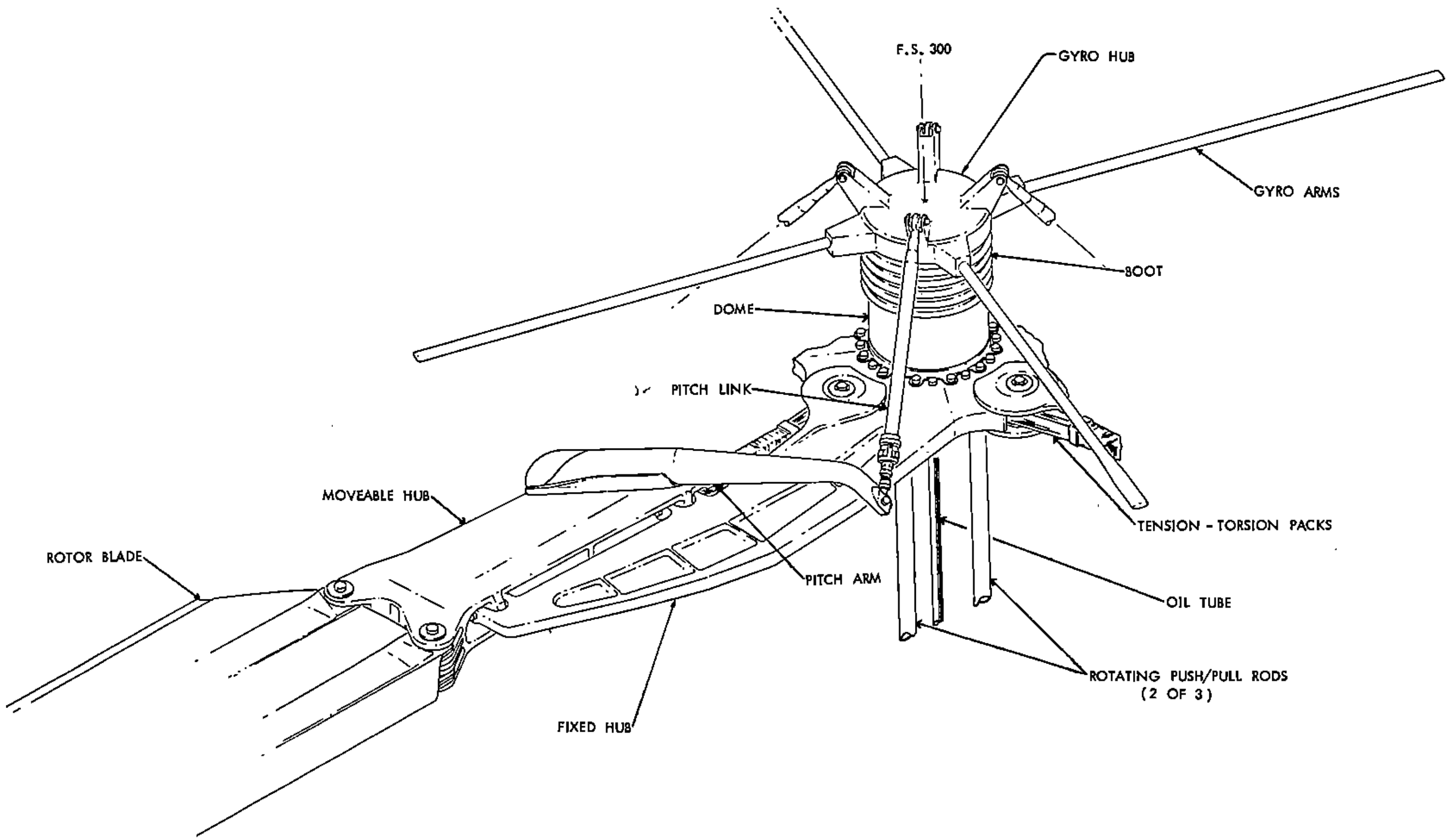


Figure 8A-1. Rotor/Gyro Diagram

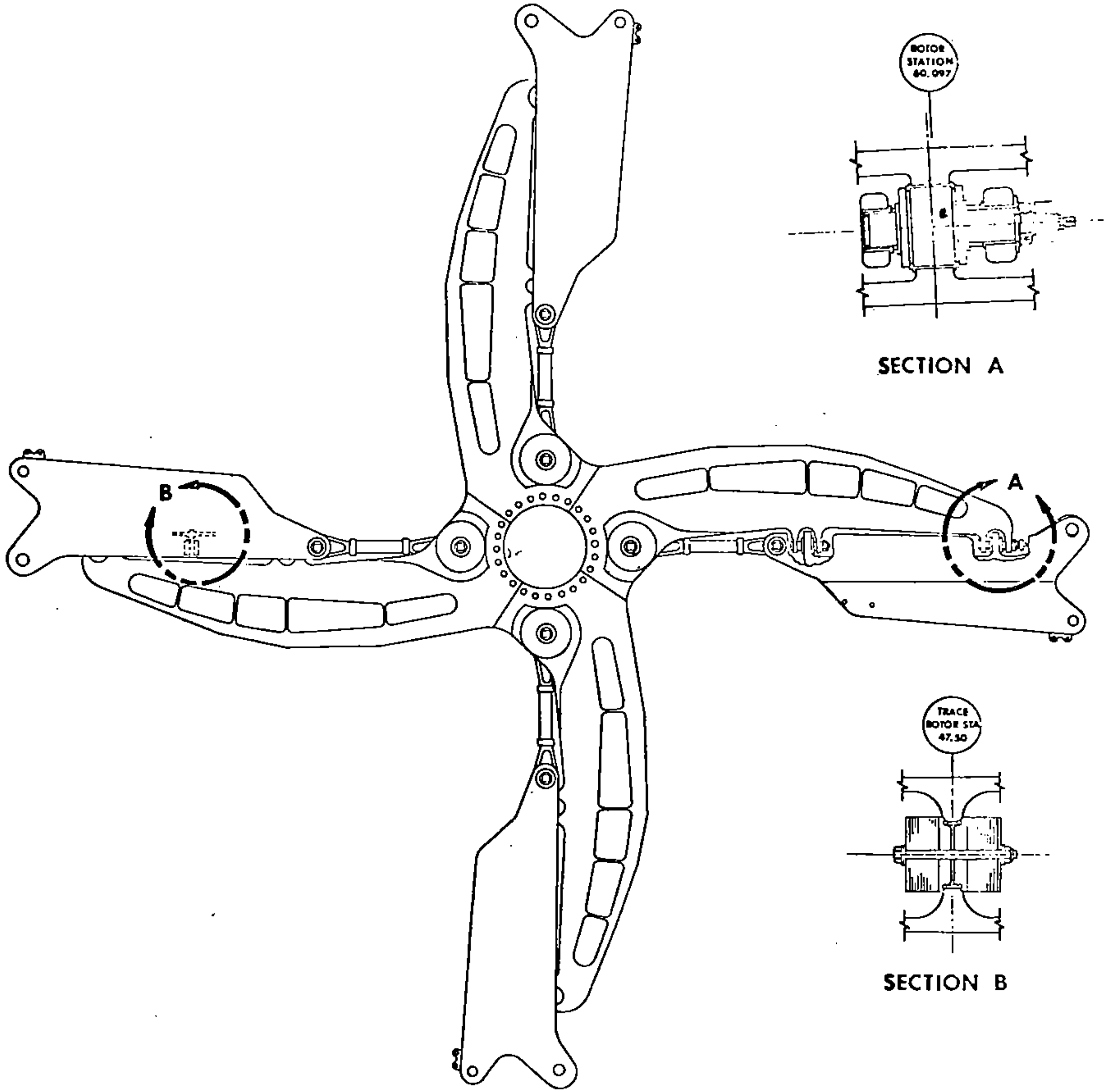


Figure 8A-2. Main Rotor Hub Assembly

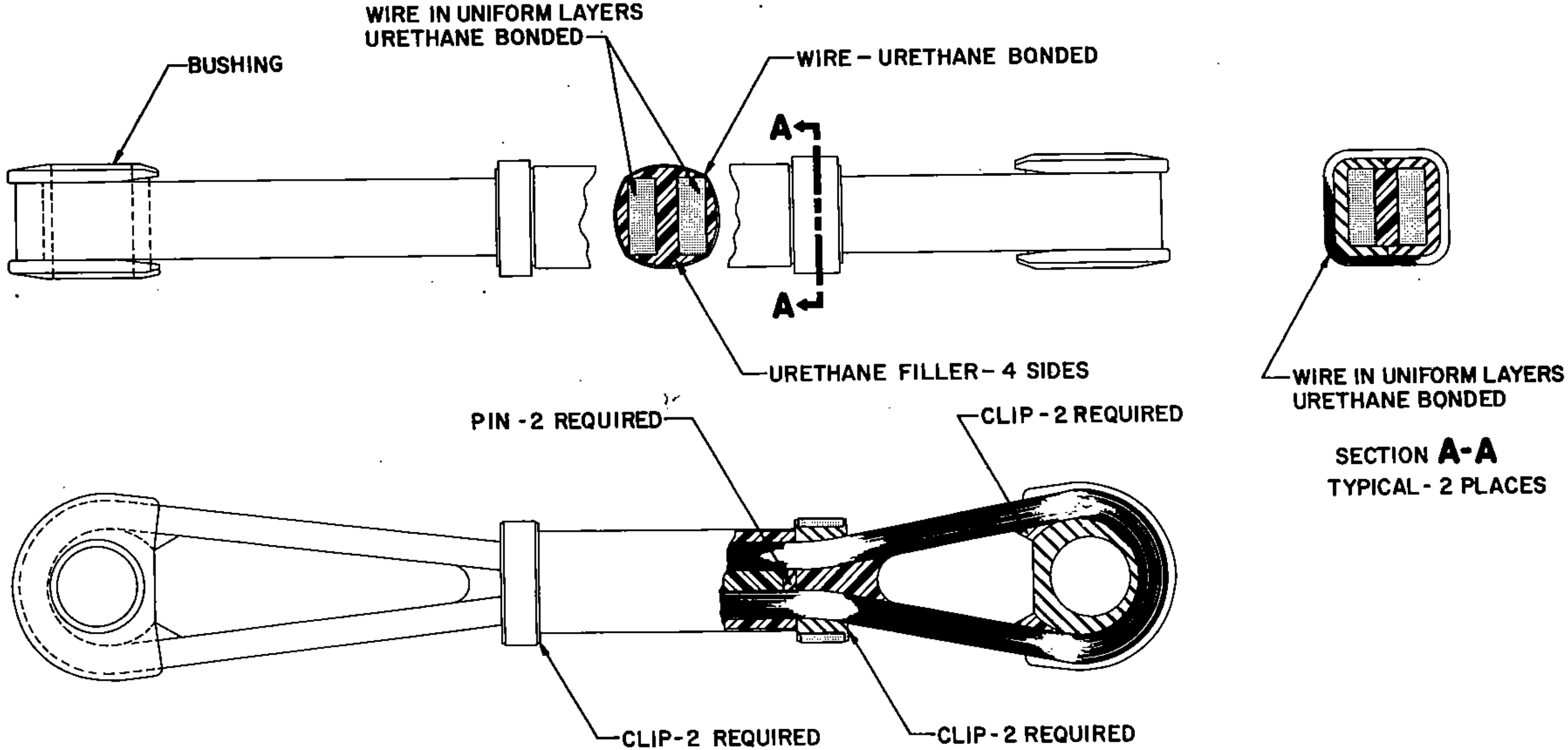


Figure 8A-3. Main Rotor Tension-Torsion Pack

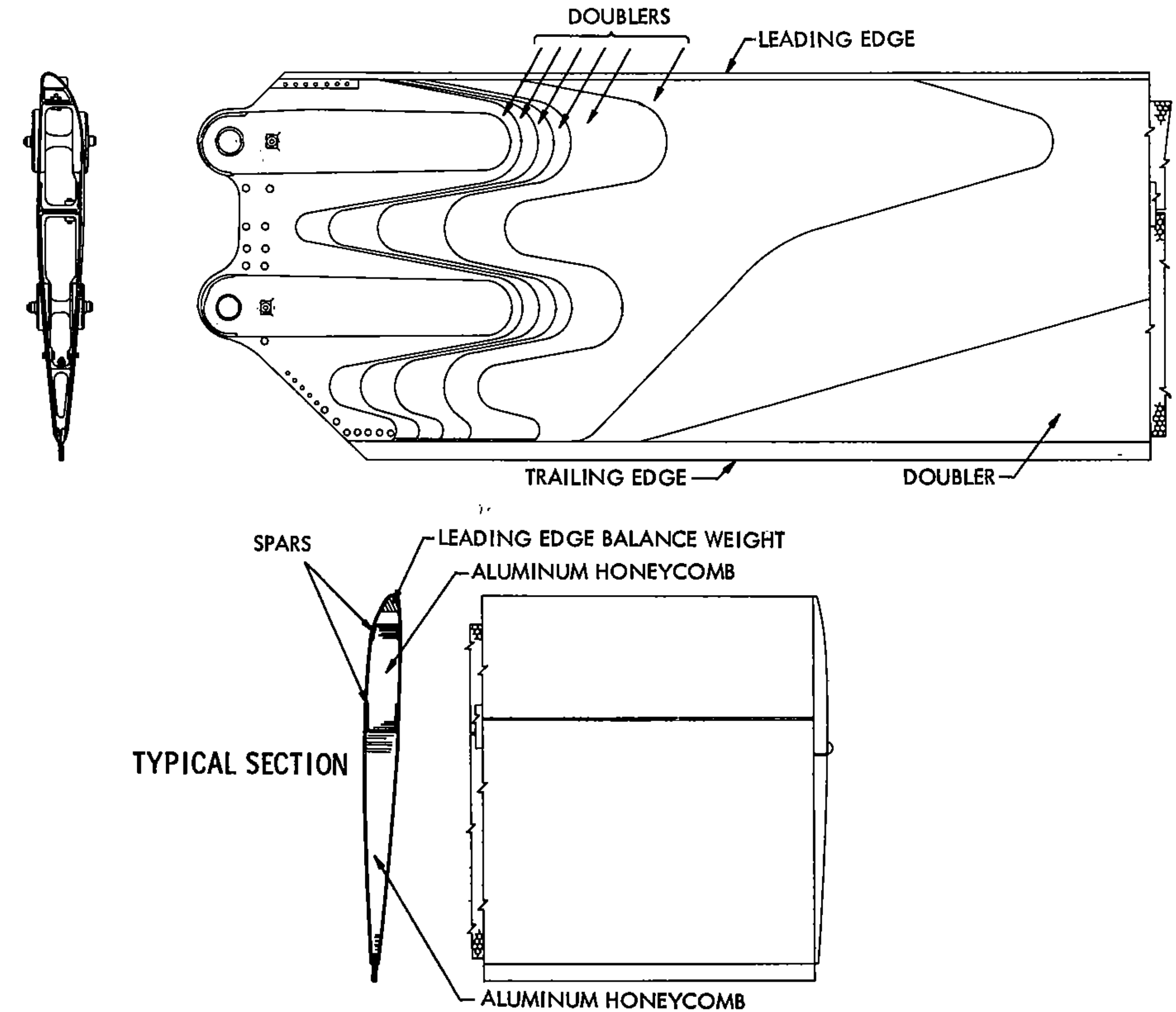
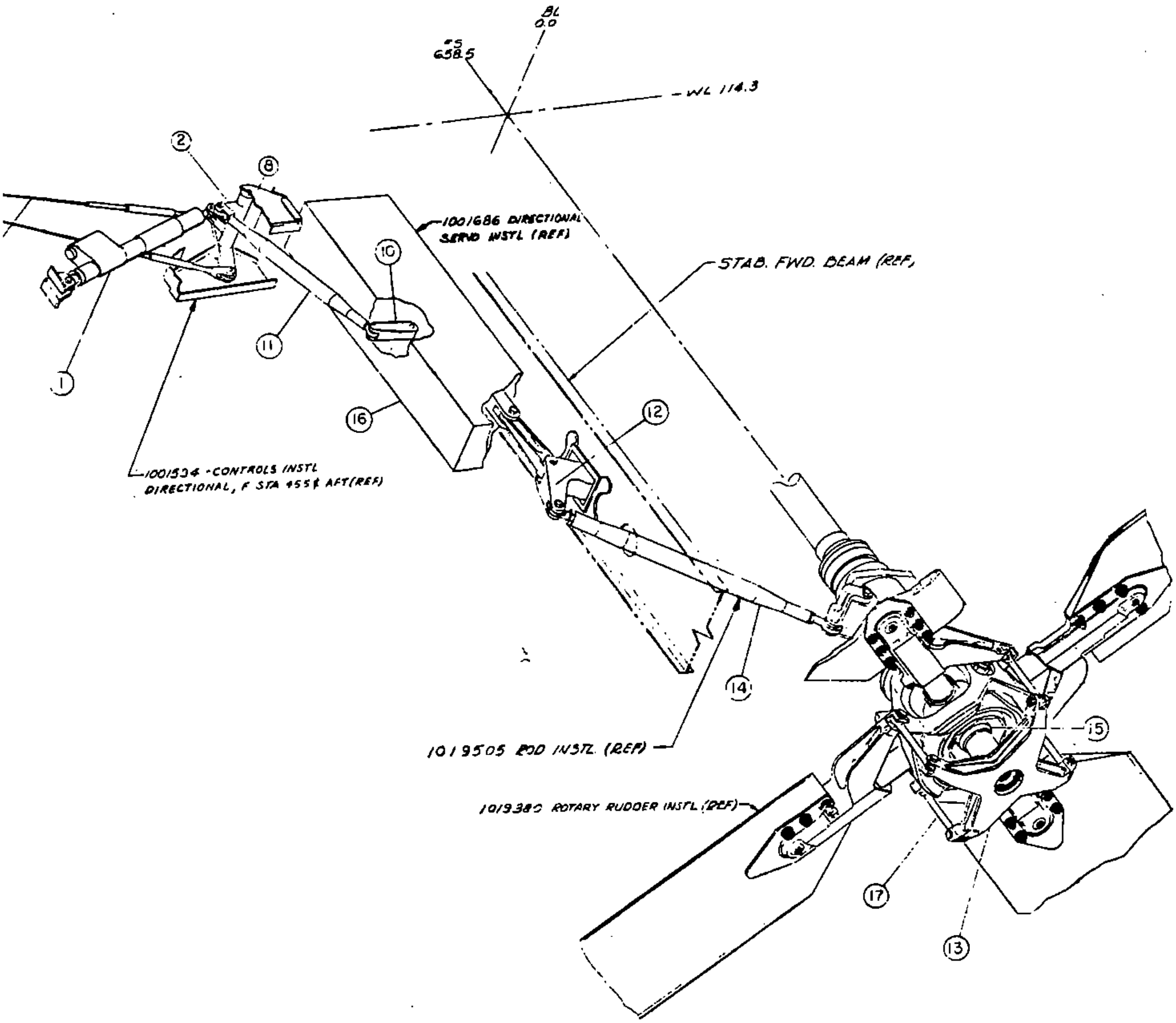


Figure 8A-4. Main Rotor Blade Cutaway



BL
0.0
FS
658.5
WL 114.3
2
8
1001686 DIRECTIONAL
SERVO INSTL (REF)
10
STAB. FWD. BEAM (REF)
11
1
16
12
1001534 - CONTROLS INSTL
DIRECTIONAL, F STA 455¢ AFT (REF)
14
15
1019505 ROD INSTL (REF)
1019380 ROTARY RUDDER INSTL (REF)
17
13

Figure 8A-5

PROPELLER

I. GENERAL DESCRIPTION

The Hamilton Standard 1311GC48-2 propeller is a three way integral gearbox pusher propeller designed for the AH56A compound helicopter. The propeller gearbox mounts on a structural bulkhead at the aft end of the fuselage. It receives its power from the engine power turbine through the engine shaft, the propeller drive section of the main transmission and a shaft extending aft to the propeller gearbox. The gearbox provides direct drive to the pusher propeller and a right angle reduction drive for the aircraft anti-torque rotor. With the engine power turbine speed at 100 percent (13,600 RPM), the propeller gearbox rotates the propeller counterclockwise (as viewed from aft of aircraft) at 1717 RPM (no reduction) and the tail rotor clockwise (top blade aft and bottom blade forward) at 1238 RPM. The power distribution to the propeller and tail rotor is:

Power input	3880 HP at 1717 RPM
Power to propeller	3830 HP at 1717 RPM
Power to tail rotor	940 HP at 1238 RPM Maximum
	1200 HP at 1238 RPM Transient

The propeller is a hydraulically actuated controllable pitch, nongoverning type with a diameter of ten feet. Blade angle control is maintained by a cockpit twist grip which is mechanically linked to the actuator input mechanism in the gearbox. The pilot can change blade angle independently of the engine power setting and has the capability to select the optimum thrust for various flight conditions. Negative thrust can be used for inflight deceleration and hovering in a tail wind.

The Hamilton Standard Model 1311GC48-2 propeller, consists of several major subassemblies, namely:

a. Hub and gearbox assembly

b. Pitch change actuator assembly

c. Propeller blades

II. COMPONENTS AND LOCATIONS

Name of Component	Number per Aircraft	Location in Aircraft
Propeller Drive Shaft	1 shaft (3 sections)	Connecting transmission to propeller gearbox
Hub and Gearbox Assembly	1	Mounted on the aft end of the tail boom
Pitch Change Actuator Assy	1	Component of gearbox assembly
Propeller Blades	3	Connect to gearbox assy.

III. MAJOR COMPONENT DESCRIPTION

A. Propeller Drive Shaft

The propeller drive shaft transmits 3880 HP at 13,600 RPM from the main transmission to the propeller gearbox. The shaft installation consists of three interchangeable shaft sections supported by two intermediate bearings and a forward and aft coupling. Power is transmitted to the shafting through the adapter mounted on the propeller gear shaft in the main transmission. Each of the shaft sections have a flange on each end with curvic coupling teeth to transmit torque. Fins are located every five inches along the tubes to prevent torsional buckling providing a lightweight design. The aft flange of each shaft length is extended axially to provide a seat for the shaft support bearing. These shafts are held to close straightness tolerances and are dynamically balanced to insure smooth operation.

The aft shaft coupling mounts of the aft end of the last shaft section and has a nitrided, silver plated crowned spline which drives the propeller gearbox. Oil from the propeller gearbox is used to lubricate the crowned spline. A lip seal is provided to prevent oil leakage from the propeller gearbox. An O-ring dam is also provided to insure that the spline is always submerged in oil.

The three shaft lengths are supported at each of their junction points by a grease lubricated, thin section, ball bearing. The bearing inner race mounts directly on the O.D. of the shaft flange. A two part hanger supports the outer race. The hanger is joined to the fuselage structure in two places by self aligning, ball type, mounts, having high radial spring rates. The mounts will align readily and can slide axially on bronze bushings. The self aligning feature compensates for fuselage motions and the axial motion takes care of differential expansion due to temperature changes. A grease fitting is provided for the bearings. These fittings are accessible from outside the fuselage. Labyrinth seals with ample grease capacity insure adequate lubrication between regularly scheduled servicings.

B. Hub and Gearbox Assembly

The gearbox housing supports the propeller shaft and contains the reduction gears for the antitorque rotor drive. It also contains the blade angle control input mechanism and provides mounting lugs for the airframe cowling supports. Lubrication for the primary gear mesh is provided by a dual element gear type pump and is self contained using MIL-L-7808 oil. The scavenge element maintains a dry sump by drawing oil past an electric chip detector, through an inlet screen and discharges the oil over a temperature limit indicator switch into the main sump. The main pump element draws oil from the sump through an inlet screen and discharges the oil through a 40-micron wire mesh filter and by a pressure relief valve and a low pressure indicator switch. Filtered oil is directed onto the input pinion thrust bearing and through a manifold to the primary gear mesh and the other major bearings. Oil from the manifold also passes through a transfer bearing into the barrel tailshaft where its flow is divided. Part of the oil is used to lubricate the actuator input system gearing while the remainder is directed forward to cool the input pinion spline. All heat is dissipated through the gearbox housing by ram air from an air scoop located in the bottom of the propeller gearbox cowling.

C. Pitch Change Actuator Assembly

The pitch change actuator provides blade angle control throughout the operating range. An input signal from the cockpit is received as a rotary motion at the input mechanism and is transmitted through a differential gear train to a concentric cylinder cam system which converts the signal to an axial translation. This axial displacement positions a distributor valve which controls the transmission of hydraulic pressure to the actuator cylinder, producing a change in blade angle. Movement of the actuator nulls the distributor valve to maintain the new blade angle setting. The scavenge element draws drainage oil from the propeller barrel through an inlet screen discharging into the main element sump. The pump main element draws oil from its sump through an inlet screen and discharges into a 40 micron wire mesh filter. Oil leaving the filter passes the high pressure relief valve and the manual pressure relief valve enroute to the distributor valve. The distributor valve is a negative overlap concentric spool valve, with the internal valve moving in response to system input and the outer spool moving with the pitch change cylinder thus nulling the input signal. The distributor valve is connected to the low and high pitch chambers of the cylinder with excess flow capacity returning to the sump by means of the negative overlap.

The propeller is equipped with three centrifugal stops (one for each blade). These units consist of a lever supported on the outside of the barrel. A connecting link transmits the counterweight force to a plunger bearing on the pitch change cylinder, when the propeller seeks negative pitch, potential energy if position is stored in the counterweights. This energy is available to return the blades to flat pitch in the event that hydraulic pressure is lost.

D. Propeller Blades

The propeller blades contain a formed hollow steel spar, nickel plated on its exterior for corrosion protection. A fiberglass shell is bonded to the spar and forms the blade airfoil. The leading edge, spar and trailing edge cavities are filled with a foamed in place polyether isocyanate foam. Fiberglass is laminated to the butt end of the cuff portion of the blades to seal exposed foam areas. The spar cavity is closed with preformed fiberglass seals cemented to the inside diameter of the spar. The outboard half of the blade leading edge has an electroformed nickel erosion strip bonded to the fiberglass shell and the entire blade is coated with a polyurethane anti-erosion and anti-static material. Lightning protectors connect blade tip to spar tip providing a discharge path through the spar to the blade shank. The blade retention is a two race ball type with raceways integrally formed with the blade butt. This retention system has been employed successfully by Hamilton Standard for previous models of fiberglass blades and thousands of steel blades.

IV. SYSTEM OPERATION

A. Propeller Delta Beta System

The function of this system is to reduce the pitch of the propeller blades to a safe level whenever the transmission experiences a negative torque condition.

The system consists of a double acting hydraulic cylinder controlled by a mechanically operated valve and an electrically operated valve. The valves are in series and actuation of both is required to actuate the cylinder. The mechanical valve is mounted on the transmission and is actuated by a sensor lever that is connected to a floating ring gear within the transmission to sense a preset value of negative rotor shaft torque. The electrically operated valve is normally energized by an electrical input when the vehicle is in a condition of less than 1.35 "g's."

In normal operation pressure is ported to both sides of the cylinder piston thereby maintaining it in the extended position.

Change of propeller blade angle is accomplished when reversal of transmitted torque, due to engine failure is sensed by the mechanical valve, actuated by the lever in the transmission, and the electrical valve is deenergized due to flight conditions of less than 1.35 g's and engine shaft speed of less than 97 percent N_R. This, in turn, dumps pressure from the extend port of the cylinder causing the piston to retract. The piston is mechanically connected to the propeller cable control system and when retracted moves the cable system to a position commensurate with a -9.2 degree or +18 degree blade angle if the propeller had been operating at a higher negative or positive blade angle.

B. Failure Detection System

Propeller Gearbox. The propeller gearbox does not provide for indicating the oil temperature and pressure, however, switches are provided to activate a caution light for high temperature and low oil pressure.

Propeller Oil Caution Light. A caution light, labeled PROP OIL, is located on the annunciator panel in each crew station. Both lights come on to provide visual caution signals when propeller gearbox oil is above the maximum temperature limit or below the minimum pressure limit. At the same time, a voice warning message is supplied to the crew and heard in their headsets.

Chip Detector Caution Light. A chip detector is installed in the propeller gearbox. When sufficient chips collect on the detector, an electrical circuit is completed to the FLAWS to cause a CHIPS caution light to illuminate and a voice warning to be heard in the crew headsets.

V. PCRS CONFIGURATION

No change.

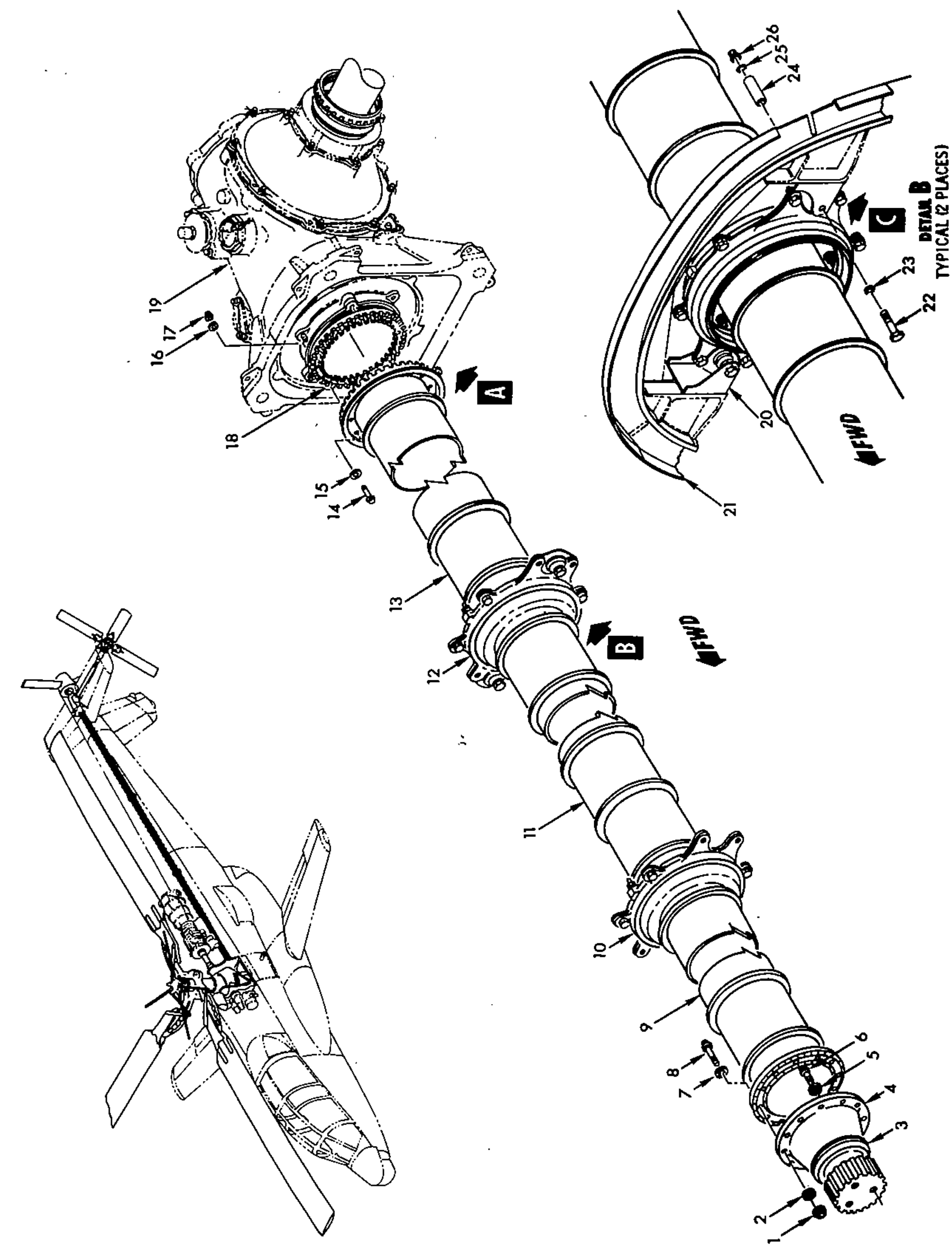


Figure 8B-1 Propeller Drive Shaft

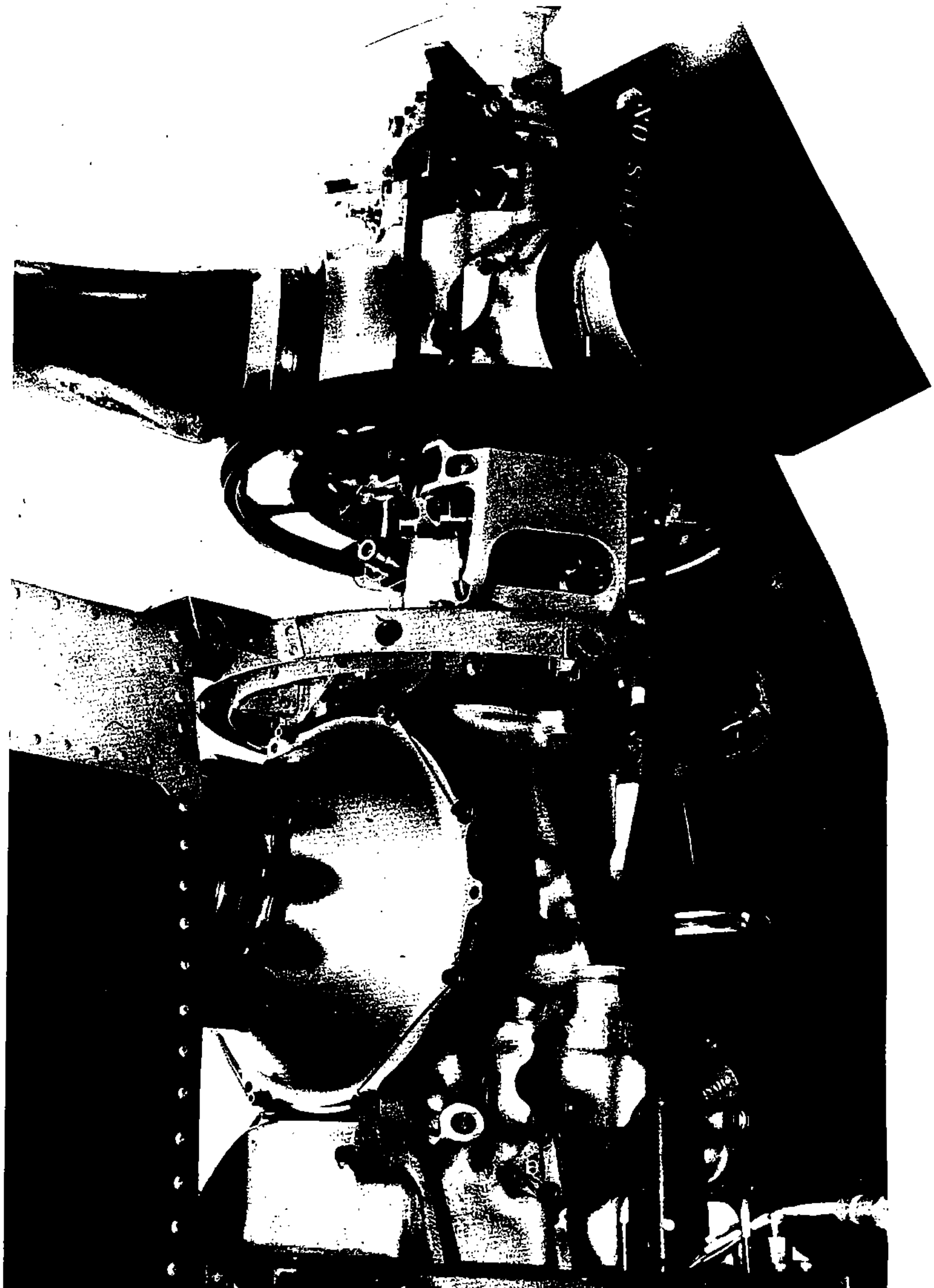


Figure 8B-2 Propeller Gearbox Installation

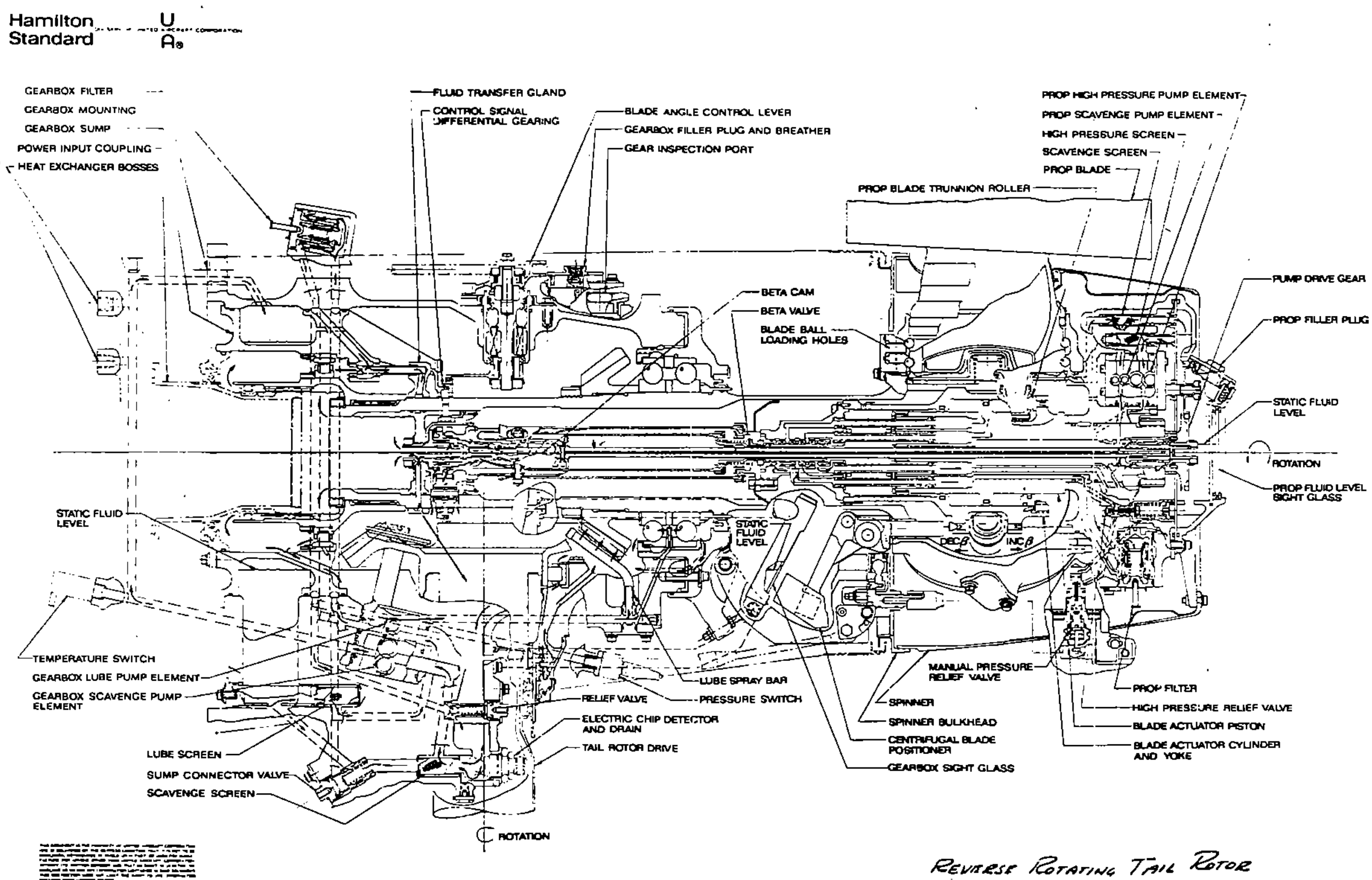


Figure 8B-3 1311GC48 Propeller (AH-56A)
Hydraulic-Mechanical Schematic

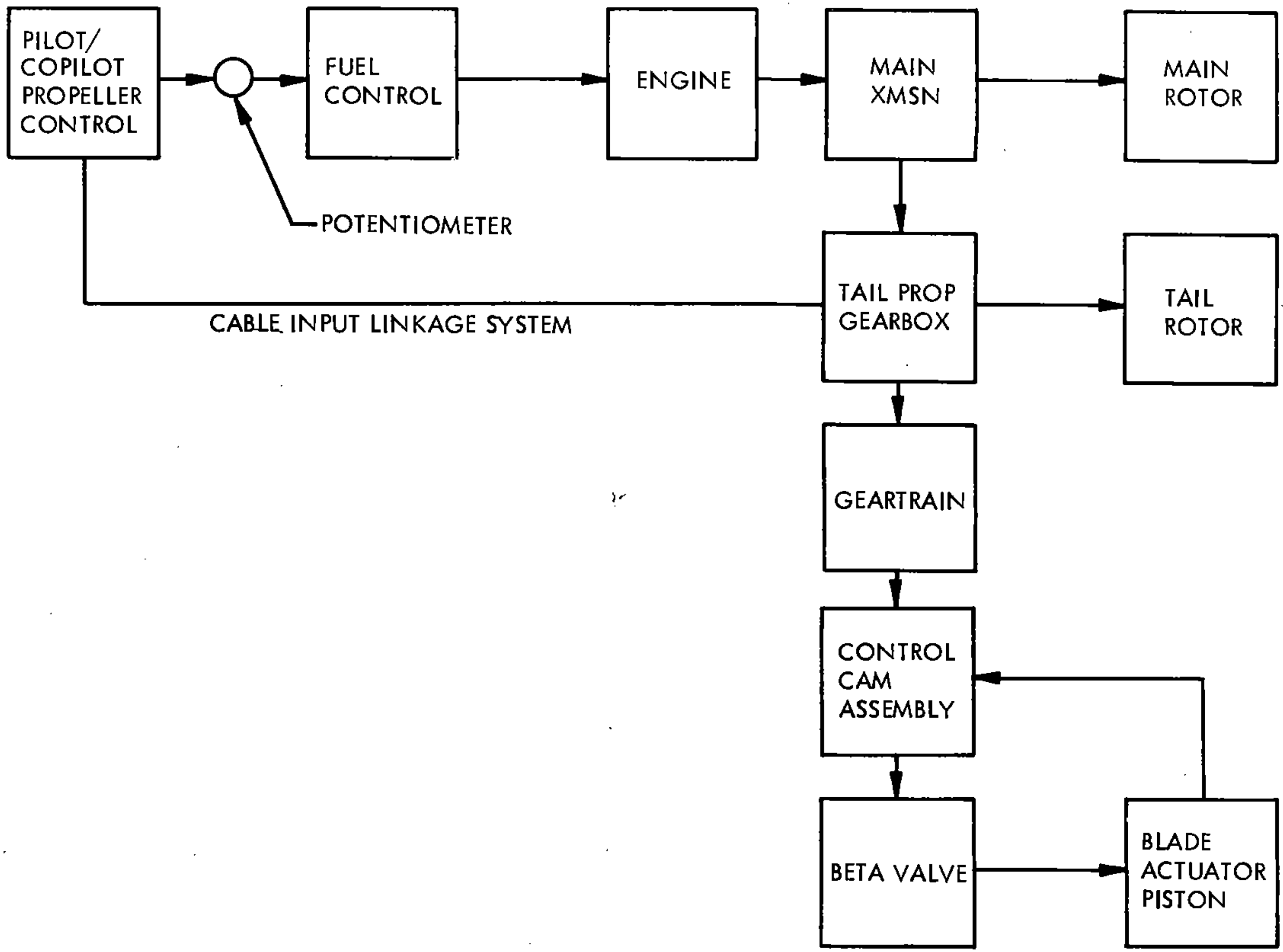


Figure 8B-4 Propeller Control Block Diagram

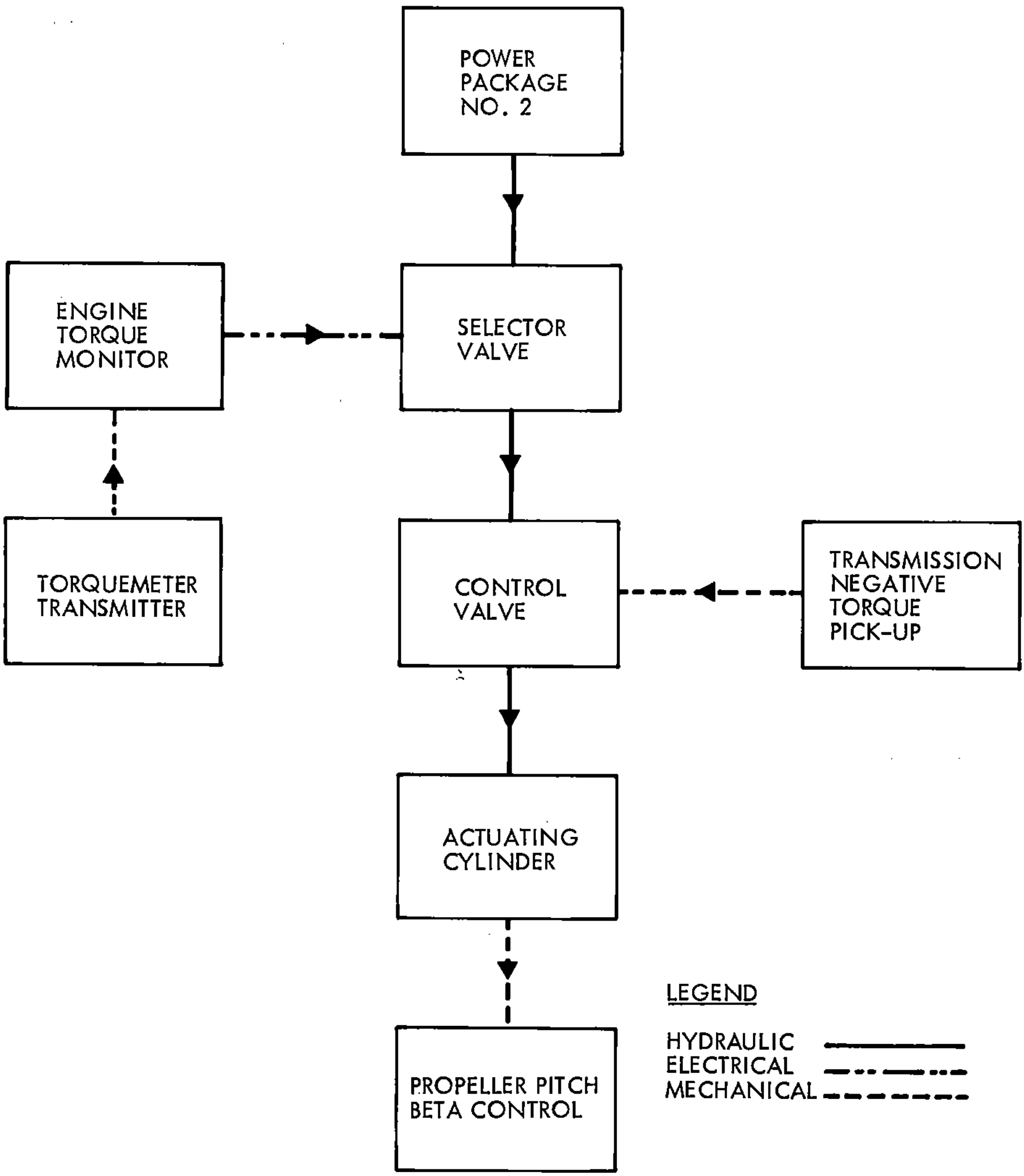


Figure 8B-5 Negative Torque System Block Diagram

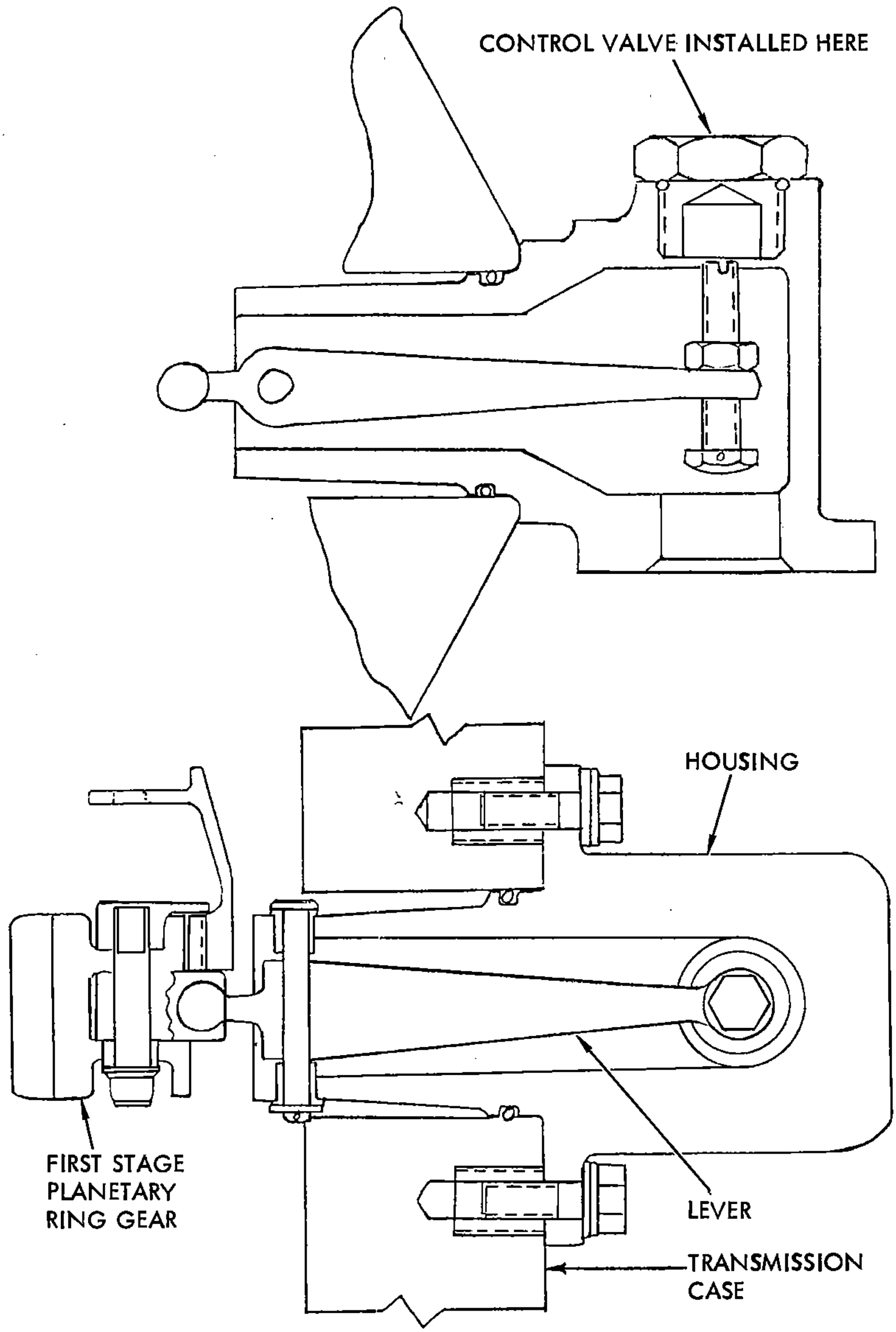


Figure 8B-6 Negative Torque Pick-Up

FLIGHT STABILITY

The AH-56A Cheyenne Compound Helicopter with its gyro-controlled rigid rotor system, provides a weapons platform with excellent handling characteristics. These characteristics are achieved as a result of the combination of the rigid rotor, wing, and propeller in a true compound helicopter configuration. This compounding gives the AH-56A handling characteristics which can not be duplicated by a conventional helicopter, winged helicopter or fixed wing aircraft.

The major advantages of compounding are:

- Pitch attitude selection at all flight speeds
- Fly similar to an airplane at high speed, like a helicopter at low speed
- Deceleration capability in a dive
- Consistantly high sustained load factor capability over a broad range of flight speeds

The major advantages of the gyro-controlled rigid rotor are:

- Rate cues vs. acceleration cues for pilot handling
- High roll and pitch damping
- Very positive short period stability
- Low control sensitivity to gross weight variation
- Low level and frequency of control activity required
- Positive pitch and roll control characteristics at all speeds and loading conditions with rotor cyclic controls only

Other advantages:

- Starts and stops in winds
- Rotor-fuselage clearance
- Complete ground taxi capability with full weight on wheels w/o rotor shaft moments
- Nose down response to power failure

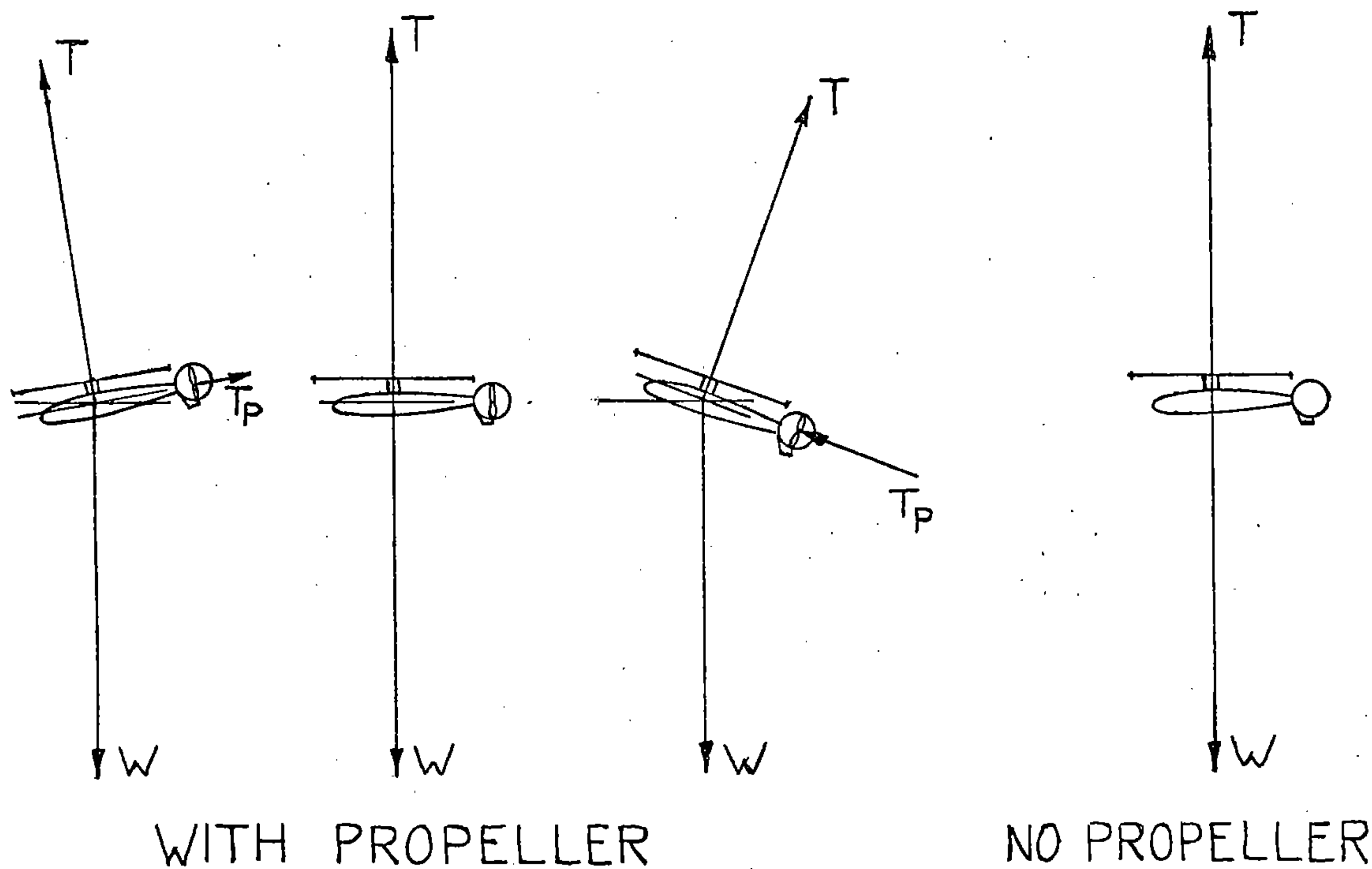


Figure 9-1. Attitude Control Hover

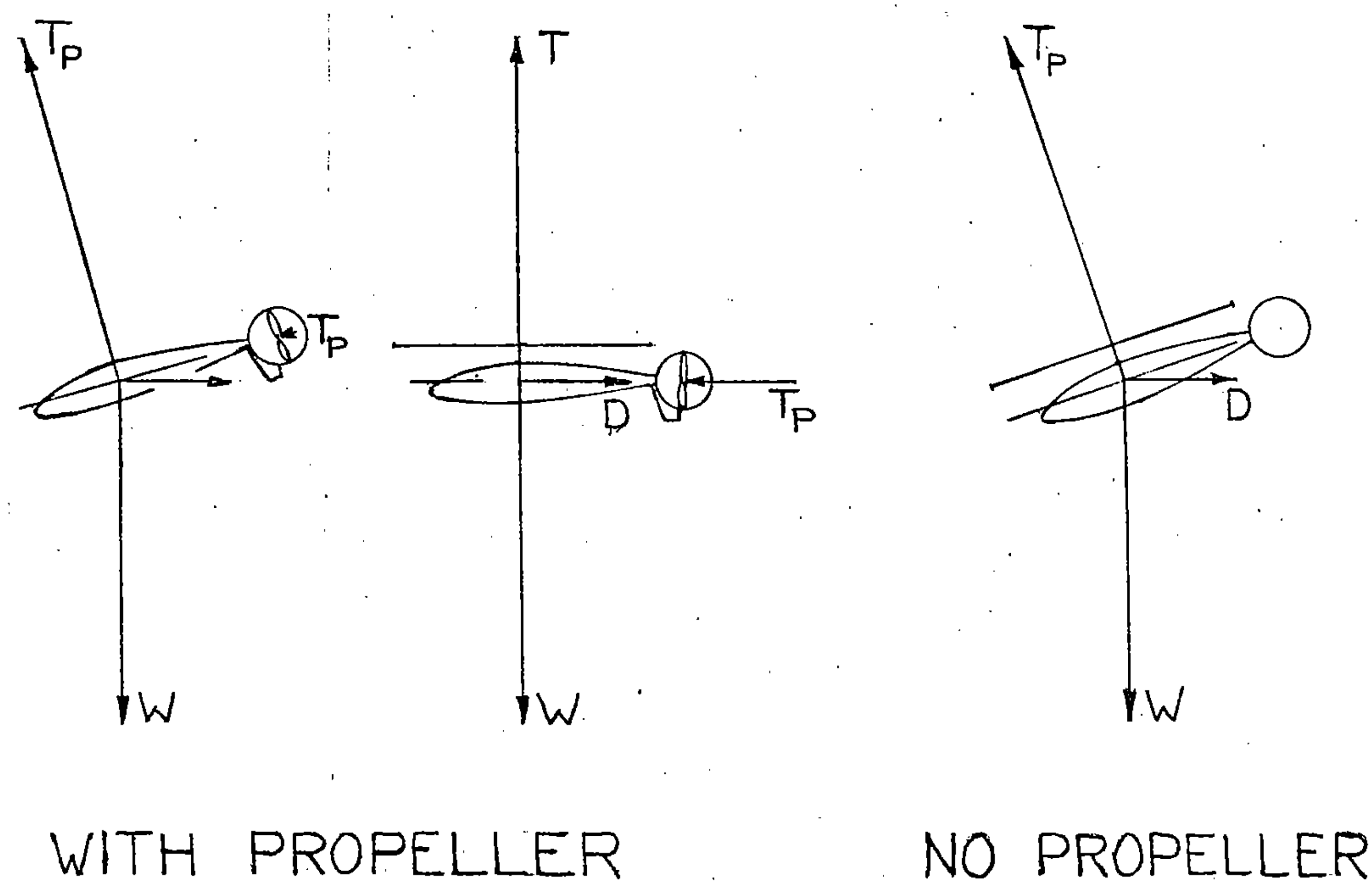


Figure 9-2. Attitude Control High Speed

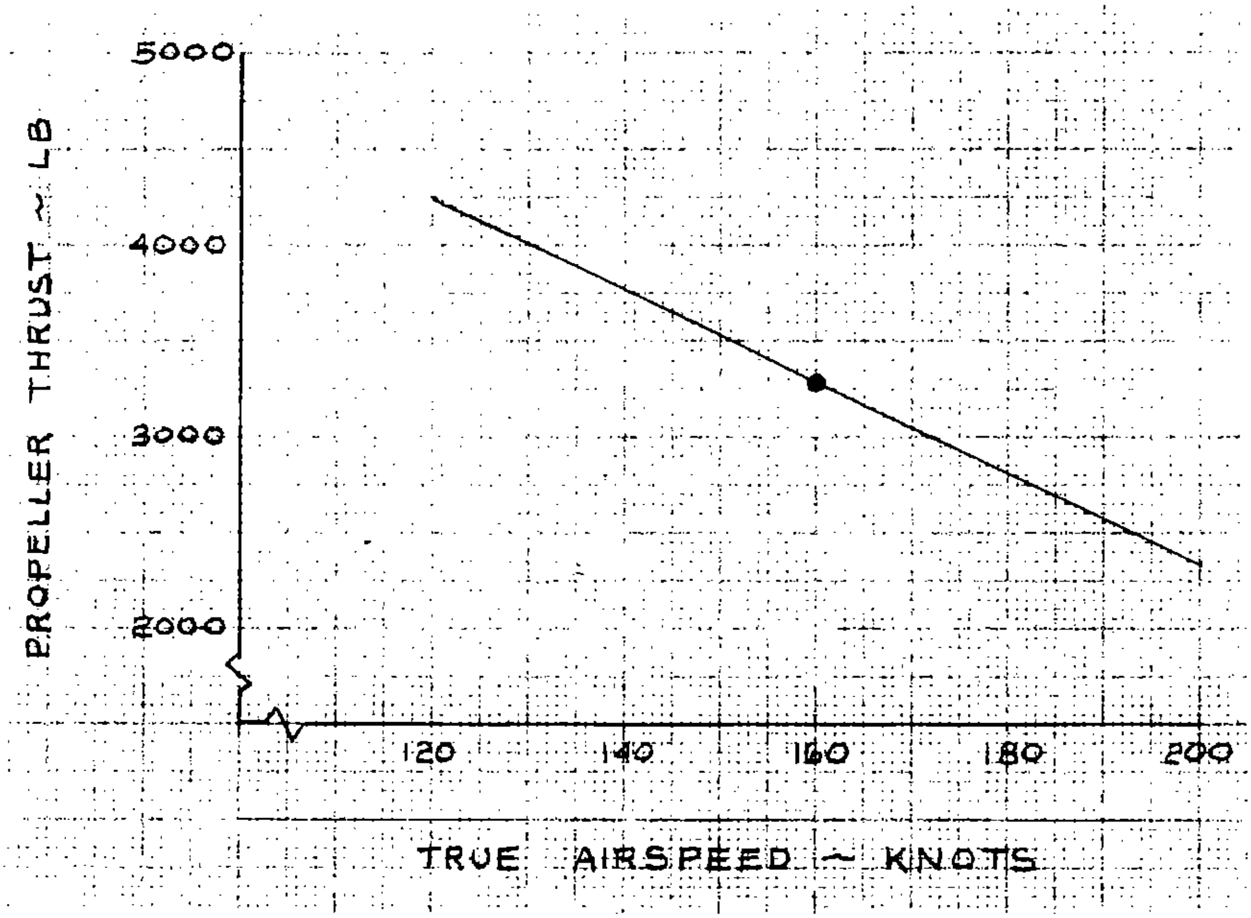


Figure 9-3. Speed Stability Propeller Thrust Variation with Velocity

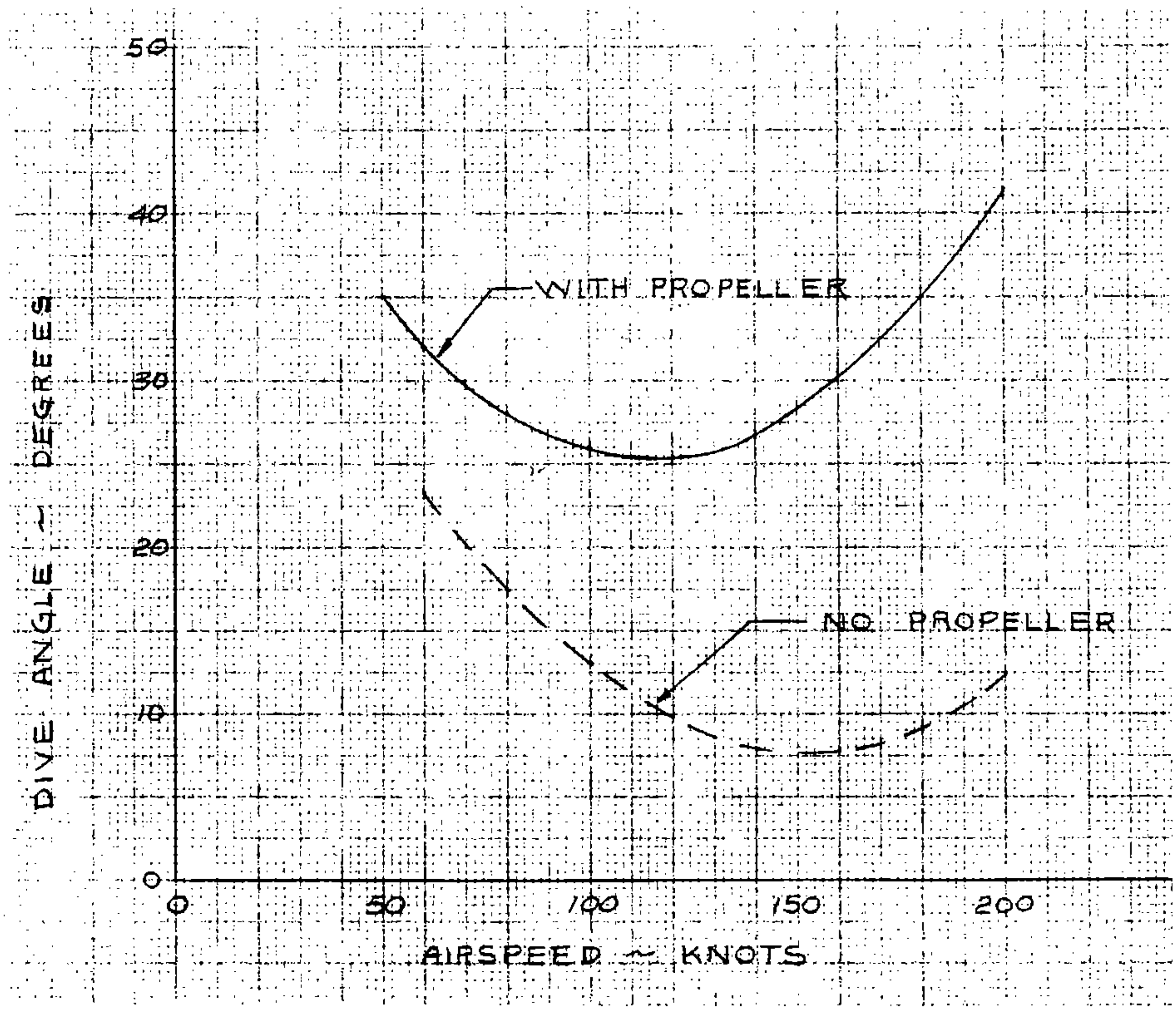


Figure 9-4. Steady Dive Angle

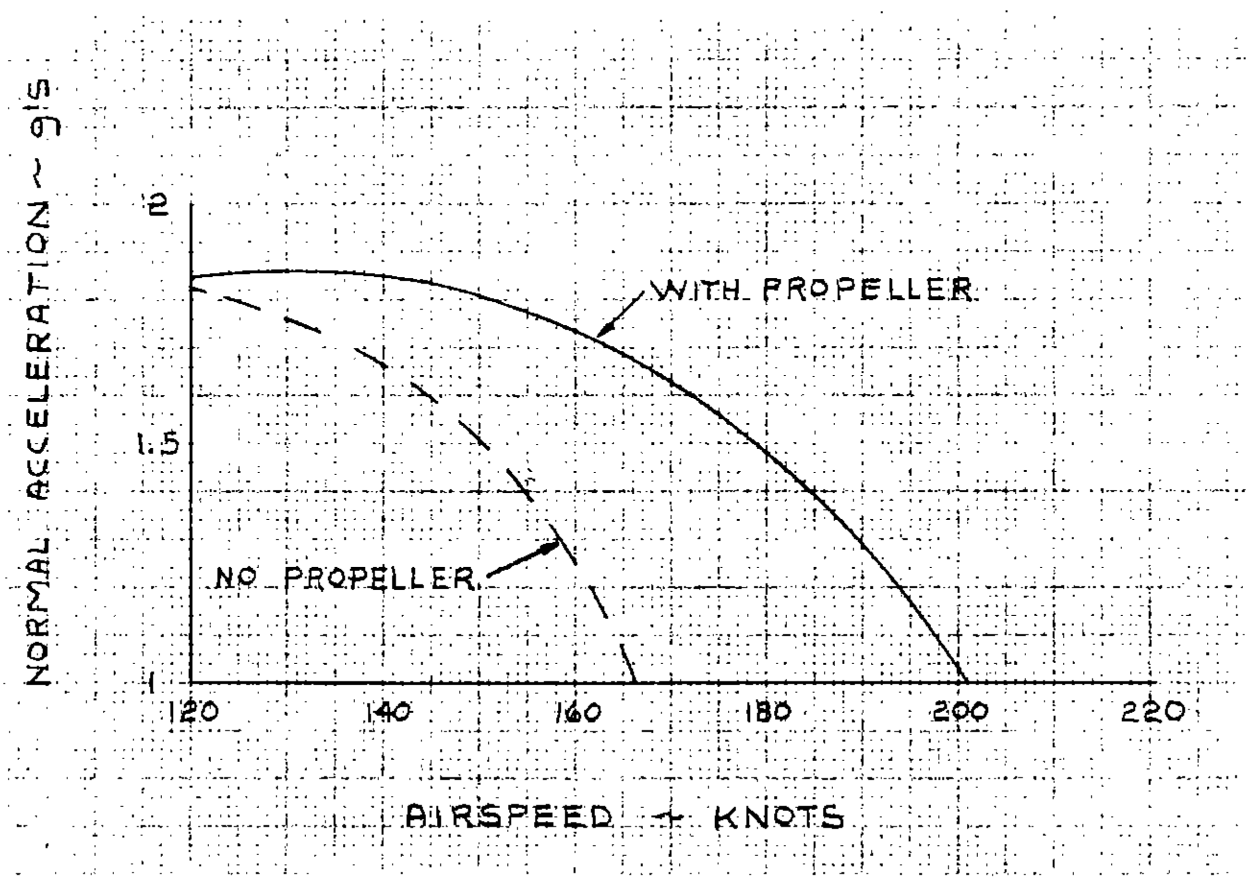


Figure 9-5. Load Factor in a Steady Level Turn

	WITH PROPELLER		NO PROPELLER	
	NORMAL FLIGHT	AUTOROTATION	NORMAL FLIGHT	AUTOROTATION
COLLECTIVE SETTING	3°	2°	15°	2°
ATTITUDE	2° nose up	2° nose up	9° nose down	2° nose up

Figure 9-6. AH-56A High Speed Autorotation Entry

	With Propeller	No Propeller
• HIGH SPEED PERFORMANCE. V_{MAX}	202 kts	166 kts
• CONSTANT SPEED CONSTANT ALTITUDE MANEUVERING @ 150 KTS	1.8 g	1.5 g
• ATTITUDE CONTROL (HOVER)	+ 12° - 5°	0°
• ATTITUDE CONTROL HIGH SPEED	+ 2° - 9°	- 9°
• CONTROLLED DIVE FLIGHT PATH ANGLE @ 150 KTS	- 28°	- 8°
• HIGH SPEED LOW ALTITUDE FLIGHT SAFETY		
AUTOROTATION ENTRY - Collective Change	1°	13°
Attitude Change	0°	11°

Figure 9-7. AH-56A Effects of Compounding Summary

	Pitch	Roll
AH-56A	47,240 Ft lbs/rad/sec	47,460 Ft lbs/rad/sec.
SPEC. MIL-H-8501A REQ.		
VFR	18,300 Ft lbs/rad/sec	20,400 Ft lbs/rad/sec.
IFR	34,400 Ft lbs/rad/sec	28,300 Ft lbs/rad/sec.

Figure 9-8. AH-56A Vehicle Pitch and Roll Damping

AIRSPEED KNOTS	PERIOD SECONDS	TIME TO 1/2 AMP - SEC($\tau_{1/2}$) AH-56A	H-8501A REQ (MAX)
Hover	2.57	.835	2.57
100	2.7	.438	2.7
160	3.12	.306	3.12
200	3.9	.236	3.9

Figure 9-9. AH-56A Short Period Damping

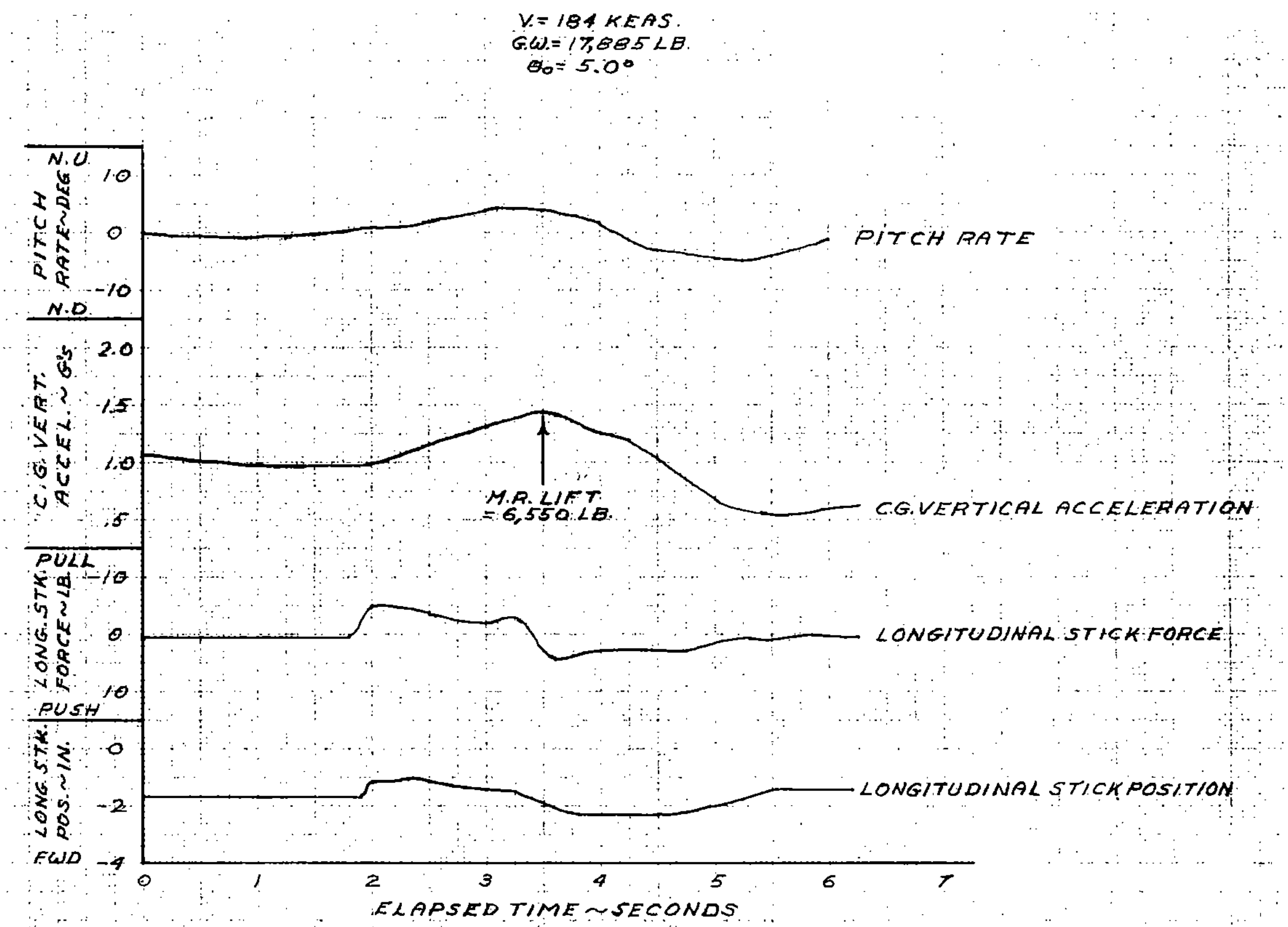


Figure 9-10. Time History of Roller Coaster

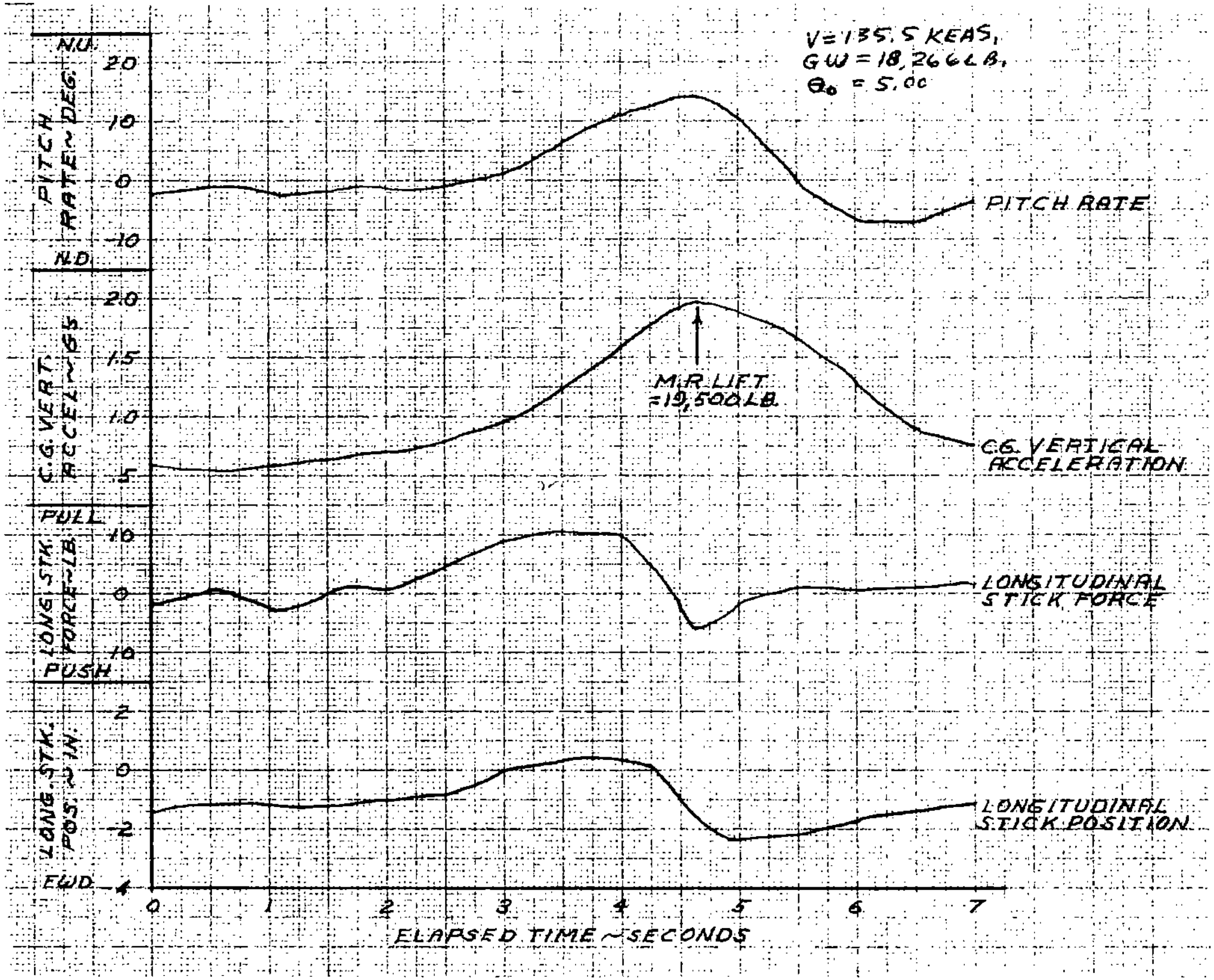


Figure 9-11. Time History of a Pull-Up

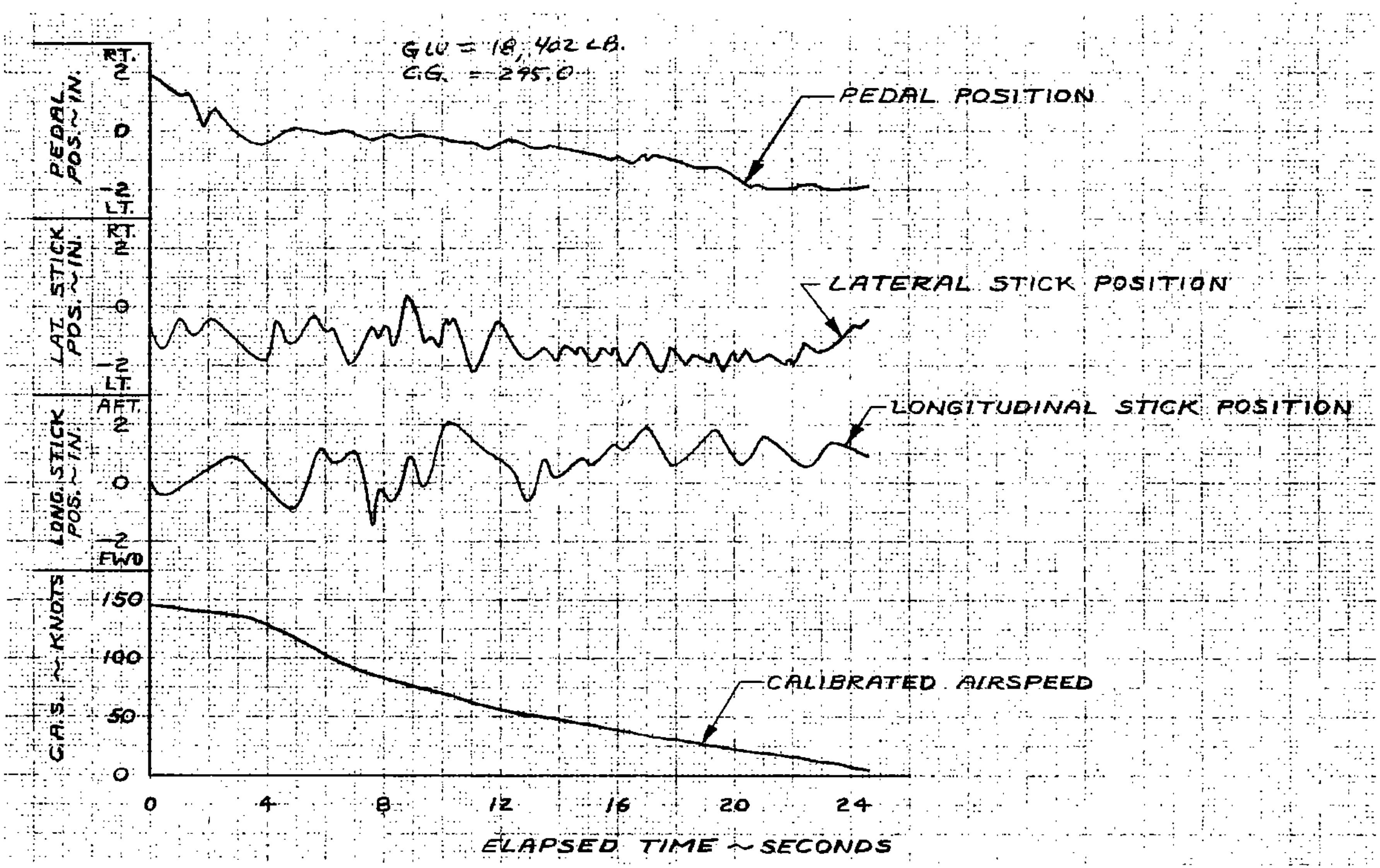


Figure 9-12. Time History of Deceleration During Level Flight

DATA POINT AND N_R	MAXIMUM EXCURSIONS			
	LONG. CYCLIC IN	LAT CYCLIC IN	PEDAL	COLLECTIVE
1 95%	.81 fwd to .69 fwd	.43 Lt to .05 Lt.	1.192 Lt to .98 Lt	12.25° max 12.18 min.
2 100%	.78 Fwd to .47 fwd	.41 Lt to .02 Rt.	.94 Lt to .60 Lt	11.33° max 11.29 min
3 105%	.78 Fwd to .34 fwd	.53 Lt to .09 Rt	.69 Lt to .48 Lt	10.55° max 10.52 min
4 95%	.90 fwd to .44 fwd	.41 Lt to .32 Rt	1.23 Lt to .83 Lt	12.21° max 12.18 min
5 100%	.87 fwd to .25 fwd	.41 Lt to .34 Rt	.92 Lt to .58 Lt	11.25° max 11.22 min
6 105%	.87 fwd to .34 fwd	.36 Lt to .09 Rt	.60 Lt to .37 Lt	10.48° max 10.44 min.

Figure 9-13. AH-56A Control Activity Hover

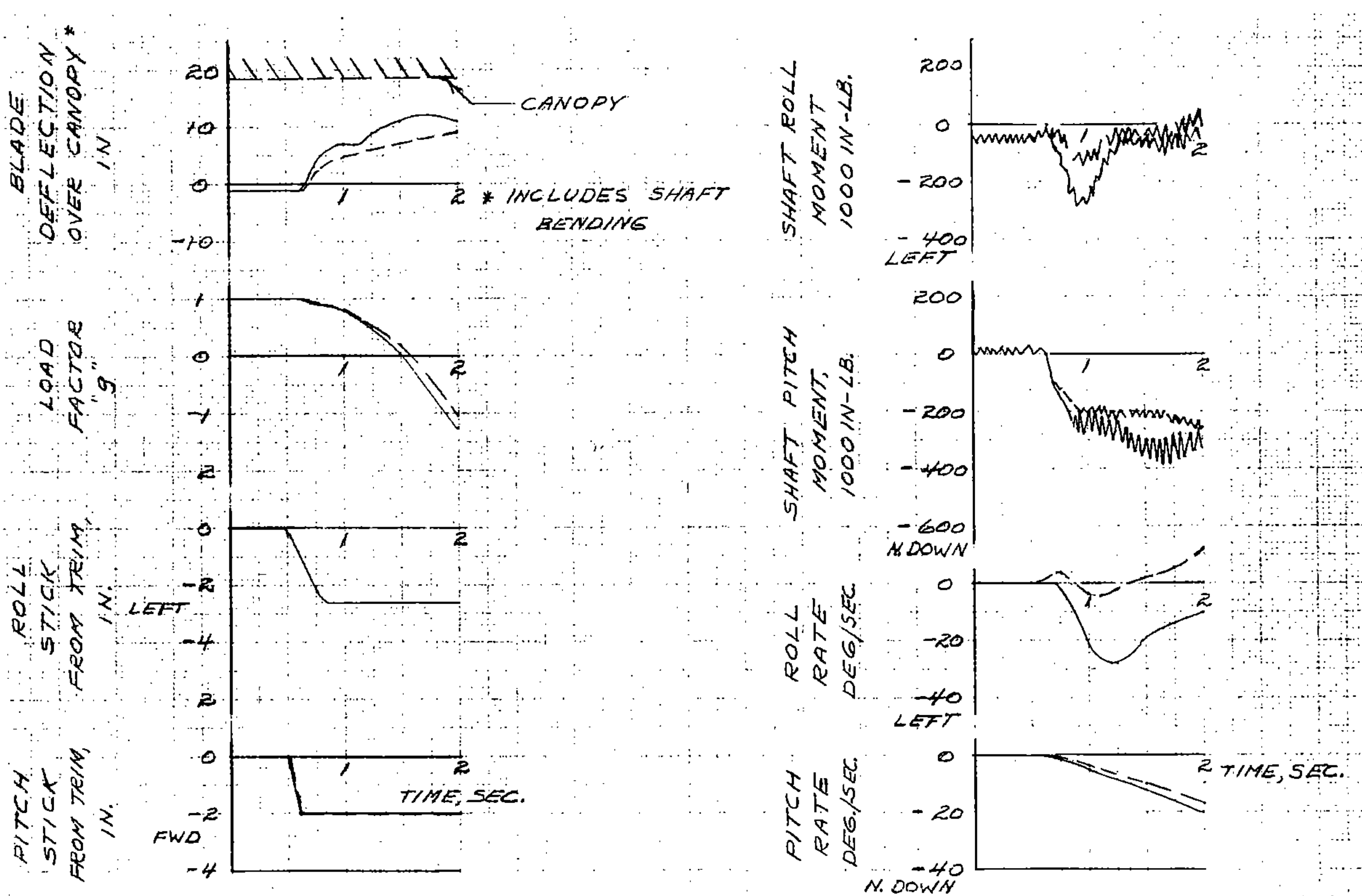


Figure 9-14. Rotor Blade/Canopy Clearance

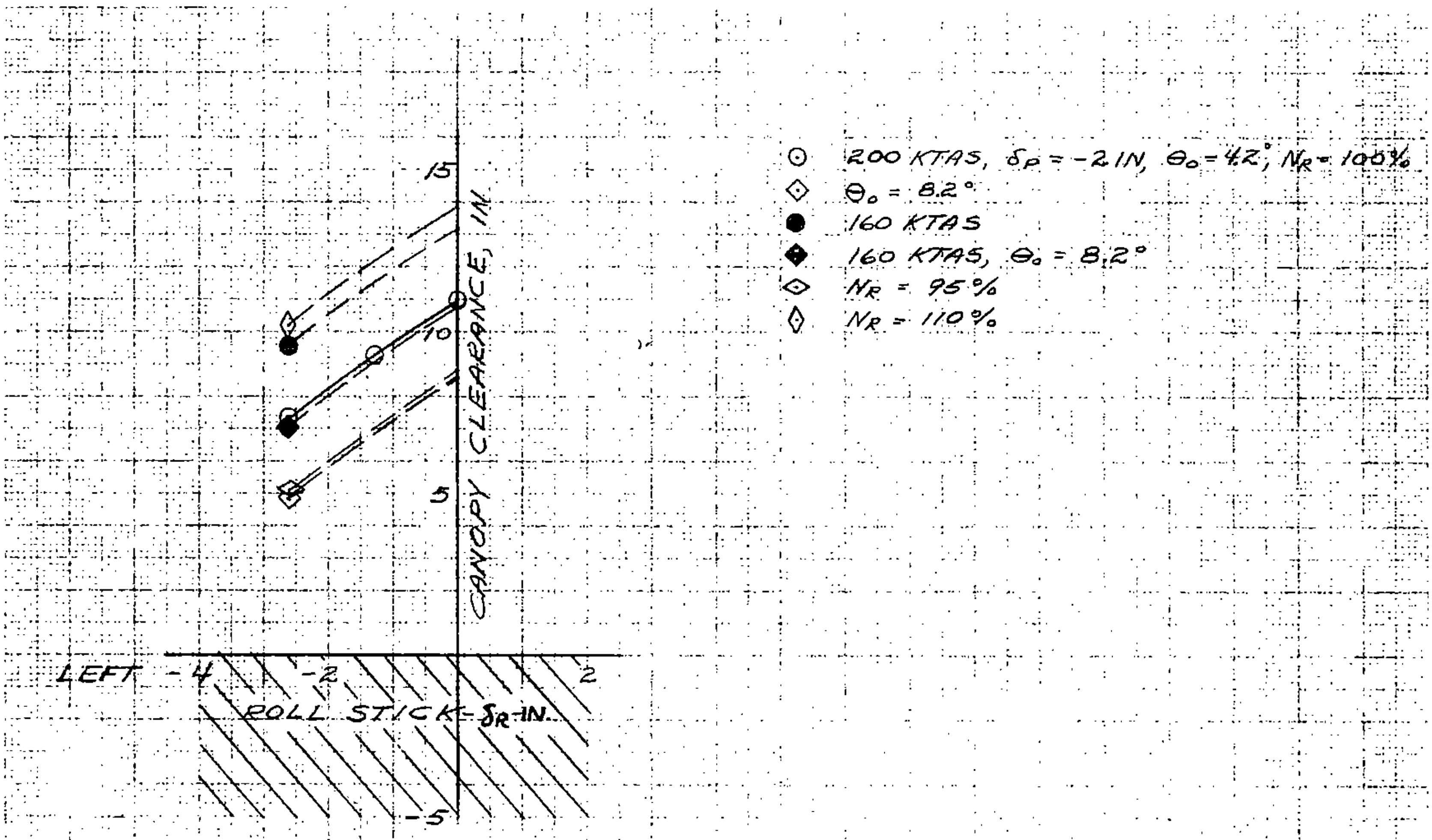


Figure 9-15. Rotor Blade/Canopy Clearance at Zero Load Factor

FUEL SYSTEM

I. GENERAL DESCRIPTION

The AH-56A fuel system provides storage capacity for the quantity of fuel required for mission accomplishment, means to transfer the fuel from storage tanks to the engine during all modes of operations, means for fuel management, and means for indicating specific functional conditions.

II. COMPONENTS AND LOCATIONS

Name of Component	Number per Aircraft	Location in Aircraft
Internal Fuel Tank (consisting of three interconnected cells)	1	Under Engine Compartment and in forward portion of each sponson
External Tanks	6	1 each located at: LH and RH fuselage pylon station LH and RH Inboard wing pylon station LH and RH Outboard wing pylon station
Fuel Boost Pump	1	RH Sponson, Midpoint
Refueling Panel	1	RH Sponson, Aft
Fuel Management Panel	1	Pilot Station

III. MAJOR COMPONENT DESCRIPTION

A. Internal Fuel Tank

The Internal Fuel Tank, with a capacity of 438 gallons, consists of three interconnected cells which provide self-sealing protection, except for the top surfaces, against penetration of 12.7 mm. projectiles. The main cell under the engine compartment contains the surge chamber in which the screened fuel outlet is located. Check valves on each end of this chamber allow fuel flow into the chamber only. Stand pipes in the chamber allow flow into the sponson tanks during fueling but prevent transfer of the fuel to the sponson tanks during nose down attitudes when fuel levels are below the top of the standpipe. A jet pump, energized by flow from the boost pump, transfers fuel from the aft end of the main tank to the surge chamber. Provisions are made in each cell for the installment of float type quantity gaging units. The main cell provides means for the installation of a gravity filler and a pressure refueling automatic level control shut-off valve.

B. External Tanks

These tanks are streamlined aluminum alloy tanks. The fuselage stations are designed for external tanks with a capacity of 300 gallons each and the wing station external tanks have a capacity of 450 gallons each. These tanks are compartmented for center of gravity control. In addition the tanks have provisions for level control shut-off during pressure refueling; provisions for capacity quantity gaging system probes; provisions for jettisoning in flight and for breakaway on crash landings.

C. Fuel Boost Pump

The fuel boost pump is a line mounted AC motor driven centrifugal flow pump with a flow rating of 16,000 pounds/hour (41 gpm). Flow control valves, electrically and manually actuated, provide for use of the pump during either suction refueling or supplying fuel to the engine.

D. Refueling Panel

The refueling panel is provided in the aft portion of the right hand sponson which contains a single point pressure refueling adapter, a suction refueling adapter, manual and electrical controls required for different modes of refueling and a boost pump control switch. This panel is readily accessible from ground level for ease of use.

E. The Fuel Management Panel

This panel located at the pilot's flight station, provides control for the emergency firewall shut-off valve, the boost pump, the external tank transfer, and external tank quantity indication selector. Emergency firewall shut-off valve control is also provided at the co-pilot's station.

IV. SYSTEM OPERATION

Gravity fueling is accomplished for the internal tanks by means of a standard refueling nozzle through a filler well in the main fuel cell on the Right Hand side of the aircraft. The tanks are full when the fuel level reaches the filler well opening. Each external tank is equipped with two filler wells, one for each compartment. These tanks are full when the fuel level reaches the filler well.

Single point pressure refueling is accomplished by use of a type D-2 fueling nozzle attached to the adapter at the refueling panel. The pressure refueling switch is placed in the "ON" position. Both internal and external tanks may be refueled by this means. Each of the tanks incorporates an automatic shut-off valve which closes when the fuel level reaches full. Suction refueling is accomplished by attaching a hose from the source of fuel to the suction refueling adapter at the refueling panel. The suction refueling switch is positioned to "ON", the Pressure refueling switch is positioned to "ON" and the boost pump switch is positioned "ON". The same automatic shut-off valves which are installed in the individual tanks perform the same function during suction refueling as they do during pressure refueling.

External fuel transfer is accomplished by selecting tanks to be transferred by actuation of the selection switch and the transfer switch at the pilot's fuel management panel. This selection initiates the transfer from the tanks in pairs, i.e., both fuselage pylon mounted tanks, both inboard wing pylon mounted tanks, etc. The energy for transfer is supplied by pressure regulated cooled engine compressor discharge air. During transfer, fuel enters the internal tank by way of the automatic fuel shut-off valve used in pressure refueling. This valve cycles on and off to maintain a full internal tank level.

Fuel feed to the engine is accomplished by use of the boost pump which draws fuel from the screened outlet in the bottom of the cell surge chamber. Control for the boost pump is by the switch mounted on the pilot's fuel management panel. From the pump, the fuel flows through the strainer, through the firewall shut-off valve and the fuel/air cooler to the engine. Fuel flows by gravity through check valves from the interconnected sponson cells into the main cell surge chamber. A portion of the boost pump flow is directed to a jet pump in the main cell surge chamber which draws fuel from the aft end of the main cell. This assumes availability of fuel to the tank outlet at extremely low fuel state and at nose up attitudes.

The internal tanks are interconnected by a vent manifold which terminates through a flame arrester on the bottom surface of Right Hand Wing near the tip. During pressure or suction refueling, venting from the external tanks is accomplished through the same vent system.

The gaging system for the internal tanks provides a totalized readout on both the pilot's and the copilot's station in pounds. The external tank gaging system, at the pilot's station only, provides a readout in pounds for the tank selected. The selection is made by means of the switch on the fuel management panel.

A float switch, in the main tank, separate from the quantity gaging system provides a warning light indication when decreasing fuel level reaches the quantity required for 20 minutes of flight at 140 knots.

A pressure switch in the boost pump outlet line provides a warning light indication if the boost pump is inoperative. The engine and the fuel system are capable of operation without the boost pump assist at altitudes up to and above 6000 feet.

A pressure switch in the external tank transfer system provides an indication when the fuel supply from the selected external tank is exhausted.

A switch within the fuel strainer senses by-pass valve position and provides a warning indication when the strainer screen has become clogged.

The firewall shut-off valve is energized by the emergency fuel and engine shut-off switch at the pilot's and co-pilot's stations. This may be used to shut-off fuel flow to the engine compartment in the event of fire or as an emergency means to shut down an engine.

Pressure relief valves are incorporated in all tanks to limit pressures to a safe level in the event of failure of the automatic shut-off valve during pressure refueling.

V. PCRS COMPARISON

The PCR study resulted in extensive changes to the fuel system. Reduction in the ferry mission requirements deletes the fuselage and the outer wing external tanks. Internal fuel capacity is increased by 30 gallons. This additional capacity is achieved by forward extension of the left hand sponson cell.

Features will be added to comply with the ballistic/crashworthy fuel system requirements. These specify that prime fuel system integrity be maintained during a survivable crash. A fuel tank wall construction having resistance to high impact and shear loads, as well as the bullet sealing capability, is substituted for the baseline bullet sealing fuel cell construction. Frangible fittings capable of separation prior to tank rupture are provided at all support interfaces between the tanks and the structure. Self sealing breakaway fittings capable of separation and sealing prior to line breakage or tank rupture are provided at each point of attachment of a plumbing line.

The suction refueling provisions have been deleted. This allows the use of a smaller submersible boost pump capable of engine flow requirements only. The single point pressure refueling provisions are replaced with "closed circuit" fueling capabilities compatible with the fueling nozzle

(FSN 4930-478-5728(A12)). This nozzle adapter is located in the position of the baseline refueling panel. Only the internal tanks are refueled with this system.

Vulnerability to gun fire is considerably reduced with this Approved PCRS configuration because of the reduction in the number of fuel accessories and fuel lines and fittings.

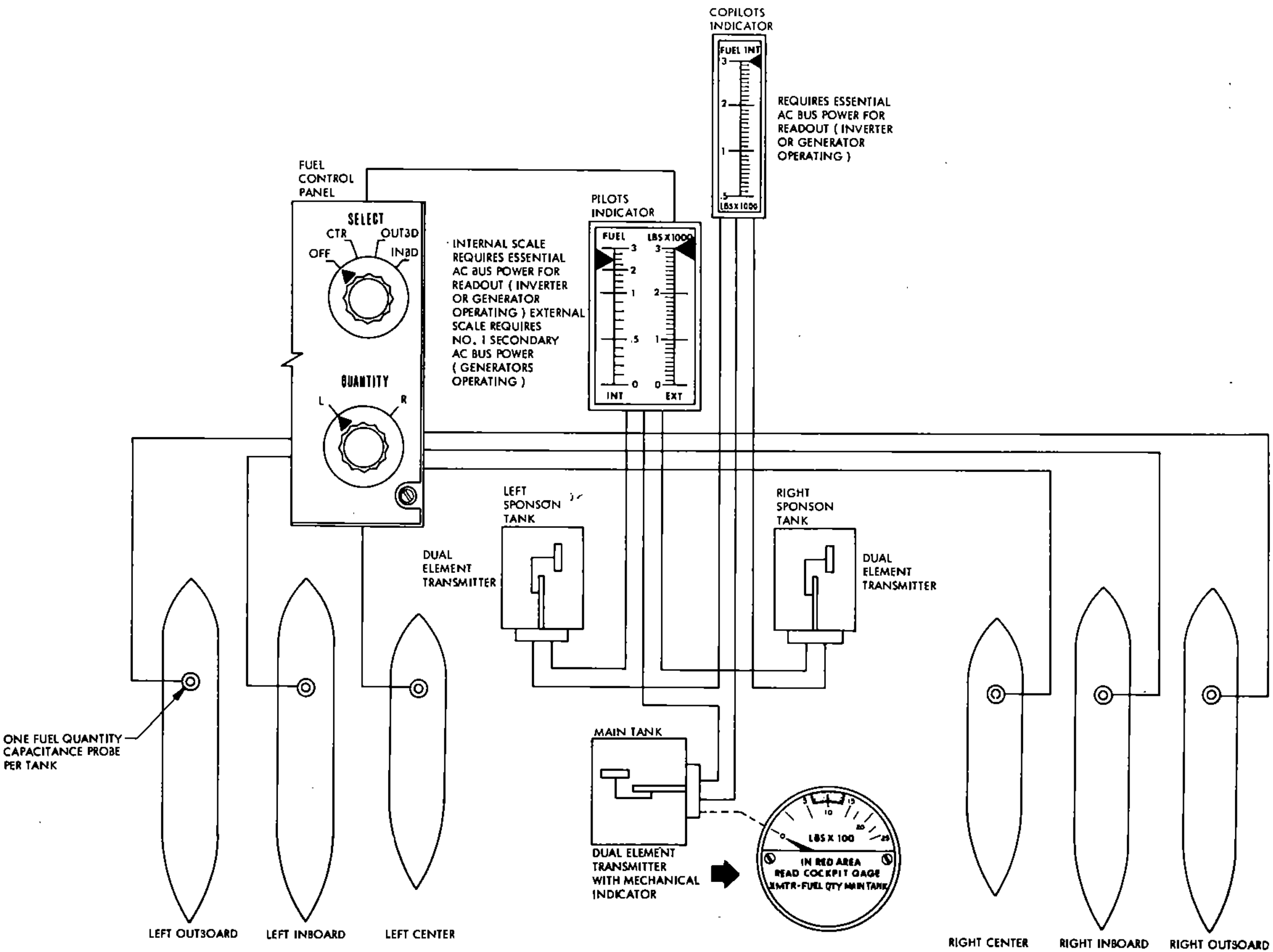


Figure 10-2. Fuel Quantity System Block Diagram

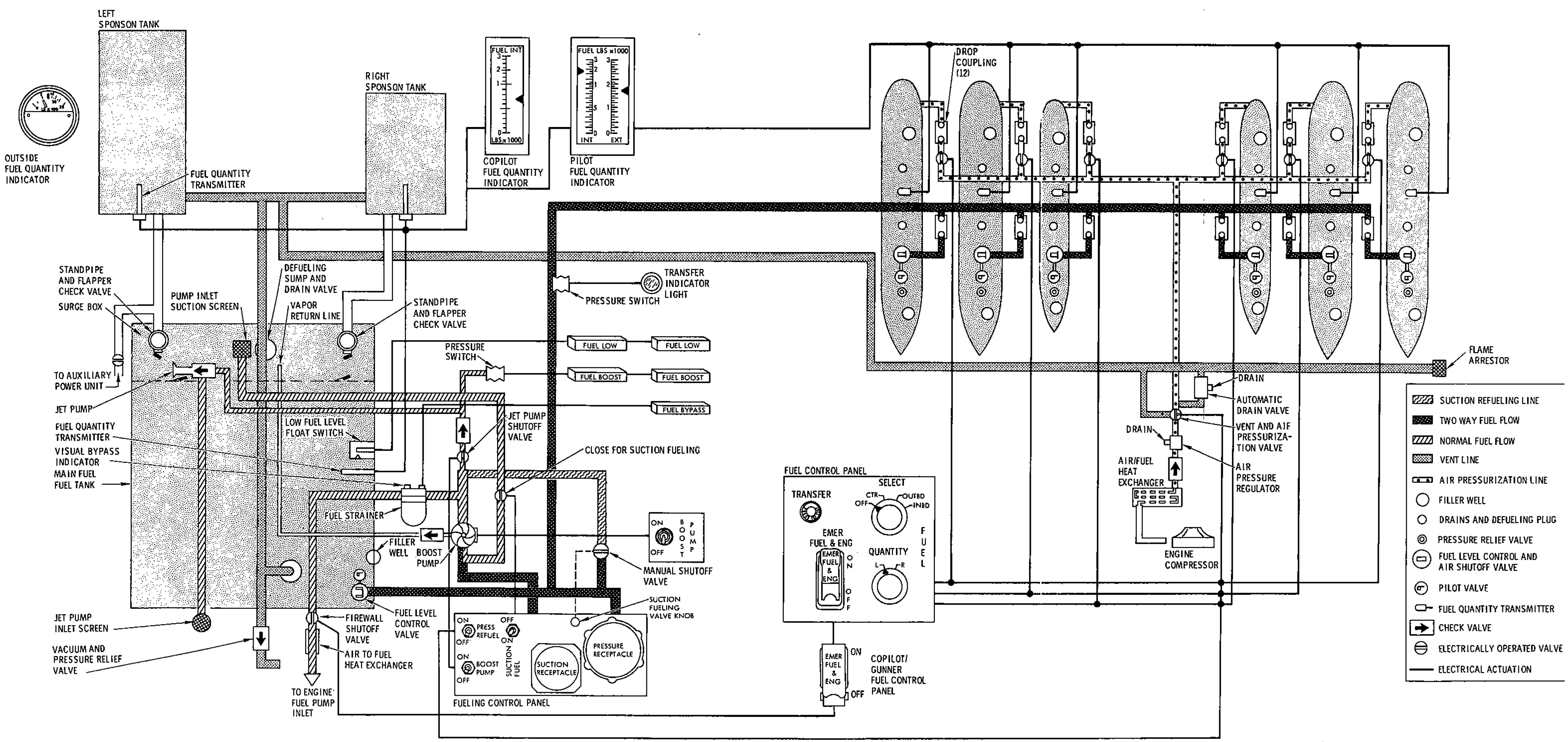


Figure 10-1. Fuel Supply System

HYDRAULIC SYSTEM

I. GENERAL DESCRIPTION

To perform the various hydraulic functions on AH-56A aircraft, two independent, 3000 psi hydraulic systems are used. Conventional red oil is the fluid media. The systems are identified as No. 1 and No. 2.

The No. 1 system is used solely for supplying power to the No. 1 side of all primary control servo units (collective, cyclic and directional). Power for the No. 1 system is supplied from a power package which operates only if the main rotor is turning.

The No. 2 system provides power for the No. 2 side of the same primary control servo units and in addition powers subsystems shown in the hydraulic schematic block diagram. Power for this system is supplied from a power package which operates whenever the APU or main rotor are operating.

A reservoir fill system common to both systems is provided. This permits bleeding the No. 1 system from the No. 2 system if desired.

Pressure switches located just downstream of pump quick-disconnect couplings cause warning lights on the pilots panel to come on if system pressures fall below 2000 psi.

Standard MS flareless fittings are used for fluid connections throughout the system. Corrosion resistant steel tubing is used for all high pressure lines and some low pressure lines. 6061 aluminum alloy tubing is used for large size return pressure lines.

II. COMPONENTS AND LOCATIONS

Name of Component	Number per Aircraft	Location in Aircraft
Hydraulic Power Package No. 1	1	Mounted on upper fwd RH side of transmission
Hydraulic Power Package No. 2	1	On lower LH side of transmission accessory case
Hydraulic Reservoir Fill Pump Unit	1	RH forward corner of transmission accessory section area
Hydraulic Rotary Selector Valves	2	Accessory section area. One in No. 1 package return line. One in between the No. 1 and No. 2 pressure lines
Hydraulic System Selector Switch	1	Pilot's collective lever
Overboard Drains	6	a. Control servo access door b. Leading edge of LH stabilizer c. Forward of aft avionics access door d. Outboard of servo access door (LH & RH)
Hydraulic System Pressure Indicator	2	Right hand side of pilot's main instrument panel
Hydraulic System Failure Light	2	Pilot's annunciator panel
Master Caution Light	2	One each in upper center portion of pilot's and copilot/gunner's main instrument panels
Fault Location Aural Warning System	1	Taped human voice message played into the crew headsets
Sight Gage	3	Installed in bleed lines at each power package and in supply line at reservoir fill pump
Hydraulic Starter Motor	1	Engine Accessory Section
Starter Selector Valve	1	Integral part of Hydraulic Valve Manifold
Rotor Brake Selector Valve Assembly	1	Left hand transmission area, lower aft bulkhead

II. COMPONENTS AND LOCATIONS (cont'd)

Name of Component	Number per Aircraft	Location in Aircraft
Rotor Brake Pressure Switch	1	Mounted just inboard and below the rotor brake selector valve assembly
Rotor Brake Switch	1	Left hand side of pilot's instrument panel
Rotor Brake Annunciator Panel Light	1	Pilot's Annunciator panel
Pilot's Landing Gear Lever	1	On pilot's left forward console landing gear control panel
Landing Gear Lever Release Button	1	On landing gear control panel. Pilot's left hand console
Landing Gear Lever Override Button	1	On landing gear control panel. Pilot's left hand console
Ground Air Safety Switches	2	Attached to MLG shock strut cylinder
Landing Gear Emergency Release Handles	2	On left hand console in each crew station
Landing Gear Position Indicators	2	In each crew station left hand console
Landing Gear Unsafe Warning Light	2	Both installed in ldg. gear knob of pilots lgd. gear lever
Main Landing Gear	2	One left hand, one right hand mounted to torsion box section
Tail Landing Gear	1	Housed in vertral fin
Negative Torque Selector Valve	1	On floor in forward left hand side of transmission area
Negative Torque Control Valve	1	Mounted on forward left hand side of transmission
Negative Torque Actuating Cylinder	1	In forward right hand side of swashplate area
Propeller Hi Pitch Control Valve	1	In aft section, above vertical stabilizer
Propeller Hi Pitch Actuating Cylinder	1	At lower center portion of bulk-head 646 aft.

II. COMPONENTS AND LOCATIONS (cont'd)

Name of Component	Number per Aircraft	Location in Aircraft
Prop Hi Pitch Selector Switch	1	On pilots fuel control panel
Stick Center Accumulator	1	On main servo package
Stick Center Selector valves	2	On main servo package
Oil Cooler Fan Motor	1	In ducting in aft avionics compartment
Oil Cooler	1	In left hand side of aft avionics compartment
Filter Scavenge Fans	1	Lower aft outboard side of filter cowl
Filter Scavenge Fan Selector Valve	1	Bottom center of transmission accessory section
Flow Control Valves	2	Bottom left hand side of transmission accessory section
XM52 Azimuth Drive	1	In XM52 belly turret
Azimuth Drive Filter	1	Lower RH forward side of ammo bay
Azimuth Drive Shut-off valve	1	Aft upper side of ammo bay

III. MAJOR COMPONENT DESCRIPTION

A. Hydraulic Power Packages

Each power package is a modular unit containing a pump, reservoir, relief valves, filters, pressure transducer and a bypass valve. The pressure transducer provides the signal for the cockpit gages mentioned previously. The bypass valve, when energized, connects pump output directly to the reservoir and drops system pressure to approximately 100 psi. Operation of both system bypass valves is controlled from a hydraulic system selector switch at the pilots station. Preflight checks of control system operation with either hydraulic system inoperative can thus be made.

B. Hydraulic Valve Manifold (No. 2 system only)

The hydraulic valve manifold is the distribution point for hydraulic pressure to components of the No. 2 hydraulic system. The valve consists primarily of an engine start selector valve and a landing gear selector valve. The Valve Manifold is located at fuselage FS 317 on the left side just aft of the transmission. The LH swash plate access door provides room for maintenance.

The valve housing has 14 ports (port No. 10 is not used). Each of the ports have fittings which use different size tube connections. Danger of wrong connections to the valve is eliminated. All ports are numbered, with pressure ports having odd numbers and the return ports even numbers.

There are three (3) electrically operated solenoids in the valve manifold. Two (2) solenoids select hydraulic pressure for the main landing gear, one (1) for gear up, and one (1) gear down positions. The other solenoid when energized, selects hydraulic pressure for the engine starting system. This solenoid, when deenergized, permits pressure supply to the oil cooler motor and engine inlet scavenge fans.

C. Rotor Brake Selector Valve Assembly

The major components of this assembly are the selector valve, accumulator, check valve and relief valve. This unit weighs approximately 3.35 pounds.

The rotor brake selector valve is a three-way, two-position solenoid operated valve. It will remain in the last selected position when electrical power is removed. A manual override provision permits rotor brake pressure to be manually released in case of selector valve failure in brake ON condition, or an electrical failure in selected brake OFF condition. The red override button is located on the forward side of the rotor brake selector valve assembly.

The accumulator is gas charged to between 775 and 1090 ±25 psi (in accordance with ambient temperature) and stores hydraulic energy sufficient for two (2) actuations of the rotor brake. It is utilized when there is no system pressure available. The accumulator head and the selector valve body are one and the same. A pressure gage is mounted at the accumulator air chamber and reads air chamber pressure direct. A check valve allows No. 2 system pressure to charge the accumulator to 3000 psi - preventing reverse flow. Therefore, 3000 psi is maintained between the check valve and the brake cylinder in the brake ON position.

A thermal relief valve, part of the closed circuit, is a protection against thermal expansion. The valve cracks at 4000 psi and seats at 3400 psi. The check valve and thermal relief valve are installed on opposite sides of the rotor brake selector valve body.

4. SYSTEM OPERATION

A. Engine Start System

Engine starting is accomplished by a hydraulic motor geared to the engine accessory driveshaft.

With APU operating, No. 2 system hydraulic power is available for starting. Engaging the start-stop switch energizes a selector valve which directs pressure to the hydraulic starter. The starter is a variable displacement unit which enables it to operate initially at maximum displacement and torque to overcome engine inertia; as the engine accelerates the starter strokes to minimum displacement enabling it to operate at higher speed for the same system flow. At approximately 5300 RPM (starter speed) the engine will normally "LITE-OFF" and continue to accelerate on its own. An overrunning clutch on the starter allows disengagement as the engine accelerates. A cutout switch on the starter de-energizes the selector valve at approximagely 5300 RPM.

B. Rotor Brake System

Used to apply braking force to the power train to reduce coast down time during engine shutdown and can be used to prevent rotation of the rotor system during engine ground idle.

The system consists of a hydraulic operated rotor brake and a selector valve. Actuation of a switch in the cockpit to brake "ON" energizes the selector valve to apply pressure to the brake. Moving the switch to "OFF" dumps pressure and releases the brake. The valve will remain in its last selected position in the event of electrical or hydraulic failure. A manual override will depressurize the brake if valve fails in "ON" position.

An accumulator provides a reserve source of pressure in the event of No. 2 system failure or for maintaining the brake in a parked position.

A pressure switch energizes a caution light on the pilot's panel if brake is "ON" at any pressure above 75 psi.

C. Landing Gear System

The aircraft landing gear is electrically controlled and hydraulically operated. The two main landing gear wheels retract by swinging aft and up, while the tail wheel retracts straight up into the ventral fin. The pilot station contains both normal and emergency landing gear controls. The main landing gear is locked up and down by over-center latching mechanisms operated mechanically by the gear linkage. The landing gear main wheels are retracted by hydraulic system pressure and held retracted by the uplocks. The tail wheel is retracted hydraulically and held retracted by pressure from hydraulic system No. 2. Dumping system hydraulic pressure allows compressed shock strut air pressure to fully extend the tail wheel; system hydraulic pressure extends the main gear. A mechanical cable lock release system plus an emergency hydraulic system ensures emergency main landing gear extension at airspeeds up to 100 knots.

D. Main Landing Gear

Each main landing gear consists of a conventional air-oil shock strut, drag strut, jury strut, spring-type bungee and an actuator. The main landing gear support is provided by the drag struts. Uplock and down-lock is accomplished by the jury strut. During emergency extension the bungees assist a hydraulic accumulator to extend the main landing gear against air loads. Power for landing gear extension and retraction is provided by hydraulic action as shown in the hydraulic schematic. Each main landing gear has provisions for the installation of a safety pin to protect against inadvertent retraction of the gear when the aircraft is on the ground.

The door retracting mechanism is entirely mechanical. A yoke attached to the bottom of the main landing gear strut cylinder contacts a hook and linkage during the last 10 percent of the retraction cycle and pulls the doors closed behind the gear. A bungee spring holds the doors open when the gear is extended.

E. Tail Landing Gear

The tail landing gear is housed in the ventral fin. It is composed of a combination shock strut and retraction cylinder. A viscous-drag shimmy damper is mounted on top of the strut. The tail landing gear is retracted and held in the up position by hydraulic pressure overcoming compressed air. When the gear is retracted the wheel is partially exposed. Release of the hydraulic pressure allows the compressed air in the shock strut cylinder to extend the gear. Normal control of the tail landing gear is electrical and automatic with the main landing gear. A hydraulically actuated pin maintains that wheel orientation during strut retraction.

F. Wheel Brakes

The wheel brake systems are manually operated systems which conform to MIL-B-8584, type IV. The brakes are disk-type units. The right and left wheel brake systems are independent of each other to allow differential braking. Pressure is generated in power boost master cylinders operated by rotation of the pilot's brake pedals. This pressure is transmitted to the wheel brake pistons which apply the braking force.

The power boost master cylinder meters the 3,000 psi system pressure, obtained from the main landing gear down pressure line, to the wheel brake units as a straight power brake valve until a pressure of 140 to 180 psi plus the system return pressure is reached. At this point the pumping valve is closed and the unit operates as a master cylinder with metered power brake pressure applied to the back side of the piston to assist the pilot effort. A parking valve operated from a tee handle in the cockpit locks pressure to the brakes for parking.

G. Propeller Delta-Beta System

This sytem provides relief for negative torque conditions resulting from power loss at high propeller blade angles. It consists of a hydraulic actuator tied to the cable control system for the propeller and two hydraulic control valves, one mechanical and one electrical. A negative torque condition is sensed within the transmission and results in actuation of the mechanical valve. If the electrical valve is de-energized the hydraulic actuator will retract and move the cable system to a low pitch propeller range. A loss in rotor RPM (to 97%) while the aircraft is below 1.25 "G's" is required to de-energize the electrical valve.

H. Propeller Dump System

This system provides a means of reducing high propeller blade angle in the event of normal cable system failure.

Propeller "dump" is initiated from a switch in the cockpit which energizes a selector valve resulting in extension of a hydraulic cylinder which through linkage dumps propeller control pressure within the prop gear box. Loss of No. 2 system means loss of propeller "dump".

I. Cyclic Stick-Center System

This system consists of two hydraulic actuators within the main servo package that act on pilot input linkage to hold it in a centered position. The actuators are controlled by two electrically operated valves (in series). Both valves must be energized to operate the system. An accumulator is used as source of pressure in the event of No. 2 system failure. This system is integral with the flight control servo package.

J. Oil Cooler Fan Drive System

No. 2 system pressure is used to operate a hydraulic motor-driven fan in the oil cooler duct. The fan operates at all times except during engine starting when the start selector valve blocks flow to the fan motor. The fan provides air flow through a heat exchanger for cooling engine, transmission and hydraulic oil.

K. Engine Inlet Scavenge Fan

The hydralic motor driven scavenge fans are used to purge debris from the bottom of each engine inlet filter cavity and discharge it outboard.

L. XM52 Belly Turret Azimuth Drive

This system provides azimuth drive control for the belly turret. Installed on and delivered with the turret is a servo valve and hydraulic drive motor. This motor is geared to the turret azimuth drive train through a 500 to 1 ratio. To protect the servo valve from fluid contamination a filter is installed just upstream of the turret. An electrically operated shut-off valve is also installed in the system to maintain the turret in a depressurized condition when not is use.

5. PCRS CONFIGURATION

The PCRS hydraulic system configuration, except for the tail gear system is identical to the system just described. For the PCRS the tail gear will not retract and consequently all hydraulic equipment and associated plumbing is deleted. The tail gear strut will be a new design. Although it has no effect on the hydraulic system the PCRS main landing gear shock strut cylinder will be made of steel rather than titanium.

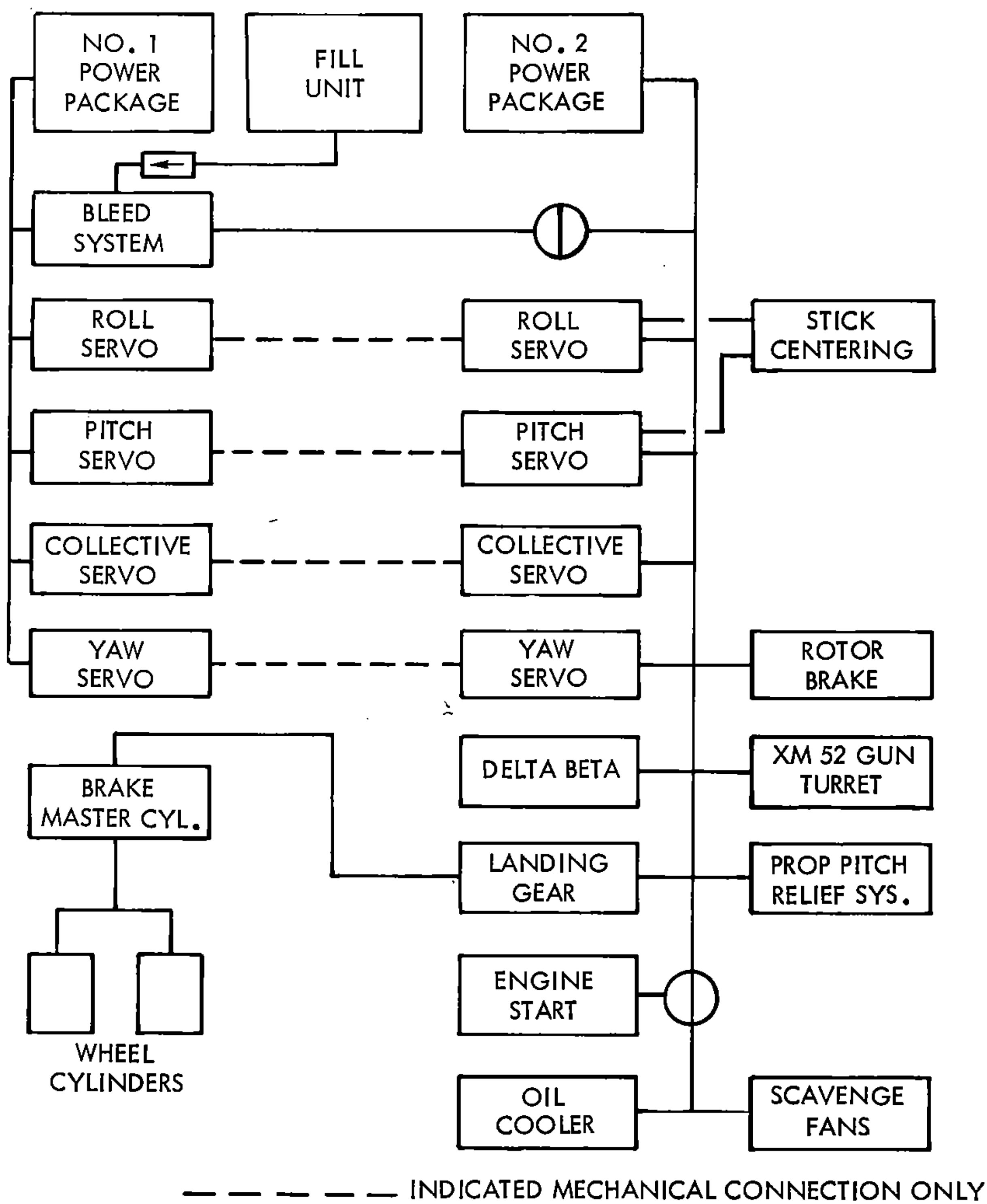


Figure 11-1. Hydraulic System Block Diagram

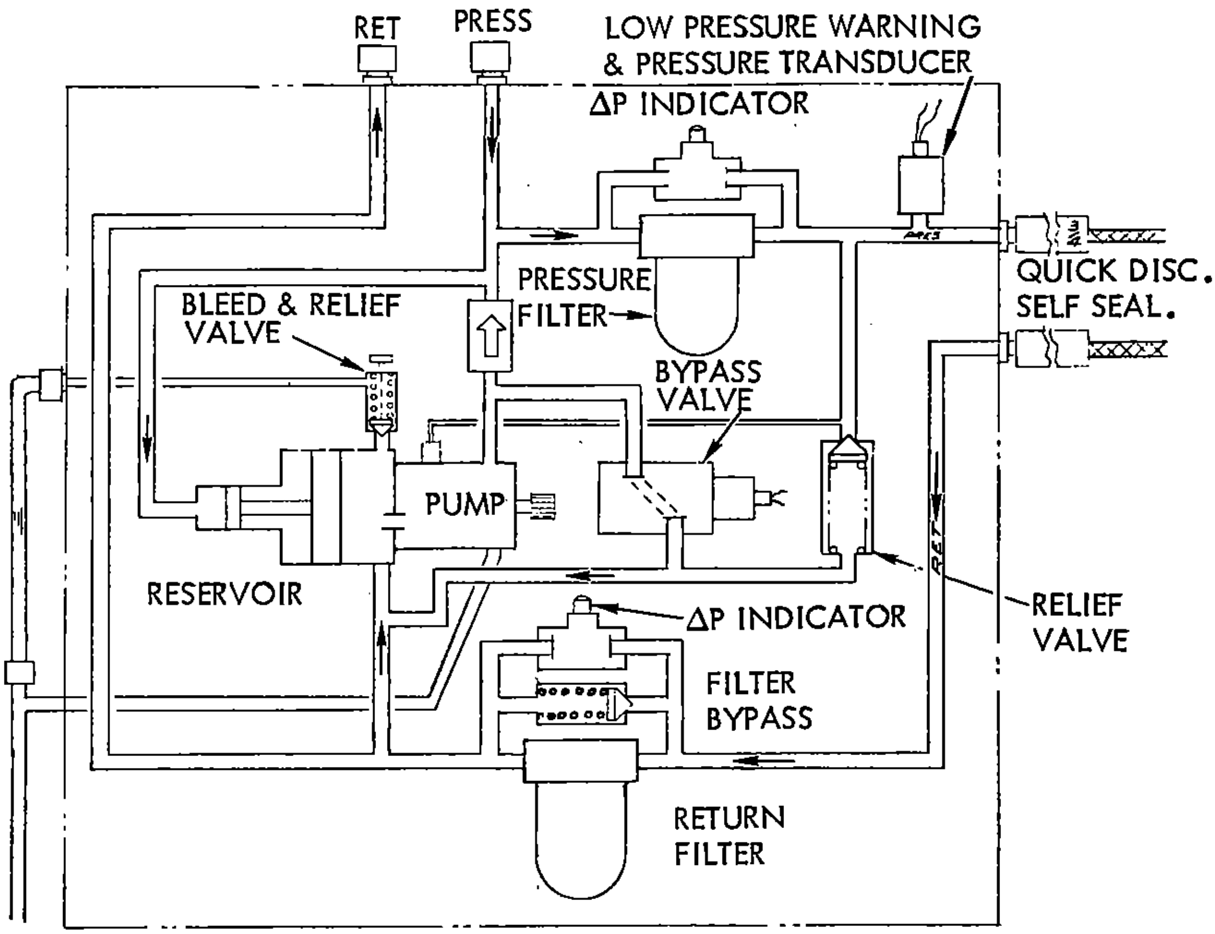


Figure 11-2. Typical Power Package

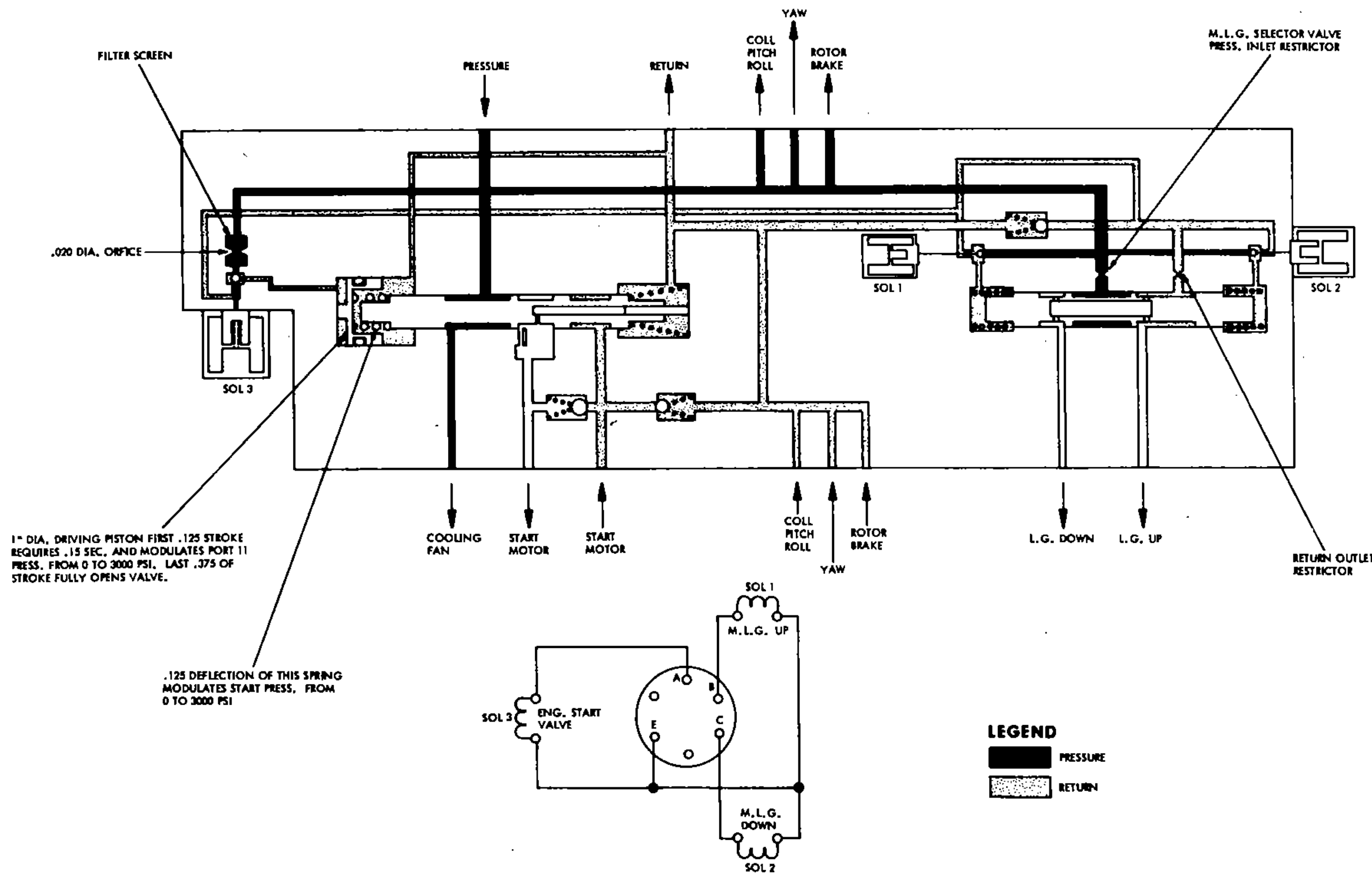


Figure 11-3. Hydraulic Manifold Valve Assembly Schematic

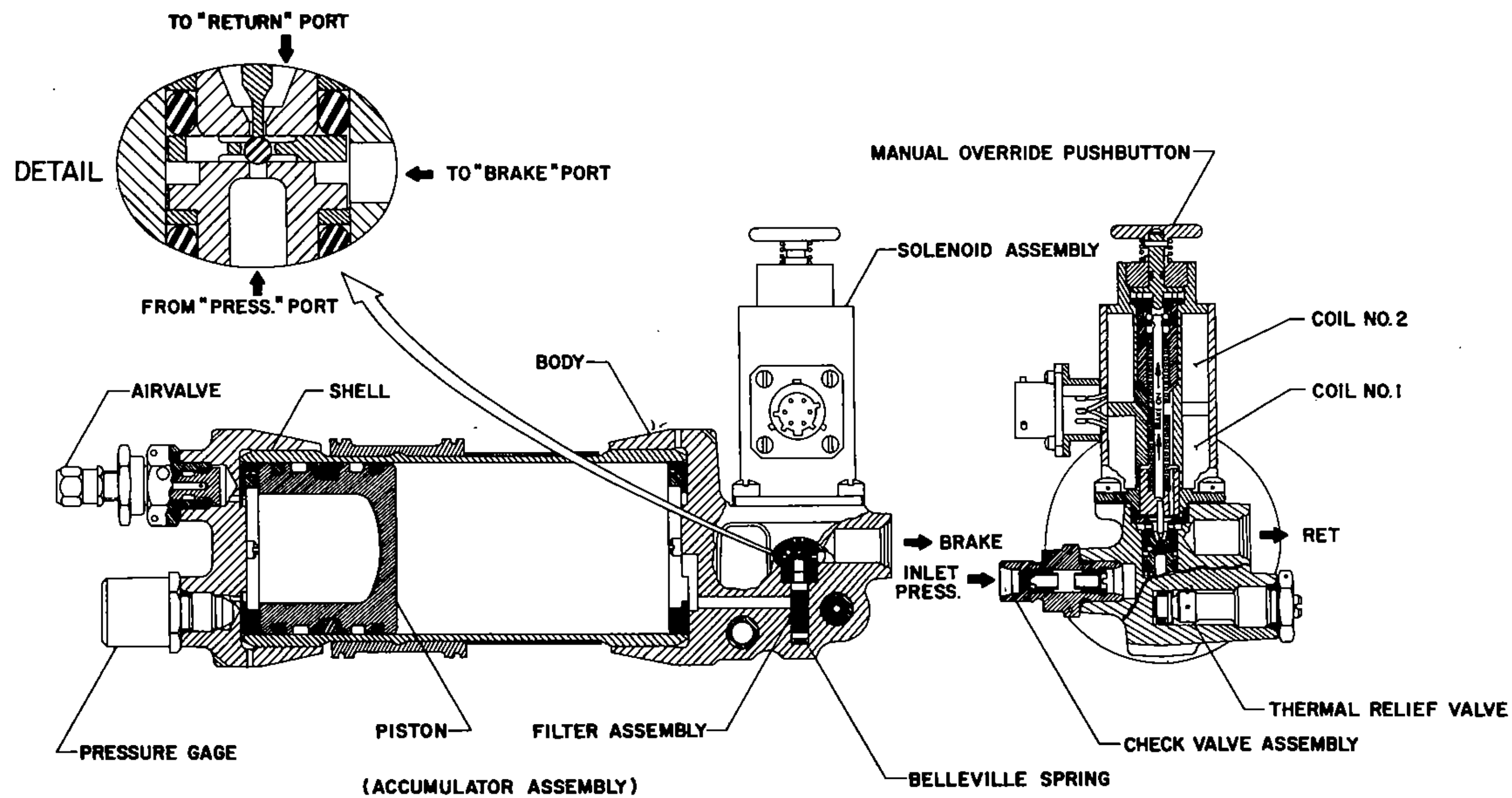


Figure 11-4. Rotor Brake Selector Valve and Accumulator Schematic

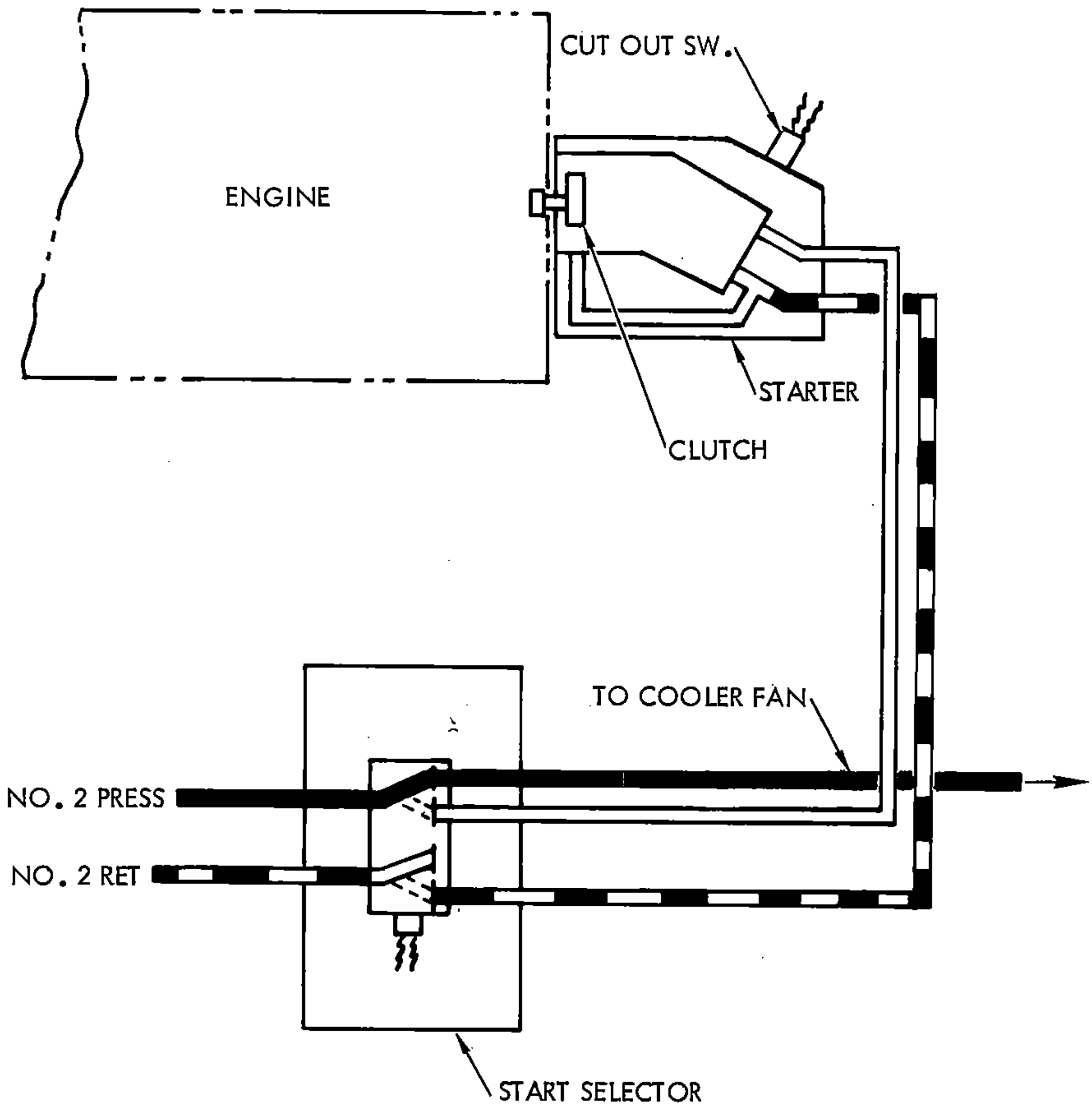


Figure 11-5. Engine Start System

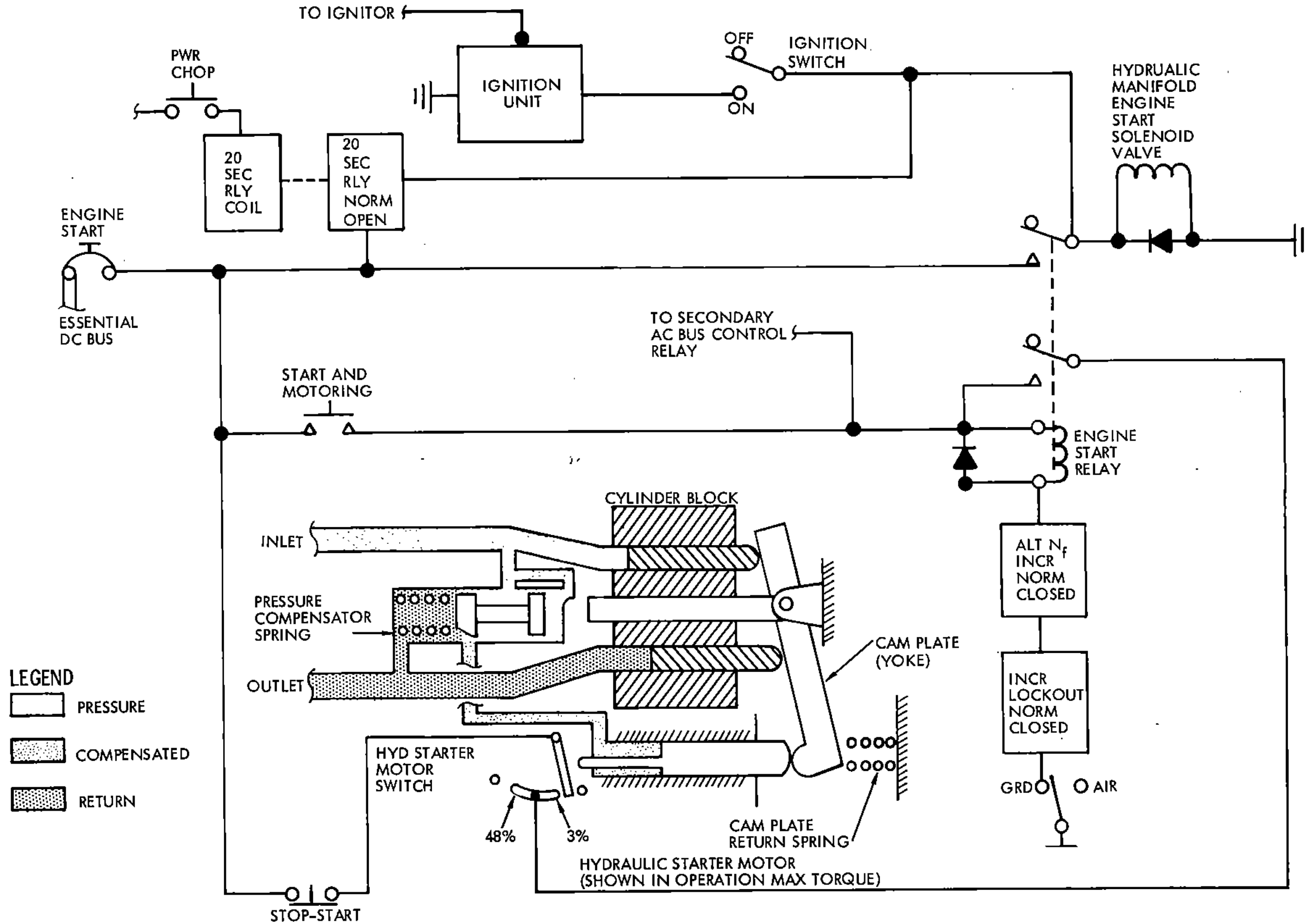


Figure 11-6. Engine Starter Circuit

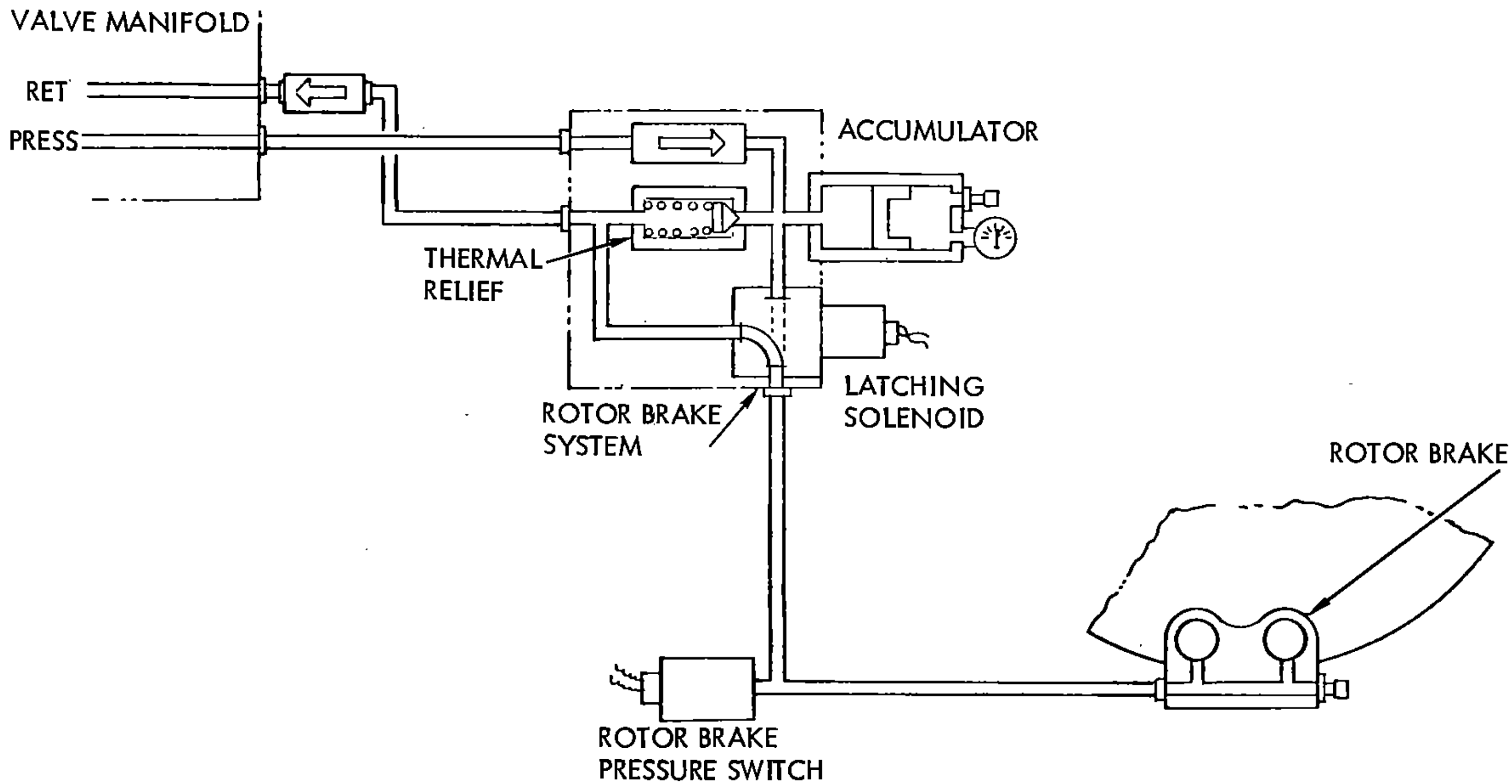


Figure 11-7. Rotor Brake System Schematic

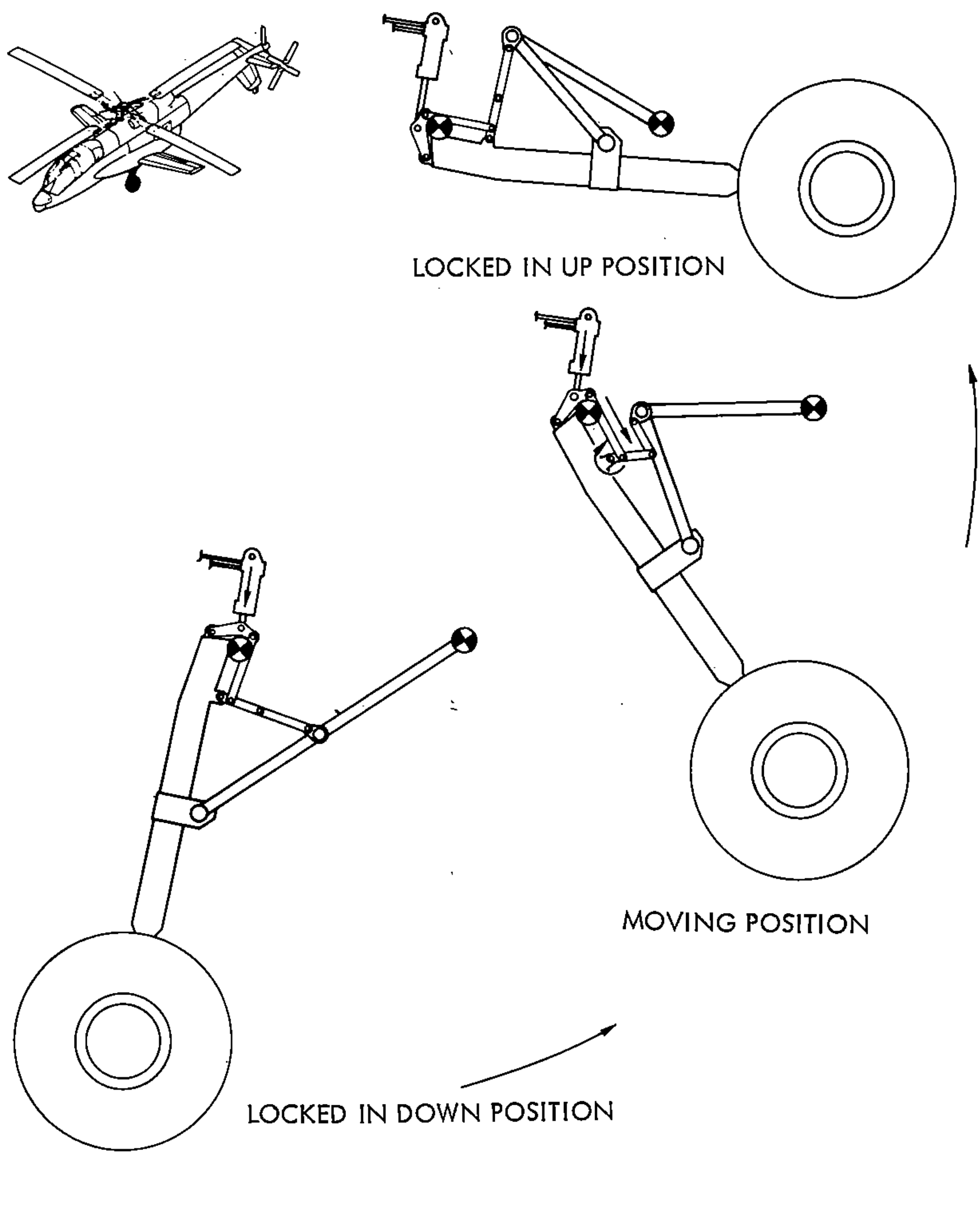


Figure 11-8. Main Landing Gear Retraction Diagram

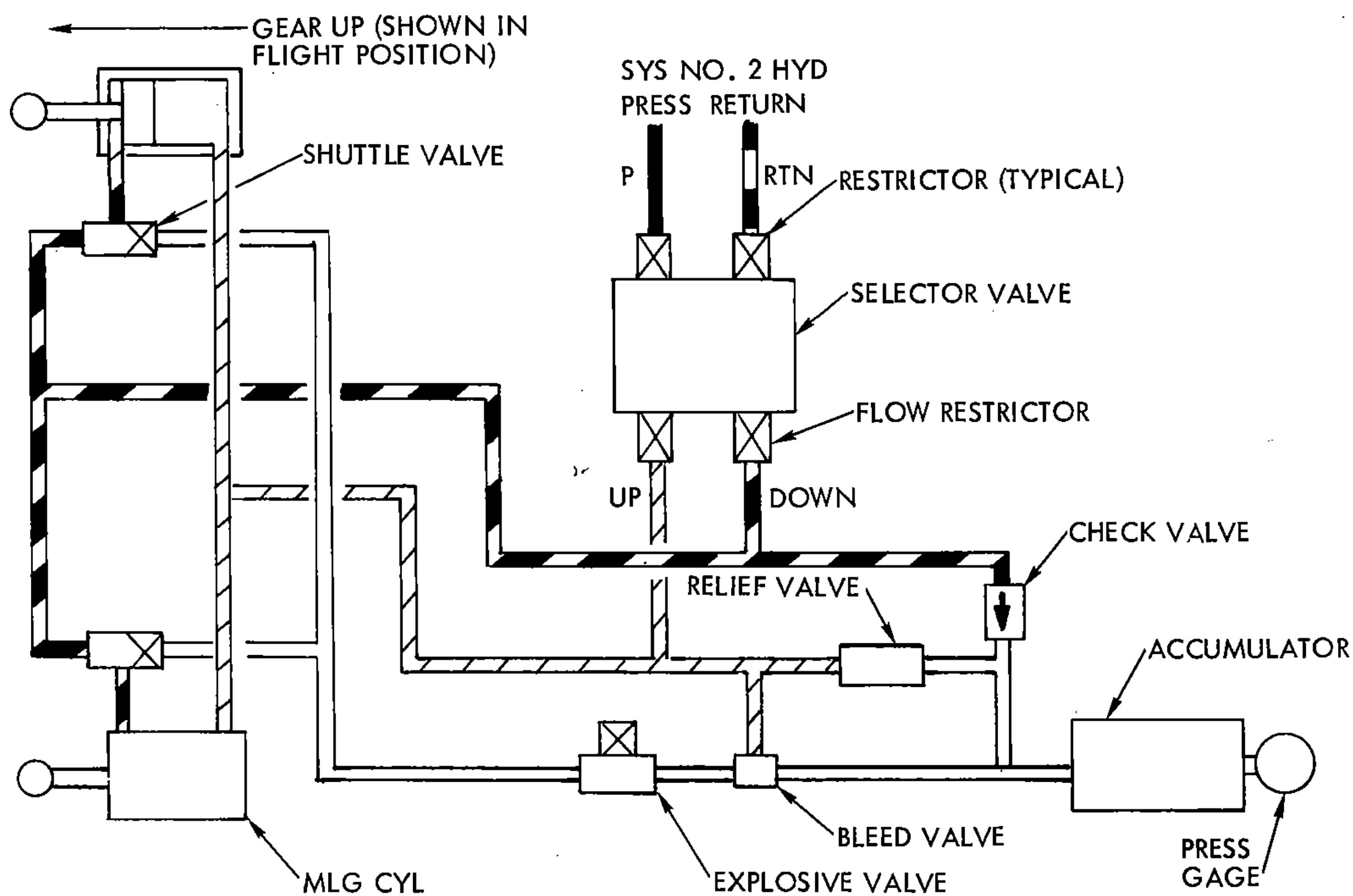


Figure 11-9. Main Landing Gear System

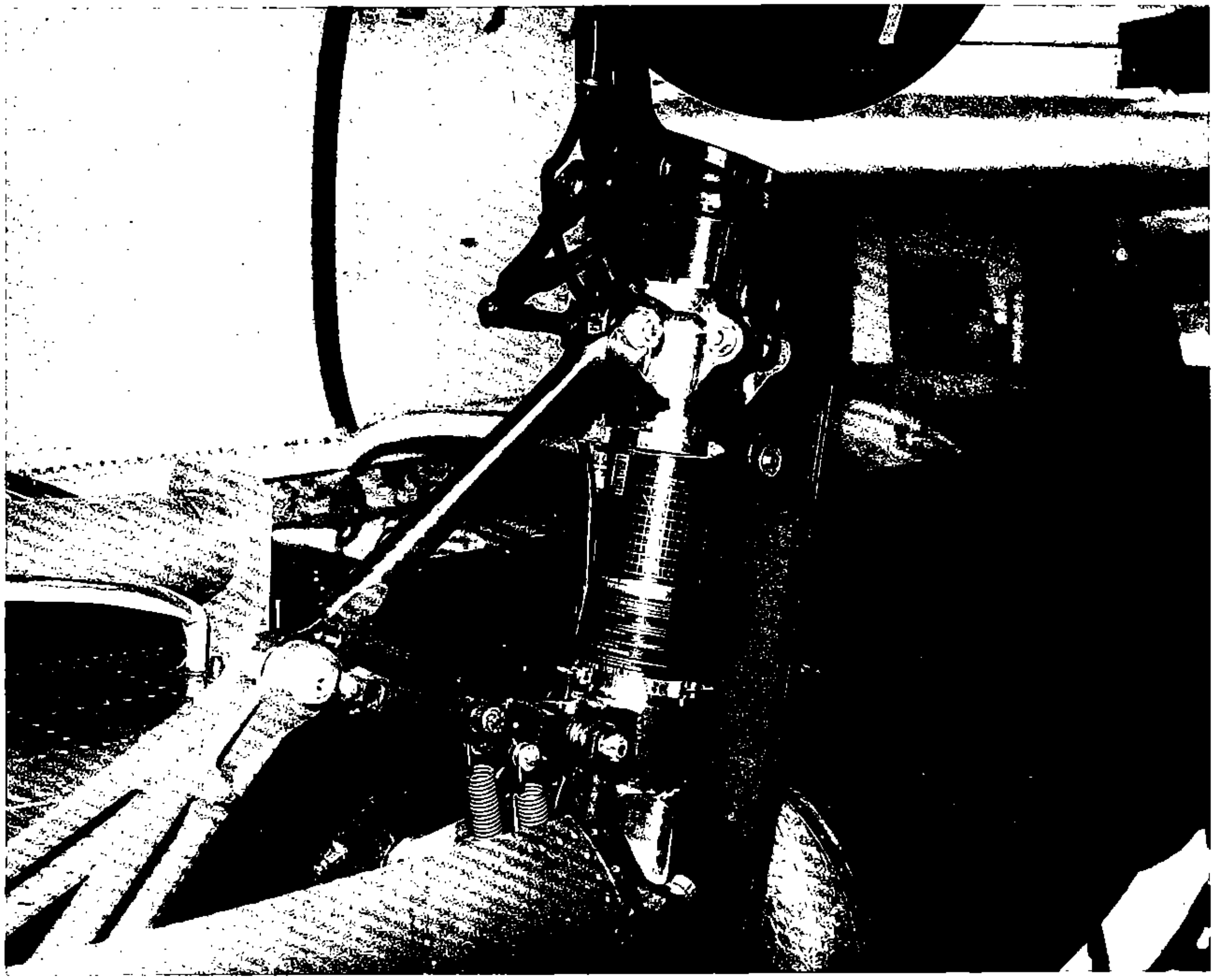


Figure 11-10. Main Landing Gear Strut

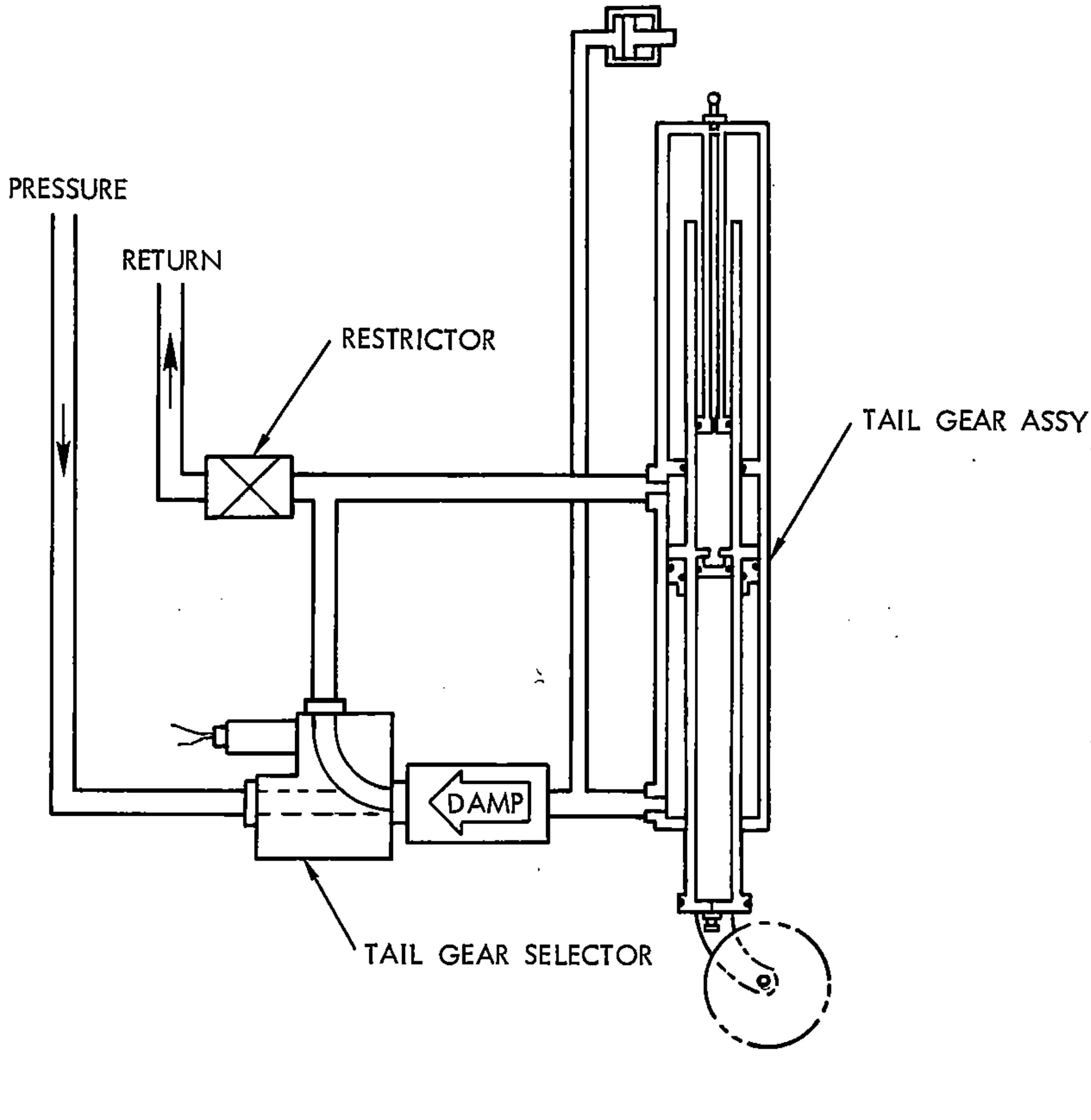


Figure 11-11. Tail Landing Gear Schematic

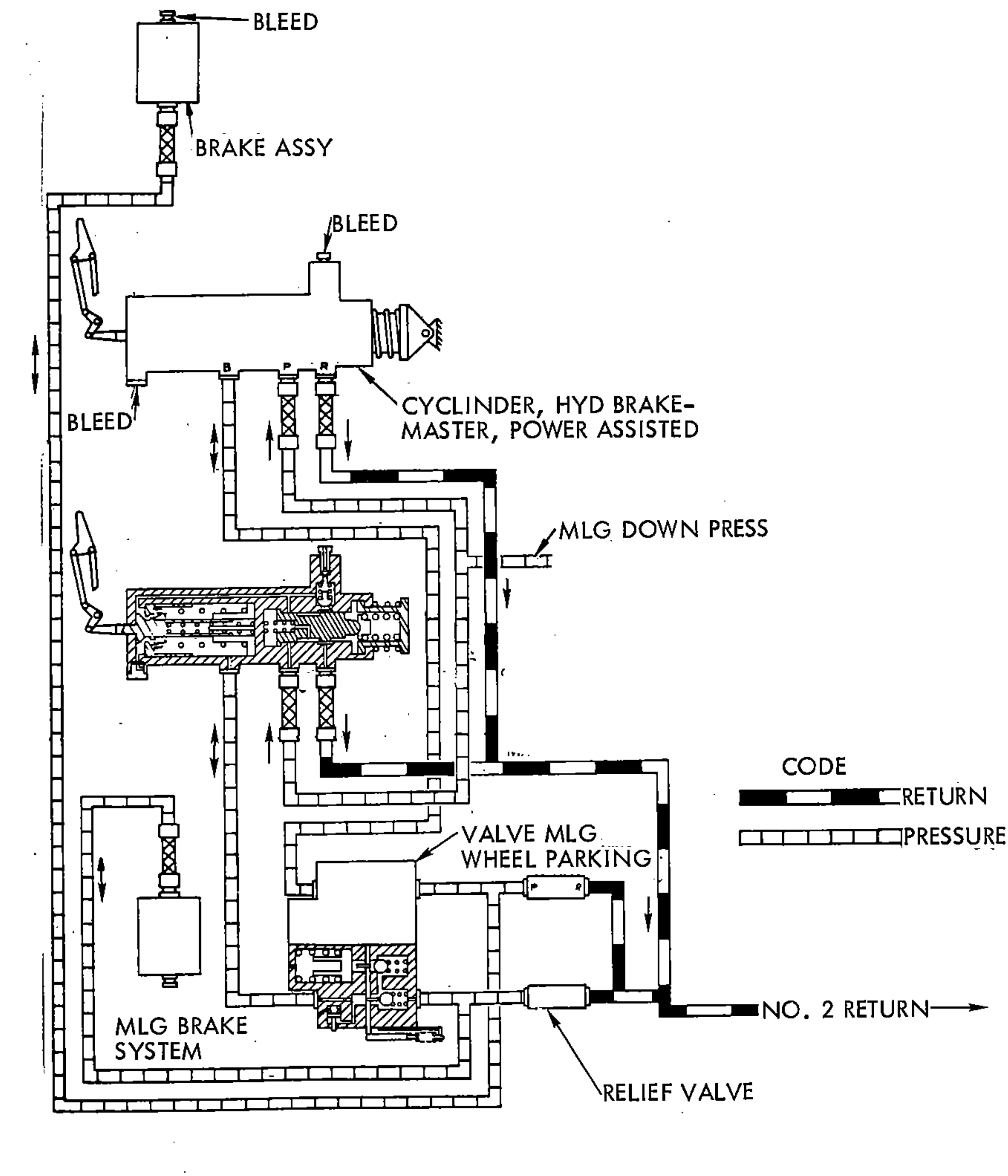


Figure 11-12. Wheel Brake System 1006 & 1007

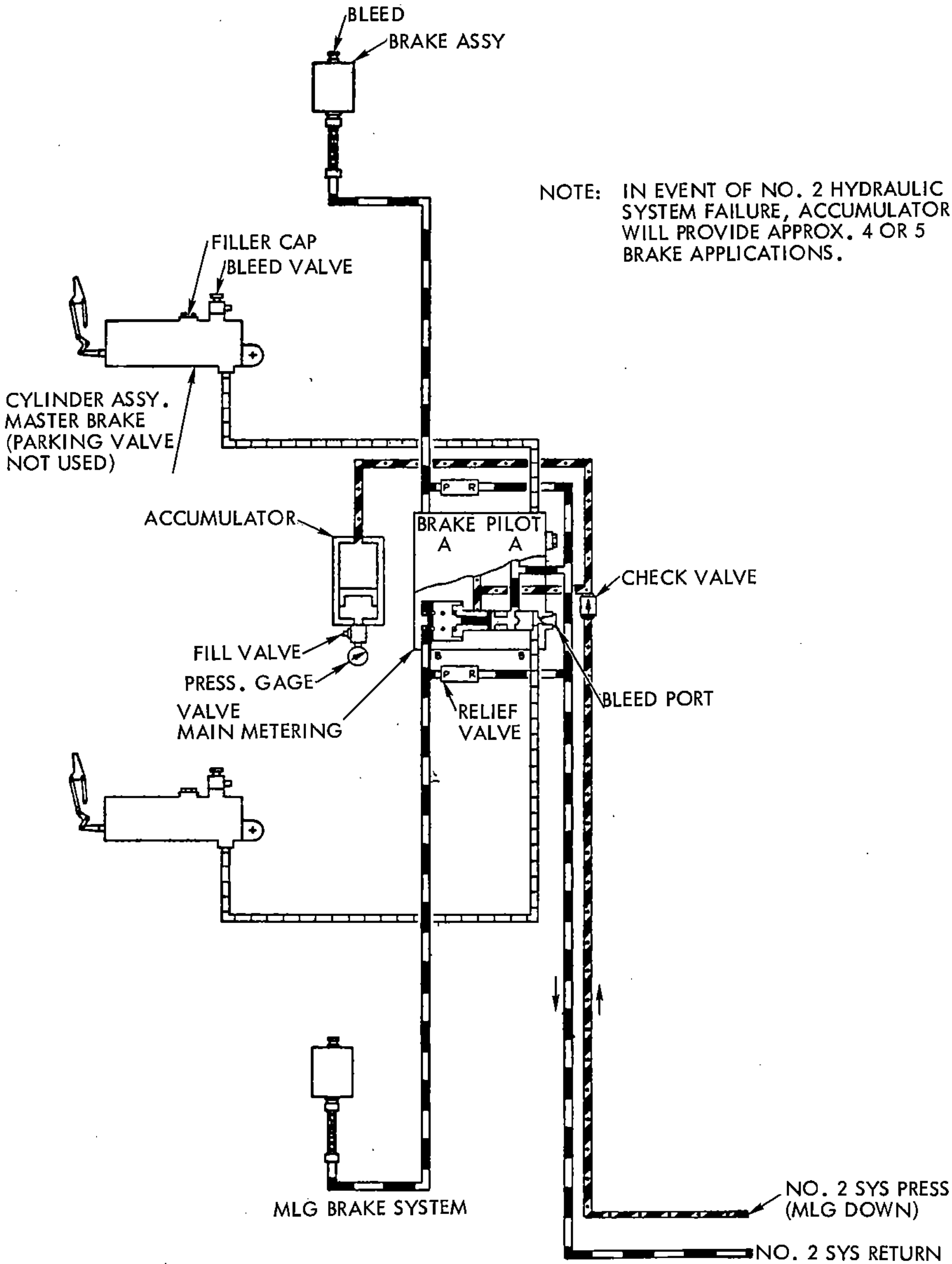


Figure 11-13. Wheel Brake System Ship 1009

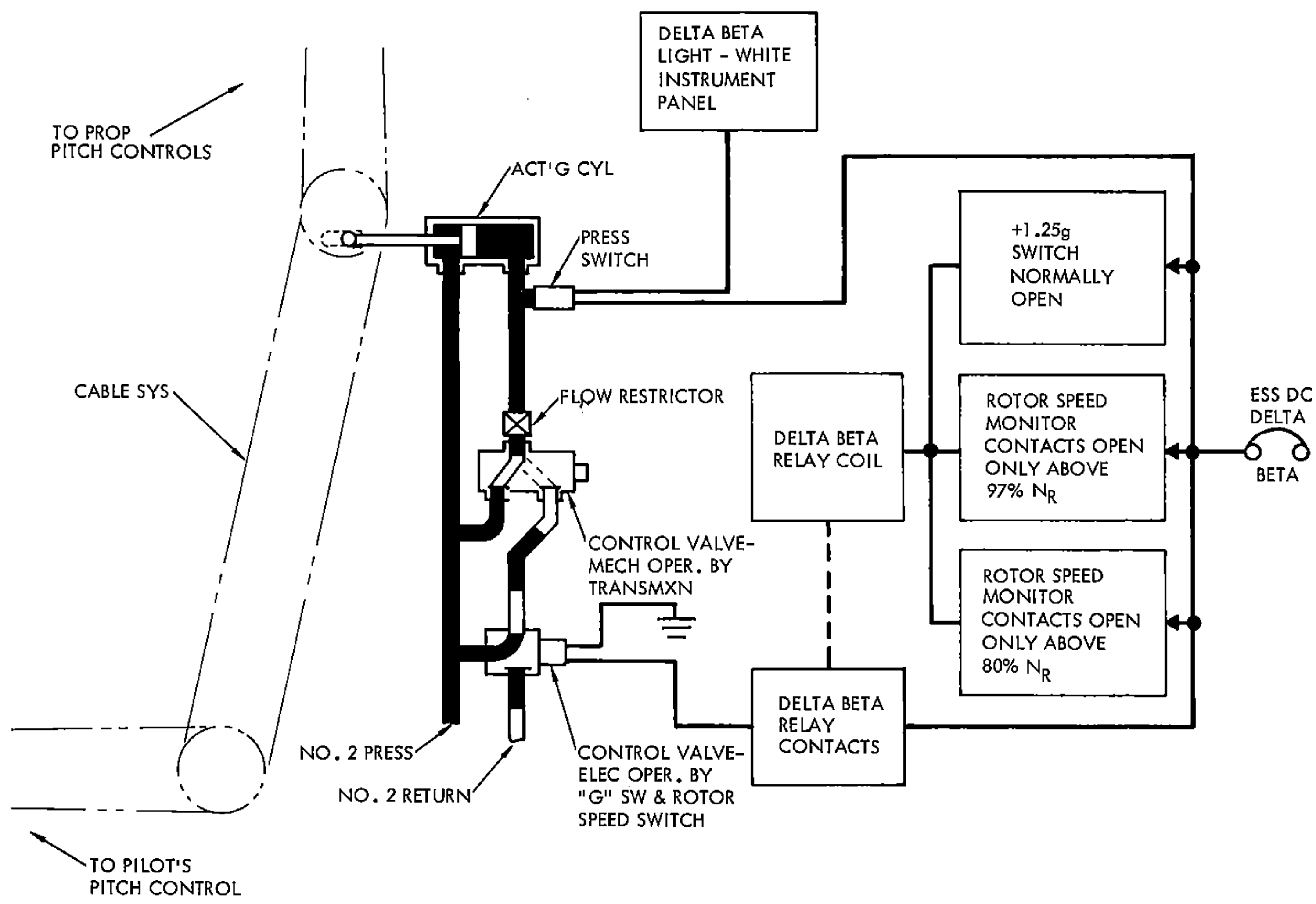


Figure 11-14. Delta Beta System

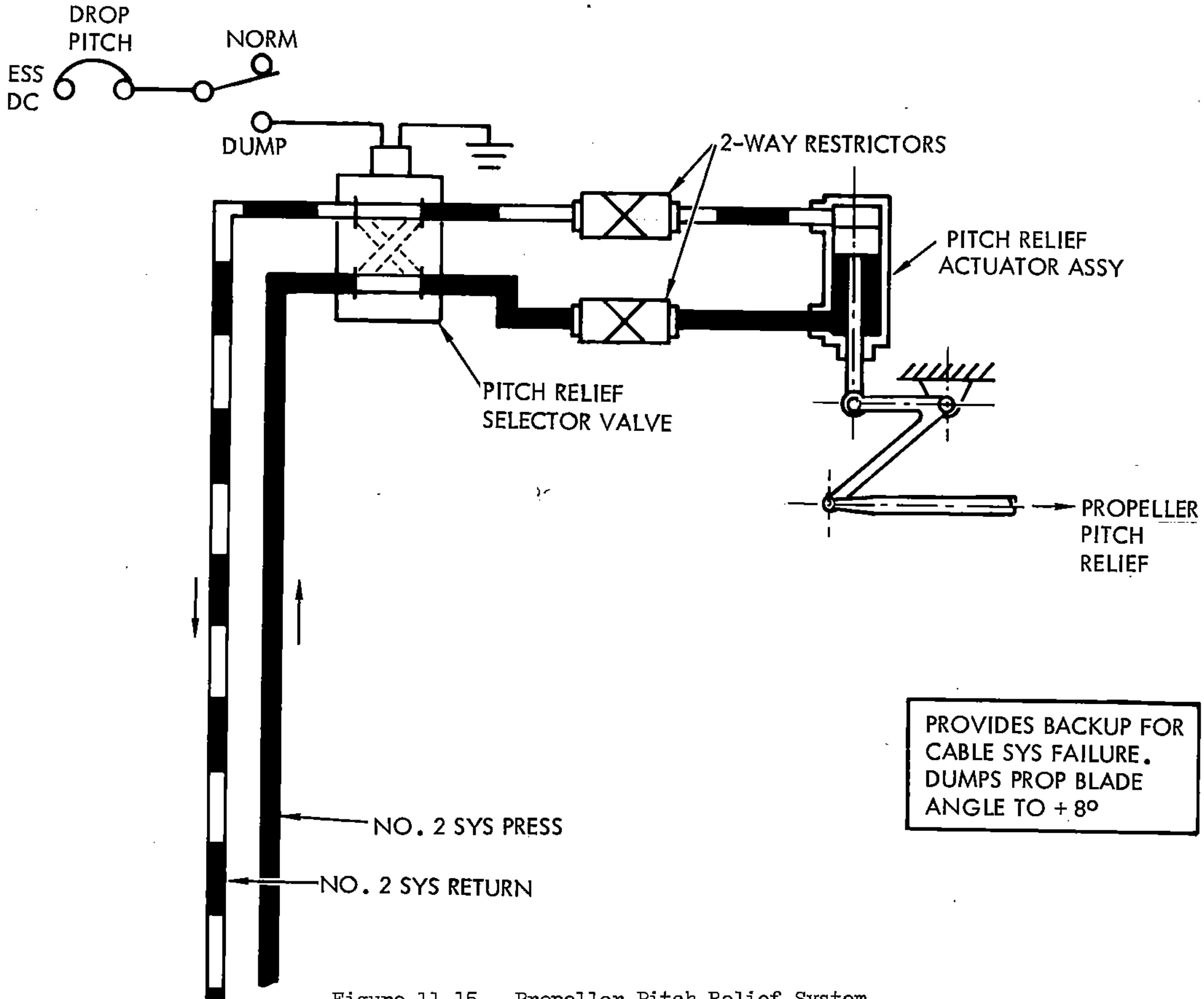
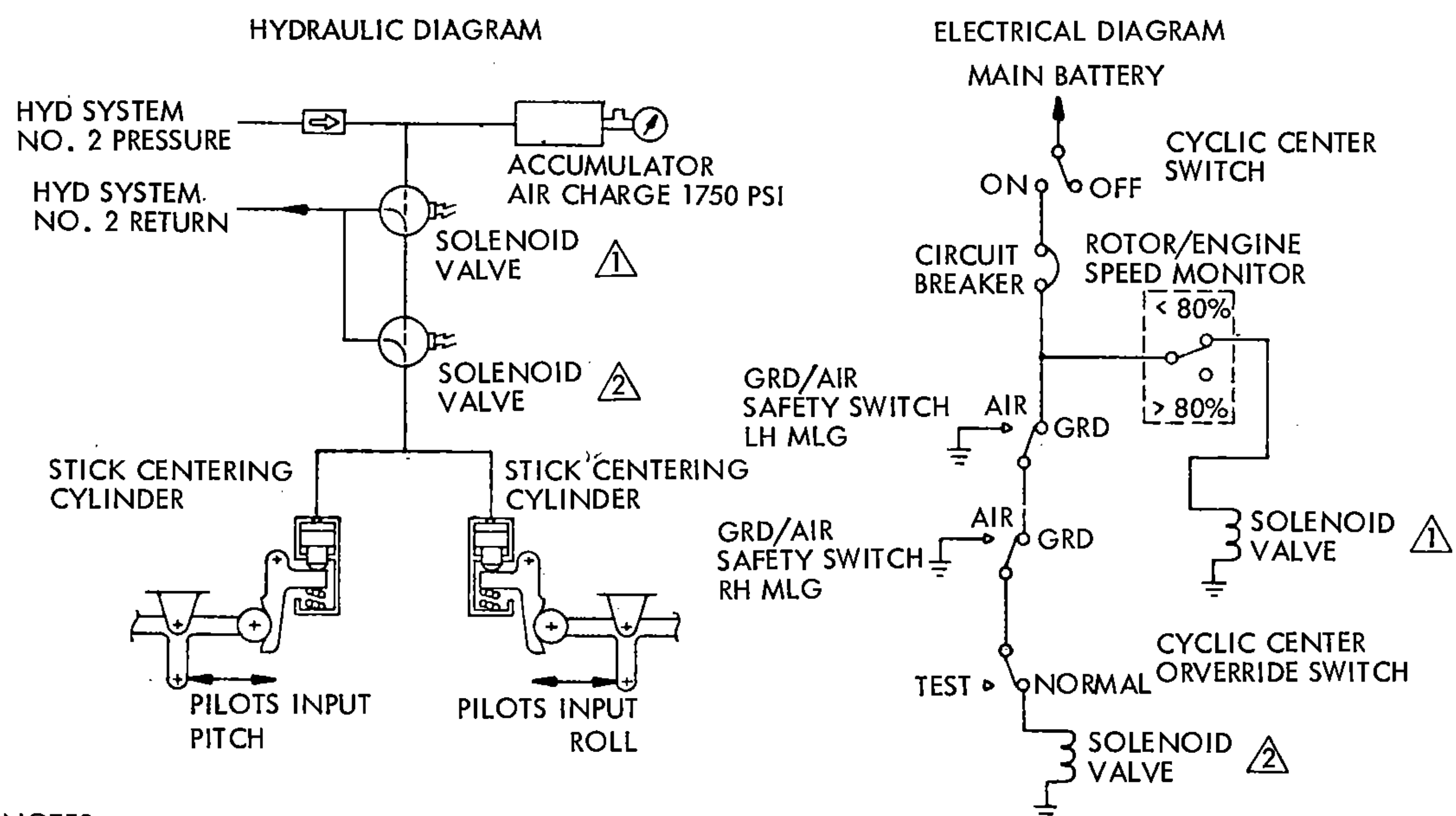


Figure 11-15. Propellor Pitch Relief System



NOTES:

1. ENERGIZED WHEN MAIN ROTOR IS BELOW 80%
2. ENERGIZED WHEN BOTH GROUND/AIR SAFETY SWITCHES ARE CLOSED

Figure 11-16. Stick Centering System

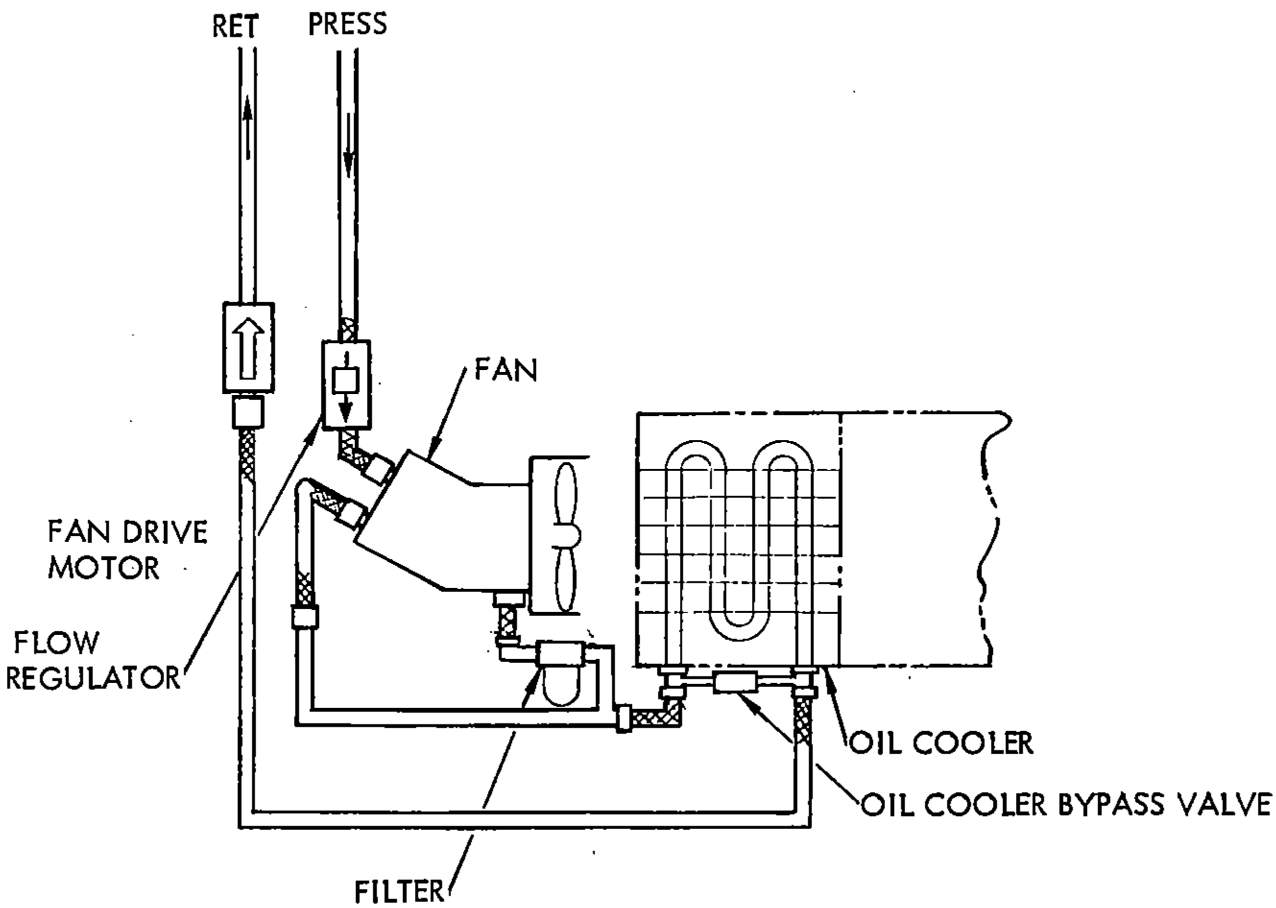


Figure 11-17. Oil Cooler System Schematic

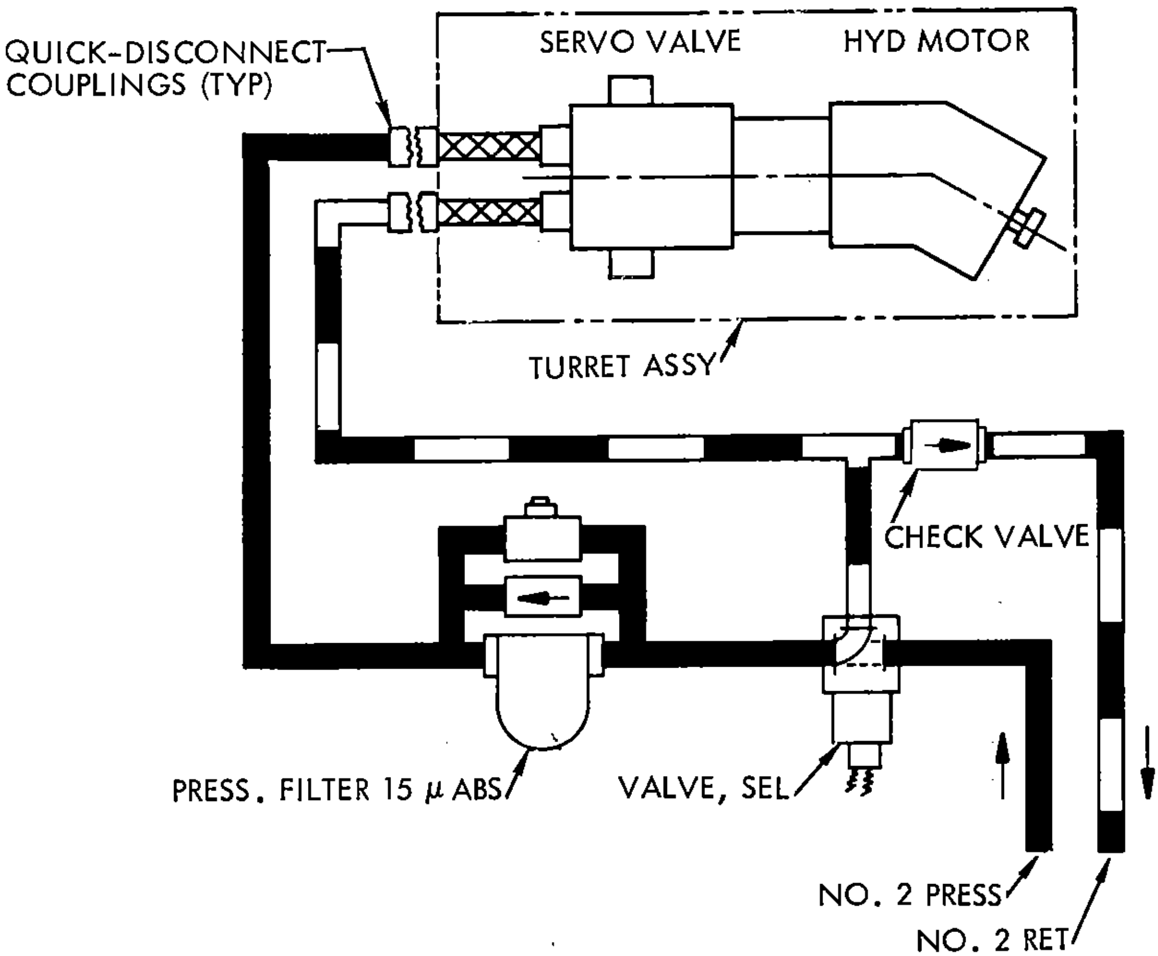


Figure 11-18. XM-52 Belly Turret Azimuth Drive System

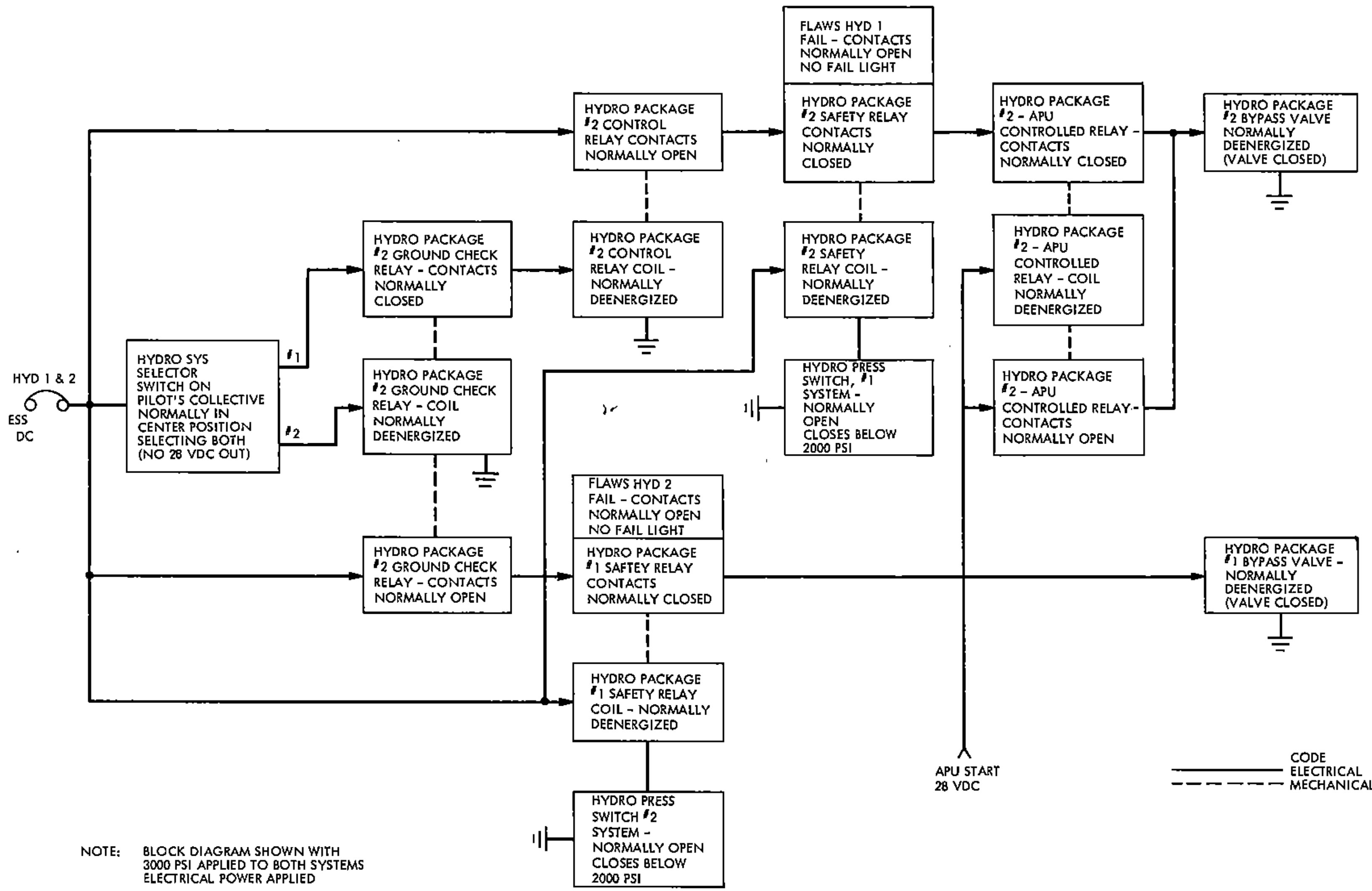


Figure 11-19. Hydraulic Pump Control

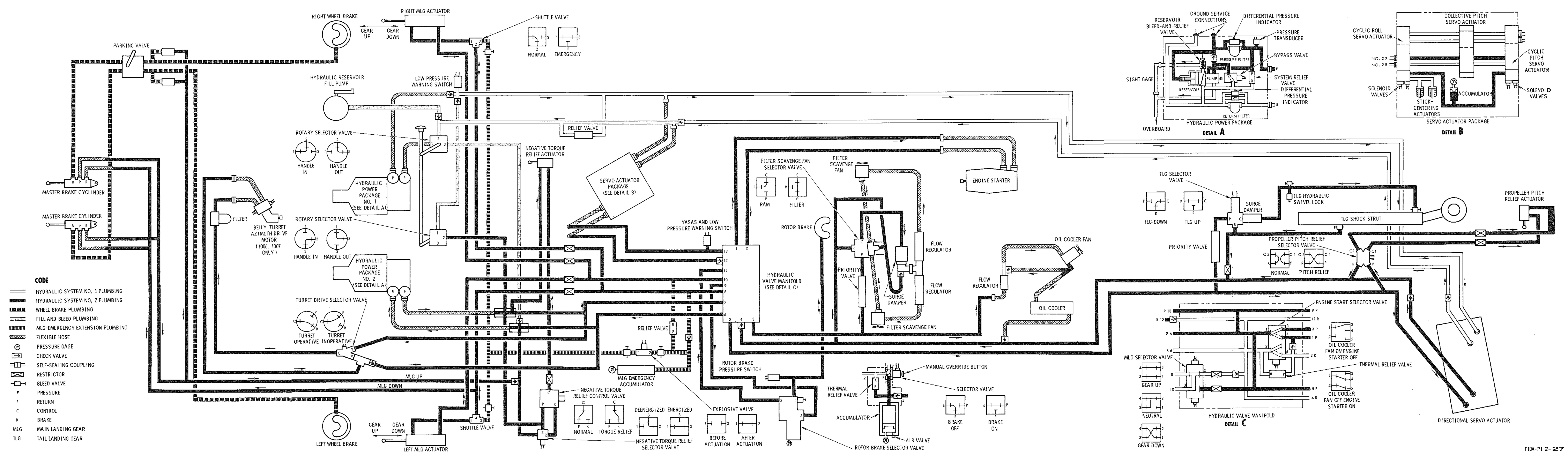


Figure 11-20. Hydraulic Power Supply Systems

ELECTRICAL POWER SYSTEM

I. GENERAL DESCRIPTION

The AH-56 A compound helicopter electrical power system consists of two AC generator power systems, a DC power system, an emergency power supply system and a sophisticated bus-type distribution system. Provisions have been made for the use of external power and for utilization of the Auxiliary Power Unit (APU) as power sources. System monitoring and fault location indicators have been designed into the system.

II. COMPONENTS AND LOCATIONS

Name of Component	Number per Aircraft	Location in Aircraft
No. 1 AC Generator	1	Transmission accessory area
No. 2 AC Generator	1	Transmission accessory area
No. 1 Regulator/Supervisory Panel	1	Electrical load center
No. 2 Regulator/Supervisory Panel	1	Electrical load center
Electric Power Panel	1	Pilots side console, left hand
External Power Monitor Panel	1	Electrical load center
No. 1 Transformer-Rectifier	1	Electrical load center
No. 2 Transformer-Rectifier	1	Electrical load center
Battery, Main Battery,	1	Battery and controls area
Battery, Auxiliary	1	Battery and controls area
Inverter	1	Battery and controls area

III. MAJOR COMPONENT DESCRIPTIONS

A. No. 1 AC Generator.

This is a four pole brushless generator, driven from the right hand transmission accessory drive. The generator is installed to a quick disconnect pad and is retained by a machined, marmon type clamp. Sealed type bearings support the rotor, and a two stage fan provides an airflow for cooling of the assembly. A five pin connector provides connection to the control circuits, and terminal posts are used for output attachment. The entire assembly weighs 25 pounds.

B. No. 2 AC Generator.

This generator is identical to No. 1 AC generator except for installation on the left hand transmission accessory drive.

C. No. 1 Regulator/Supervisory Panel.

This is a solid state electrical assembly approximately 3" x 5" x 6" in size. It provides voltage control of the generator, and prevents generator output not within specified values from being applied to the bus system. One multi-pin receptacle provides means for electrical connections. Two BITE indicators provide indication upon fault; one for generator fault, the remaining one for regulator/supervisory panel fault. An elapsed time meter provides indication of accumulated operating time of the component.

D. No. 2 Regulator/Supervisory Panel.

This component is identical to the No. 1 regulator/supervisory panel.

E. Electric Power Panel.

This is an indirect illuminated, removable panel of the pilots left hand side console which provides control for the complete electrical system.

F. External Power Monitor Panel.

This is a solid state assembly containing three modules for monitoring of external power. A receptacle provides electrical connection.

G. No. 1 Transformer-Rectifier.

The assembly is cylindrical in shape, approximately six inches in diameter and thirteen inches in length. Internally the transformer is both wye and delta connected and wound on a wye shaped core for efficient cooling. Twelve diodes are provided for rectifying the AC to DC. An integral fan provides air flow for cooling. A sensor in the assembly provides a signal in the event of an output failure or an overheat condition. The AC input to the transformer-rectifier is through a receptacle type connector and the DC output connectors are terminal studs.

H. No. 2 Transformer-Rectifier.

This transformer-rectifier is identical to the No. 1 transformer-rectifier, but in addition to its normal operational functions, it provides the source of DC power for recharging the battery.

I. Battery, Main.

The main battery is a nominal 24 VDC, 22 ampere-hour nickel cadmium storage battery. The 55 pound unit is installed on the maneuver gradient platform in the battery compartment. It is made up of 19 individual cells housed in a stainless stell case with a stainless steel cover. This cover is retained by snap fasteners. The electrical connections are made by means of two-pin male connecters.

J. Battery, Auxiliary.

This battery is a small 24 VDC, 0.3 ampere-hour unit with a self-contained AC powered trickle charger. This unit functions to provide emergency power for alternate engine control and for engine shut-off circuitry in the event of a total failure of all other DC power sources. The unit is maintained in a standby mode until the pilot manually selects the AUX position on the battery select switch.

K. Inverter.

The inverter is a 250 volt ampere motor generator 9 1/4 inches long, 6 1/4 inches by 4 3/4 inches wide. A solid state voltage regulator is housed on the inverter as an integral component. Two connectors, one for input, the other for output are installed in the housing. Also installed in the house are test jacks and a voltage adjustment. A frequency adjustment is mounted within the housing. A fan installed on the rotor provides movement of air for cooling. The inverter weighs 11.8 pounds and is installed adjacent to the battery on the maneuver gradient platform.

IV. SYSTEM OPERATION

A. Normal Operation

Two engine driven 20 KVA AC generators feed the respective AC buses through the Regulator/Supervisory panels and the generator contactors. Generator number 1 supplies power to the No. 1 Primary and Secondary AC buses, the essential AC bus, and the No. 1 transformer-rectifier which provides DC power to the No. 1 DC bus. Generator No. 2 normally supplies power to the No. 2 AC bus and to the No. 2 Transformer-Rectifier which supplies DC power to the No. 2 DC bus. The essential DC bus is supplied DC power from both the No. 1 and No. 2 DC buses. The main battery is connected to the main battery bus which is connected to the No. 2 DC bus through the battery contactor. The Auxiliary Battery is floating in a standby position during the normal operational mode.

B. Redundancy

System redundancy is provided within the design of the electrical power system by providing generators with sufficient capacity to power all AC buses from one engine driven/APU driven AC generator in the event of failure of the other generator. In addition, one transformer-retifier can power all DC buses through a pilot selected bus tie in the event of failure of the other transformer-rectifier.

C. Emergency

In case of the failure of both AC generators, a 250 volt ampere standby inverter turns on at about 94 percent Main Rotor RPM. At about 88 percent of Main Rotor RPM when the AC generators go off the line, the essential AC bus is transferred to the Standby Inverter. The Main Battery powers the Standby Inverter and provides essential DC power. Finally in the event of a failure of all main AC and DC power sources, an Auxiliary Battery is available, upon selection, to provide power to the engine alternate control system and to the engine emergency shut-off system.

D. Ground Operation

The Electrical power system can be operated on the ground in a Normal mode using the Auxiliary Power Unit (APU) to drive the two AC generators through the main transmission. In addition, the system can be operated from an external source through the external power receptacle. This power is controlled and monitored through the external power monitor panel.

When external power is connected to the aircraft bus system, the external power monitor panel removes this power if not within the range of 100 to 130 volts or 370 to 430 Hz. The panel prevents application of incorrect phase rotation to the system.

E. FLAWS

The pilot is provided visual, and when applicable, aural warning by the FLAWS system when a critical operation or failure occurs in operation of the electrical power supply system. The FLAWS, fault location and aural warning system, provides visual indication by use of a status panel, annunciator panels, and master caution lights.

The electrical power supply system provides signals to the FLAWS for five conditions:

(1) When the inverter is furnishing power to essential AC bus.

(2) When the No. 1 transformer-rectifier fails to furnish output or is operating in overheat condition.

(3) When the No. 2 transformer-rectifier fails to furnish output or is operating in overheat condition.

(4) When the No. 1 generator is not connected to the bus system.

(5) When the No. 2 generator is not connected to the bus system.

V. PCRS CONFIGURATION

The basic electrical system design as used in the Baseline Model is also used for the Approved PCRS Configuration except as follows:

- The Auxiliary (24 VDC, 0.3 Ampere-Hour) Battery has been deleted.
- The requirement for this redundant power source for emergency operation of the engine power control shaft operation has been eliminated by the installation of a mechanical control system for the engine power control shaft.

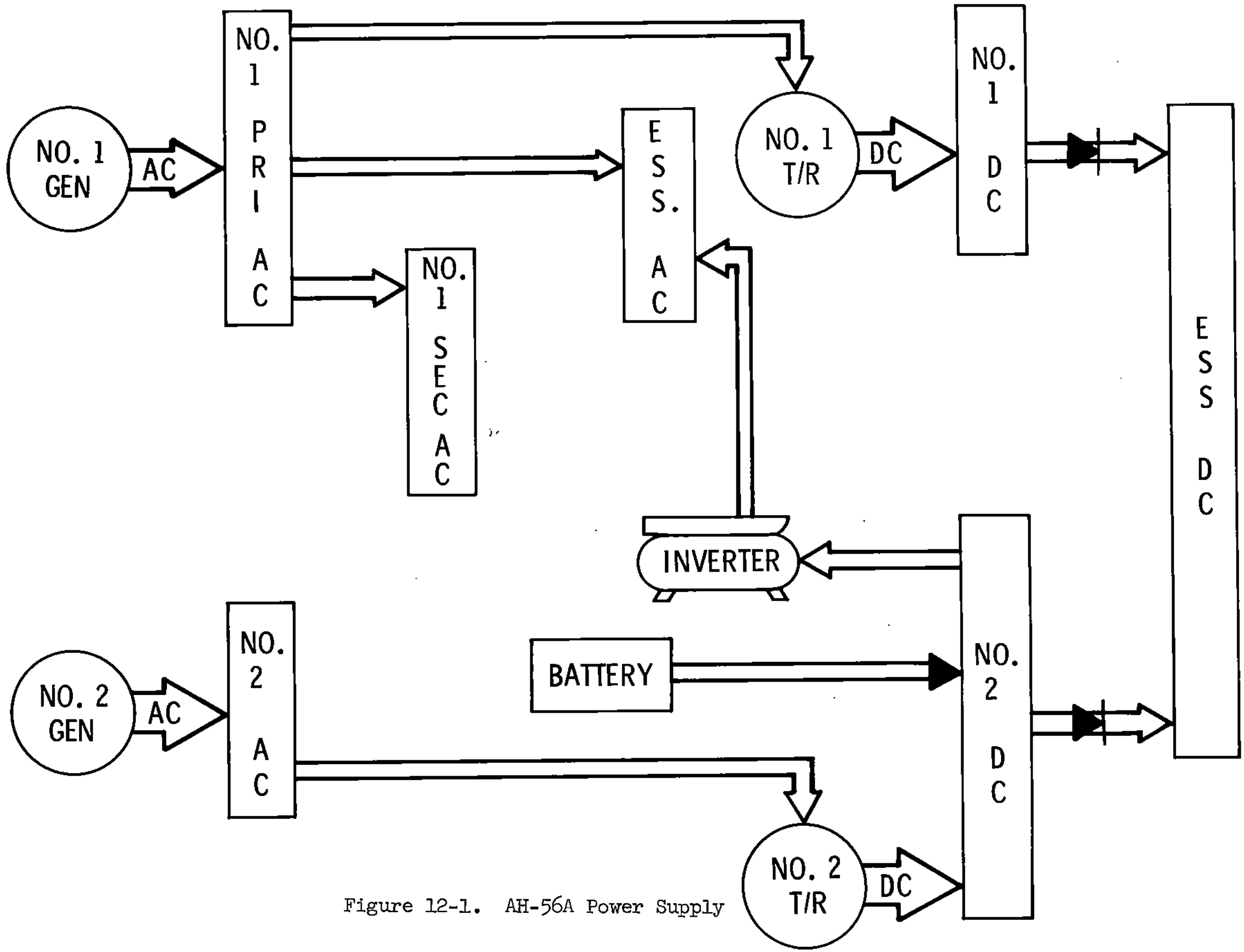


Figure 12-1. AH-56A Power Supply

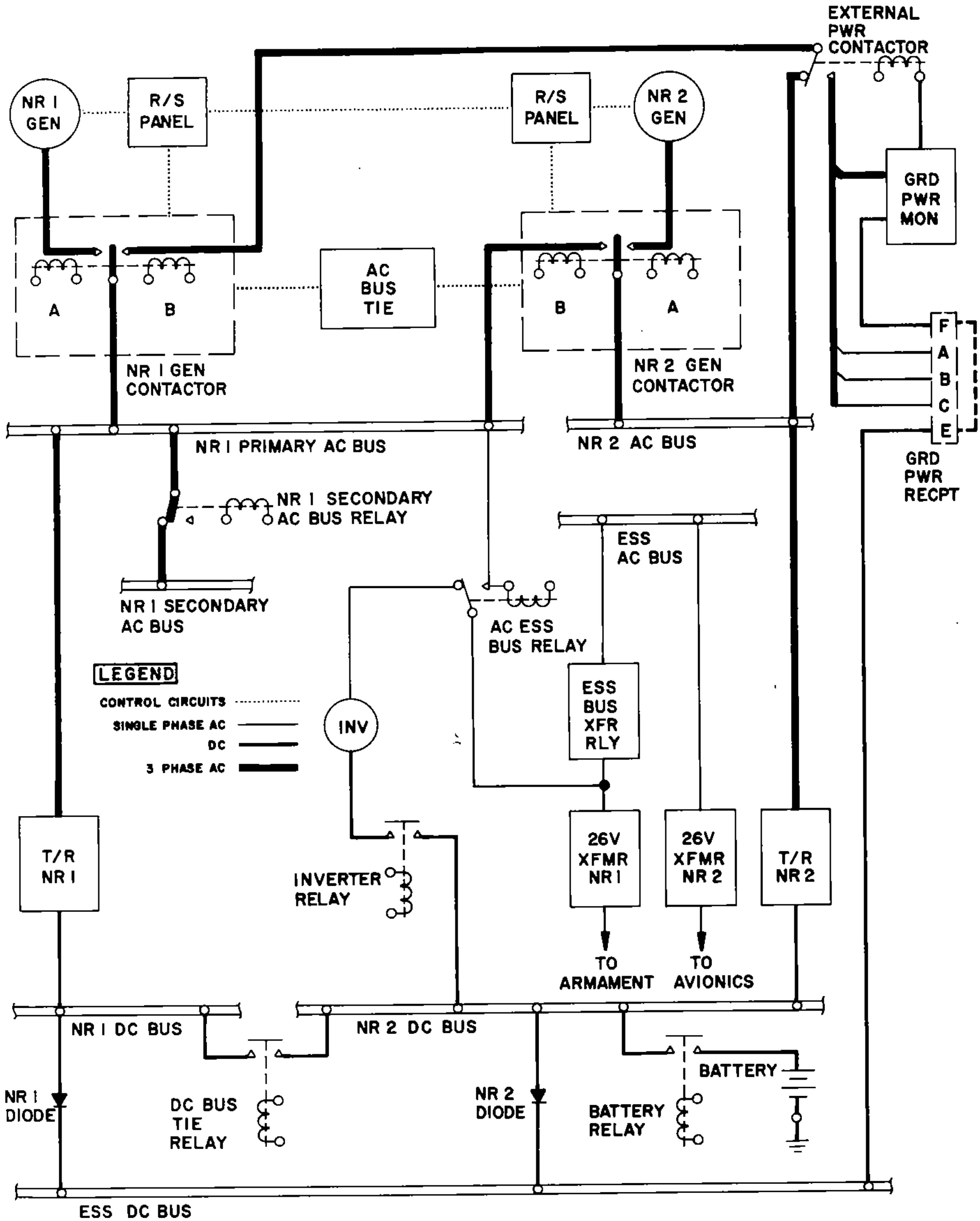
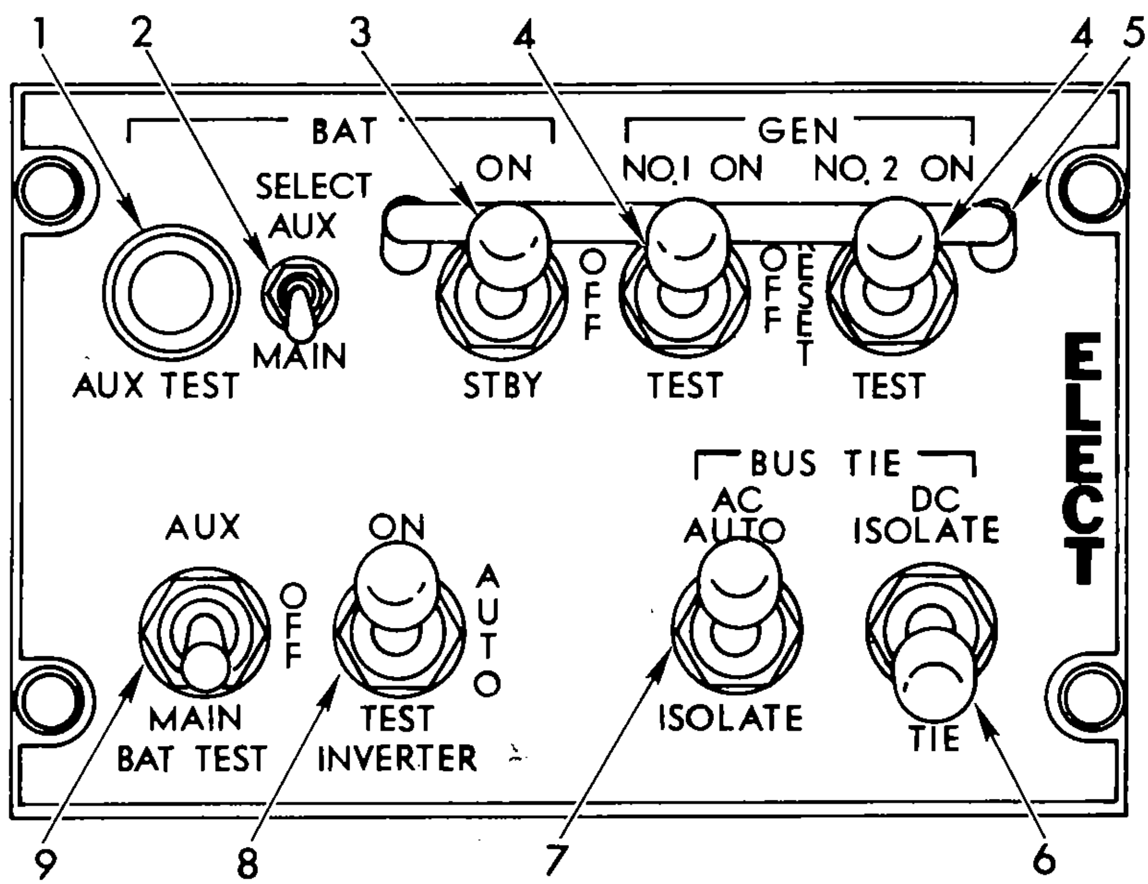


Figure 12-2. AC and DC Bus Diagram



1	Battery condition test light	6	DC bus tie switch
2	Battery select switch	7	AC bus tie switch
3	Battery switch	8	Inverter switch
4	Generator switch	9	Battery condition test switch
5	Power shutoff gang bar		

Figure 12-3. Electrical Power Control Panel

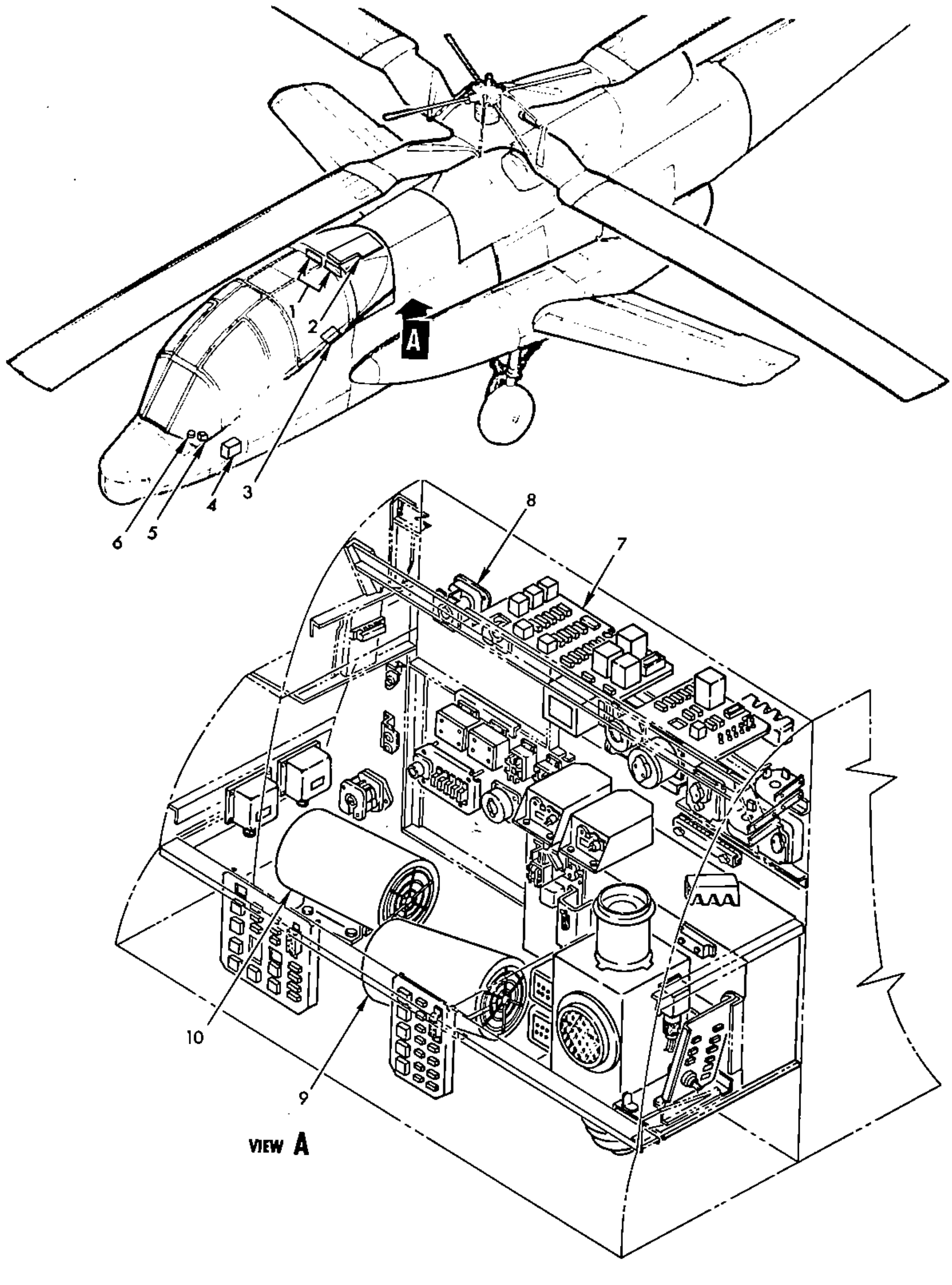


Figure 12-4. DC Power System Components

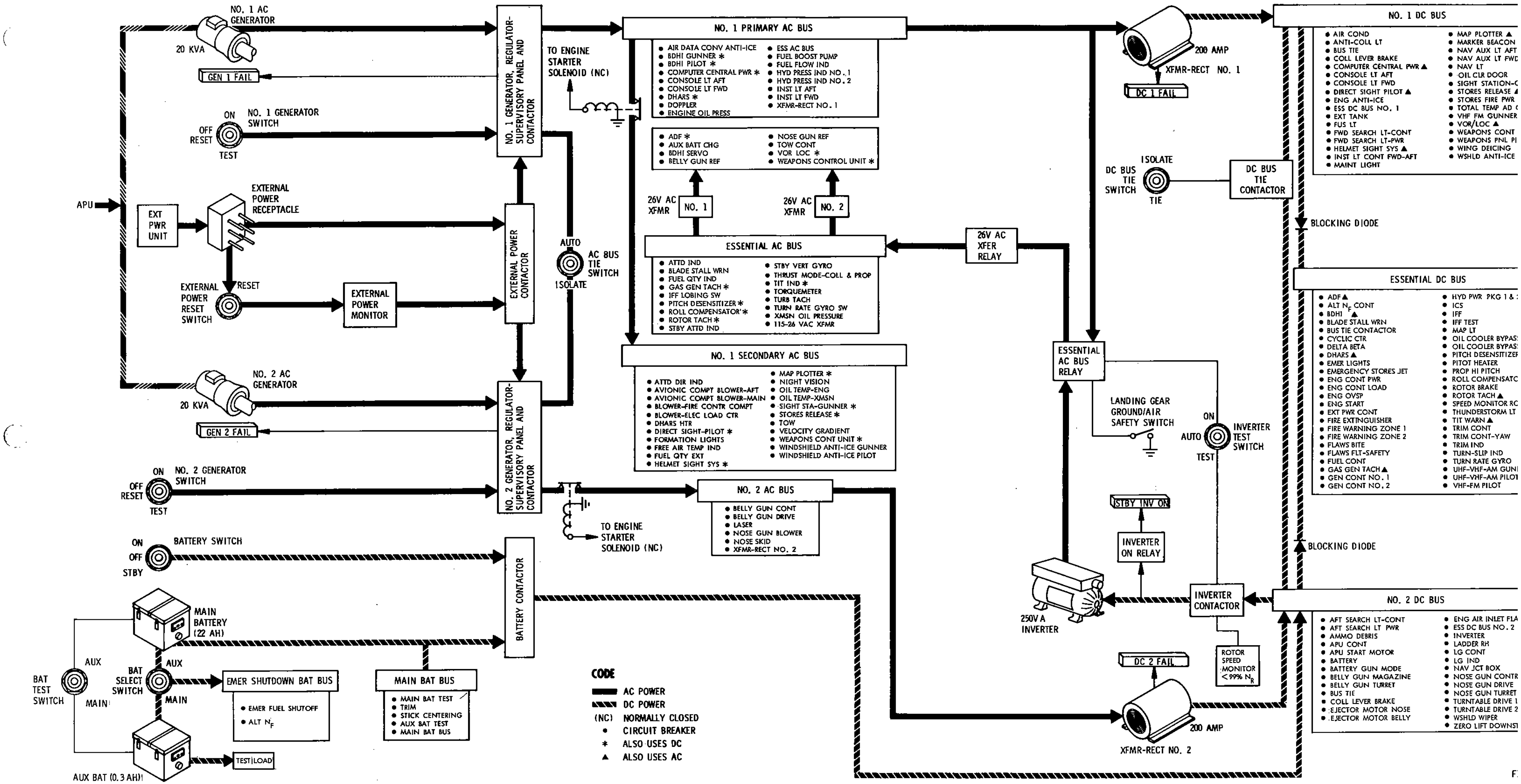


Figure 12-5. Electrical System Schematic

LOCKHEED CALIFORNIA COMPANY

LIGHTING

I. GENERAL DESCRIPTION

The AH-56 compound helicopter is equipped with interior and exterior lighting. Interior lighting is provided within each cockpit for illumination of instruments, consoles, push legend, high-intensity and utility lighting. The exterior lighting systems include, navigation, formation, anti-collision, fuselage and the forward and aft searchlights.

The pilot and copilot/gunner each have illumination intensity control of their instrument and control panel lights. The instruments incorporate integral lamps which provide red lighting of the dial. The control panels are edge lighted plastic panels with red lighting.

High intensity lights in both crew stations provide instrument lighting during a thunderstorm condition. A utility lamp is installed in each station to provide a portable source of light with controlled intensity. Five maintenance lights, two domelights and three utility floodlights, are used to illuminate compartments in the aircraft by crewmen performing ground maintenance.

One navigation light is installed on each wingtip, one in the ventral fin trailing edge, and the fourth in the top of the empennage. Flash or steady operation can be selected as well as intensity controlled (bright or dim). A formation light is located in the top of each wing and one in the top of the fuselage. The top empennage navigation light can function as an additional formation light if the navigation lights are not in use. Each wingtip contains an anti-collision light. Two white fuselage lights and two controllable searchlights are also provided.

II. COMPONENTS AND LOCATIONS

Name of Component	Number per Aircraft	Location in Aircraft
Pilot Station Inerior Lighting Control Panel	1	Pilot's left console, aft panel

II. COMPONENTS AND LOCATIONS (Cont'd)

Name of Component	Number per Aircraft	Location in Aircraft
Copilot/Gunner Station Interior Lighting Control Panel	1	Copilot/gunner's left forward panel
Exterior Lights Control Panel	1	Pilot's left conole, aft
High Intensity Lights	3	Right & left side of pilot's canopy - left side of copilot/ gunners canopy
Utility Lights	2	Pilot's left console, aft and left side of copilot/ gunner
Maintenance Lights	5	Fwd ammo compartment, transmission accessory area, and aft avionics compartment extension lights plus battery compartment and aft ammo compartment dome lights
Navigation Lights	4	Right & left wingtip, ventral fin, and top of empennage
Formation Lights	3	Left & right upper wing surface, and top of fuselage
Anti-Collision Lights	2	Left & right wingtip
Fuselage Lights	2	Fuselage top and bottom
Searchlights	2	Bottom of fuselage - fwd & aft

III. MAJOR COMPONENT DESCRIPTION

A. Pilot Station Interior Lighting Control Panel

This panel contains a rheostat PUSH LEGEND switch to control the pushbutton-indicator legends. Rotating the switch clockwise increases legend brightness. A BDHI LEGEND toggle switch selects dim or bright illumination of legends on the BDHI instruments in both crew stations. The INSTRUMENTS rotating switch controls instrument light brightness. The HIGH INTENSITY toggle switch energizes and selects maximum or minimum brightness of thunderstorm lights. A rotating CONSOLES light switch selects light brightness of all console station lights.

B. Copilot/Gunner Station Interior Lighting Control Panel

This panel contains three rheostat light switches; INST, CONSOLES, and PUSH LEGEND. They function the same as their respective pilots station interior lighting control panel switches.

C. Exterior Lights Control Panel

This panel contains a conventional NAVIGATION lights FLASH-STEADY switch and a BRIGHT-DIM switch. Also, a COLLISION and a FUSELAGE ON - OFF switch. The fifth switch is a rheostat FORMATION light switch. It controls intensity of illumination of formation lights and upper navigation position light.

D. Utility Lights

Two detachable utility (map lights), equipped with extension cords are installed in the aircraft. One light is mounted on the aft end of the left console in the pilot station and the other in the copilot/gunner station on the left side aft of the seat. A rotary type switch on the end of the light has three positions: OFF, DIM, and BRIGHT for selection of light intensity.

E. Maintenance Lights

These lights provide illumination for inspection, maintenance and servicing in their pertinent work areas: fwd ammo compartment, aft ammo compartment, transmission accessory area, aft avionics compartment and battery compartment.

F. Navigation Lights

For night-time recognition, navigation position lights are installed on the aircraft. They consist of a red light mounted in the left wingtip, a green light mounted in the right wingtip, and two white lights, one in the lower trailing edge of the ventral fin and the other forward of the propeller on the aircraft centerline above the horizontal stabilizer.

G. Formation Lights

Three flush panel-mounted electroluminescent formation lights are mounted on the top panel of each wing inboard from the wingtip and on the fuselage top directly aft of the pilot.

H. Anti-Collision Lights

An oscillating-type anti-collision light is located in each wingtip.

I. Fuselage Lights

Two white lights are provided. One is mounted on the bottom of the fuselage aft of the sponsons. The other is mounted on the top of the fuselage aft of the main rotor mast.

J. Searchlights

Two searchlights are installed on the aircraft. The forward searchlight, controlled by the copilot/gunner, is located on the bottom of the fuselage forward of the periscope. The aft searchlight, controlled by the pilot, is located on the bottom of the fuselage aft of the Doppler antenna.

IV. SYSTEM OPERATION

In general, the pilot has control of the exterior lights and the lights in the aft flight station. The copilot/gunner has control of the instrument lighting in the forward station. Each crew member has control of an exterior searchlight.

V. PCRS CONFIGURATION

No change.

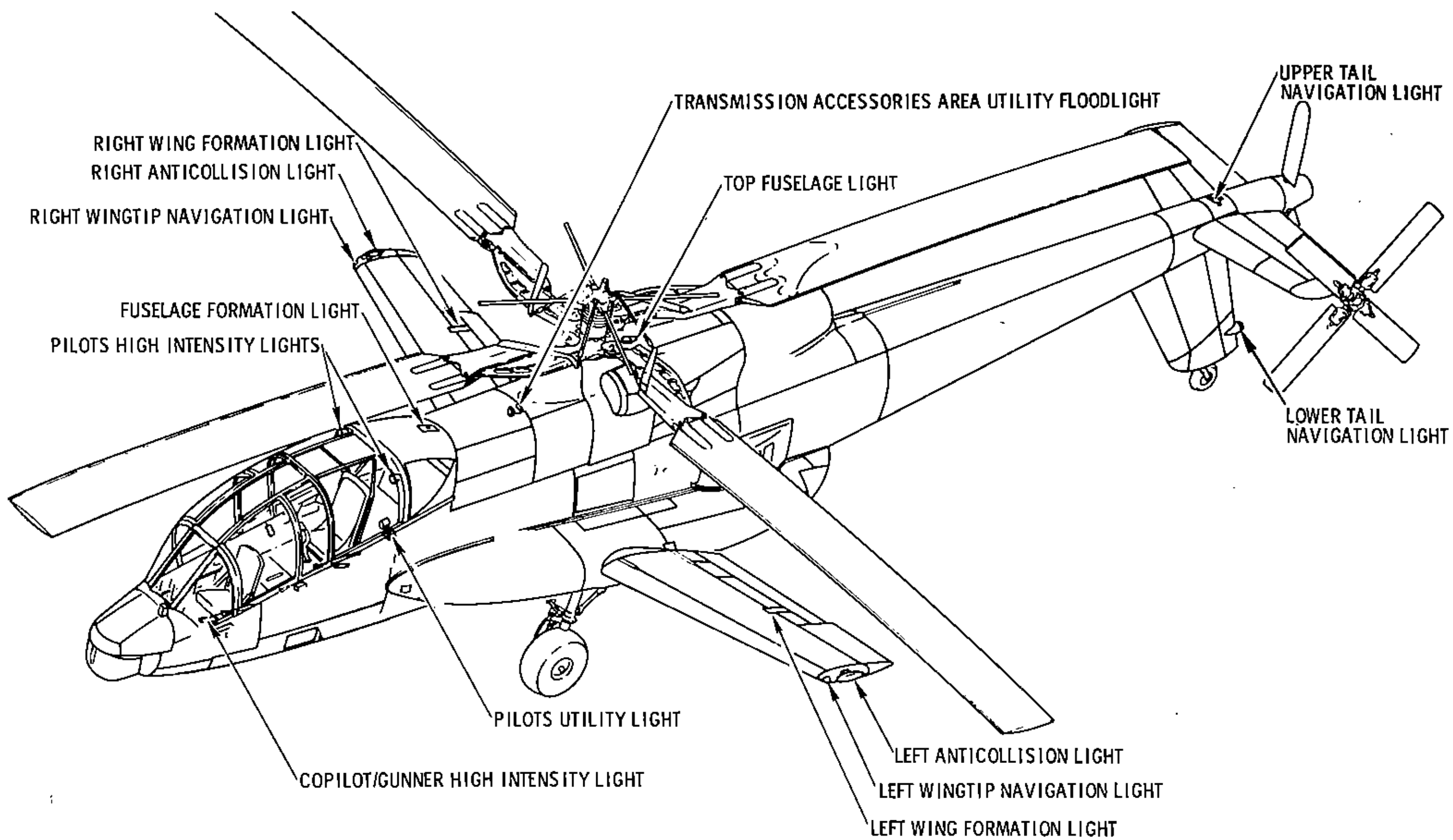


Figure 12B-1. Aircraft Lights (Sheet 1 of 2)

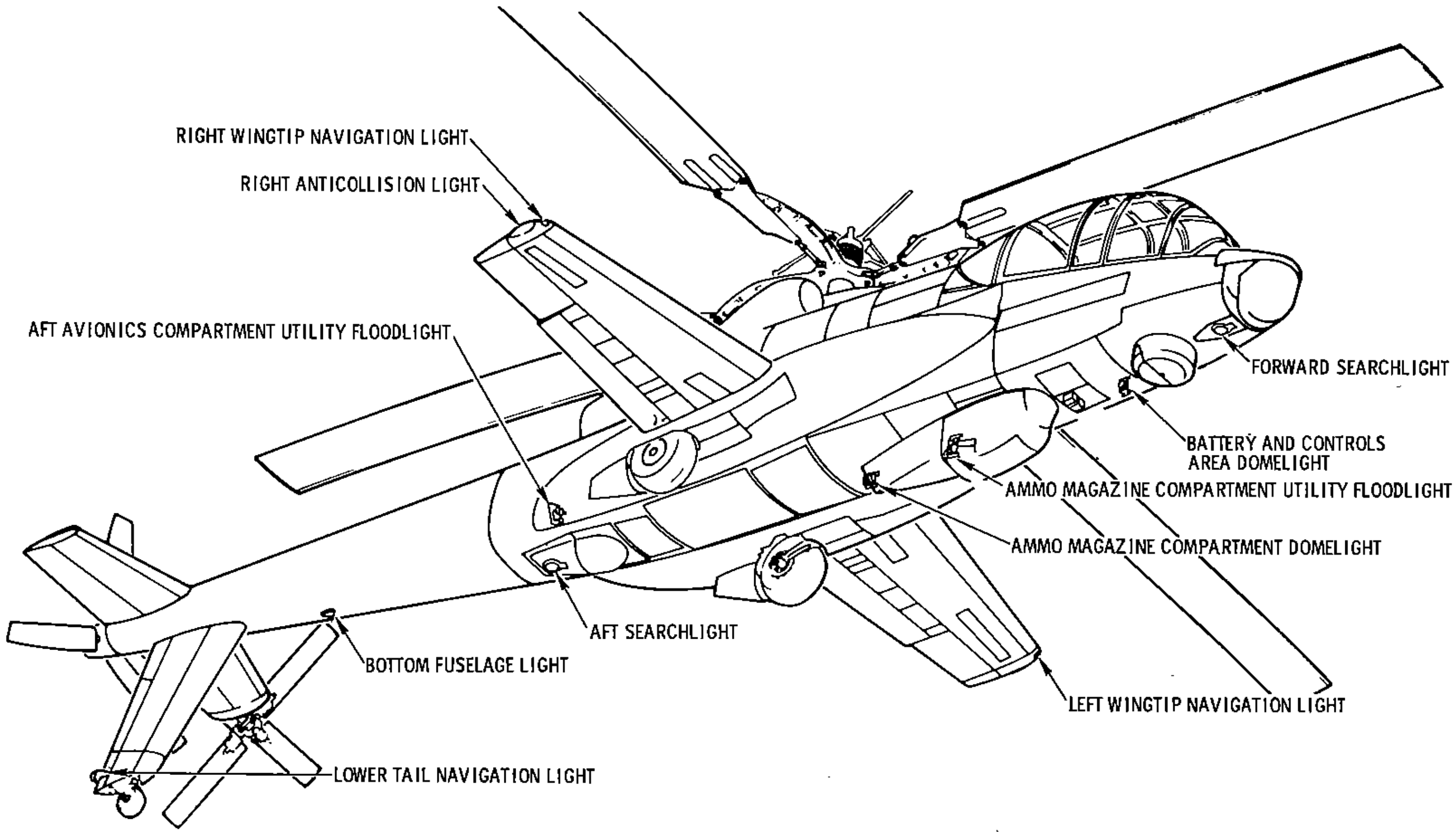


Figure 12B-1. Aircraft Lights (Sheet 2 of 2)

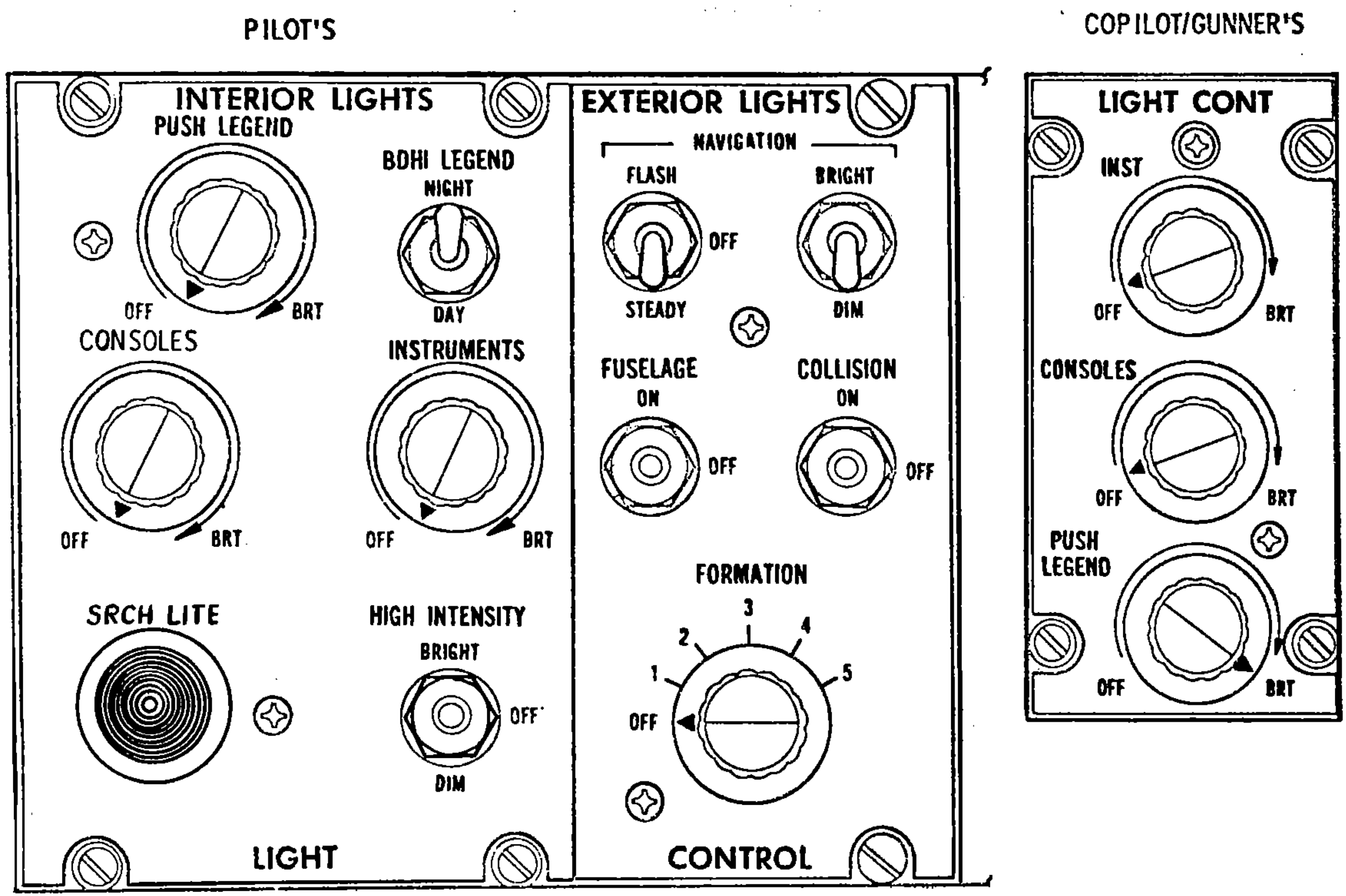


Figure 12B-2. Lighting Control Panels

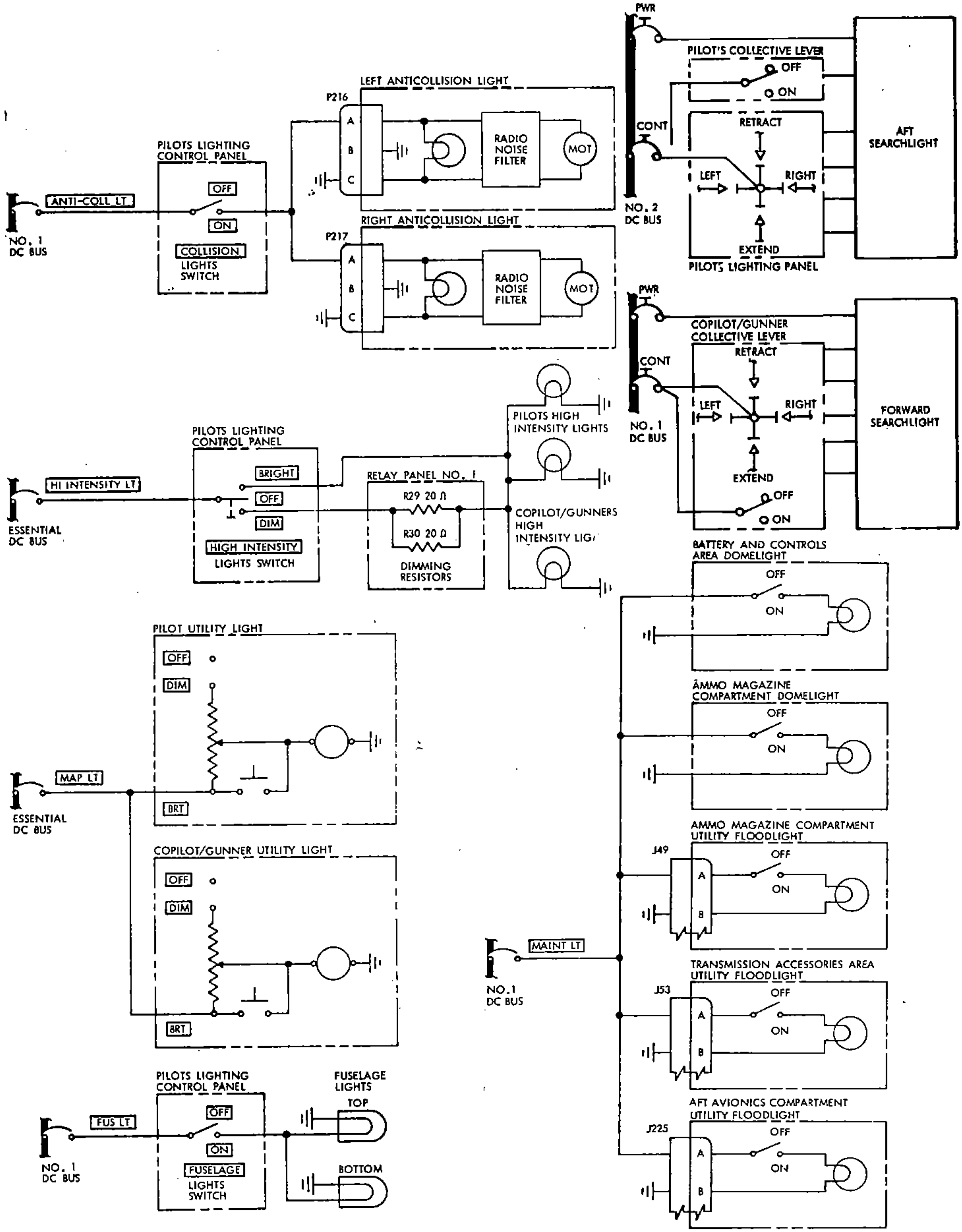


Figure 12B-3. Lighting Subsystems

INSTRUMENT SYSTEMS

I. GENERAL DESCRIPTION

The AH-56A compound helicopter incorporates instrument systems which provide the pilot and the copilot/gunner with much of the essential information necessary for efficient operations of the weapons and the aerial vehicle. The instruments are grouped into systems and subsystems which readily provide a readable and understandable display of the condition of selected critical operational parameters. The instruments may be classed as Flight and Navigation Instruments, Engine and Power Train Instruments, Operational Systems Instruments, and Miscellaneous Instruments.

II. COMPONENTS AND LOCATIONS

Name of Component	Number per Aircraft	Location in Aircraft
A. Flight and Navigation Instruments		
1. Attitude Director Indicator (ADI)	1	Pilot Instrument Panel
2. Turn and Slip Indicator	1	Copilot Instrument Panel
3. Bearing Distance Heading Indicator (BDHI)	2	Pilot Instrument Panel Copilot Instrument Panel
4. Standby Attitude Indicator	1	Pilot Instrument Panel
5. Radar Altimeter	2	Pilot Instrument Panel Copilot Instrument Panel
6. Barometric Altimeter	2	Pilot Instrument Panel Copilot Instrument Panel
7. Airspeed Indicator	2	Pilot Instrument Panel Copilot Instrument Panel
8. Vertical Velocity Indicator	2	Pilot Instrument Panel Copilot Instrument Panel

II. COMPONENTS AND LOCATIONS (Cont'd)

Name of Component	Number per Aircraft	Location in Aircraft
A. Flight and Navigation Instruments (Cont'd)		
9. Clock	1	Copilot Instrument Panel
10. Standby Compass	2	Pilot Flight Station Copilot Flight Station
B. Engine and Power Train Instruments		
1. Free Turbine and Main Rotor Tachometer	2	Pilot Instrument Panel Copilot Instrument Panel
2. Gas Generator Tachometer	2	Pilot Instrument Panel Copilot Instrument Panel
3. Engine Torquemeter	2	Pilot Instrument Panel Copilot Instrument Panel
4. Turbine Inlet Temperature Indicator	2	Pilot Instrument Panel Copilot Instrument Panel
5. Blade Angle Indicator	2	Pilot Instrument Panel Copilot Instrument Panel
6. Beta Angle Indicator	2	Pilot Instrument Panel Copilot Instrument Panel
C. Operational Systems Instruments		
1. Oil Pressure Indicator	1	Pilot Instrument Panel
2. Oil Temperature Indicator	1	Pilot Instrument Panel
3. Auxiliary Engine Oil Temperature Indicator	1	Pilot Instrument Panel
4. Hydraulic Pressure Indicator	1	Pilot Instrument Panel
5. Fuel Quantity Indicator	3	Pilot Instrument Panel Copilot Instrument Panel Fuselage Fueling Station

II. COMPONENTS AND LOCATIONS (Cont'd)

Name of Component	Number per Aircraft	Location in Aircraft
C. Operational Systems Instruments (Cont'd)		
6. Fuel Flow Indicator	1	Pilot Instrument Panel
D. Miscellaneous Instruments		
1. Accelerometer	1	Pilot Instrument Panel
2. Trim Position Indicator	1	Pilot Instrument Panel
3. Free Air Temperature Indicator	1	Pilot Instrument Panel
4. Auxiliary Power Unit (APU) Tachometer	1	Auxiliary Power Control Panel, Pilot Left Console
5. Auxiliary Power Unit (APU) Exhaust Gas Temperature (EGT) Indicator	1	Auxiliary Power Control Panel, Pilot Left Console

III. MAJOR COMPONENT DESCRIPTION

A. Flight and Navigation Instruments

The flight and navigation instruments are arranged to provide the pilot with full panel capability. The primary flight and navigation instruments are located on the pilot's panel so as to provide an immediate display of all pertinent flight information in the middle of the instrument panel and to simplify the pilot scan pattern for instrument flight operations. The copilot/gunner's instrument panel is arranged to provide a simplified scan pattern for flight capability sufficient to conduct VFR flight and limited instrument flight if required in an emergency.

1. Attitude Director Indicator

An attitude director indicator (ADI) provides the pilot with the following displays: pitch and roll attitude indications, turn rate and slip-skid indications, and three modes of aircraft steering information. Pitch and roll attitude indications and turn rate indications operate from signals supplied by the heading and attitude reference equipment in the Doppler heading attitude reference system (DHARS). Power is supplied by the No. 1 secondary ac bus.

2. Turn and Slip Indication

The copilot/gunner is provided an electrically operated turn and slip indicator. The indicator provides the rate and direction of a turn and skid of the aircraft.

3. Bearing, Distance, Heading Indicator

A bearing, distance, heading indicator (BDHI) is located on the instrument panel in the pilot and copilot/gunner stations. The BDHI receives directional and distance inputs from the DHARS, ADF radio set, VOR and ILS receivers, VHF FM homing, and the CCC. The instrument displays aircraft heading on a rotating compass card (directional input from the CCC) and relative bearing indications displayed on movable pointers (inputs from the navigation system). Power to both BDHI is from the essential dc bus through the BDHI circuit breaker, and from the No. 2 26-vac transformer through the BDHI servo circuit breaker. Additional power is provided to the BDHI in the pilot station from the essential ac bus through the BDHI PILOT circuit breaker, and to the BDHI in the copilot/gunner station from the No. 1 secondary ac bus through the BDHI GUNNER circuit breaker. This instrument will be discussed more fully in the section on Navigation Systems.

4. Standby Attitude Indicator

A standby attitude indicator provides the pilot with standby aircraft pitch and roll attitude indications. The indicator presents aircraft attitude through 360 degrees of roll and

±84 degrees of pitch. The indicator display consists of a fixed miniature aircraft, a bank angle pointer, bank angle indices, a movable attitude sphere, and a power-off warning flag. Signals for operation of the indicators are supplied by the standby vertical gyro in the heading and attitude reference system. The indicators receive power from the essential ac bus.

5. Radar Altimeter

Accurate indications of absolute aircraft altitude over land and water is provided by radar altimeters. The altimeters operate from Doppler radar inputs from the DHARS. Each altimeter is calibrated from zero to 5000 feet and contains a pointer on the bezel of the instrument which may be positioned to select a specific altitude. Two light displays are associated with the radar altimeters to advise of information source and to warn of a low altitude condition. The altimeters receive power from the No. 1 secondary ac bus.

6. Barometric Altimeter

The pilot and the copilot/gunner are each provided a barometric type altimeter at their stations to indicate the altitude of the aircraft in feet above or below sea level. Static (atmospheric) air pressure is supplied to the indicators by a pitot-static system.

7. Airspeed Indicator

The pilot and copilot/gunner are each provided a differential pressure type airspeed indicator. The indicators indicate speed of the aircraft in relation to the air through which it is flying. The pitot-static system supplies the signals for operation to both indicators.

8. Vertical Velocity Indicator

The pilot and copilot/gunner are each provided a differential pressure type vertical velocity indicator to indicate rate of climb or descent in feet per minute. The pitot-static system provides atmospheric pressure to both indicators for operation.

9. Clock

The copilot/gunner is provided a spring driven chronoflight type clock. The clock has four hands, two to continuously indicate time and two elapsed-time hands which may be stopped and reset.

10. Standby Compass

The pilot and copilot/gunner are each provided a standby compass to indicate the magnetic heading of the aircraft. The compasses are used as emergency backup instruments in event of failure of other navigational instruments.

B. Engine and Power Train Instruments

The engine and power train instruments, which provide the pilot with readable indication of the operating condition of the engine, transmission, main rotor, and propeller, are all conveniently grouped to provide easy identification of irregular engine or power train performance. The copilot/gunner's instruments are arranged to provide the primary engine and power train parameters necessary to monitor performance for limited operations of an emergency nature.

1. Free Turbine and Main Rotor Tachometer

The pilot is provided a dual indicating tachometer indicator. One indication scale identified as N_f provides indication of engine power turbine speed, and the other scale identified N_r provides indication of main rotor speed. The copilot/gunner is provided a single scale indicator to indicate main rotor speed. A rotor speed magnetic pickoff provides the signal to operate N_r scale of the pilots indicator and the copilot/gunners indicator. The N_f scale of the pilots indicator is provided a signal from the engine torquemeter system. The circuit is furnished power from the essential ac bus and essential dc bus.

2. Gas Generator Tachometer

The pilot and copilot/gunner are each provided a gas generator tachometer to indicate engine gas generator speed. A gas generator pickoff furnishes the speed signal to both indicators and the circuit is supplied power from the essential dc bus and essential ac bus.

3. Engine Torquemeter

The pilot and copilot/gunner are each provided an engine torquemeter indicator to indicate amount of engine torque transmitted to the main transmission. A torquemeter magnetic pickoff provides the signals for operation. The circuit is provided power from the essential ac bus.

4. Turbine Inlet Temperature Indicator

The pilot and copilot/gunner are each furnished a power turbine inlet temperature indicator to indicate power (free) turbine inlet temperature. Fourteen engine installed thermocouples provide a signal for operation. The circuit is supplied power from the essential ac bus and essential dc bus.

5. Blade Angle Indicator

A blade angle indicator is located on each instrument panel. The indicator is a rectangular scale indicator with two independent indicating channels. One channel indicates propeller pitch (angle at which the propeller blades have been set by adjustment of the propeller pitch grips). The other channel indicates collective pitch (angle at which the main rotor blades have been set by adjustment of the collective levers). Both indicating channels operate from signals supplied by control position potentiometers which are mechanically linked to the propeller and collective pitch control system. Power is supplied to the indicators by the essential ac bus through two circuit breakers.

6. Beta Angle Indicator

A beta angle indicator is located on each instrument panel to provide an auxiliary means of determining the propeller blade angle (beta angle). The indicators convert synchro signals, from the propeller gearbox pulley, into indications of propeller blade angle shown in degrees. Scale range is from -20° to + 40°. Power is supplied by the 115/26 vac No. 1 transformer.

C. Operational Systems Instruments

The pilot's operational system instruments are conveniently grouped to provide a rapid and ready reference to the operational condition of each of the essential systems requiring an instrument/indicator display for effective monitoring. The copilot/gunner's instrument has the limited operational system monitoring. The only significant operational system instrument available is the internal fuel quantity indicator.

1. Oil Pressure Indicator

The pilot is furnished a dual oil pressure indicator to indicate engine oil and transmission oil pressure. An engine oil pressure transmitter provides a signal to operate the engine oil pressure indicator and a transmission oil pressure transmitter controls the transmission oil pressure indicator. The circuit is supplied power from the No. 1 secondary ac bus.

2. Oil Temperature Indicator

The pilot is furnished a dual oil temperature indicator to indicate engine oil temperature and transmission oil temperature. An engine oil temperature bulb provides a signal to operate the engine oil temperature indicator portion of indicator and a transmission oil temperature bulb provides a signal to operate the transmission oil temperature portion. The circuit is supplied power from the No. 1 secondary ac bus.

3. Auxiliary Engine Oil Temperature Indicator

The pilot is furnished an auxiliary oil temperature indicator to provide a means of determining engine oil temperature if temperature exceeds 150° C which is the maximum shown on the oil temperature indicator. The indicator has a range of 0° to 225° C and is selectable by using a switch on the pilot instrument panel. Power is supplied by the No. 1 dc bus.

4. Hydraulic Pressure Indicator

The pilot is furnished a dual hydraulic pressure indicator to indicate the pressure of the No. 1 and No. 2 hydraulic systems. A hydraulic pressure transducer on each hydraulic package provides a signal to operate the respective portion of the indicator. The circuit is furnished power from the No. 1 primary ac bus.

5. Fuel Quantity Indicator

The pilot is furnished an indicator for indication of fuel quantity. The indicator is a dual indicator, one portion for internal fuel quantity and the other for external fuel quantity. The copilot/gunners indicator indicates internal fuel quantity only. A float actuated dual potentiometer transmitter is installed in the left hand sponson tank, another in the main tank, and one in the right hand sponson tank. They provide the control signals to operate the indicators. The circuit is supplied power from the essential ac bus.

The main tank fuel quantity also incorporates a direct reading gage which may be read at the right side of the fuselage. The location permits quantity readings to be taken of internal fuel quantity from outside the aircraft during refueling operations.

6. Fuel Flow Indicator

The pilot is furnished a fuel flow indicator. This provides indication of rate of fuel flow being consumed by the engine. A fuel flow transmitter plumbed to the engine fuel supply provides the signals to operate the indicator. The circuit is supplied power from the No. 1 primary ac bus.

D. Miscellaneous Instruments

The miscellaneous instruments discussed below are some of the more familiar and more significant instruments which the pilot and copilot/gunner may require to provide a clear indication of weapon system conditions. Other types of indicators and displays will be covered in other sections of this training manual.

1. Accelerometer Indication

The pilot is provided an accelerometer to indicate the gravitational and acceleration forces acting along the vertical axis of the aircraft.

2. Trim Position Indicator

The pilot is provided a trim position indicator to indicate the trim settings of the cyclic roll, cyclic pitch and directional control systems. A potentiometer incorporated in a trim/feel spring actuator for each of the channels, furnishes a signal to operate the respective indication. The circuit is supplied power from the essential dc bus.

3. Free Air Temperature Indicator

The free air temperature indicator provides outside air temperature indications. A temperature bulb, located in the left side of the fuselage aft of the electrical load center, senses changes in the outside air temperature and transmits them to the indicator. Scale graduations on the indicator dial are from -50° to +50° C (-58° to +122° F) in increments of 5° C. Power is supplied the indicator by the No. 1 secondary ac bus.

4. APU Tachometer

An APU tachometer is installed in the pilots station to indicate percent of rpm operation of the APU. A signal is originated by an APU monopole transducer on the APU is passed to an APU speed and fault indicating control of the APU. This signal is converted and applied to operate the APU tachometer.

5. APU Exhaust Gas Temperature Indicator

An APU exhaust gas temperature indicator is installed in the pilots station to indicate the exhaust gas temperature of the APU. A thermocoupld on the APU provides a signal to operate the indicator and a resistor is incorporated in the circuit to adjust for correct circuit resistance.

IV. SYSTEM OPERATIONS

The discussion of the operations of the various electrical, mechanical, and barometric type instruments will be included within the section describing the specific subsystems for which the instruments provide display.

V. PCRS CONFIGURATION

Flight testing, development and producibility cost reduction studies have resulted in the recommendation that all but one rectangular scale instruments be replaced by round scale indicators. Following is a tabulation of insturments affected and their final approved PCRS configuration.

A. PILOTS COCKPIT

1.	ACCELEROMETER	DELETED
2.	THRUST MODE INDICATOR	REPLACED BY BLADE ANGLE INDICATORS
3.	FREE TURBINE & MAIN ROTOR TACHOMETER INDICATOR	REPLACED BY ROUND SCALE INDICATOR
4.	TORQUEMETER INDICATOR	REPLACED BY ROUND SCALE INDICATOR
5.	GAS GENERATOR TACHOMETER	REPLACED BY ROUND SCALE INDICATOR
6.	TURBINE INLET TEMPERATURE INDICATOR	REPLACED BY ROUND SCALE INDICATOR
7.	FUEL FLOW INDICATOR	REPLACED BY ROUND SCALE INDICATOR
8.	OIL PRESSURE INDICATOR	REPLACED BY SECONDARY SYSTEMS PANEL
9.	OIL TEMPERATURE INDICATOR	REPLACED BY SECONDARY SYSTEMS PANEL
10.	HYDRAULIC PRESSURE INDICATOR	REPLACED BY SECONDARY SYSTEMS PANEL
11.	FUEL QUANTITY INDICATOR	REPLACED BY SECONDARY SYSTEMS PANEL
12.	FREE AIR TEMPERATURE INDICATOR	REPLACED BY SECONDARY SYSTEMS PANEL
13.	TRIM POSITION INDICATOR	REPLACED BY A ROUND YAW TRIM POSITION INDICATOR
14.	BEARING DISTANCE HEADING INDICATOR	REPLACED BY A SMALLER INDICATOR
15.	ATTITUDE DIRECTOR INDICATOR	REPLACED BY A SMALLER INDICATOR

B. COPILOT/GUNNERS COCKPIT

1.	CLOCK	DELETED
2.	THRUST MODE INDICATOR	REPLACED BY BLADE ANGLE INDICATOR

B. COPILOT/GUNNERS COCKPIT (Cont'd)

3.	FREE TURBINE & MAIN ROTOR TACHOMETER INDICATOR	REPLACED BY ROUND SCALE INDICATOR
4.	TORQUEMETER INDICATOR	DELETED - REPLACED BY A 100% TORQUE LIGHT
5.	GAS GENERATOR TACHOMETER	DELETED
6.	TURBINE INLET TEMPERATURE INDICATOR	REPLACED BY ROUND SCALE INDICATOR
7.	FUEL QUANTITY INDICATOR	DELETED
8.	BEARING DISTANCE HEADING INDICATOR	REPLACED BY A SMALLER INDICATOR
9.	RADAR ALTITUDE INDICATOR	DELETED

C. PCRS Components Added

1. Secondary Systems Panel

a. Engine Oil

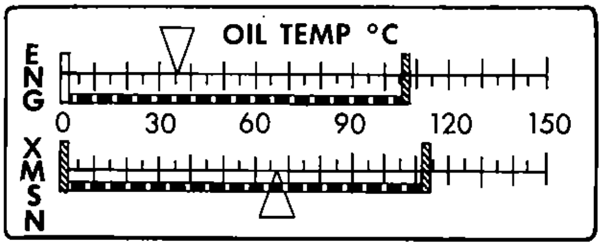
The pilot is furnished a dual oil pressure and temperature indicator to indicate the condition of the engine oil system. It is located on the pilot's right hand console. The engine oil transmitter provides a signal to operate the pressure indicator and the engine oil temperature transmitter provides a signal to operate the engine oil temperature portion of the indicator. Electrical power is supplied to the circuit from the No. 1 secondary ac bus.

b. Transmission Oil

The pilot is furnished a dual oil pressure and oil temperature indicator to indicate the condition of the transmission lubrication system. A transmission oil temperature bulb provides a signal to operate the temperature section of the indicator and a transmission oil pressure transmitter provides a signal to operate the oil pressure segment of the indicator. Electrical power is supplied to the circuit from the No. 1 secondary ac bus.

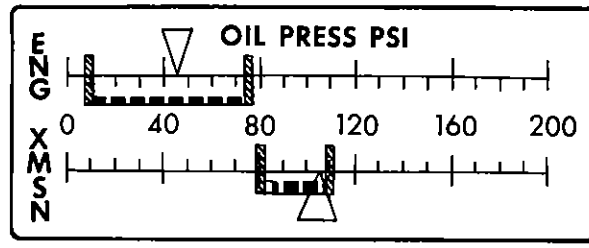
c. Hydraulic Pressure

The pilot is furnished a dual hydraulic system indicator to indicate the condition of each system. A separate pressure transducer in each hydraulic package provides a signal to operate each segment of the indicator. Electrical power is supplied from the No. 1 secondary ac bus.



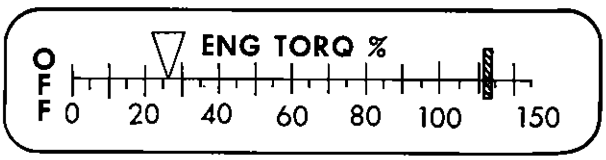
OIL TEMPERATURE INDICATOR (P)

(TOP)
CENTERED ON 107
RANGE 0 TO 107
(BOTTOM)
CENTERED ON 0
CENTERED ON 113
RANGE 0 TO 113



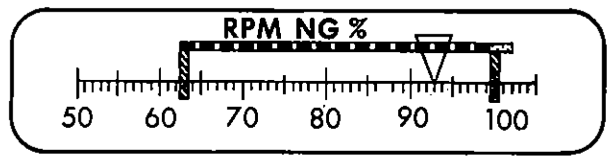
OIL PRESSURE INDICATOR (P)

(TOP)
CENTERED ON 10
CENTERED ON 75
RANGE 10 TO 75
(BOTTOM)
CENTERED ON 80
CENTERED ON 110
RANGE 80 TO 110



ENGINE TORQUEMETER

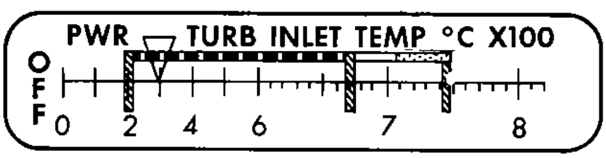
CENTERED ON 114



GAS GENERATOR TACHOMETER

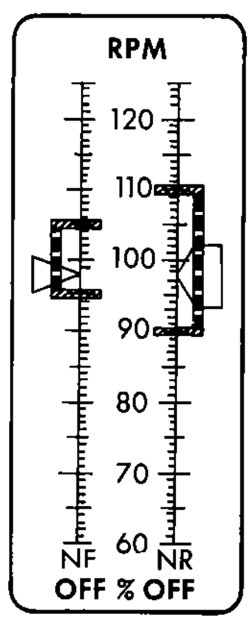
CENTERED ON 63
CENTERED ON 101.5
RANGE 63 TO 100
RANGE 100 TO 101.5

FUEL GRADE
MIL-T-5624
GRADE JP-4
OR GRADE
JP-5



POWER TURBINE INLET TEMPERATURE INDICATOR

CENTERED ON 200
CENTERED ON 671
CENTERED ON 743
RANGE 200 TO 671
RANGE 707 TO 743



POWER TURBINE AND MAIN ROTOR TACHOMETER

(LEFT)
CENTERED ON 105
CENTERED ON 95
RANGE 95 TO 105
(RIGHT)
CENTERED ON 90
CENTERED ON 110

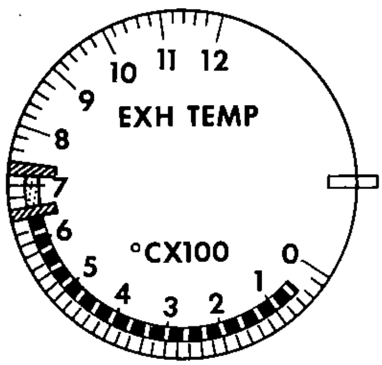
RANGE 90 TO 110

LEGEND
RED
GREEN
YELLOW

INSTRUMENTS MARKED (P) ARE LOCATED ON THE PILOTS INSTRUMENT PANEL ONLY. INSTRUMENTS MARKED (CP/G) ARE LOCATED ON THE COPILOTS/GUNNERS INSTRUMENT PANEL. ALL OTHER INSTRUMENTS ARE FOUND ON BOTH PANELS

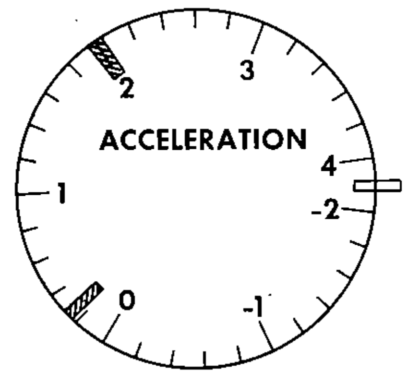
Figure 12C-1. Instrument Markings (Sheet 1 of 3)

FUEL GRADE
MIL-T-5624
GRADE JP-4
OR GRADE
JP-5



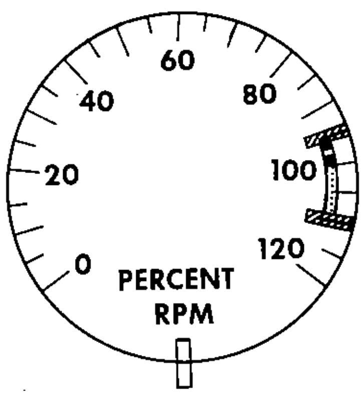
APU EXHAUST GAS
TEMPERATURE INDICATOR (P)

CENTERED ON 650
CENTERED ON 732
RANGE 50 TO 650
RANGE 650 TO 732



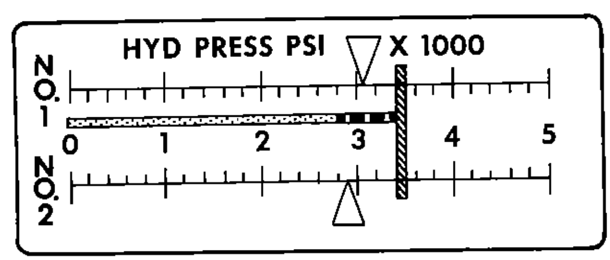
ACCELEROMETER (P)

CENTERED ON 0.25
CENTERED ON 2.0



APU TACHOMETER (P)

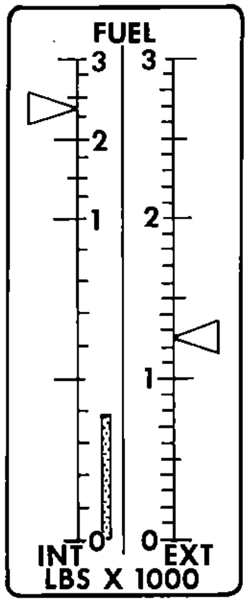
CENTERED ON 95 AND 110
RANGE 95 TO 100
RANGE 100 TO 110



HYDRAULIC PRESSURE INDICATOR (P)

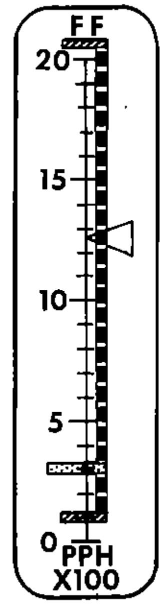
CENTERED ON 3400
RANGE 2800 TO 3400
RANGE 0 TO 2800

Figure 12C-1. Instrument Markings (Sheet 2 of 3)



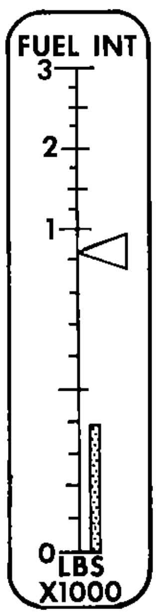
FUEL QUANTITY INDICATOR (P)

RANGE 0 TO 390



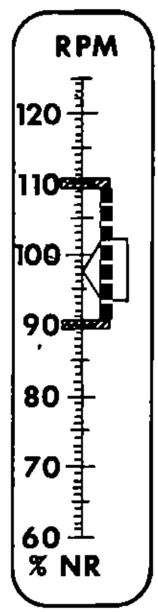
FUEL FLOW INDICATOR (P)

CENTERED 2060
CENTERED ON 110
CENTERED ON 350
RANGE 110 TO 2060



INTERNAL FUEL QUANTITY INDICATOR (CP/G)

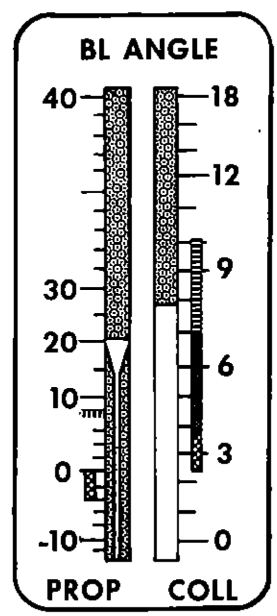
RANGE 0 TO 390



MAIN ROTOR RPM (CP/G)

CENTERED ON 90
CENTERED ON 110

RANGE 90 TO 110



BLADE ANGLE INDICATOR

(RIGHT)
RANGE 2.5 TO 3.7
RANGE 3.7 TO 7
RANGE 7 TO 10
(LEFT)
RANGE 0 TO -4
RANGE AT 8

LEGEND
RED
GREEN
YELLOW

LEGEND
ORANGE
LT YELLOW
DK GREEN

Figure 12C-1. Instrument Markings (Sheet 3 of 3)

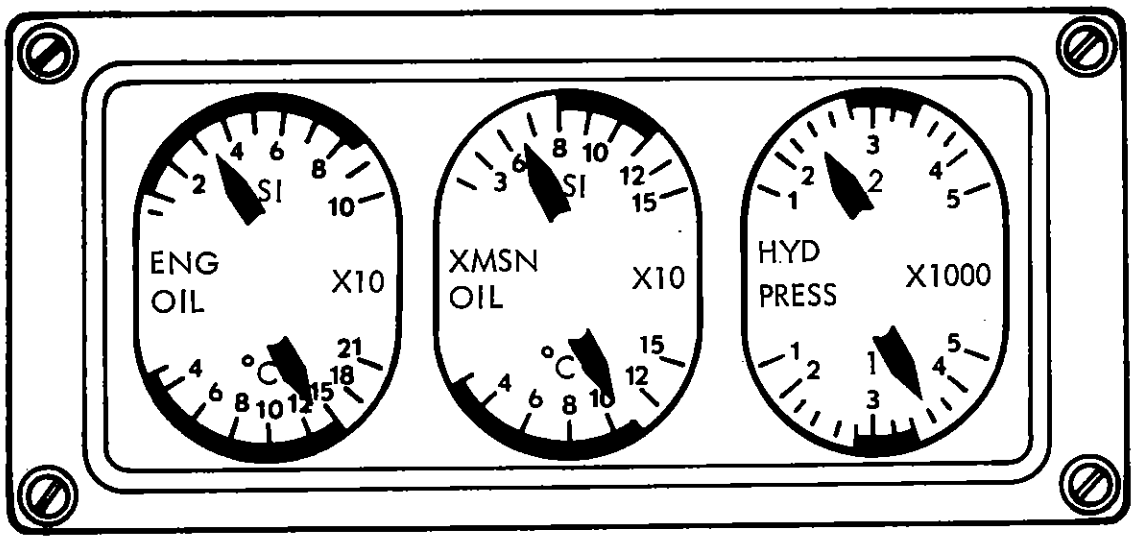


Figure 12C-2. Secondary Systems Panel

ENVIRONMENTAL CONTROL SYSTEM

I. GENERAL DESCRIPTION

The environmental control system is provided to supply the cockpit and the electrical and avionics compartments with air properly conditioned for equipment cooling and crew comfort. The system is capable of keeping the cockpit at 85°F on a 115°F ambient day and at 40°F on a -65°F ambient day.

The AH-56A armed helicopter is equipped with a Stratos Model NURH 40-1 air cycle cockpit cooling and heating system. Pneumatic in operation, the system is powered by bleed air from the engine compressor.

The principle components are:

- Flow Control and Shutoff Valve
- Environmental Control Unit
- Water Separator
- Duct Temperature Selector/Sensor

Figure 13-1 shows a view of the aircraft and the area where the air conditioning package is installed. Figure 13-2 shows the location of the environmental control unit in the forward right sponson. High temperature high pressure engine bleed air which is extracted from the engine at the 14th compression stage enters the environmental control system at the flow control and shutoff valve. The bleed air is then ducted to the environmental control unit. Bleed air, having had its temperature reduced in the pre-cooler and heat exchangers, expands through the turbine section of the turbine-fan assembly. The work extracted in the turbine causes a further reduction in the bleed air to an effective temperature of -20°F. Conditioned air is then heated in the mixing muff utilizing bypass air to temperature selected in cockpit. Ambient cooling air is moved through the pre-cooler and heat exchanger by the axial-flow fan mounted on a common shaft with the axial-flow turbine. The fan also serves to load the turbine as well as to draw the cooling air through the heat exchangers.

To reduce the moisture content, all air flow entering the cockpit is directed through a water separator located downstream of the turbine-fan assembly. Cockpit temperature is selected by the pilot and is then regulated by pneumatic temperature control components.

The avionics and electrical compartment cooling air system is interfaced with the cockpit cooling and heating system in that cockpit air is used to provide cooling air for the fire control compartment, the main avionics compartment swivel gun station, and computer compartment. Ambient air is used to cool the electrical load center and the aft avionics compartment.

Completing the system are the necessary inter-connecting ducts, flanges, couplings, and other miscellaneous hardware.

II. COMPONENTS AND LOCATIONS

Name of Component	Number per Aircraft	Location in Aircraft
Flow Control and Shut-off Valve	1	Mounted in the engine air plenum chamber on the right hand side
Environmental Control Unit	1	Forward area of the right sponson
Temperature Control Valve	1	In the environmental control unit
Fan and Turbine Assembly	1	In the environmental control unit
Pre Cooler	1	In the environmental control unit
Heat Exchanger	1	In the environmental control unit
Thermal Fuse	1	In the environmental control unit
Servo Air Tube and Filter	1	In the environmental control unit

Name Component	Number per Aircraft	Location in Aircraft
Pressure Switch	1	In the environmental control unit
Environmental Control Exhaust Stack	1	Attached to the ECU and trimmed to the contour of the access door skin surface
Temperature A Anticipator	1	In the moisture separator
Overheat Switch	1	In the moisture separator
Temperature Selector/ Sensor	1	In the cockpit air duct
Temperature Control Panel	1	Right pilots console
Water Separator	1	Between the ECU and cockpit duct
Fan, Fire Control Area	2	Below cockpit floor, B.L.18.0 L&R FS 185
Fan, Electrical Load Center	1	Electrical load center
Fan, Computer Com- partment	1	Mounted on a vertical bulkhead directly below the computer compartment.
Fan, Main Avionics Compartment	1	Main avionics compartment
Fan, Aft Avionics Compartment	1	Aft avionics compartment

III. MAJOR COMPONENT DESCRIPTIONS

A. Flow Control and Shutoff Valve (FCV)

The flow control and shutoff valve is located in the bleed air supply duct upstream of the environmental control unit (Figure 13-2)

It is located on the forward bulkhead, right hand side of the engine air inlet plenum chamber. This FCV acts as a pressure regulator which maintains a downstream pressure schedule as a function of air craft altitude and engine inlet pressure. A sonic venturi section downstream of the regulator limits maximum flow when the Temperature Control Valve is fully open.

A solenoid is mounted on the FCV. It is normally closed and opens on application of 28 Volt DC, permitting the FCV's butterfly valve to open and operate in the pressure regulating mode.

B. Environmental Control Unit (ECU)

The ECU, located in the forward end of the right hand sponson, consists of the following components:

1. Heat exchanger-air to air, aluminum, plate-fin, three-pass, cross counter flow design.
2. Pre-cooler - air to air, stainless steel, tubular, single pass, cross flow design.
3. Valve, temperature control.
4. Turbine fan assembly, air cycle.
5. Servo air tube and filter assembly.
6. Switch, pressure.
7. Fuse, thermal.
8. Meter, time totalling.

The ECU receives pressure regulated, high temperature, engine bleed air. The bleed air passes first through the pre-cooler and then through the heat exchanger and into the turbine inlet of the turbine fan assembly. Bleed air is cooled by (1) cooling air drawn over the "RAM" side of the pre-cooler and heat exchanger by the fan wheel of the turbine fan assembly, and (2) expansion in the turbine side of the turbine fan assembly. Turbine exit air is then mixed with bleed air, through the operation of the temperature control valve, in the mixing chamber where it then exits.

Auxiliary devices are also mounted on the ECU. The time totaling meter operates whenever 28 volts DC is present, the pilots on/off switch is on and the pressure switch is energized by the presence of sufficient bleed pressure in the exhaust header of the heat exchanger. A thermal fuse, mounted on the heat exchanger will interrupt DC voltage to the solenoid valve if an over temperature condition occurs. Control air pressure is supplied to the control system by the control air filter and pressure source, mounted at the turbine inlet.

C. Water Separator

The water separator (Figure 13-2) is located in the forward right hand sponson downstream of the ECU. It is a self-contained unit, capable of removing 70-80% of the entrained moisture which is condensed when the bleed air is expanded through the turbine. The separator consists of a plenum which provides the volume necessary for the driplets to grow in size. Located in the plenum is the separator which consists of a coalescer and a collector. The coalescer is made of a Teflon felt-like material which serves to coalesce the small water droplets. If the coalescer becomes excessively contaminated with dirt, etc. the pressure in the plenum could rise to unacceptable levels. To prevent this possibility a bypass valve opens at higher than normal operating pressure levels and the cabin supply air by-passes the coalescer.

The collector section consists of turning vanes which whirls the air and thereby centrifuges the water droplets to the collector walls. The water is then drained out of the separator.

Also located in the plenum wall is the anticipator which is part of the pneumatic temperature controls and an over-temperature switch which closes the system down if the temperature rises above 225°F.

D. Temperature Selector/Sensor

The temperature selector/sensor (Figure 13-3) is installed in the cockpit ducting directly below the pilots right console and is linked to the selector knob on the console by a flexible drive shaft (Fig ure 3).

Selection of the desired cabin temperature is made in the cockpit and the temperature selector/sensor then senses the outlet temperature of the environmental control unit in the cockpit ducting. The sensor element acts to increase/decrease an orifice, as a function of delivered air temperature around the setting or quiescient point established by the pilot. These orifice variations meter control air to the temperature control valve to maintain the delivered air temperature. (The anticipator performs the same function for short-term temperature transients since its sensor reacts only to change the Delivered Air Temperature.)

E. Temperature Control Panel

The temperature control panel (Figure 13-3) located on the pilots right console and consists of a two position toggle switch and a temperature control knob.

The two position (OFF-ON) switch energizes the shutoff solenoid on the flow control valve and activates the time totalizer through the overheat switch. The time totalizer indicates total operating time of the system.

The temperature control know is mechanically linked to the temperature selector/sensor in the cockpit ducting below the cockpit floor and therefore controls the cockpit temperature.

F. Fire Control Area Fans

The two fire control area fans provide cooling air for the components installed in the fire control area located below the cockpit floor. The fans are electrically controlled by the FIRE CONT COMPT BLOWER circuit breaker on the No. 1 secondary ac bus circuit breaker panel. The fans are mounted to the underside of the pilots floor at FS 185 and BL 18 L & R. Inlets inside the consoles are screened to prevent debris from entering the fan.

G. Electrical Load Center Fan

The electrical load center fan provides circulated ambient air for the electrical compartment. The fan is installed on a plenum which is located in the aft area of the electrical compartment at FS 252. The fan is electrically controlled by the ELEC LOAD CTR BLOWER circuit breaker on the No. 1 secondary ac bus curcuit breaker panel.

H. Computer Compartment Fan

The computer compartment fan provides air from the cockpit for cooling the computer compartment. A fiber-glass duct connects the fan to a screened opening in the cargo compartment. The fan is located on a vertical bulkhead directly below the inboard bulkhead of the computer compartment. A duct leads from the fan up to the deck of the compartment where cooling air is dispersed into the base of the computer. The fan is electrically controlled by the COMPUTER CENTRAL PWR circuit breaker on the No. 1 primary ac bus circuit breaker panel.

I. Main Avionics Compartment Fan

The main avionics compartment fan provides air from the cockpit for cooling the equipment installed in the main avionics compartment. The fan is attached to the underside of the forward lower shelf in the compartment. A fiber glass duct connects the fan to a screened opening in the cockpit. The fan is electrically controlled by the MAIN AVIONICS COMPT BLOWER circuit breaker on the No. 1 secondary AC bus.

J. Aft Avionics Fan

The aft avionics compartment fan provides circulated ambient air for cooling the avionics equipment installed in the aft avionics compartment. The fan is electrically controlled by the AFT AVIONICS COMPT BLOWER circuit breaker on the No. 1 secondary ac bus circuit breaker panel. The fan is located directly above the left side of the aft avionics compartment access door at FS 407. The fan is installed between an inlet duct and a deflector assembly.

IV. SYSTEM OPERATION

A. Normal

High temperature (839°F), high pressure (170 psia) engine bleed air is extracted from the 14th compression stage of the engine (Fig 13-4) and enters the environmental control system at the flow control and shutoff valve.

The shutoff valve is controlled by an OFF-ON switch on the pilots temperature control panel. The system is activated when the switch is positioned to the ON position. This supplies 28 volt DC power to the solenoid valve and permits the flow control valve to open and regulate the bleed pressure which results in the hot (839°F) high pressure (91 psia) bleed air flowing to the environmental control unit. A sonic venturi in the flow control valve limits the flow to 40 lb/min. The temperature of the bleed air is reduced to 607°F by passing first through the precooler and to 157°F after passing thru the heat exchanger. The bleed temperature and pressure are then both reduced (35°F and 15 psia) when the bleed air passes through the turbine. The energy extracted by the turbine is used by the fan to pull ambient air (approx. 127 pounds per minute) through the heat exchanger and then push it overboard through the precooler.

Upon leaving the turbine, the cooled bleed air passes through the water separator where between 70 and 80 percent of the entrained moisture is removed. Entrained moisture results when, on humid days, water condenses as the air is expanded through the turbine. The air is now delivered to the crew station via the duct distribution system.

The temperature control system is pneumatically operated by servo air bleed from the aluminum heat exchanger. It consists of a temperature sensor selector which is operated by the pilot together with a temperature control valve (TCV) and anticipator. The TCV allows the hot bleed air to bypass the ECU to control the delivered air temperature to the value selected by the pilot. The anticipator serves to moderate the rate of temperature change when the system is undergoing temperature transients caused by a change in the temperature selector setting, or engine bleed air conditions.

The system is protected against overheating by a thermal fuse located in the aluminum heat exchanger and also by an over-temperature switch located in the water separator. If higher than normal operating temperatures are experienced the over-temperature switch will electrically close the FCV. If a blockage occurs between the heat exchanger and the water separator, the thermal fuse will open and cause the FCV to close.

The environmental system is interfaced with the avionics and fire control compartments in so far as the air drawn by the swivel gun station, main avionics compartment, fire control compartment and computer compartment fans, is cockpit exhaust air. The remainder of the fans which supply air to the electrical load center and aft avionics compartment draw upon ambient air for cooling purposes. The aircraft cooling system block diagram (Figure 13-5) and the compartment airflow diagram (Figure 13-6) show the relationship of the systems.

B. Emergency

In case of ECS failure, a small sliding window located in the Pilot's L.H. canopy and a small ram air scoop, located in the fuselage, just above the Copilot/gunner's left console provides a source of ambient air. This air source coupled with the fact that all of the electronic fans operate continuously whenever the AC bus is energized allows the exchange of cockpit air - approximately four times per minute.

V. PCRS CONFIGURATION

The basic ECS system design as used in the Baseline Model is also used for the approved PCRS Configuration.

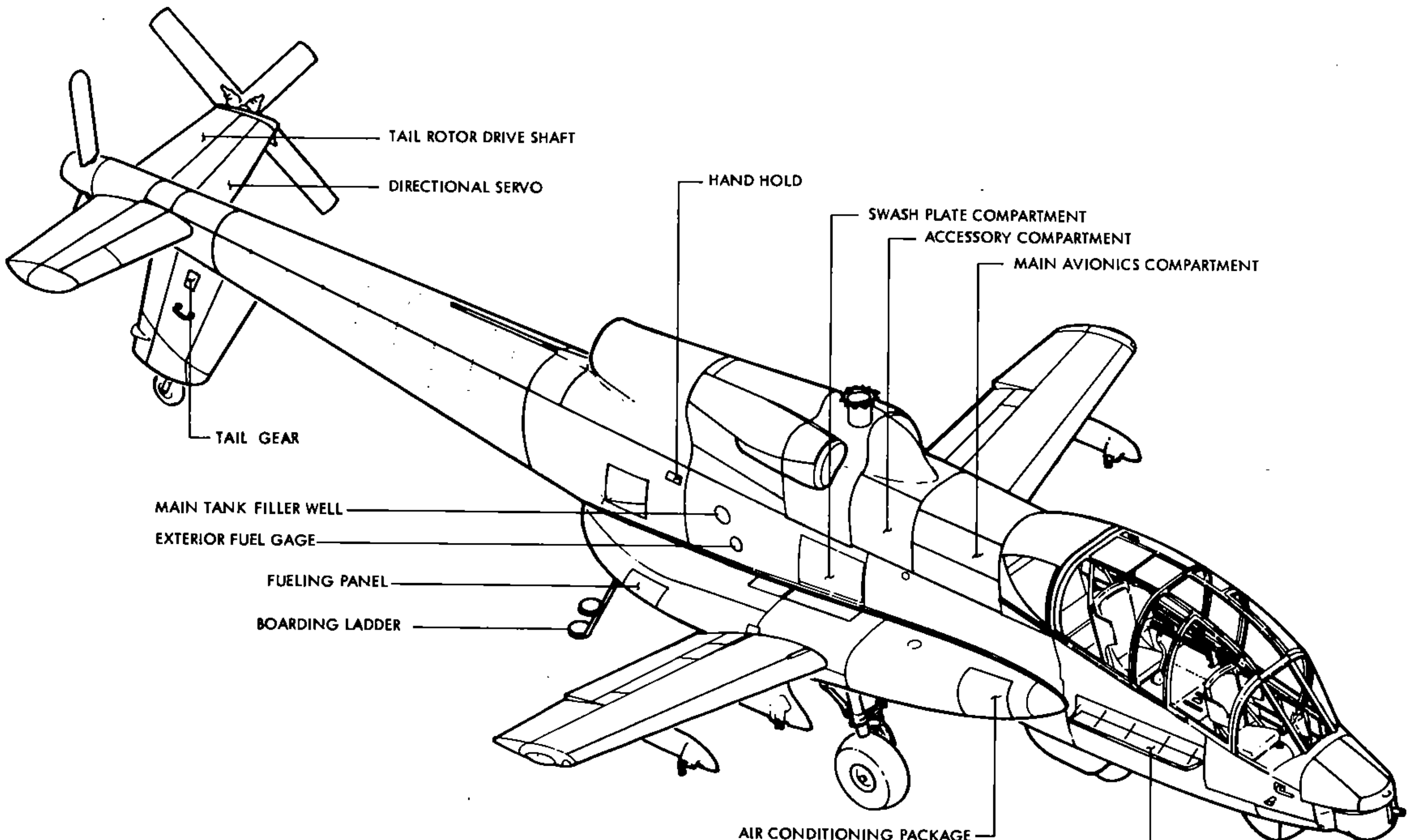


Figure 13-1. Right Hand Exterior View of AH-56A

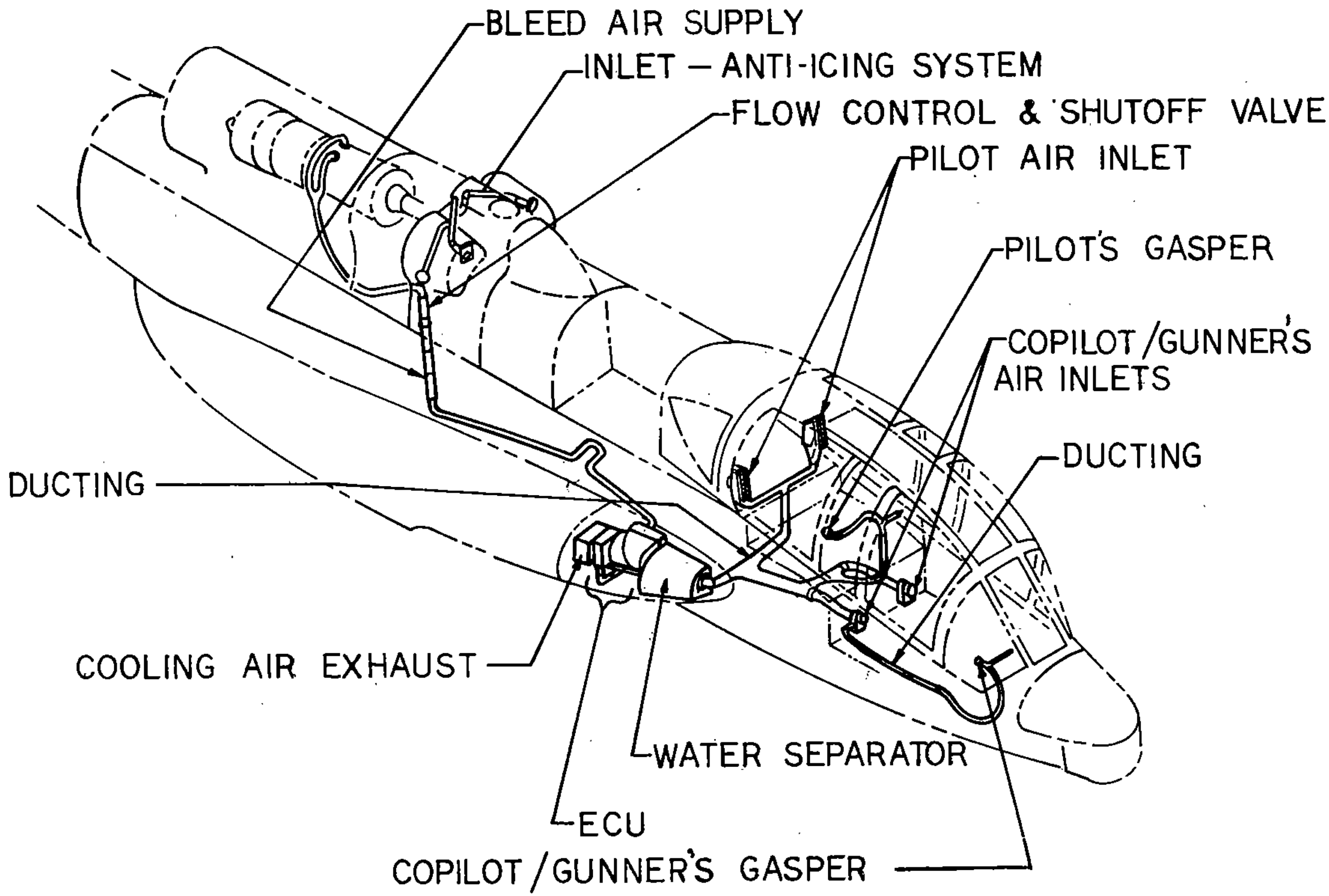


Figure 13-2. ECSC Supply & Distribution Ducting

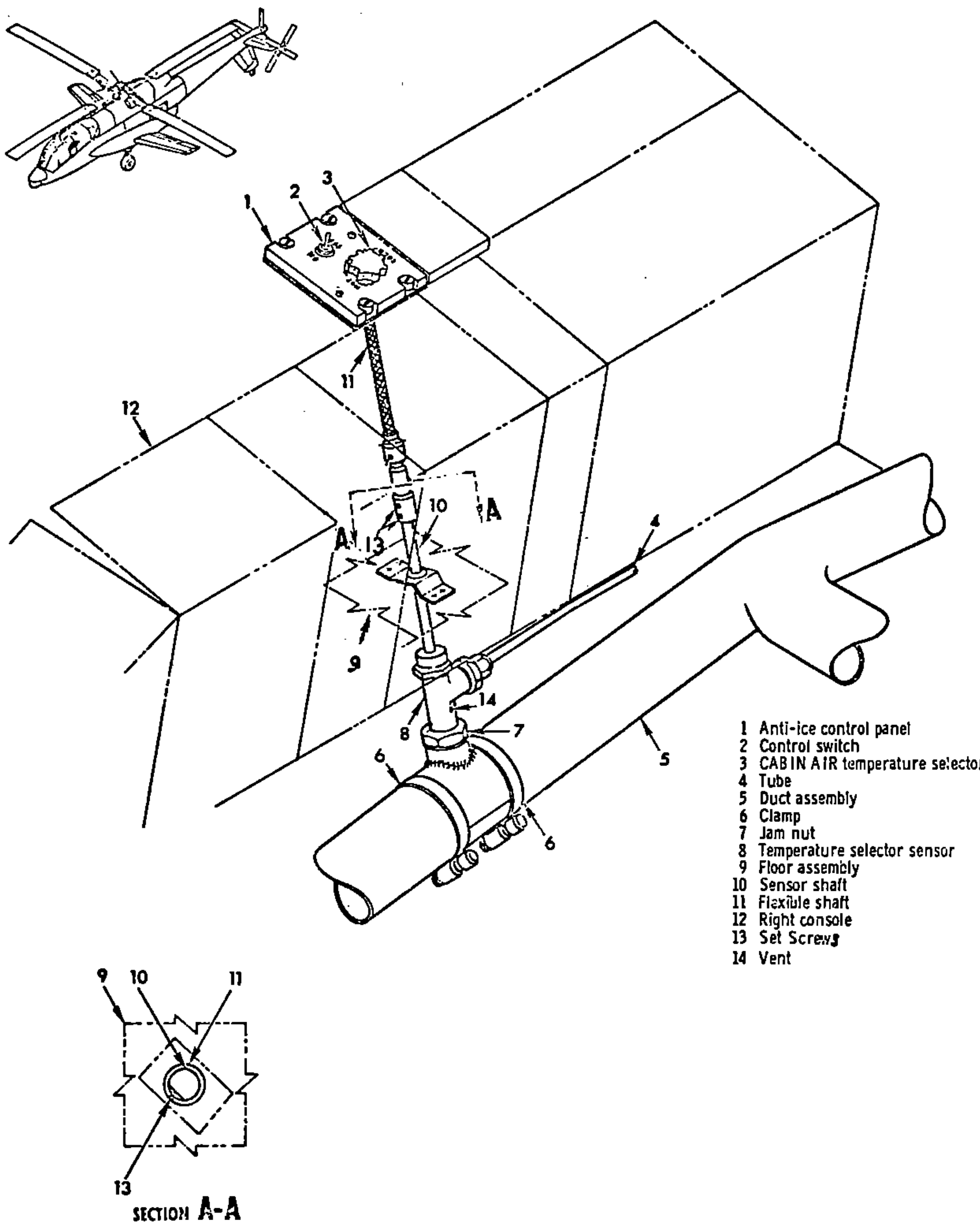


Figure 13-3. Temperature Selector/Sensor Installation

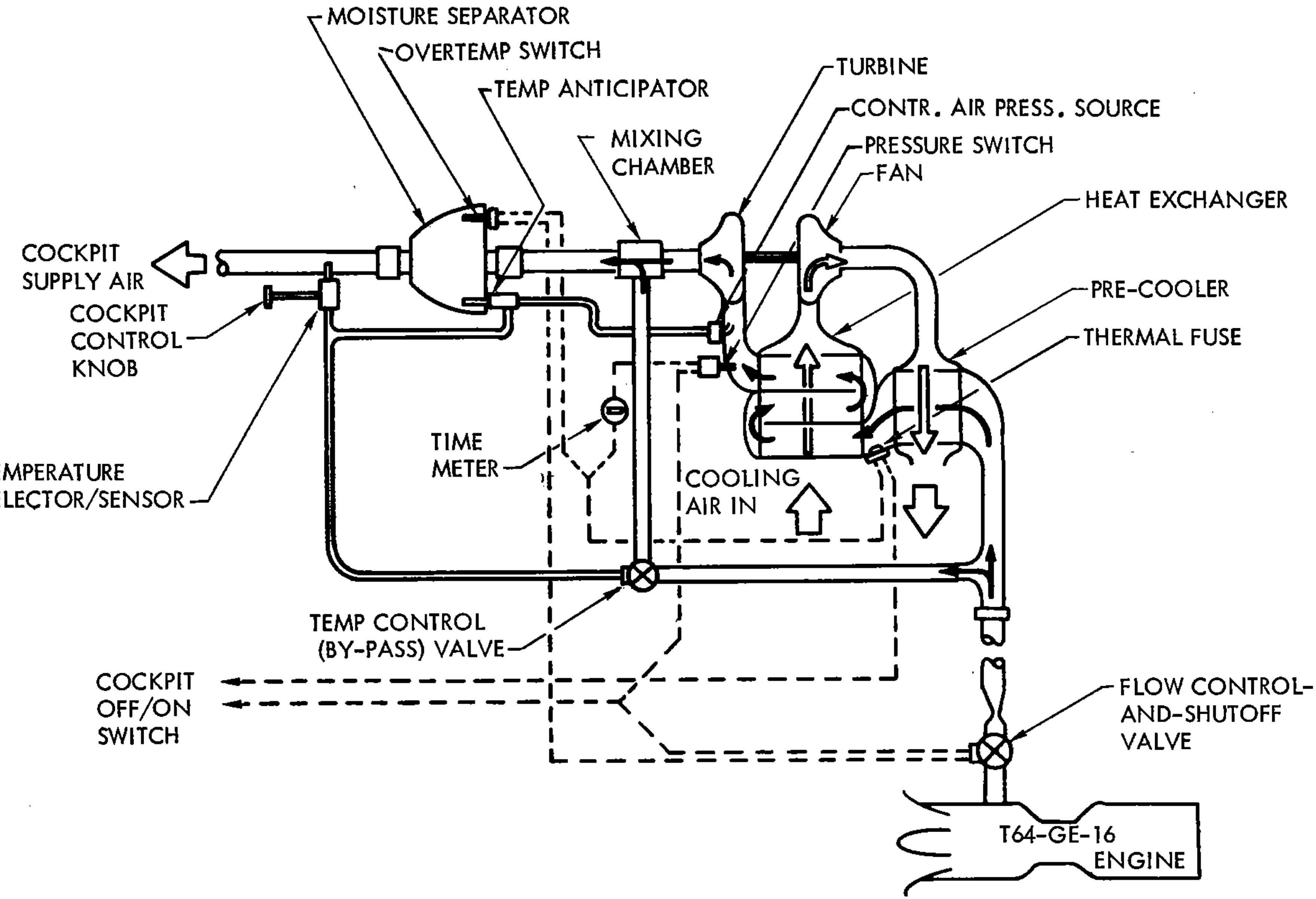


Figure 13-4. Environmental Control System-Cockpit

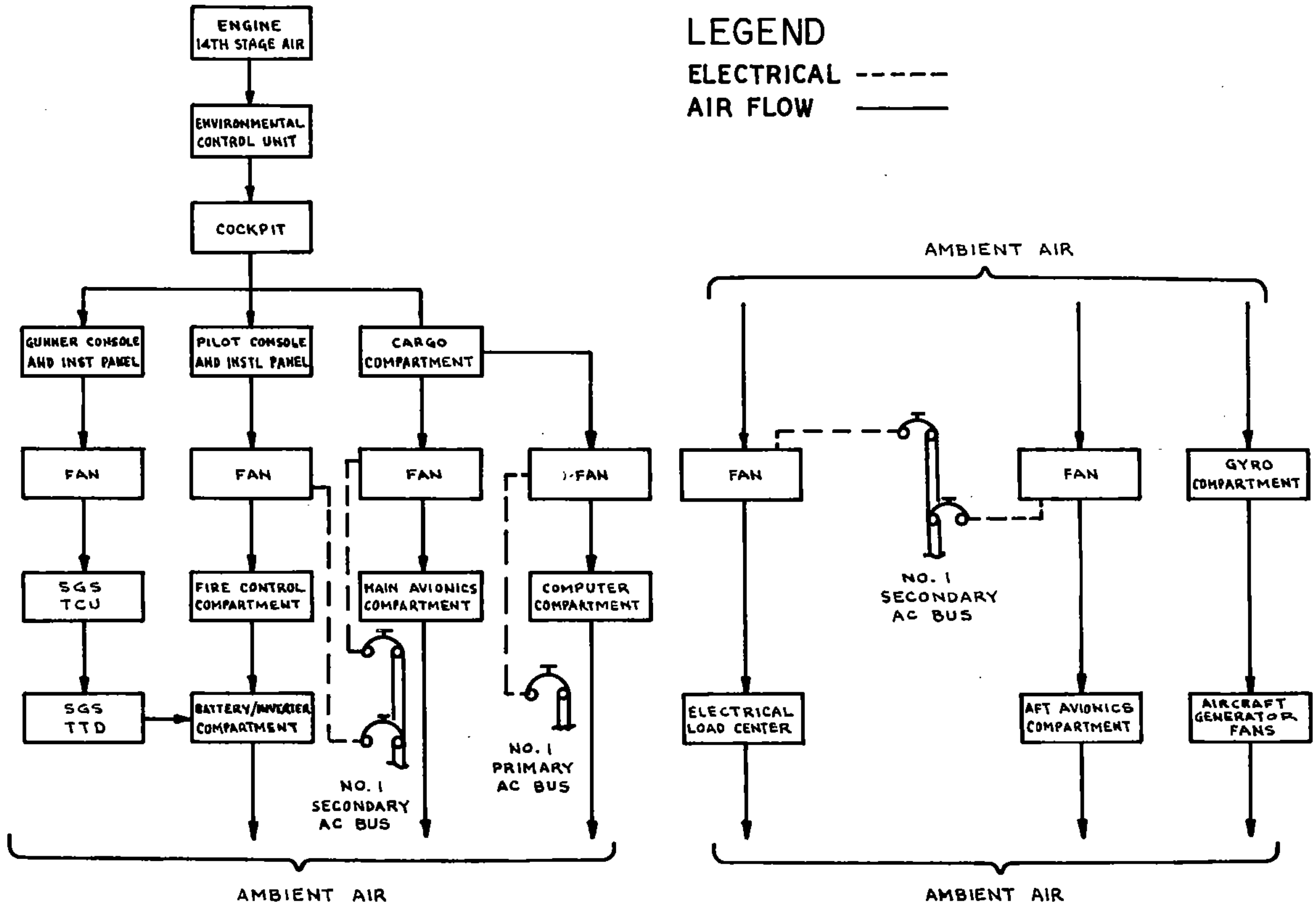


Figure 13-5. Block Diagram of Aircraft Cooling System

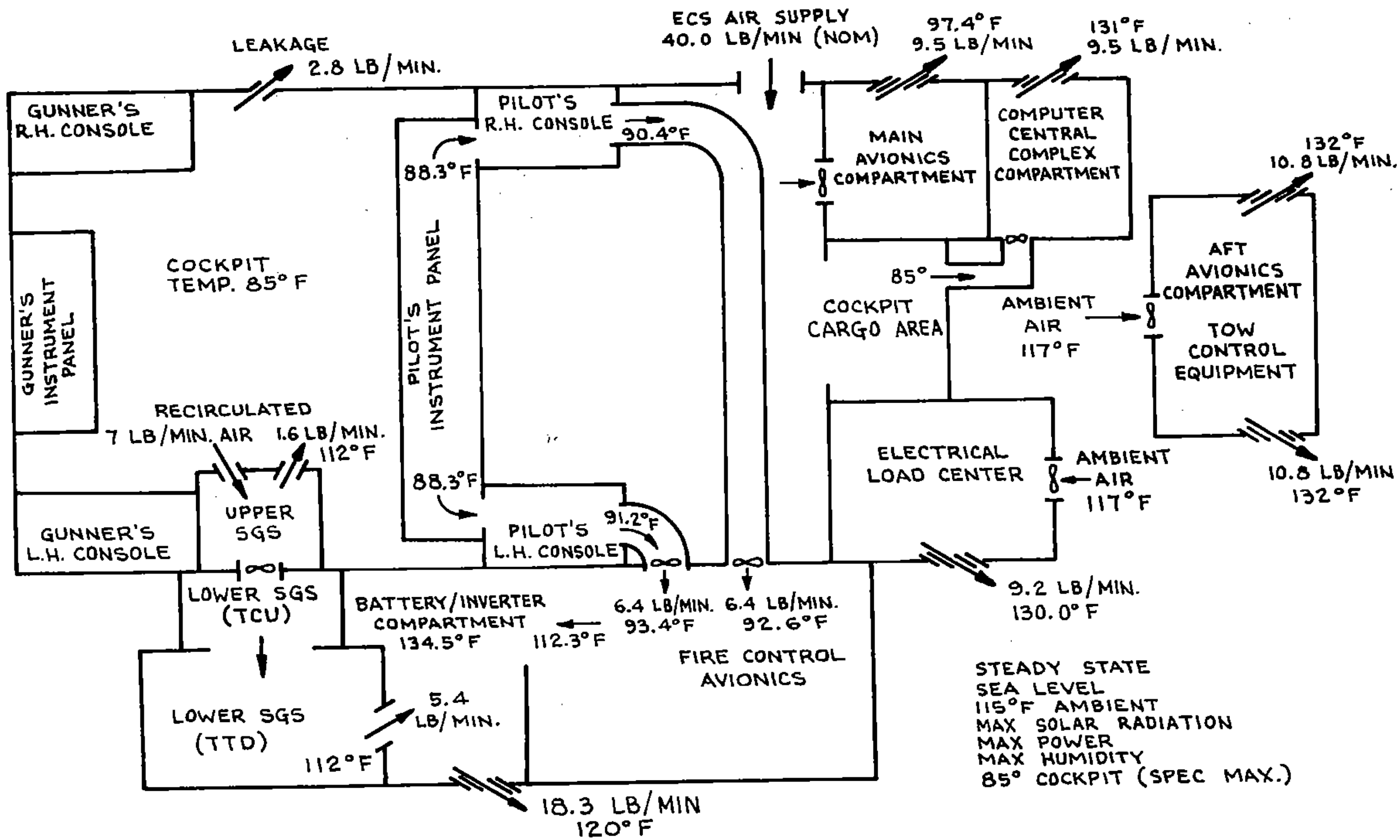


Figure 13-6. Airflow & Compartment Temperatures ECSC for NVS/TOW (Development)

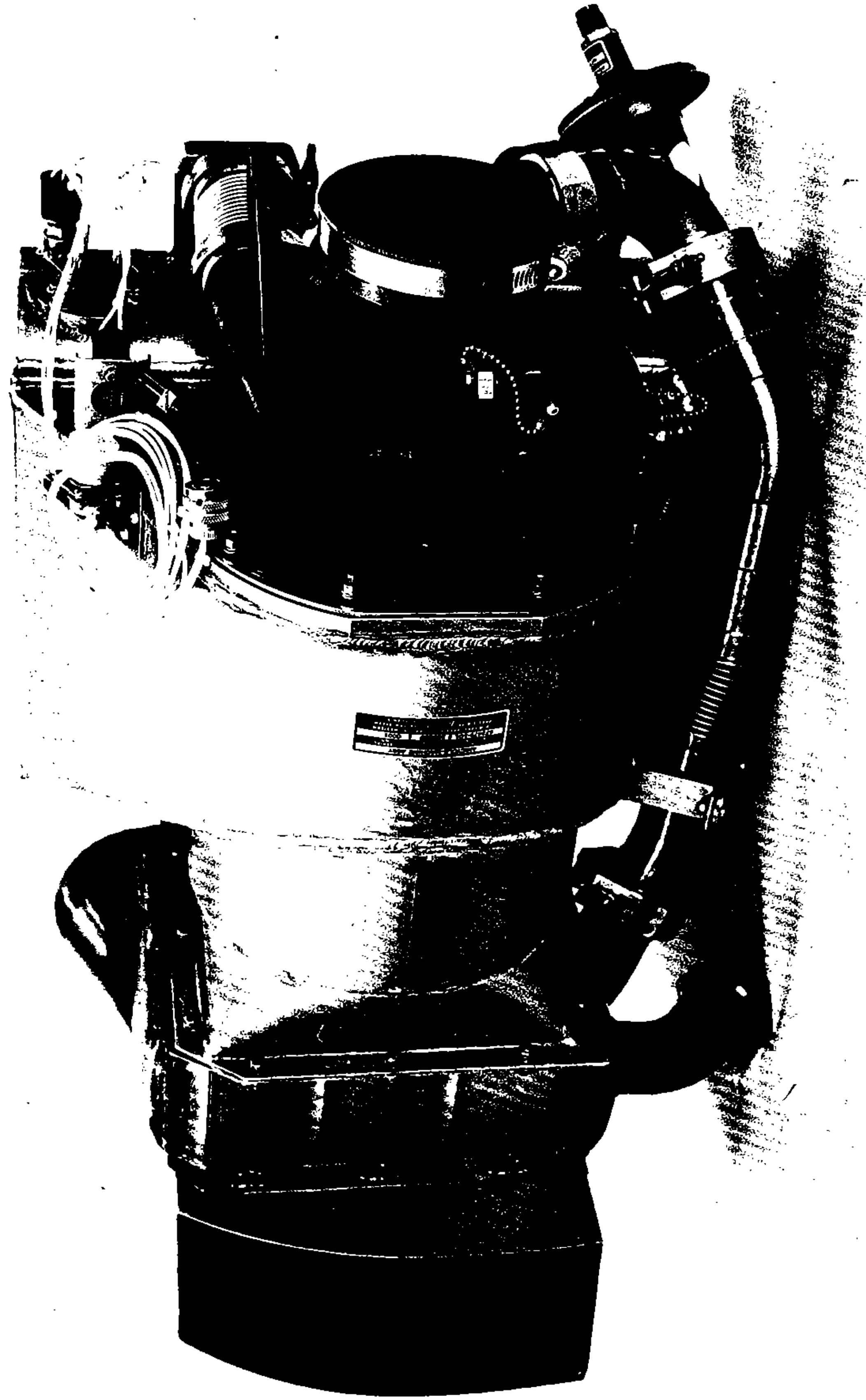


Figure 13-7.

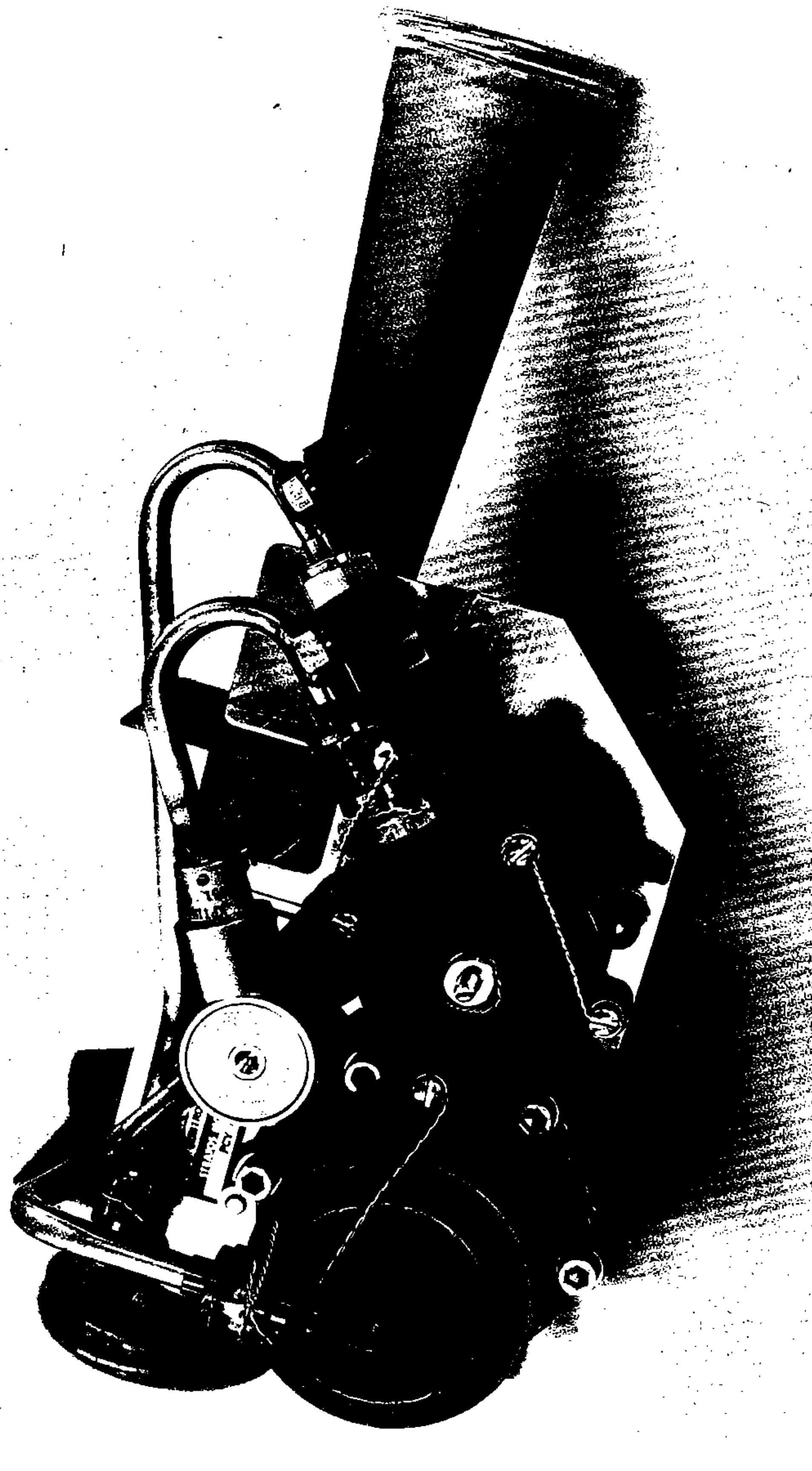
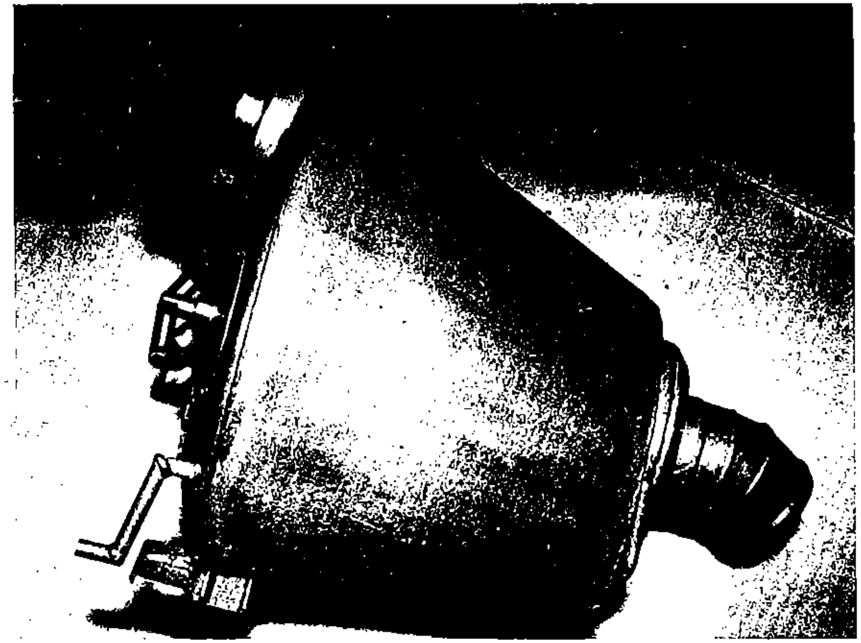
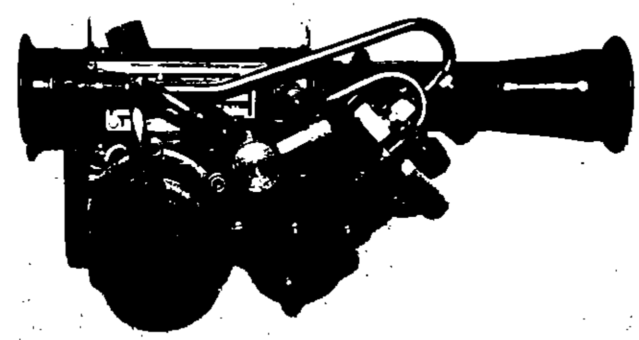


Figure 13-8.



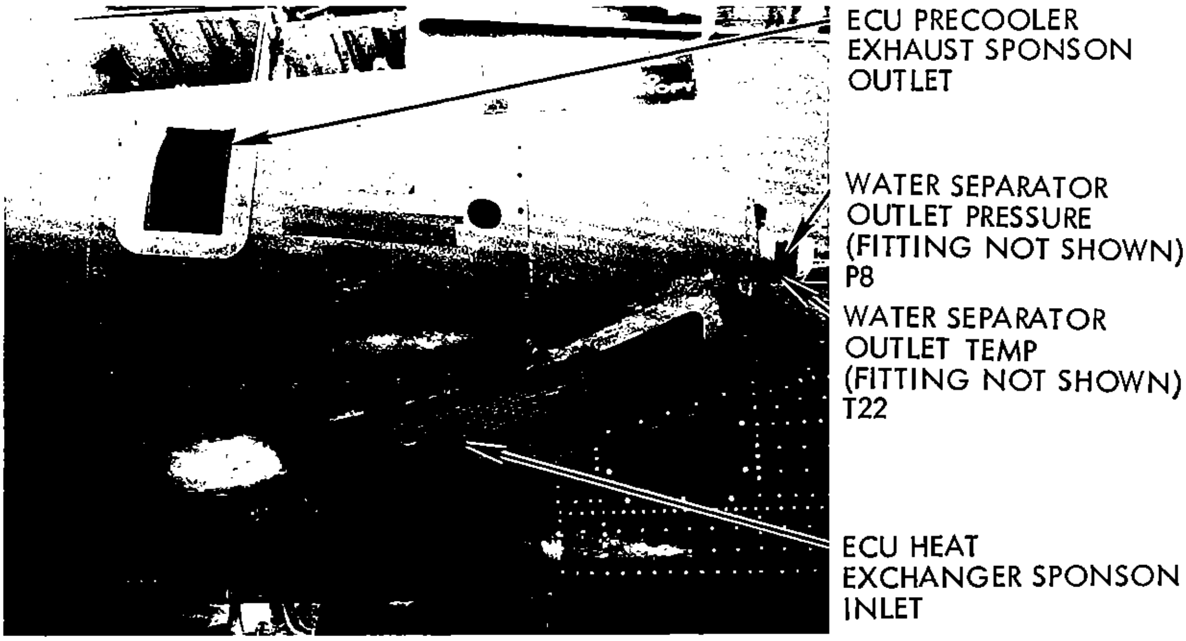
MOISTURE SEPARATOR

TEMPERATURE SELECTOR/SENSOR

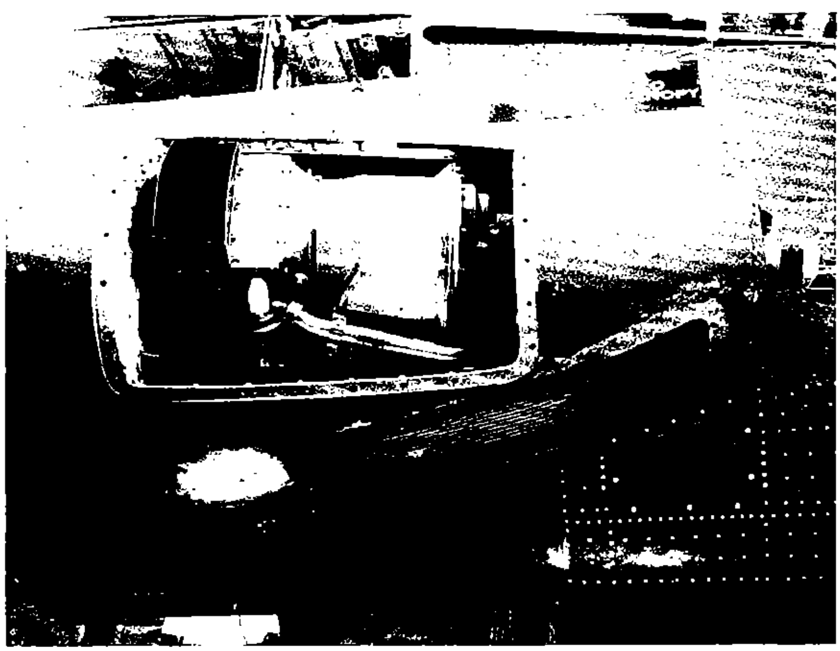


FLOW CONTROL-AND-SHUTOFF VALVE

Figure 13-9. Environmental Control System Components



40 LB/MIN ECU Package Enclosed by Sponson Cowl



40 LB/MIN ECU Package with Sponson Cowl Removed

Figure 13-10.

COMMUNICATIONS

I GENERAL DESCRIPTION

The AH-56A communications sytems provide the crew with the following capabilities:

A. FM tactical communications with ground forces on any two channels, simultaneously

B. FM homing to a tactical ground unit

C. UHF communication channels compatible with air and naval forces

D. VHF communications compatible with existing air traffic control frequencies

E. IFF responding capability compatible with military and civil interrogators

F. Retransmission capability between one FM unit and either the UHF or the VHF unit

G. Intercommunication capability between the cockpits and between the ground crew and the flight crew

H. Monitoring capability, for each cockpit, of all onboard radio transceivers at all times

II COMPONENTS AND LOCATIONS

The major systems of the communications installation in the AH-56A compound helicopter are outlined in the following list:

A. AN/ARC-114, VHF/FM Radio Sets, installed in each of the two crew stations, which provide "line of sight" voice communications with tactical forces. The pilot's VHF/FM also provides an FM homing function.

B. AN/ARC-115, VHF/AM Radio Set, installed in the copilot's station, which provides non-tactical "line of sight" voice communication primary for aerial navigation and traffic control at civil installations.

C. AN/ARC-116, UHF/AM Radio Set, installed in the pilot's station, which provides non-tactical "line of sight" voice communications primarily for air traffic control at United States Air Force installations.

D. Intercommunications System, which consists of an ICS "J" Box located in the ceiling of the debris bay and ICS control panels in both the pilot's and copilot's station, provides audio communications between the crew members.

E. Retransmission System, installed in the pilot's station, allows retransmission of command data on the copilot/gunners FM set or the pilots VHF or UHF (whichever is installed).

F. The AH-56A is supplied with the AN/ARC-116, UHF/AM Radio Set installed in the pilot's station and the AN/ARC-115, VHF/AM Radio Set installed in the copilot's station. The units are directly interchangeable, however antenna and ICS "J" Box reconnections must be made.

G. AN/APX-72 IFF transponder system, installed in the main avionics compartment, provides identifying replies to military and civilian interrogators.

III MAJOR COMPONENTS DESCRIPTION

A. AN/ARC-114, VHF/FM Radio Set

Each of the AN/ARC-114, VHF/FM Radio Sets provides voice communications on any one of 920 channels in the very high frequency range of 30 megahertz to 76 megahertz. Each of the VHF/FM radio sets transmits a nominal power of 10 watts and has a nominal effective range of approximately 70 nautical miles at a helicopter altitude of 5000 feet above the terrain. Each of the VHF/FM radio sets incorporates a retransmission capability. Fixed "guard" channel monitoring at 40.5 megahertz is provided. Each of the VHF/FM radio sets is of solid-state, single replaceable unit design. Two ferrite cavity antennae and two logic tuning boxes, one for each of the VHF/FM sets, provide the excitation for the ventral fin to radiate as an antenna.

The pilot's AN/ARC-114, VHF/FM #1 Radio Set is installed in the pilot's left console. Transmission on the pilot's VHF/FM #1 is enabled by selection of XMTR SELECTOR position #1 on the pilot's ICS Control Panel, and keyed by the RADIO/ICS switch on the pilot's cyclic control stick. Audio monitoring of the received VHF/FM #1 is enabled by selection of the #1 receiver switch on the pilot's ICS Control Panel, or is automatically enabled when the XMTR SELECTOR is selected to position #1.

The copilot's AN/ARC-114, VHF/FM #2 Radio Set is installed in the left console of the copilot's station. Transmission on the copilot's VHF/FM #2 is enabled by selection of position #1 of the XMTR SELECTOR on the copilot's ICS Control Panel, and is keyed by the RADIO/ICS switch on the swivelling gunner's station left-hand grip or by the copilot's RADIO/ICS switch on the copilot's cyclic control stick. Audio monitoring of the received VHF/FM #2 is automatically enabled by selection of the XMTR SELECTOR position #1 on the copilot's ICS Control Panel, or is enabled by selection of receiver switch #1 on the copilot's ICS Control Panel.

The RCVR TEST pushbutton on the VHF/FM Control Panel verifies proper operation of the receiver section when a modulated tone is heard in the headset. Presence of a proper sidetone during a test transmission verifies proper operation of the transmitter.

The <u>pilot's</u> VHF/FM radio set <u>only</u> provides an FM "homing" function utilizing the "towel bar" antenna directly aft of the pilot's station on the top of the fuselage. Automatic switching to the "homing" antenna is accomplished when HOMING is selected on the VHF/FM Control Panel in the pilot's station. FM homing information is displayed on the Bearing Distance Heading Indicator (BDHI) "course offset" needle, which is rotated to alignment with the "lubber line" when FM STEER is selected on the pilot's Mode Select Panel. The course indicator on the BDHI displays actual heading flown in the FM STEER mode of operation.

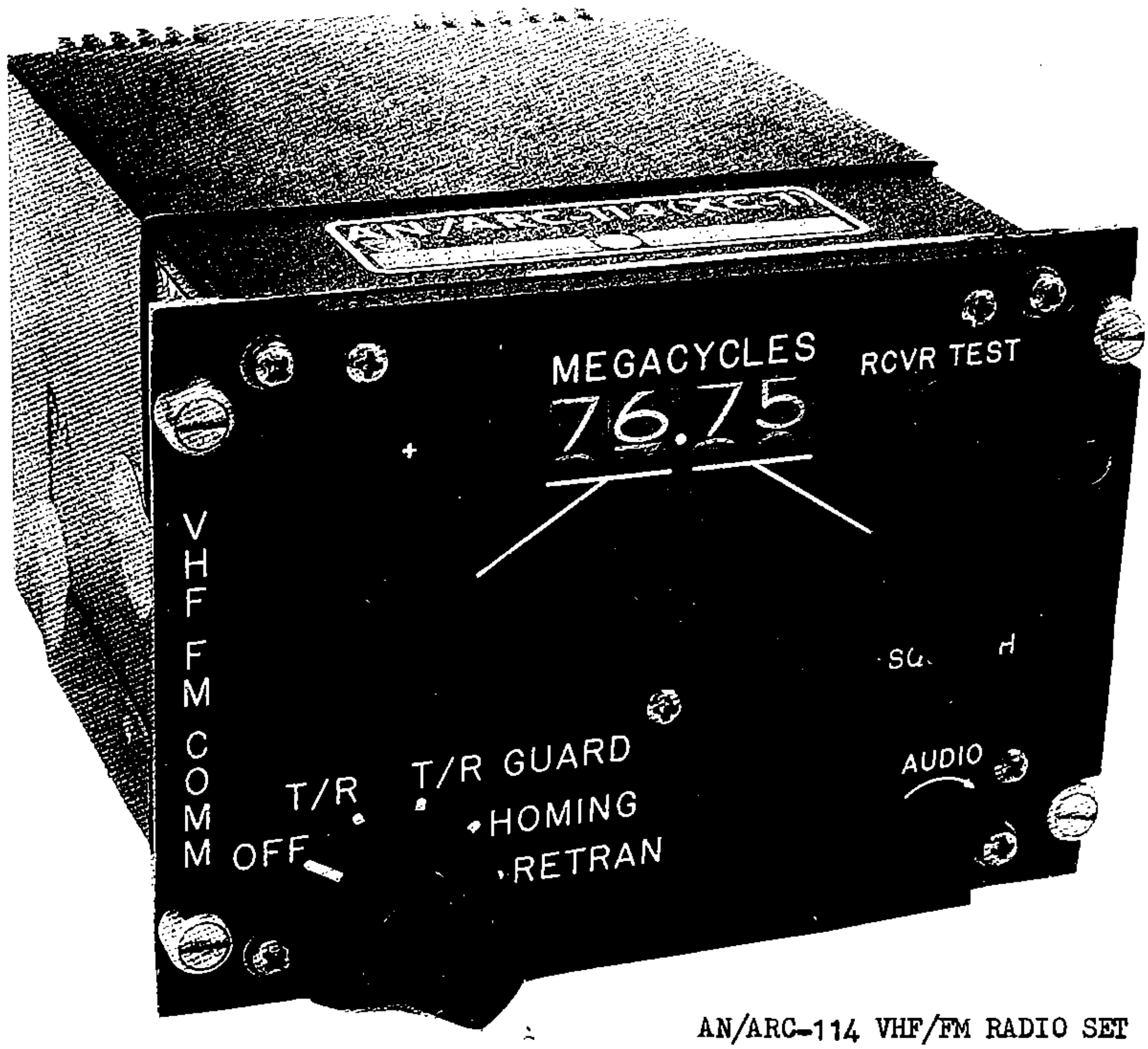
B. AN/ARC-115, VHF/AM Radio Set

The AN/ARC-115, VHF/AM Radio Set (figure 12) provides radio communications on any one of 1360 channels in the range of 116 megahertz to 150 megahertz. The VHF/AM radio set transmits a nominal power of 10 watts and has a nominal effective range of 70 nautical miles at a helicopter altitude of 5000 feet above the terrain. Fixed "guard" channel monitoring at 121.5 megahertz is provided. Retransmission capability is included only when the VHF/AM is installed in the pilot's station (the AH-56A is supplied with the VHF/AM Radio Set installed in the copilot's station). The direction finder (D/F) capability is not used.

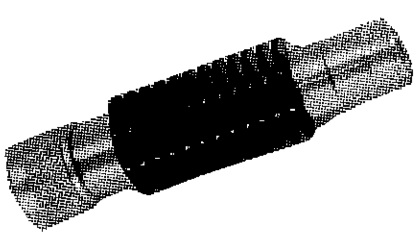
The copilot's AN/ARC-115, VHF/AM Radio Set is installed in the right console of the copilot's station and is of solid-state, single replaceable unit design. The AN/ARC-115, VHF/AM Radio Set shares the AT-1108 "broadband AM Antenna" on the underside of the aft fuselage section with the pilot's AN/ARC-116, UHF/AM Radio Set. The copilot's AN/ARC-115, VHF/AM Radio Set is directly interchangeable with the pilot's AN/ARC-116, UHF/AM Radio Set after antenna reconnections and ICS "J" Box reconnections are accomplished interchanging two ICS "J" Box jumper plugs to an opposite configuration.

Transmission on the copilot's VHF/AM radio set is enabled by selection of XMTR SELECTOR position #3 on the copilot's ICS Control Panel and keyed by the RADIO/ICS switch on the swivelling gunner's station left-hand grip or by the RADIO/ICS switch on the copilot's cyclic stick. The pilot can also transmit on the copilot's VHF/AM radio set by selecting XMTR SELECTOR position #3 on the pilot's ICS Control Panel and keying the VHF/AM using the RADIO/ICS switch on the pilot's cyclic control stick.

The RCVR TEST pushbutton on the VHF/AM Control Panel verifies proper operation of the receiver section when a modulated tone is heard in the headset. Presence of a proper sidetone during a test transmission verifies proper operation of the transmitter section of the copilot's VHF/AM radio set.



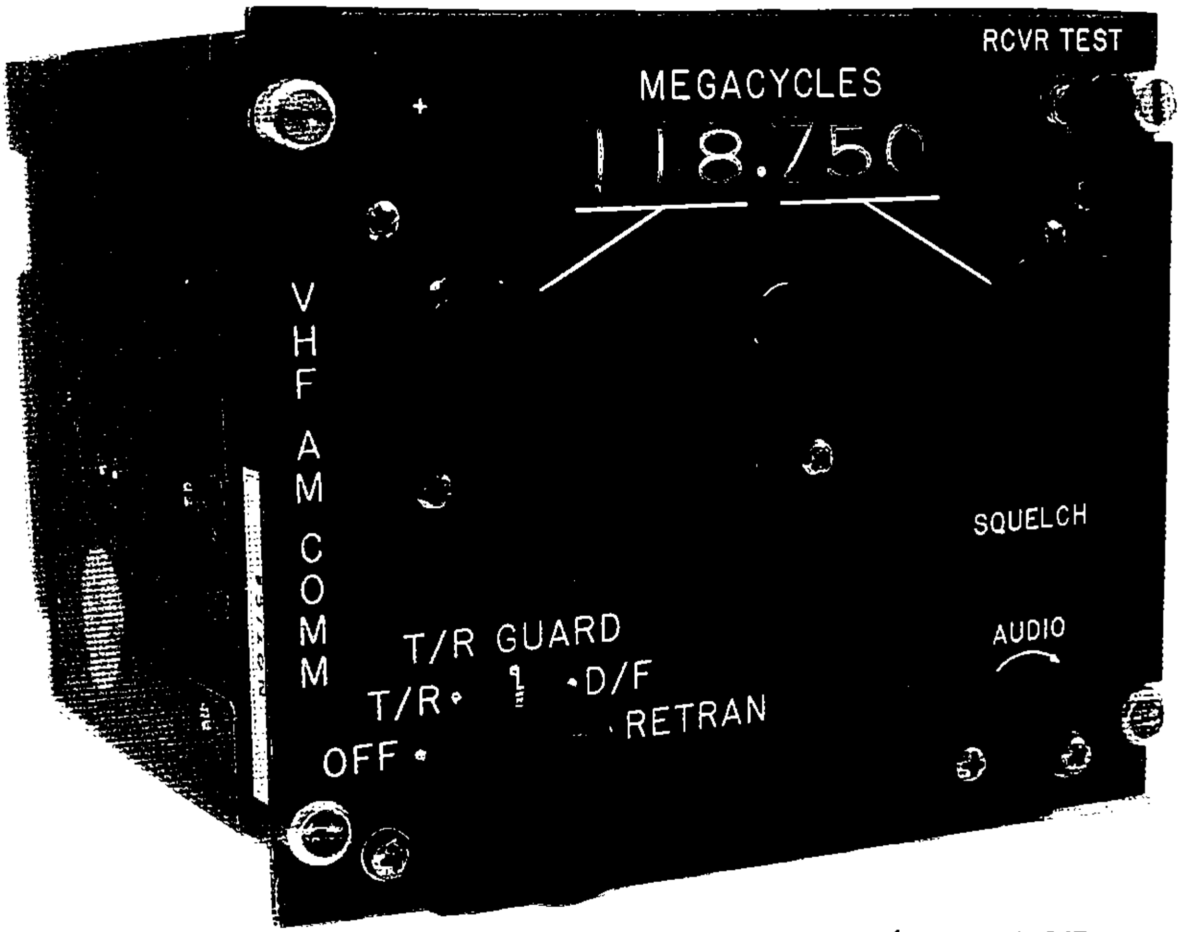
AN/ARC-114 VHF/FM RADIO SET



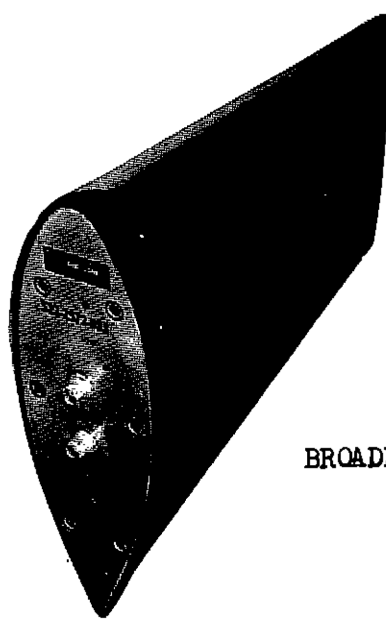
ATTENUATOR

VHF/FM HOMING ANTENNA

Figure 14-1



AN/ARC-115 VHF/AM RADIO SET



BROADBAND AM ANTENNA

Figure 14-2

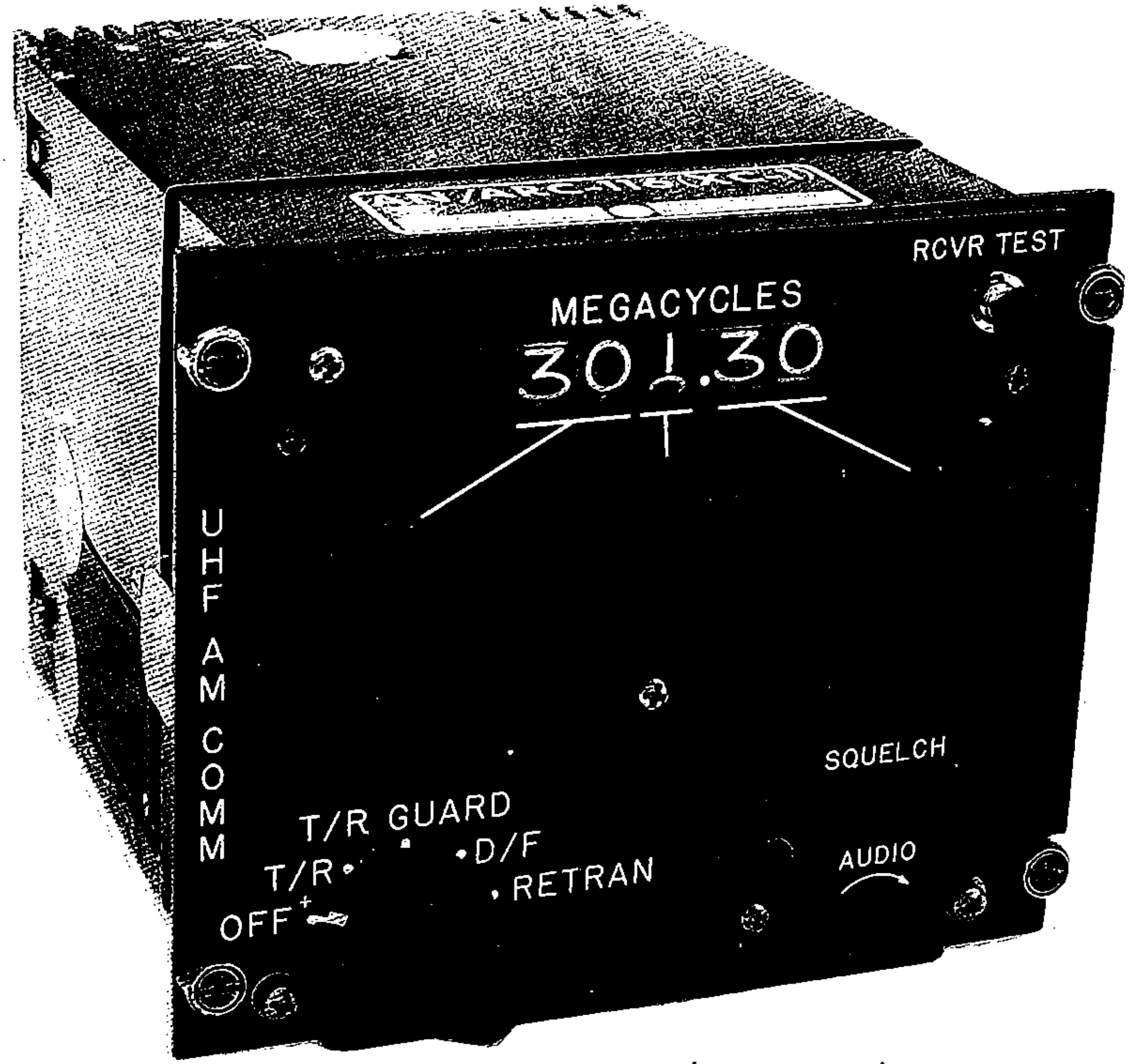
C. AN/ARC-116, UHF/AM Radio Set

The AN/ARC-116, UHF/AM Radio Set provides radio communications on any one of 3500 discrete channels in the range of 225 megahertz to 400 megahertz. The UHF/AM radio set transmits a nominal power of 10 watts and has a nominal effective range of 70 nautical miles at a helicopter altitude of 5000 feet above the terrain. Fixed "guard" channel monitoring at 243 megahertz is provided, and retransmission capability can be utilized when the UHF/AM is installed in the pilot's station. The direction finder (DF) capability is not used.

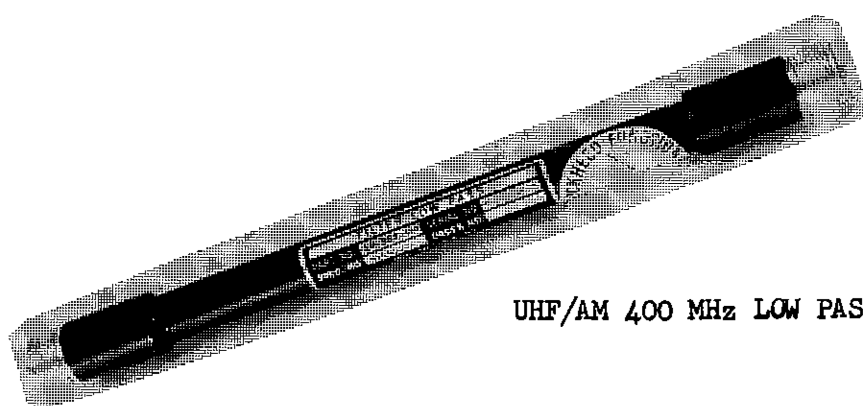
The pilot's AN/ARC-116, UHF/AM Radio Set is installed in the right console of the pilot's station and is of solid-state, single replaceable unit design. The AN/ARC-116, UHF/AM Radio Set shares the AT-1108 "Broadband AM Antenna" on the underside of the aft fuselage section with the copilot's AN/ARC-115, VHF/AM Radio Set. A low-pass filter installed in the aft avionics bay, is connected into the UHF/AM cabling to provide signal separation from the AN/APX-72, IFF. The pilot's UHF/AM radio set is directly interchangeable with the copilot's VHF/AM radio set after antenna reconnections and ICS "J" Box reconnections are made.

Transmission of the pilot's UHF/AM radio set is enabled by selection of XMTR SELECTOR position #2 on the pilot's ICS Control Panel and keyed by the RADIO/ICS switch on the pilot's cyclic stick. The copilot can transmit on the pilot's UHF/AM radio set by selection of XMTR SELECTOR position #2 on the copilot's ICS Control Panel and keying using either the copilot's cyclic control stick or the swivelling gunner's station left-hand grip.

The RCVR TEST pushbutton on the UHF/AM Control Panel verifies proper operation of the receiver section when a modulated tone is heard in the headset. Presence of a proper sidetone during a test transmission verifies proper operation of the transmitter section of the pilot's UHF/AM radio set.



AN/ARC-116 UHF/AM RADIO SET



UHF/AM 400 MHz LOW PASS FILTER

Figure 14-3

D. AN/ARC-102, HF/SSB Radio System

A high frequency/single sideband (HF/SSB) radio system was installed in the helicopter and tested satisfactorily in flight above 3 MHz to distances of 500 miles. Subsequent testing was curtailed and the system was removed from the vehicles at customer direction.

The pilot's HF/SSB radio system provided radio communications on any of 28,000 discrete channels in the range of 2 megahertz to 30 megahertz.

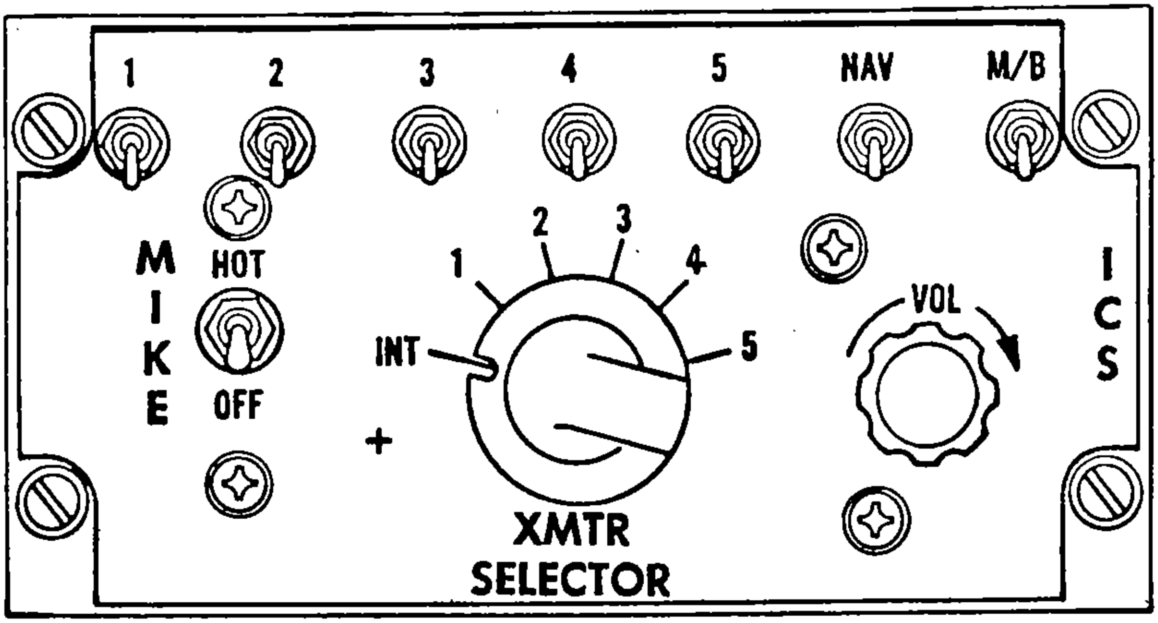
E. Intercommunication System

The Intercommunication System installed in the AH-56A provides intercommunication capabilities between the pilot and copilot, and between the crew members and ground personnel. Two Intercommunication System (ICS) Control Panels are installed in the helicopter, one in each of the crew stations. The pilot's ICS Control Panel is located in the left console of the pilot's station, and the copilot's ICS Control Panel is installed in the left console of the copilot's station. Each of the ICS Control Panels is interconnected with the Intercommunication System Junction (ICS "J") Box which is installed in the ceiling of the ammo debris bay. The ICS "J" Box contains the distribution circuits to provide the required audio interfaces.

The ICS Control Panel in each crew station contains a rotary XMTR SELECTOR switch which enables the keyline for the selected transmitter, a MIKE switch which enables "hot" microphone operation, a VOLume control, and selector switches which enable headset audio from the selected receiver. Numbered switch positions and their corresponding radio set selection are indicated for the pilot's ICS Control Panel in the following list:

(1) Switch 1: selects receiver audio for the pilot's AN/ARC-114, VHF/FM Radio Set.

(2) Switch 2: selects receiver audio for the pilot's AN/ARC-116, UHF/AM Radio Set.



PILOTS ICS PANEL CONTROL FUNCTIONS

AUDIO SWITCHES

1 VHF/FM -1*
2 UHF/AM
3 VHF/AM
4 HF-SSB
5 VHF/FM-2*
NAV ADF,VOR/LOC,
M/B MARKER BEACON

MIKE

PROVIDES HOT MIKE OPERATION IN INTERPHONE MODE ONLY

XMTR SELECTOR

INT INTERPHONE
1 VHF/FM-1*
2 UHF/AM
3 VHF/AM
4 HF-SSB
5 VHF/FM -2*

VOL

MASTER VOLUME CONTROL FOR SWITCHED AUDIOS.

UNSWITCHED AUDIOS

- INTERPHONE
- FLAWS
- ECM
- IFF

* COPILOT/GUNNERS PANEL CONTROL FUNCTIONS ARE IDENTICAL EXCEPT FOR ITEMS MARKED WITH AN ASTERISK. THESE ITEMS ARE REVERSED IN POSITION.

Figure 14-4. ICS Control Panel

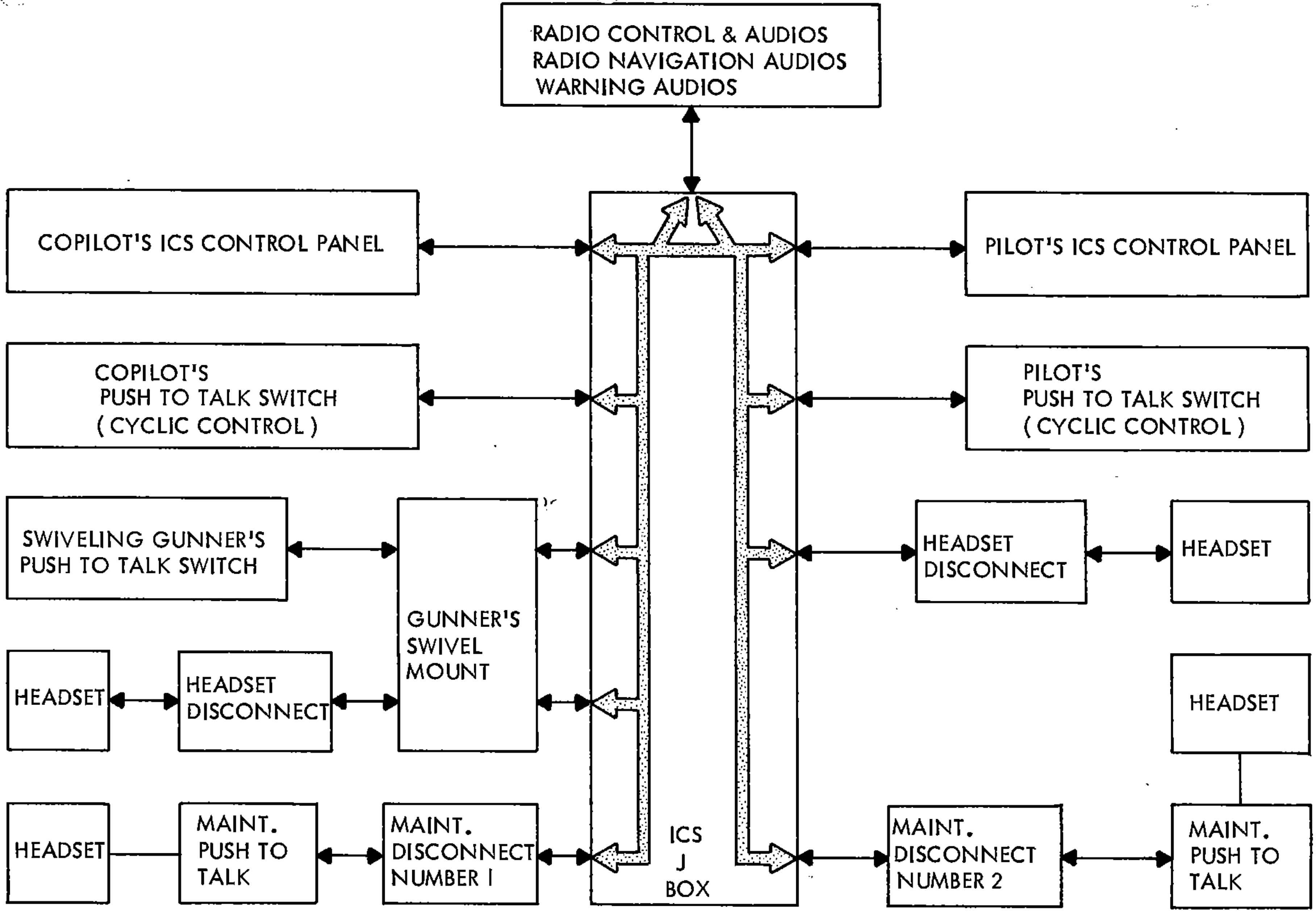


Figure 14-5. ICS Block Diagram

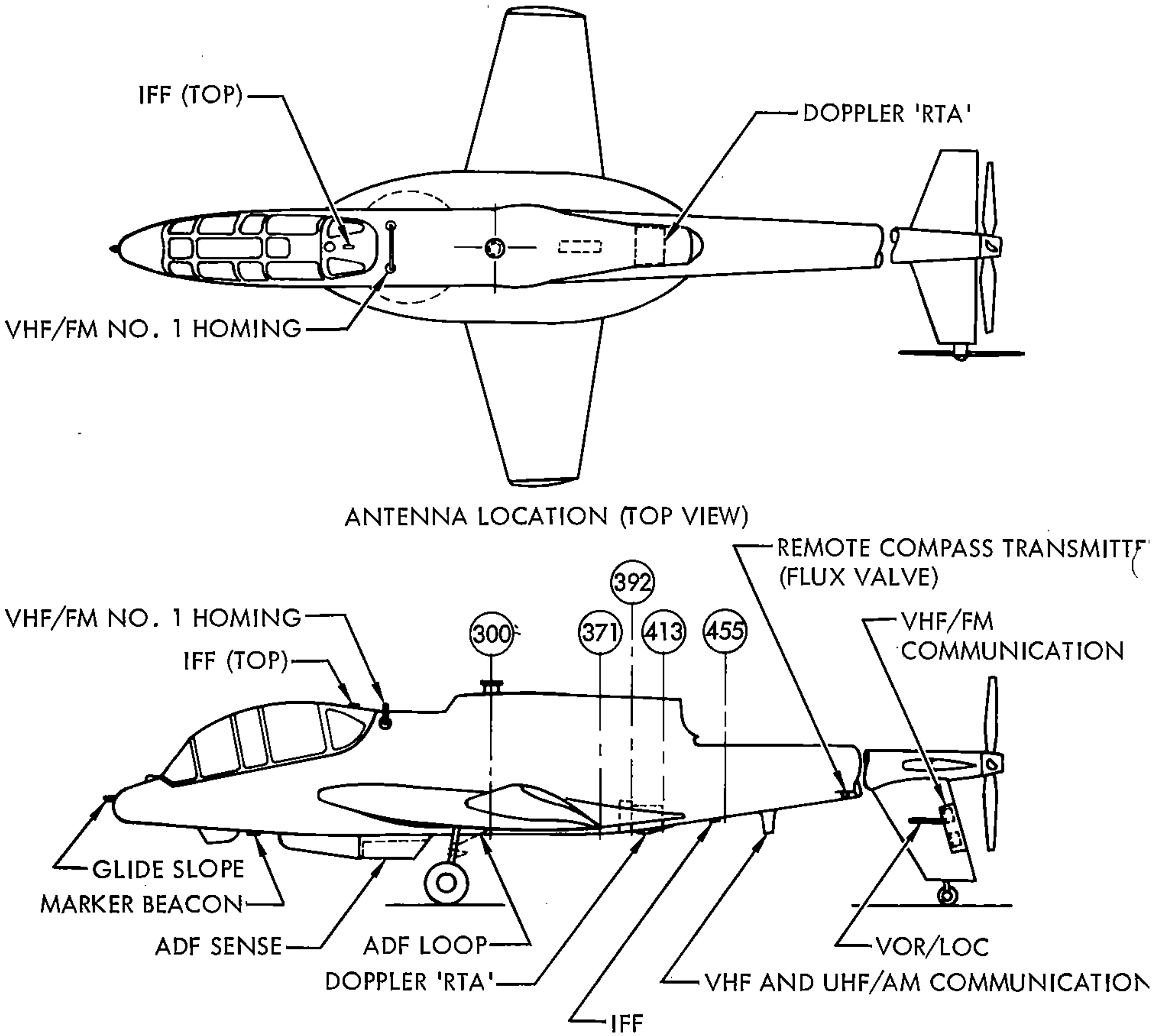
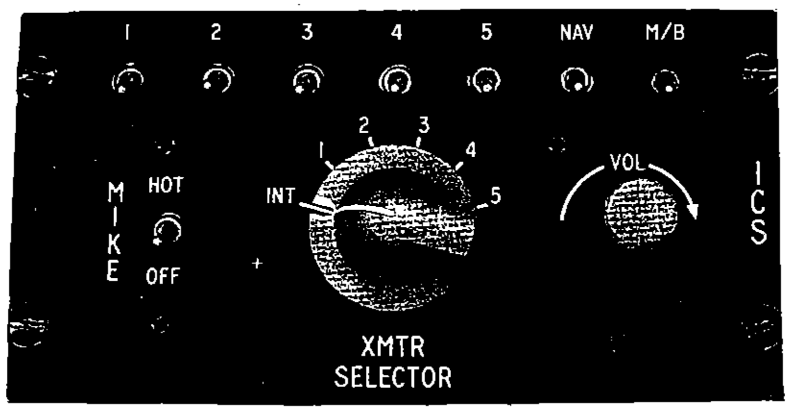
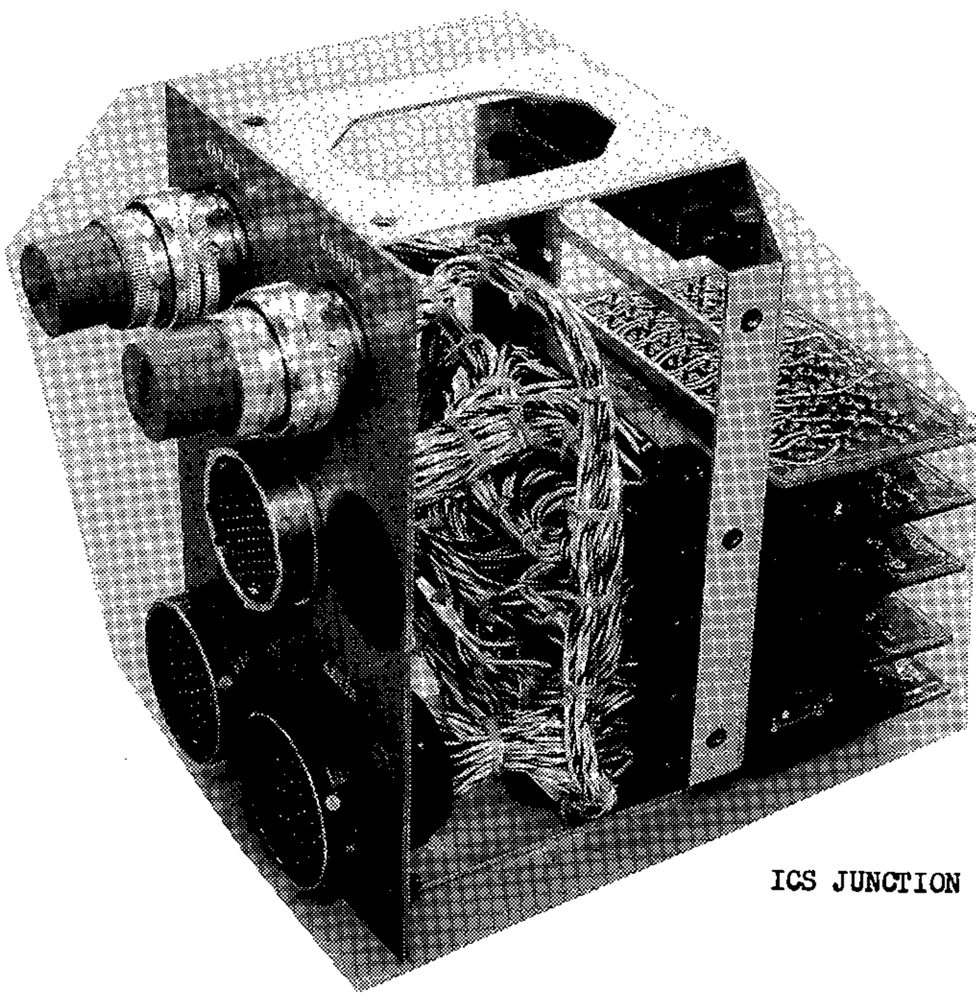


Figure 14-6. Antenna Locations



INTERCOM SYSTEM CONTROL PANEL



ICS JUNCTION BOX

Figure 14-7.

(3) Switch 3: selects receiver audio for the copilot's AN/ARC-115, VHF/AM Radio Set.

(4) Switch 4: selected receiver audio for the pilot's AN/ARC-102, HF/SSB Radio System (no longer used).

(5) Switch 5: selects receiver audio for the copilot's AN/ARC-114, VHF/FM Radio Set.

(6) NAV: selects station identification audio for the AN/ARN-82, VOR/LOC Receiver, and the AN/ARN-89, ADF Receiver.

(7) MB: selects identification tone for the AN/ARN-58, GS/MB Receiver.

Switch positions #1 and #5 are reversed in the copilot's station so that position #1 on the copilot's ICS panel selects receiver audio for the copilot's AN/ARC-114, VHF/FM Radio Set.

Ground crew headset connectors, labeled INTERCOM RECP on the helicopter, are provided at the battery vent access panel on the left forward fuselage section and at the hand hold above the aft portion of the right sponson.

F. Retransmission System

The Retransmission System installed in the AH-56A compound helicopter is a RETRANSMIT SELECT switch located at the forward portion of the fuel control panel in the left console of the pilot's station. The retransmission selector switch controls "feed" networks in the ICS "J" Box and provides a limited "radio relay" capability for the helicopter.

When the RETRANSMIT SELECT switch is positioned to "2 and 3" or to "5", command communications received on the pilot's AN/ARC-114 Radio Set are automatically retransmitted on the selected radio set if the mode selector on the control panel of the selected set has been positioned to RETRANS. Conversely, communications received on the

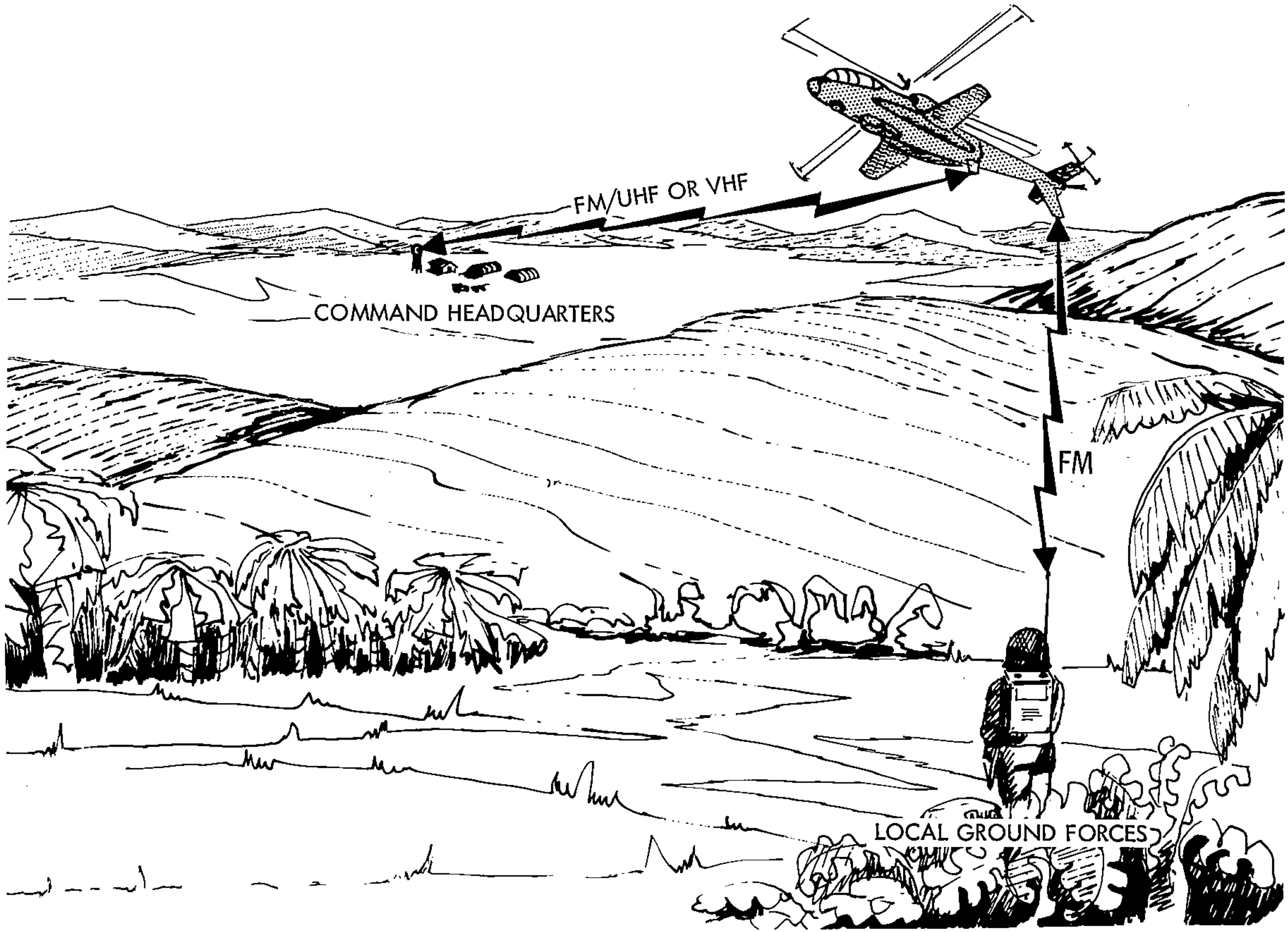


Figure 14-8. Retransmit Function - Radio Relay

selected radio set are automatically retransmitted on the pilot's VHF/FM Radio Set if the mode selector of the pilot's AN/ARC-114 has been positioned to RETRANS.

RETRANSMIT SELECTOR switch position, and the associated radio set selected for the retransmission link, are indicated in the following list:

(1) OFF: retransmission capabilities are not provided when OFF is selected.

(2) "2 and 3": retransmission capability is enabled on the pilot's AN/ARC-116, UHF/AM Radio Set (installed in the pilot's station) when the mode selector on the UHF/AM Control Panel is positioned to RETRANS. If the AN/ARC-115, VHF/AM is installed in the pilot's station, retransmission capability is enabled on the VHF/AM when the mode selector is positioned to RETRANS.

(3) "4": provisions for HF/SSB retransmission capability. The AN/ARC-102, HF/SSB Radio System previously installed in the helicopter did not have a retransmission capability.

(4) "5": retransmission capability is enabled on the copilot's AN/ARC-114, VHF/FM Radio Set if the mode selector on the copilot's VHF/FM has been positioned to RETRANS.

G. AN/APX-72, IFF System

The Identification Friend or Foe (IFF) System installed in the helicopter provides electronic radar identification of the helicopter for a friendly interrogating radar. After interrogation by a radar beacon, the helicopter's IFF System transmits a suitable replay to identify the helicopter as "friend".

Figure 14-9. IFF Control Panel

Figure 14-10. AN/APX-72 IFF Transponder

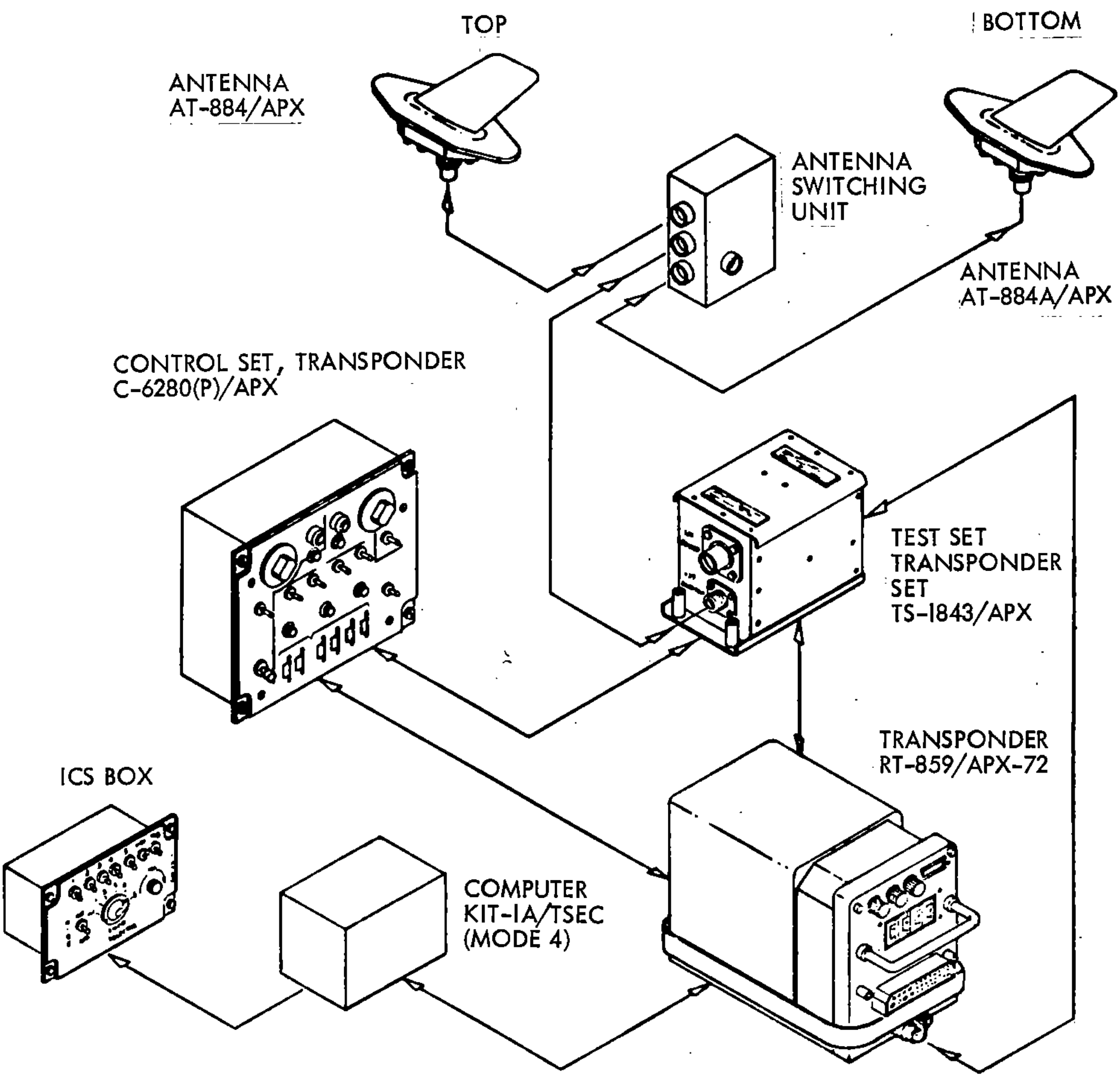


Figure 14-11. IFF System

The major components of the AN/APX-72, IFF System installed in the AH-56A are indicated in the following list:

(1) IFF Control Panel, which is installed in the right console of the pilot's station, provides code selection for Mode 1 reply and for Mode 3 reply, mode selection, identification point operation, and self test indications.

(2) AN/APX-72, IFF Transponder, which is installed in the main avionics compartment, provides reception, decoding, encoding, and transmission of the IFF signals. The IFF transponder is a solid-state, hermetically sealed unit and requires pressurization. Mode 2 code selection is made at the front panel of the IFF transponder.

(3) TS-1843, IFF Test Set, which is installed in the main avionics compartment, provides a complete ground or in-flight self-test for the IFF transponder.

(4) IFF Lobing Switch, which is installed in the main avionics compartment, provides automatic and continuous switching between the two IFF Antennae.

(5) Two IFF Antennae are installed. The top IFF antenna is installed directly behind the pilot's station on the top fuselage, and the bottom IFF antenna is located aft of the access to the aft avionics compartment on the underside of the fuselage.

(6) Three modes of operation, indicated in the following list, are used. Mode 4 capabilities can be incorporated by the addition of a special Mark XII Computer in the main avionics compartment. Mode C (altitude reporting capability) is provided in the transponder, but is not used in the present vehicle configuration.

a. Mode 1: 32 specific coded replies for security identification.

b. Mode 2: 4096 specific coded replies for personal (helicopter) identification.

c. Mode 3/A: 4096 specific coded replies for traffic control identification.

IV SYSTEM OPERATION

Each of the SLAE communications transceivers is a self contained unit. Their operation is identical to all other previous communications systems. The absence of the receiver test tone or transmitter sidetone indicates unit failure.

The intercommunication system contains no BITE or self-test capability but any failures are readily recognizable.

The IFF system is fully compatible with the FAA's air traffic control system. A vehicle installed test set allows for monitoring of proper replying by the transponder <u>and</u> provides the capability of performing a complete end-to-end test of the transponder.

The retransmit capability of the AH-56A utilizes the SLAE transceivers and their operational status may be monitored at all time during the automatic relaying sequence.

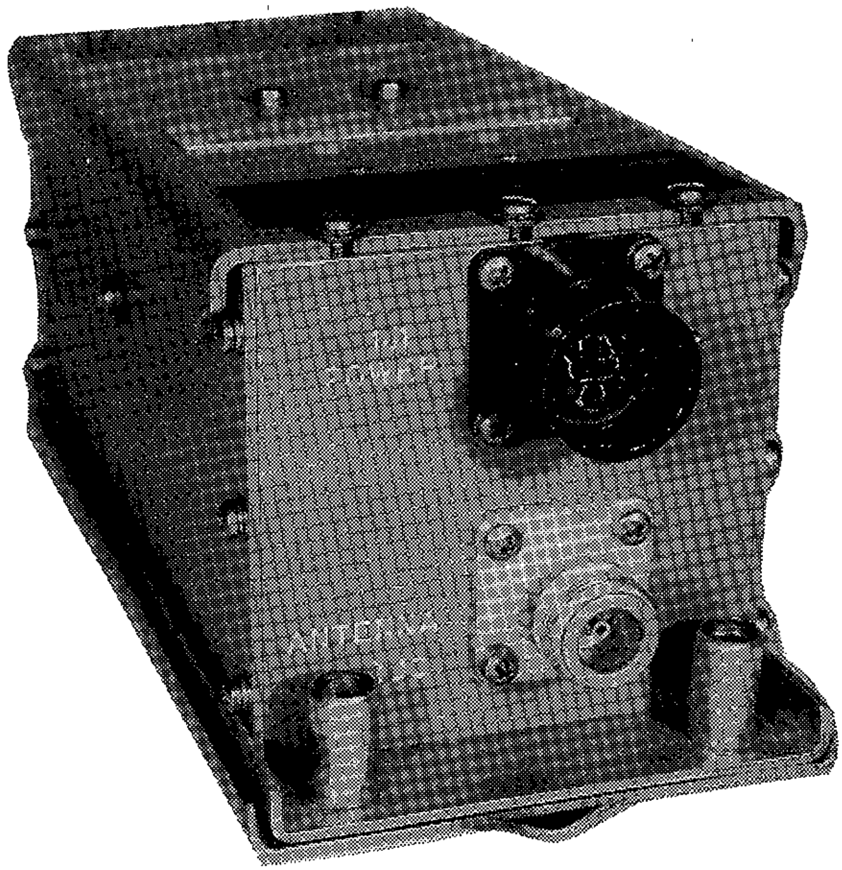
V P/CRS CONFIGURATION

The communications system in the approved producibility cost reduction configuration has remained basically unchanged. Following are the changes envisioned:

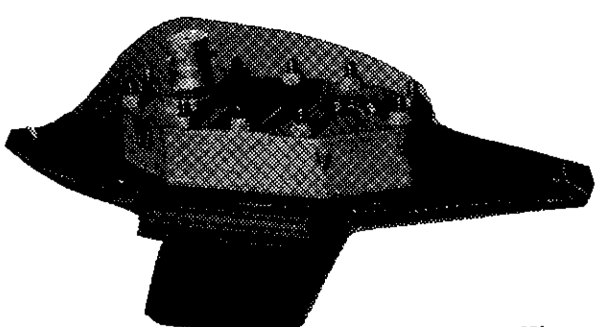
A. Replace Lockheed developed ICS control panel with Army developed SLAE ICS control panel.

B. Delete HF/SSB and its provisions (has been done at customer direction on existing AH-56s).

C. Delete the copilot/gunner's VHF/FM from the forward station <u>but</u> retain all space and wiring provisions.



TS-1843 IFF TEST SET



IFF ANTENNA

Figure 14-12.

COMPUTER CENTRAL COMPLEX

I. GENERAL DESCRIPTION

A. The Computer Control Panel (Figure 15A-1) is the primary external control of the Computer Central Complex. The pilot initializes the computer and enters the necessary information for a particular mission through the use of this panel. The Computer Control Panel is also used to modify the mission or weapon system status in flight.

B. The Computer Central Complex (CCC) is a lightweight, airborne, digital computer that is programmed to perform navigation, flight display and fire control computations. The CCC consists of two types of computing devices and the necessary input/out devices. One computing device, the Signal Transfer Unit (STU), utilizes whole number computational techniques similar to those of a general purpose (GP) business or scientific computer. The second computing device, the Central Processor Unit (CPU) uses incremental or digital differential analyzer (DDA) computing techniques.

Prime requirements of the CCC are (1) high reliability, (2) functional modularity (wherein the failure of one unit does not affect the operation of other units), and (3) a high degree of maintainability. The first requirement is satisfied by the complete use of integrated circuits and the use of triply redundant computing hardware. The second and third requirements are met in the STU by the use of multiple-module program memories and in the CPU by using a group of modules, all of one type -- the General Purpose Computing Module (GPCM) -- each of which has the capability of performing any of the various computational functions.

The purpose of the STU is to perform computations such as decision making, initialization of the CPU, and also many mathematical computations which are more amenable to whole word computation. The CPU computational units perform arithmetic calculations in real time using incremental techniques. Three GPCM's are used for navigation

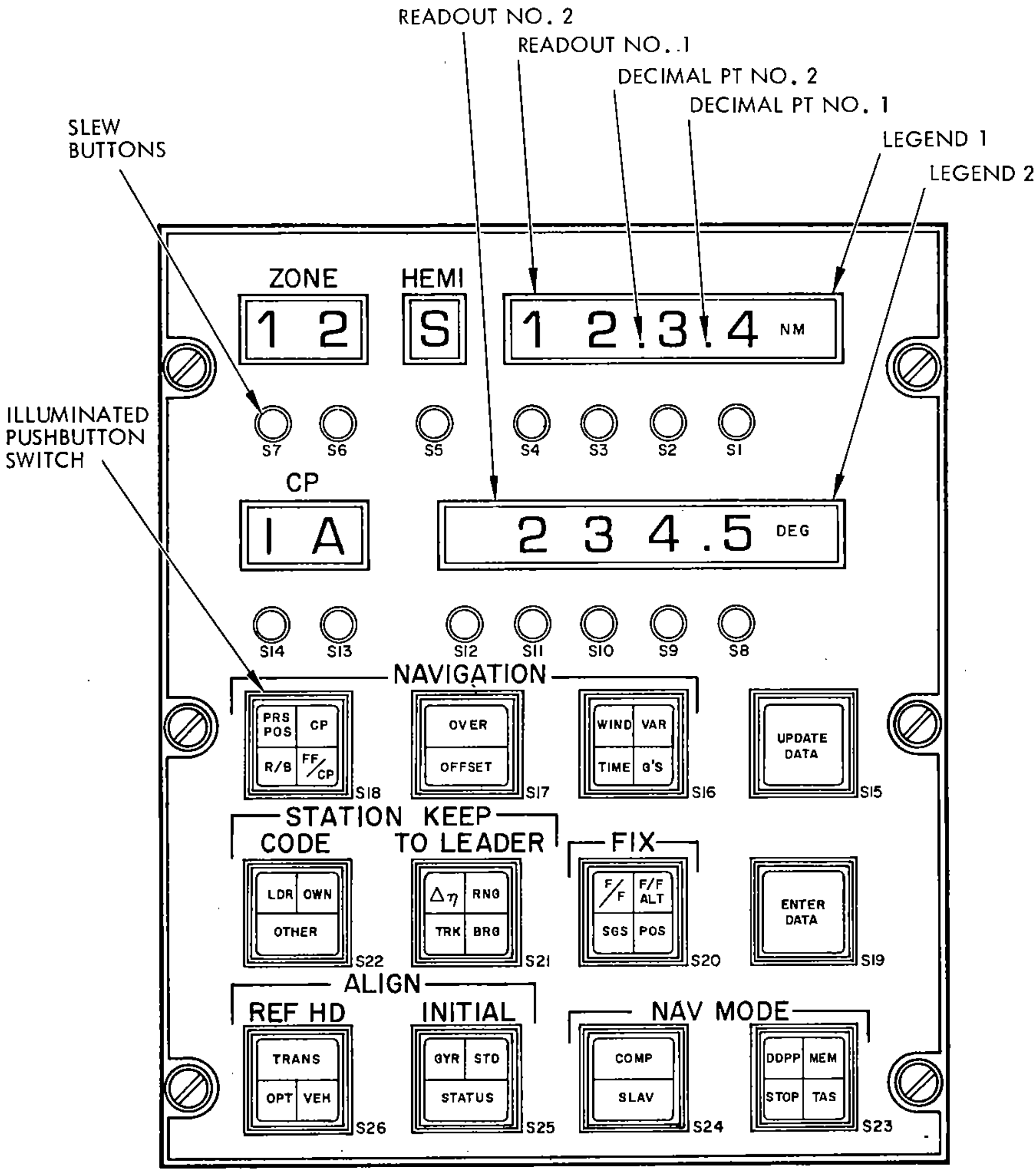


Figure 15A-1. Computer Control Panel

LOCKHEED CALIFORNIA COMPANY

computations, and the remaining twenty-one are devoted to fire control functions. There is a third unit in the CCC, the Digital Interface Unit (DIU), which is the primary input/output device for the system.

II. COMPONENTS AND LOCATIONS

A. The Computer Control Panel (CCP) is installed in the pilot's right console.

B. Computer Central Complex Components (see Figure 15A-2).

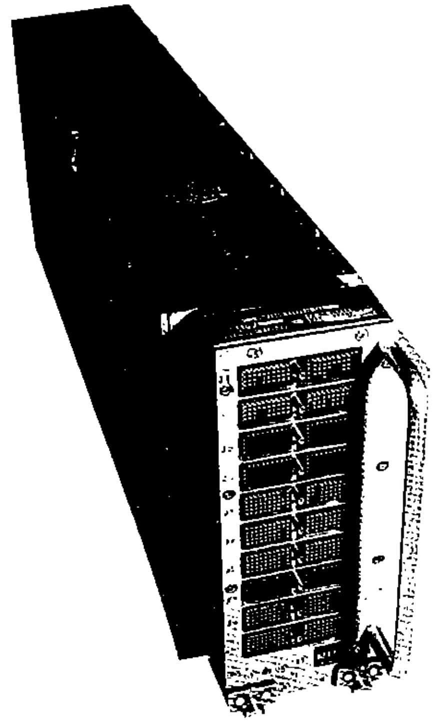
Name of Component	Number per Aircraft	Location in Aircraft
Signal Transfer Unit	1	Mounted on rack in main avionics compartment
Central Processor Unit	3	Mounted on rack in main avionics compartment
Digital Interface Unit	1	Mounted on rack in main avioncs compartment

These units mounted in a special rack which provides interconnection, cooling, and power distribution.

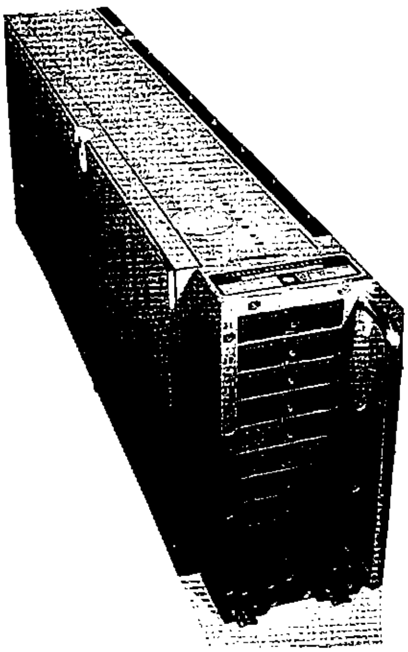
III. MAJOR COMPONENT DESCRIPTION

A. The computer control panel contains indicators, pushbutton switches, and illuminated pushbutton switches.

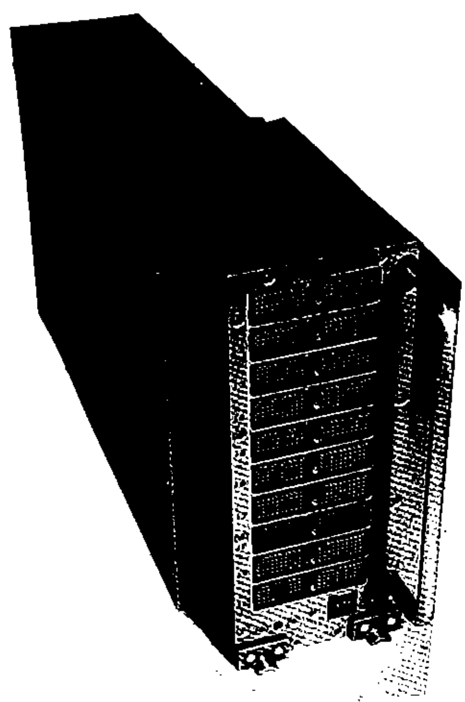
1. Indicators and Pushbuttons. The panel contains alpha, numeric, and legend indicators. The pushbutton switch below each alpha or numeric indicator is used to provide a discrete signal for the computer to advance the associated indicator to the next position. (Advances one position for each switch depression.) The six and eight position legend indicators are associated with Readouts 1 and 2 respectively. The legend indicators display the



SIGNAL TRANSFER UNIT



CENTRAL PROCESSOR UNIT



DIGITAL INTERFACE UNIT

Figure 15A-2

units to be associated with the number displayed in Readouts 1 and/or 2. Positioning of the legend indicator is by computer command, and is dependent on the logic implemented by the panel illuminated pushbutton switch selection(s).

2. Illuminated Pushbutton Switches. The twelve illuminated pushbutton switches control the selection of all functions within the capability of the Computer Control Panel. Actuating a switch (or combination of switches) will provide signals to the Computer Control Panel shift register(s) to store the switch commands and enable the computer to input this information on its next cycle. A single switch may implement from one to four separate operations. On multisegment switches, a single switch depression will enable the next clockwise switch function and will be indicated by the switch's illuminated display. The appropriate display will be illuminated when a signal from the computer is received by the panel, indicating that the computer has received and understands the switch commands. Several illuminated switches in the ALIGN and NAV MODE have segmented display which are annunciators only and are not controlled by the normal associated switching relationship. The following is a list of displays which function as annunciators:

   STATUS
   COMP
   DOP
   MEM
   TAS

3. Through the use of the CCP, the pilot can select the method of alignment of the inertial platform, the mode of operation of the navigation equipment, and various other functions. Also, he may select a number of quantities stored in and/or computed by the CCC to be displayed on the two readouts of the CCP, including vehicle present UTM map coordinates, distance to a checkpoint, etc.

B. CCC Major Components

1. Signal Transfer Unit. The STU comprises an arithmetic and control section, an input/output section, and a memory section. The STU computational elements are entirely micro-electronic (except for magnetic core storage) and feature "triple-redundant" mechanization. That is, there are three arithmetic sections always simultaneously processing the same problems, and the results are "voted" upon in such a way that two of the three sections must agree (have the same answer) before the results are used for further computations and/or for an output from the STU.

   The STU (in fact the entire CCC system) does not include operating controls or indicators, except for the built-in test equipment described below. However, two avionics equipment units external to the CCC do exercise direct control over the STU. The pilot's MODE SELECT panel, located on the pilot's left forward panel contains the on-off switch for the CCC. The computer control panel (CCP), located on the pilot's right hand console is both the primary control and the readout (indicator) for the CCC navigation functions.

   The CCC, including the STU, contains extensive self-testing mechanisms. A self-test program checks the operation of the arithmetic and memory sections of the STU; built-in test equipment (BITE) monitors the operation of many of the individual modules of the CCC. For the STU, there are individual BITE indicators to indicate the status, on a go/no-go basis, of the memory section (one BITE indicator set by the self-test program), the arithmetic and control section (three BITE indications, one for each of the module pairs which make up the "triple redundant" section), the power supply (one BITE indicator), and one unit BITE indicator which is set if any of the above are set.

   The unit BITE indicator on the STU is monitored by the Fault Location Aural Warning System (FLAWS).

2. Central Processor Unit. CPU employs one type of module only, designated a general purpose computer module (GPCM). The equation solved by each GPCM is determined by wiring external to the GPCM but internal to the CPU. This feature makes the GPCM's fully interchangeable.

   The three CPU's are identical (physically and functionally interchangeable). Two "node points" (two-out-of-three majority "voting" elements) are included in each GPCM, allowing the three identical CPU's to be used in a triple-redundant configuration. There are no operating controls associated with the CPU. It is controlled entirely by the STU program. The only indicators on, or associated with, the CPU are the BITE indicators. There is one BITE indicator for the CPU power supply, one for each of the 24 GPCM's, and one unit BITE indicator which is set if any of the 25 module BITE indicators are set.

   The three CPU unit BITE indicators are monitored by the FLAWS.

3. Digital Interface Unit. The DIU has two functions in the CCC: it contains the primary power supply for all units (designated the "pre-regulator"), and it is the primary input/output device for the entire system. All modules except the power supply are devoted to input-output functions.

   The only indicator on the DIU is the unit BITE indicator, which is set by a failure of the power supply (pre-regulator). The DIU unit BITE indicator is monitored by the FLAWS.

IV. SYSTEM OPERATION

A. Computer Control Panel

1. Functions performed through the use of this panel are:

   a. Preflight alignment of the HARS.
   b. Selection of navigation mode.
   c. Selection of navigation idle mode (stopmode).
   d. Navigation data readout.

e. Navigation data entry.

f. Target fixing.

g. Special.

2. The Computer Control Panel has three basic operating modes:

   a. Display - the desired mode of operation is selected on the illuminated pushbutton switches. The appropriate data is displayed in Readouts 1 and/or 2.

   b. Direct data entry using Readouts 1 and/or 2 and the push-button switches.

   c. Data entry from a source other than the Computer Control Panel readouts. In some modes of operation (e.g., FF) two sets of data are required for a solution, necessitating double depression of ENTER DATA. Following initial entry the readouts are blanked and UPDATE remains illuminated to indicate that insufficient data has been provided and another entry required.

B. Computer Central Complex

The CCC has no specific "operational procedures". Assuming the CCC circuit breakers have been set, computer operation will commence when the computer ON/OFF switch on the pilot's Mode Select Panel is set to ON. The computer will cycle through its program routines as a function of the system and modes selected by the crew or by the status of peripheral equipments themselves. These are described in the following paragraphs. A simplified system block diagram is shown in Figure 15A-3.

1. The Computer Central Complex utilizes both General Purpose (GP) and Digital Differential Analyzer (DDA) techniques to solve the navigation, display and fire control problems of the AH-56A. A general purpose digital computer operates on whole words; i.e., each data word is complete and meaningful in itself, and is independent of past values of the data. A DDA computer is an

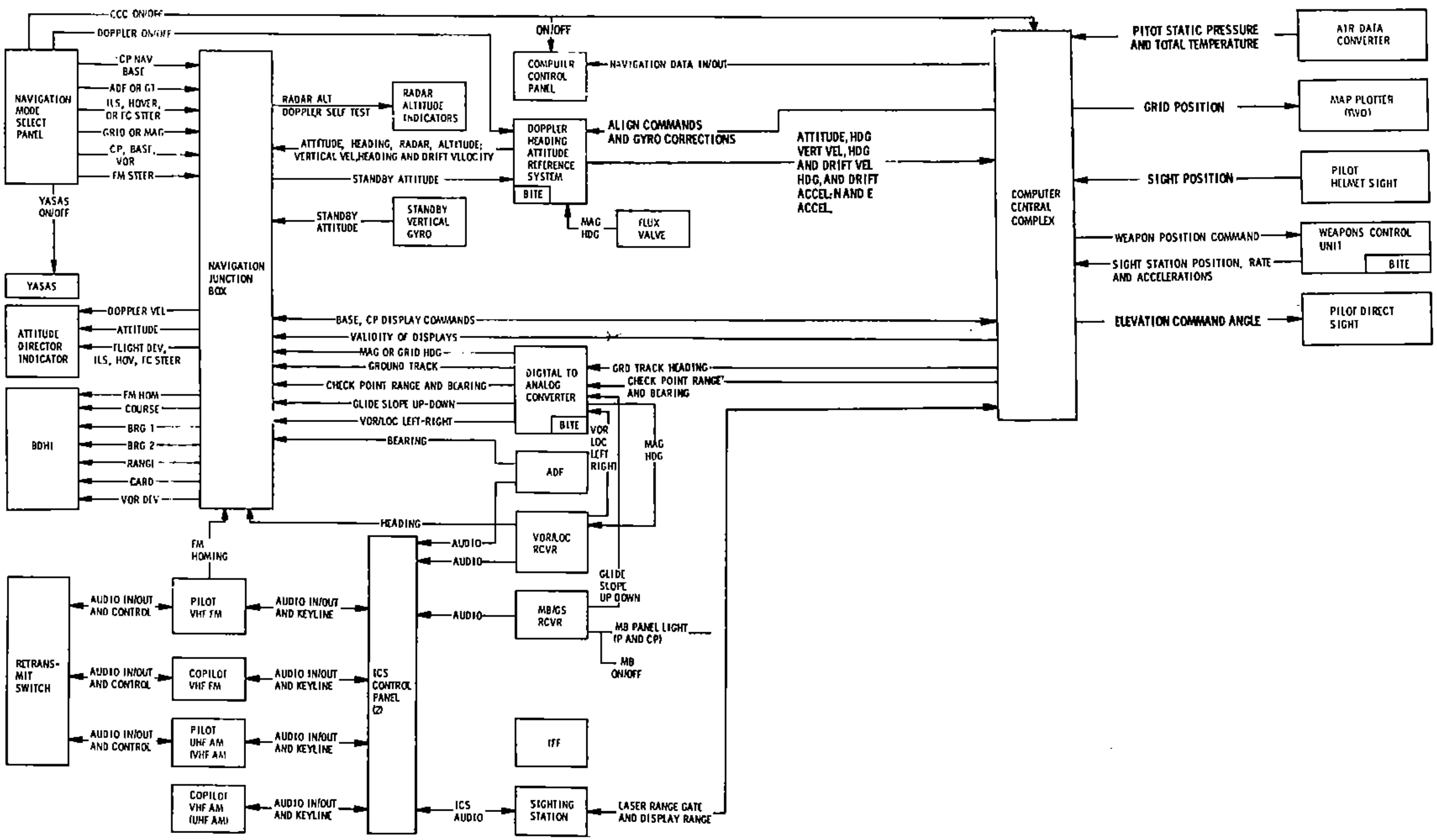


Figure 15A-3. Simplified System Block Diagram incremental device. Each new bit of data used is <u>not</u> equal to the value of a variable; rather, it is equal to the change in the value of the variable from the last time it was sampled and/or operated upon. A DDA computer is especially well suited to the solution of differential equations. Also, the solution of coordinate transformations and rotations, where high angular rates require a high computation rate, are best handled by DDA incremental techniques.

The CCC is employed to provide very accurate, high speed solutions to the following group of problems.

a. Coordinate transformation. It is continually necessary to perform transformations among earth coordinates, aircraft coordinates, gunline coordinates, map grid system coordinates, pilot's helmet sight coordinates, and doppler navigator coordinates. For example, doppler navigator input data must be transformed into map grid system coordinates before it can be used to compute distance traveled.

b. Linear motion compensation. When aiming at a ground target, the copilot/gunner's line of sight must be continuously corrected to compensate for changes in aircraft present position.

c. Ballistic solutions. The gunline angle must be continuously determined with respect to the sightline angles (in both azimuth and elevation) to compensate for gravity drop, aircraft motion, target motion, weapon aerodynamic characteristics, etc.

d. Platform stabilization. Three axis corrections must continuously be made to the navigation system's stable platform to account for earth rotation, vehicle movement and coriolis accelerations.

e. Display logic. The CCC controls many navigation and fire control displays such as distance to a check point, present position coordinates, heading, etc.

2. More specifically, the basic functioning of the CCC may be broken down into the following five major sections, corresponding to the five major subroutines of the STU program.

   a. Executive. This program subroutine determines which other subroutines are to be processed, and in what order of priority. The CCC is forced to process, upon pilot and/or co-pilot command, fire control computations twenty times per second, and navigation and Computer Control Panel (CCP) computations and display functions five times per second. When computer time is available, including but not limited to those times when no fire control processing is demanded, computer self-test routine is run. In addition, program "interrupts" are controlled by the executive program. These interrupts, which are explained in detail in Section V, are: power on, power failure, fire control, and STU malfunction. When any one of these interrupt conditions exists, a particular executive subroutine must be commenced.

   b. Navigation. The total navigation problem is subdivided into three sections: platform alignment, Heading and Attitude Reference System (HARS) servicing, and present position solution.

      (1) Alignment comprises timing the sequence through which the stable platform must proceed in order to achieve alignment, and providing proper servicing of the platform during each of the stages of this sequence. There are several modes of alignment, including coarse alignment, fine leveling, gyrocompassing, and optical alignment. During these modes, tests are made by the CCC to determine when new modes can be entered, and many feedback gains are changed as a function of mode.

(2) HARS servicing includes accepting acceleration data from the two sections of the stable platform, and converting them to two sets of velocities. These are compared with Doppler derived velocities to provide Doppler damping of inertial velocities. The damped velocity terms are used to compute the gyro torquing rates needed by the platform, and are used by the Doppler radar system to obtain inertial-damped Doppler velocities. The gyro torquing rates consist of earth rate, coriolis rate, and vehicle rate terms.

(3) Present position computation is normally accomplished using the inertially - damped Doppler velocities. However, the CCC program is set up so that in case of Doppler and/or inertial failure, Air Data Converter derived signals, pilot entered wind data, and back-up heading information will be used. The CCC solves a detailed and complicated set of equations to compute present position in Universal Transverse Mercator (UTM) coordinates.

c. Fire Control. The CCC is able to solve ballistic equations and provide coordinate conversions in order to aim two weapons simultaneously and independently. Sighting angles, to which ballistic functions are added, are derived from a pilot's helmet sight, pilot's direct sight, or the copilot-gunner's stabilized seat/sight (swiveling gunner's station). The CCC drives the stabilized seat/sight to compensate for aircraft linear motion. It also solves ballistic equations for two guns, grenade launcher, rockets, and attitude functions for a missile. In addition, the CCC is used to normalize resolver outputs, process laser ranging data, filter and mix velocities, and smooth target motion as subfunctions of the target tracking and prediction process.

d. Display and Control. The CCC provides an extensive group of navigation displays to the Computer Control Panel (CCP). It solves equations for and/or causes the display of vehicle present position, range and bearing to a selected check-point, wind speed and direction, ground speed, time-to-go to a check point, magnetic variation, ground coordinates of a sighted position, etc.

One of the most interesting and complex CCP/CCC operations is the "target fix" function. For this function, the CCC accepts swiveling gunner's station (SGS) sight angles and laser range and provides:

(1) The capability to update vehicle present position data without flying over a check point.

(2) Display on the CCP the range and bearing between two sighted ground points.

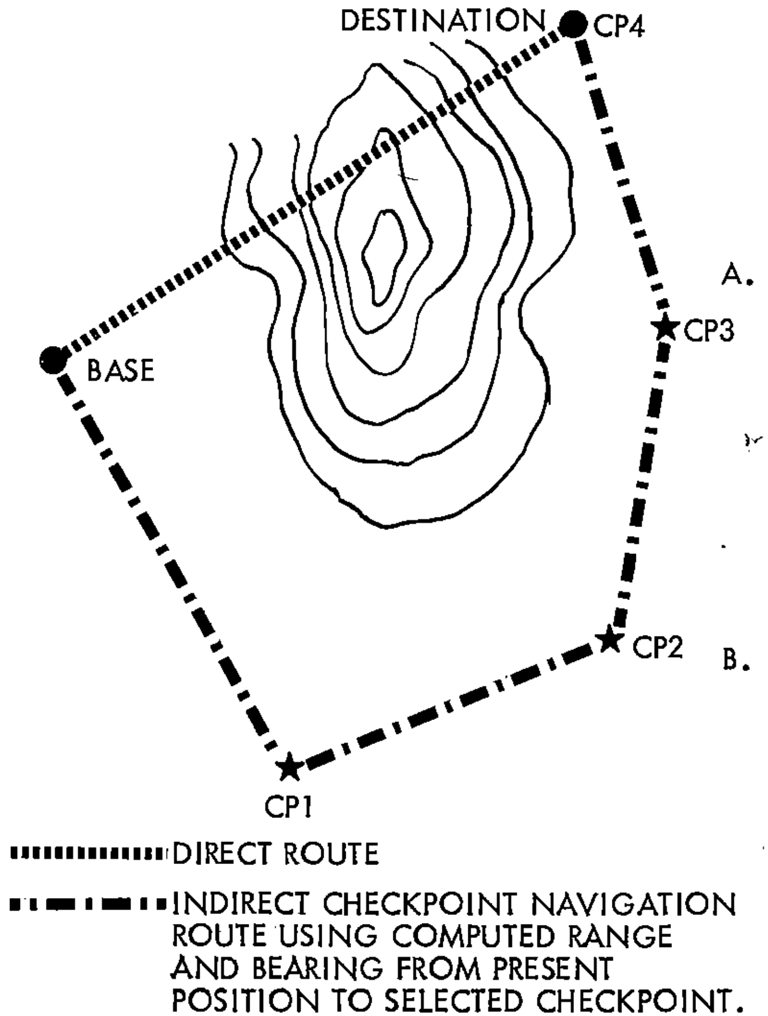
(3) The capability to determine the UTM coordinates of any sighted point.

(4) The capability to determine the position of a sighted point when the Laser Range Finder is inoperative, by using two sightings and triangulation computations.

These operations are illustrated in Figures 15A-4 through 15A-9.

Furthermore, most of the CCP switching and control logic functions are performed by the CCC. For example, the CCC program is structured such that only compatible combinations of CCP switch positions will be accepted as valid input commands. This feature reduces the complexity, and hence the physical size, of the CCP. In addition, the CCC drives two map plotters to display vehicle present position, and a Digital-to-Analog Converter which supplies a Bearing, Distance, Heading Indicator (BDHI) with heading, bearing to a check-point, range to the same checkpoint, and ground track angle.

PURPOSE: TO NAVIGATE FROM POINT-TO-POINT VIA STORED CHECKPOINTS, AND TO DETERMINE PRESENT POSITION RELATIVE TO BASE OR SELECTED CHECKPOINT.



STEPS

A. CHECKPOINT NAVIGATION

1. SEQUENCE NAV SWITCH TO R/B (RANGE BEARING) OR CP.
2. ACTUATE CP (CHECKPOINT) SWITCH UNTIL APPROPRIATE CHECKPOINT NUMBER IS DISPLAYED.
3. READ ON CCP AND/OR BOTH THE RANGE AND BEARING FROM PRESENT POSITION TO SELECTED CHECKPOINT OR UTM COORDINATES OF SELECTED CHECKPOINT.
4. SET VEHICLE COURSE TO CHECKPOINT & NAV. TO CHECKPOINT
5. REPEAT 2 THRU 4 UNTIL DESTINATION CHECKPOINT IS REACHED.

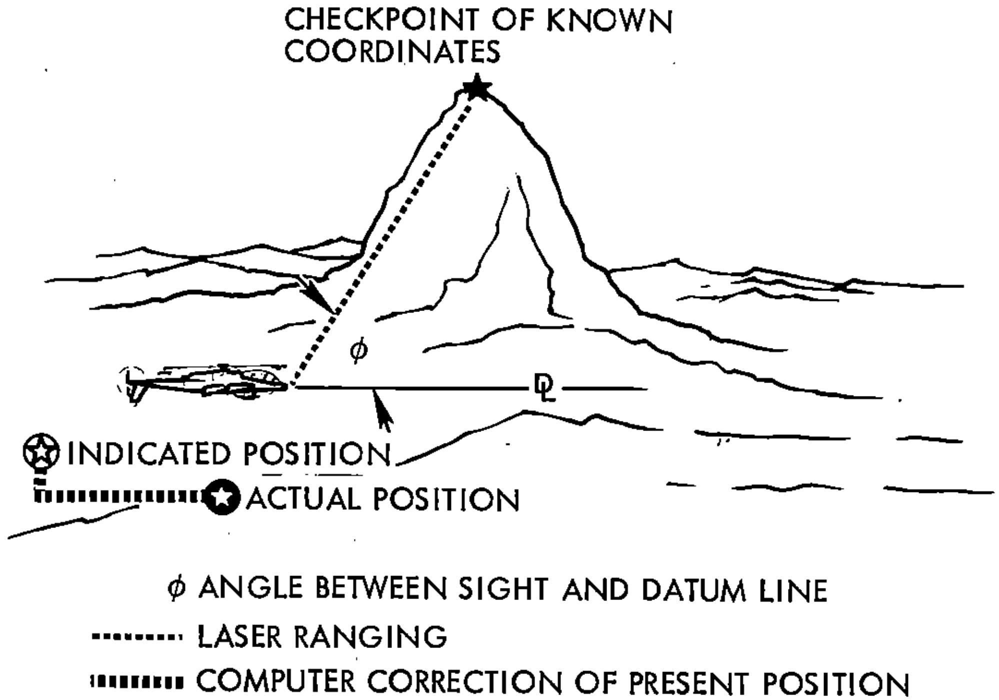
B. POSITION FIXING

1. SEQUENCE NAV SWITCH TO POS(POSITION).
2. READ PRESENT POSITION IN UTM COORDINATES FROM CCP.

NOTE:
BDHI NEEDLE NO. 1 CAN BE SELECTED TO GROUND TRACK AND BDHI NEEDLE NO. 2 CAN BE SELECTED TO CHECKPOINT BEARING. BY MARRYING NEEDLES, THE MOST DIRECT ROUTE TO CHECKPOINT WILL BE FLOWN.

Figure 15A-4. Checkpoint Navigation - Position Fixing

PURPOSE: ELIMINATION OF NAVIGATION ERRORS WHEN UNABLE TO FLY OVER CHECKPOINT OF KNOWN COORDINATES



STEPS

1. SEQUENCE NAV SWITCH TO POS
2. ACTUATE UPDATE DATA SWITCH
3. ACTUATE OFFSET SWITCH
4. ENTER ZONE, HEMI & COORDINATES OF KNOWN CHECKPOINT INTO CCP
5. SLEW SGS UNTIL CHECKPOINT IS IN SIGHT
6. OPERATE LASER
7. ACTUATE ENTER DATA SWITCH ON CCP
8. CORRECTED POSITION COORDINATES WILL BE STORED IN COMPUTER MEMORY AND DISPLAYED ON CCP

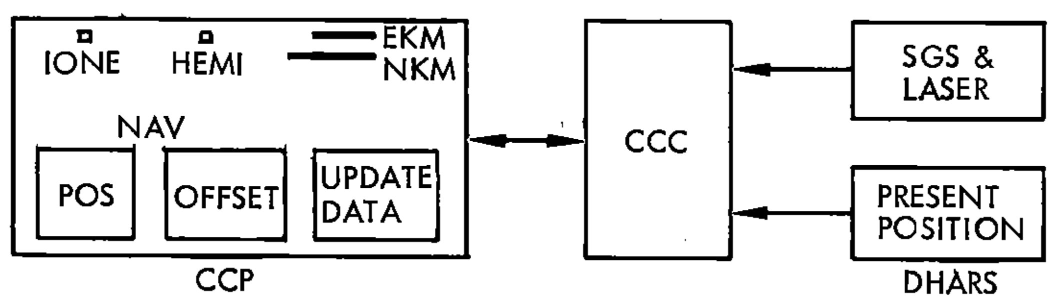
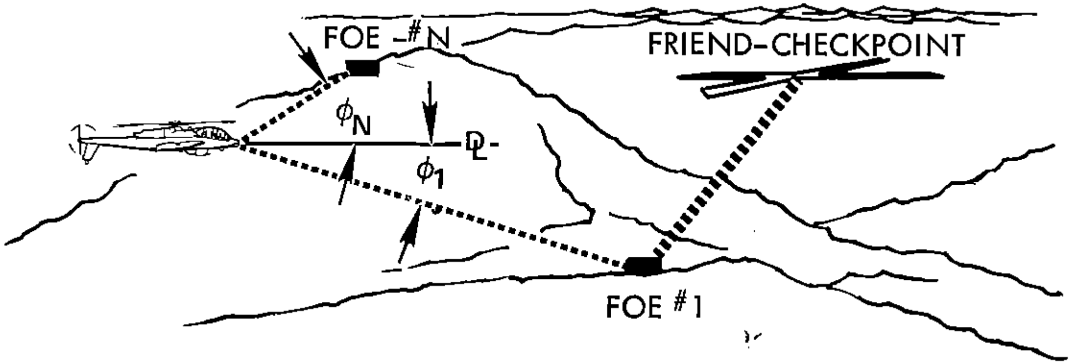


Figure 15A-5. Present Position Correction - Offset Update

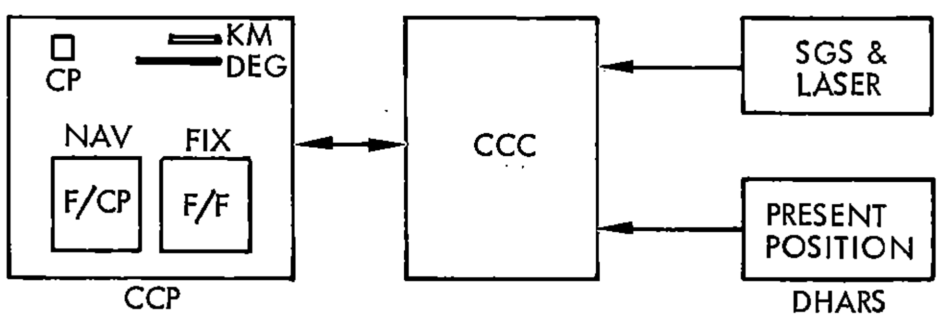
PURPOSE: DETERMINATION OF RANGE & BEARING TO A TARGET-FOE FROM FRIENDLY POSITION OF KNOWN UTM COORDINATES STORED AS A CHECKPOINT.



ϕ ANGLE BETWEEN SIGHT AND DATUM LINE

.... LASER RANGING

IIIIIIIIII COMPUTED RANGE & BEARING OF FOE RELATIVE TO FRIEND



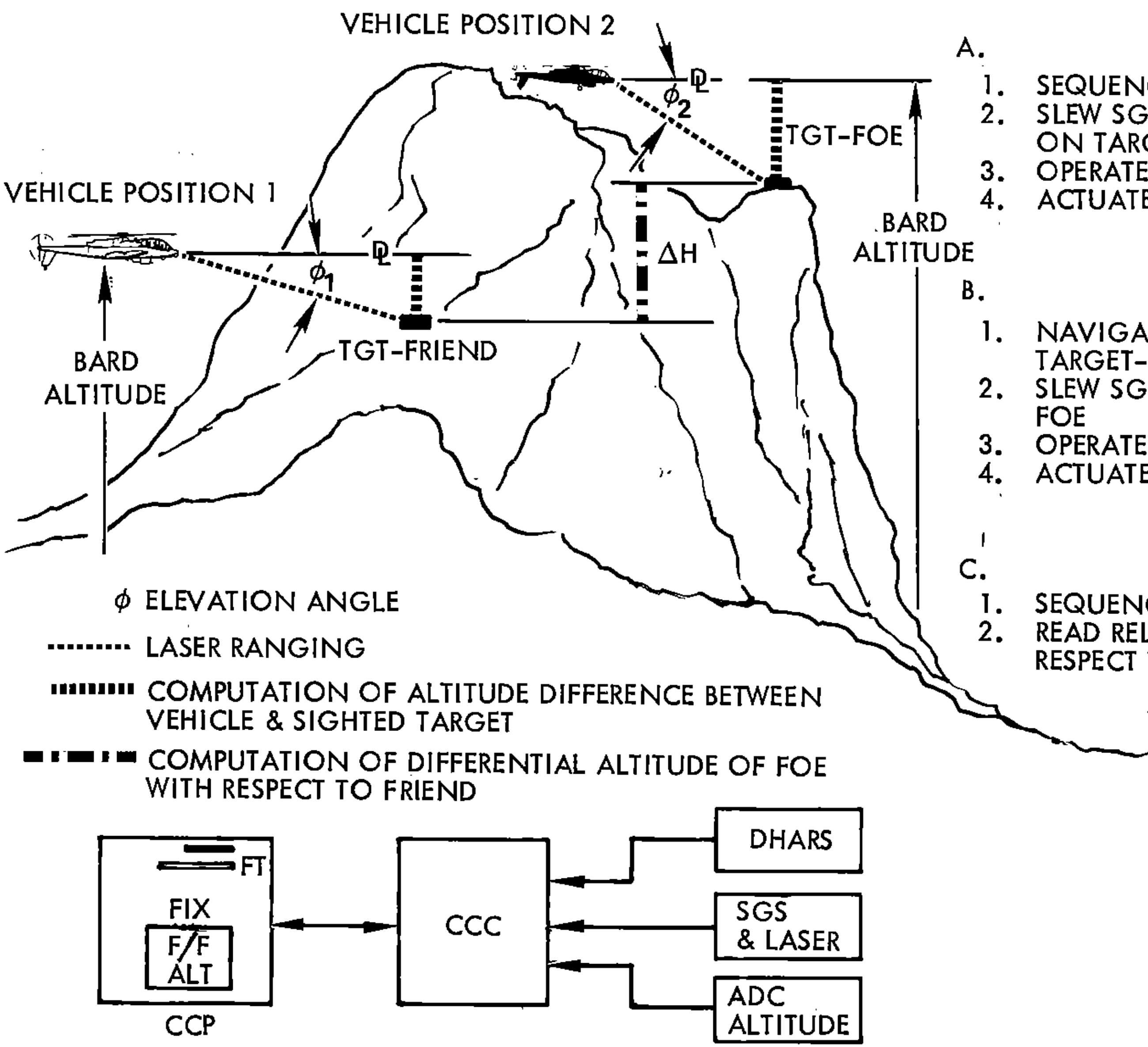
STEPS

A. 1. SEQUENCE FIX SWITCH TO F/F (FRIEND OR FOE)
   2. SEQUENCE NAV SWITCH TO F/CP (FRIEND/CHECKPOINT)
   3. ACTUATE CP SWITCH UNTIL APPROPRIATE CHECKPOINT NUMBER APPEARS IN CP WINDOW
   4. ACTUATE ENTER DATA SWITCH ON CCP WHICH INSERTS COORDINATES OF FRIEND-CHECKPOINT
   5. SLEW SGS UNTIL SIGHT IS ON TARGET-FOE
   6. OPERATE LASER
   7. ACTUATE ENTER DATA SWITCH.
   8. READ RANGE AND BEARING OF TARGET-FOE FROM FRIEND FROM CCP

B. 1. REPEAT A.5, A.6, A.7 & A.8 FOR N TARGET-FOES

Figure 15A-6. Target Fixing - Friend/Foe (Stored Checkpoint)

PURPOSE: DETERMINATION OF RELATIVE ALTITUDE OF FOE WITH RESPECT TO FRIEND



STEPS

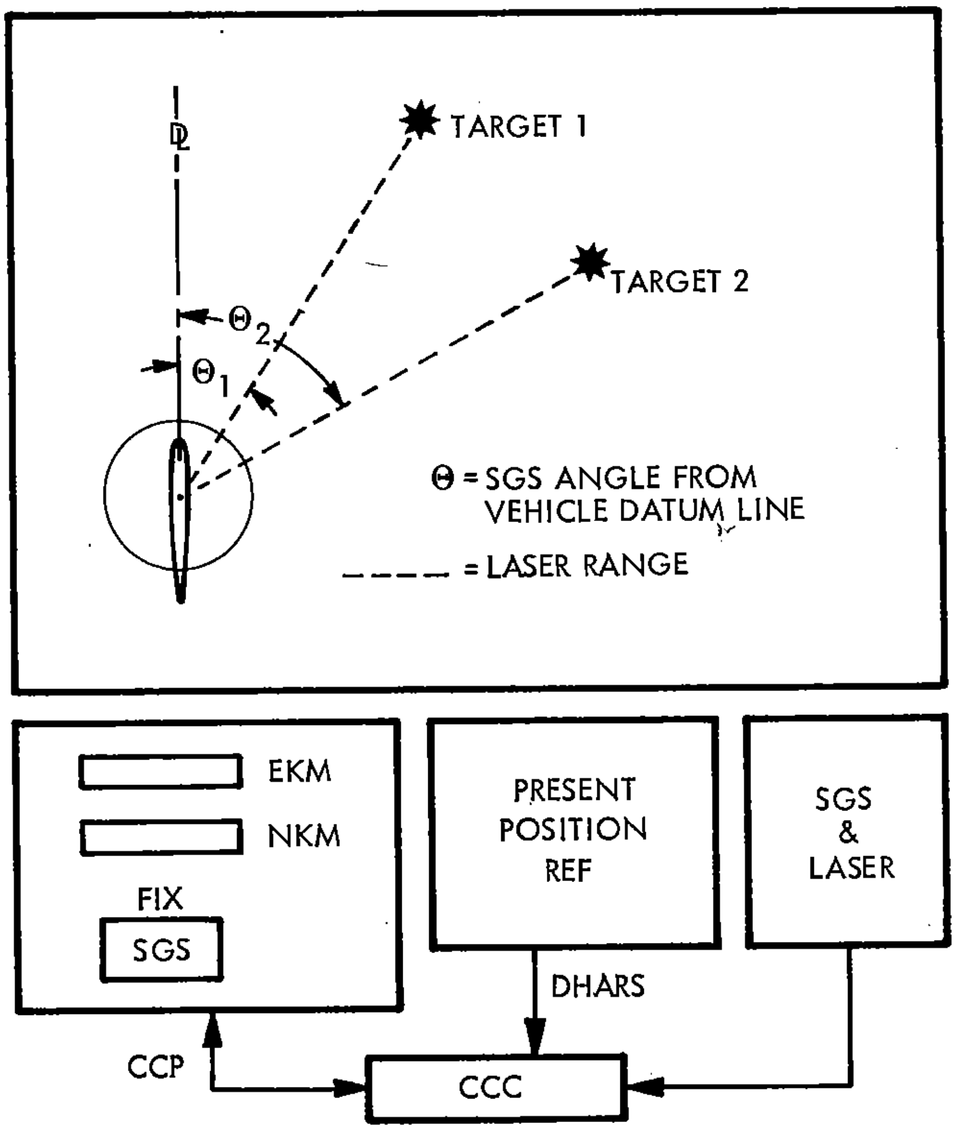
A.
1. SEQUENCE FIX SWITCH TO F/F
2. SLEW SGS UNTIL SIGHT IS ON TARGET-FRIEND
3. OPERATE LASER
4. ACTUATE ENTER DATA SWITCH ON CCP

B.
1. NAVIGATE TO NEW POSITION WHERE TARGET-FOE CAN BE SIGHTED
2. SLEW SGS UNTIL SIGHT IS ON TARGET-FOE
3. OPERATE LASER
4. ACTUATE ENTER DATA SWITCH ON CCP

C.
1. SEQUENCE FIX SWITCH TO F/F ALT
2. READ RELATIVE ALTITUDE OF FOE WITH RESPECT TO FRIEND ON CCP.

Figure 15A-7. Target Fixing - Friend/Foe Altitude

PURPOSE: DETERMINATION OF UTM GRID COORDINATES OF A TARGET WITH SGS AND LASER.



STEPS

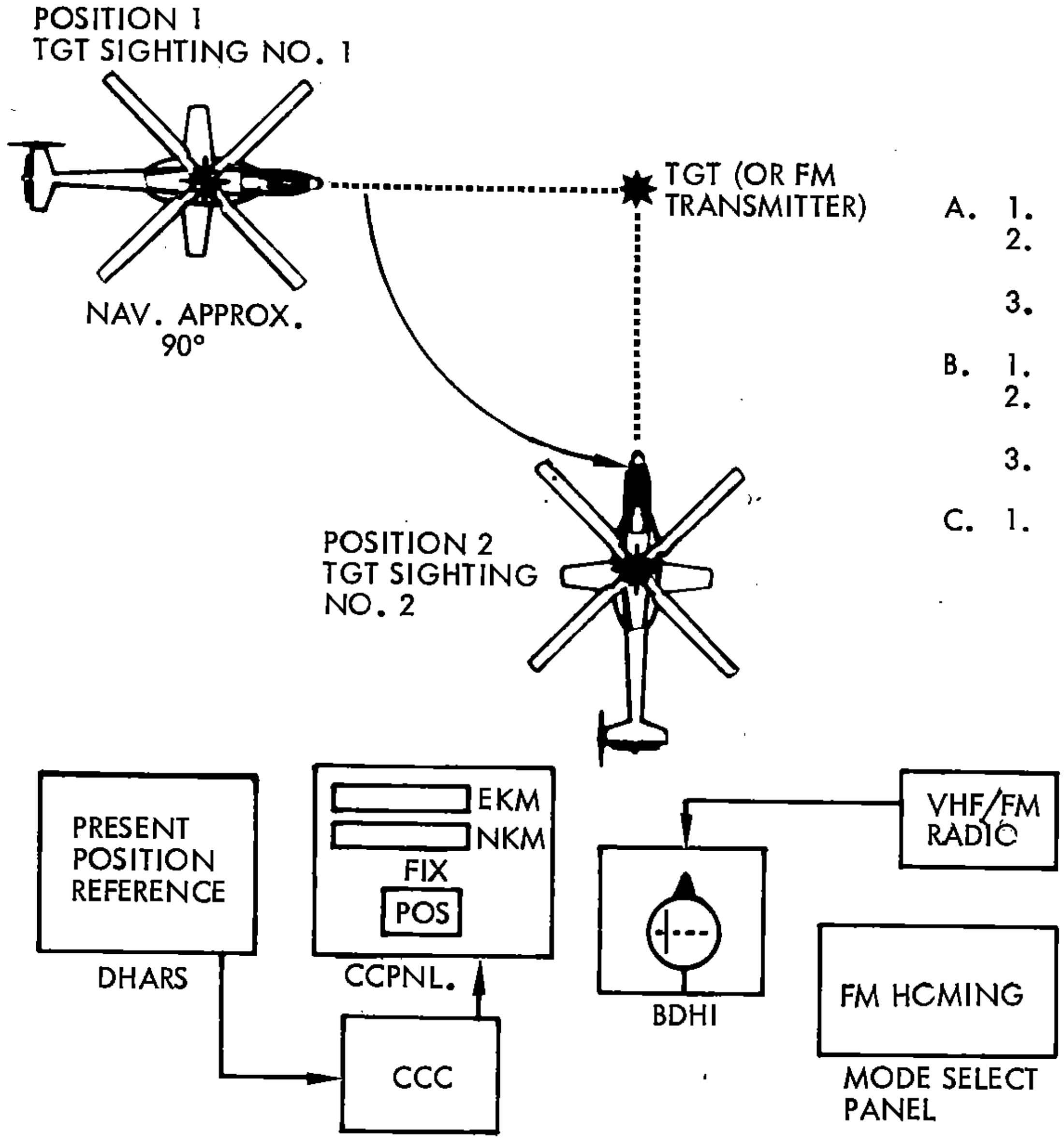
A. 1. SEQUENCE FIX SWITCH TO SGS (SWIVELING GUNNERS STATION)
   2. SLEW SGS, POSITION SIGHT ON TARGET AND OPERATE LASER
   3. ACTUATE ENTER DATA SWITCH FOR COMPUTER COMPUTATION.

B. READ UTM GRID COORDINATES OF TARGET FROM CCP.

C. REPEAT FOR OTHER TARGETS

Figure 15A-8. Target Fixing - SGS and Laser

PURPOSE: DETERMINATION OF UTM GRID COORDINATES OF A TARGET WITH FIXED SIGHT OR FM HOMING



STEPS

A. 1. SEQUENCE FIX SWITCH TO POS (POSITION)
   2. AUGN AIRCRAFT DATUM LINE (OR ZERO FM HOMING NEEDLES - BDHI)
   3. ACTUATE ENTER DATA SWITCH

B. 1. NAVIGATE TO POSITION 2
   2. ALIGN AIRCRAFT DATUM LINE (OR ZERO FM HOMING NEEDLES)
   3. ENTER DATA FOR COMPUTER COMPUTATION

C. 1. READ UTM GRID COORDINATES OF TARGET (FM TRANSMITTER) FROM THE CCP.

Figure 15A-9. Target Fixing - Fixed Sight or FM

V. PCRS CONFIGURATION

A. The PCRS Computer Central Complex is basically a repackaged Baseline computer, but with improvements obtained from the experience of recent programs. Notably, these improvements include a higher digital computer clock rate, more versatile input/output which provides signal conversion capability, a new bulk memory, improved power supply, and improved front panel connectors. Table 1 provides a comparison of characteristics. The advantages obtained from these changes are addressed in the following paragraphs:

1. Clock Rate. The digital computer clock rate has been increased from 750 KHz to 1.2 MHz. This has the immediate effect of increasing computer speed (except IOC instructions) by 60 percent. In terms of hardware, this means the digital computer has time to perform some computations now done by the DDA processor unit. Eight (8) GPCM are now required instead of twenty-four (24). The disadvantages of the inflexible "wired-in" DDA method of programming are reduced by the reduction in the number of GPCMs used.

   One computer unit is eliminated by combining the converter unit and and the remaining eight GPCM in one 20-inch long box. Attendant benefits are a reduction in computer weight, power dissipation and cooling air requirements.

2. Input/Output. The I/O section of the computer will include the analog-to-digital and digital-to-analog conversion functions. This will eliminate the need for separate conversion units.

3. Memory. A bulk memory will replace the DRO 1024-word memory modules. The benefits are improved reliability and reductions in power, size, and weight.

4. Power Supply. An improved power supply now in use on another system will replace the converter power supply. Advantages obtained are increased reliability, higher efficiency, and shorter computer-to-operation time following prime power

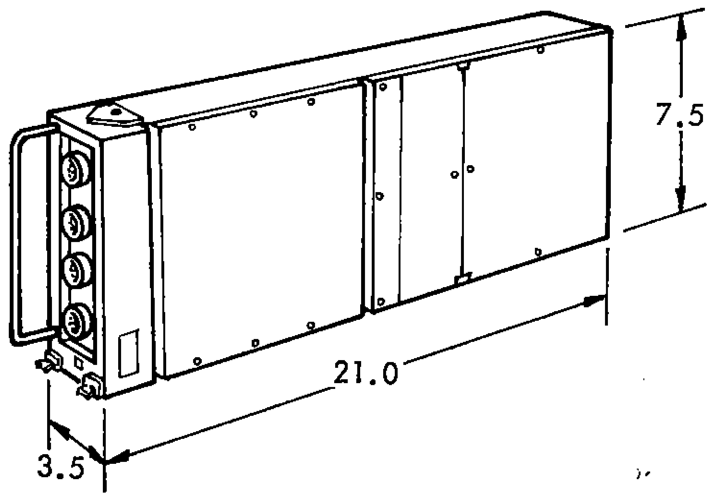
Table 1. Summary of Computer Characteristics

Machine Organization	PCR and Baseline	
Type	General purpose, stored program, digital computer	
Number System	Fractional binary, fixed point, two's complement	
Operation	Full parallel (20 bits)	
Data Word Length	20-bit word including sign, 40-bit double-word including sign	
Instruction Word Length	20 bits	
Number of Instructions	29	
Registers	9 (U, L, X1, X2, X3, PC, IR, MAR, MIR Registers)	
Operation Times	**PCR**	**Baseline**
Typical Execution Times		
Add	7.5 μs	12 μs
Add double	15 μs	24 μs
Multiply	22.5 to 39.2 μs	36 to 66 2/3 μs
Divide	39.2 μs	62 2/3 μs
Main Storage	**PCR**	**Baseline**
Type	3D lithum ferrite magnetic core, nonvolatile, random access, DRO	3D lithum ferrite magnetic core, nonvolatile, random access, DRO
Capacity	16,384 20-bit words (max)	8192 - 20-bit words (max)
Modularity	4,096/8,192 20-bit words per pluggable module	1024 - 20-bit words per pluggable module
Cycle time	3.3 μs (write or read restore)	5.33 μs
Access time	1.7 μs	3.04 μs
Electronics	Monolithic	Monolithic
Special features	3D organization	3D
Clock	1.2 MHz	750 KHz

application, either at initial turn-on or after a power interrupt. Computer recovery time is reduced from 800 milliseconds to 100 milliseconds.

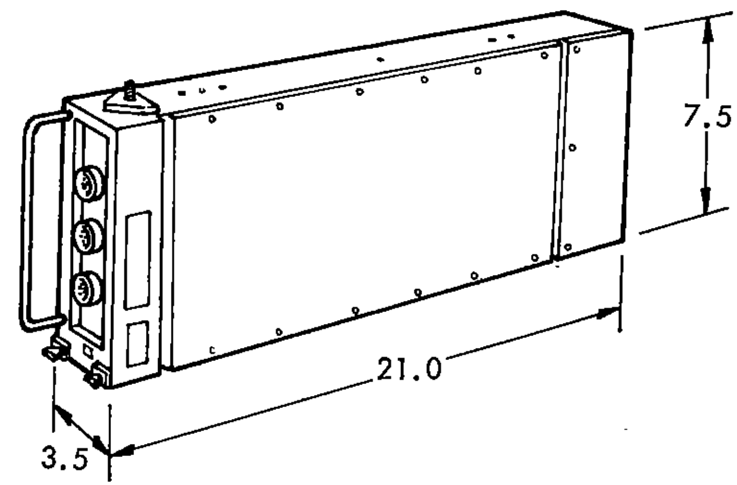
B. Hardware Configuration

The PCRS computer will make some use of Baseline computer hardware, maintaining continuity with the Baseline computer. Additionally, the higher clock rate of 1.2 MHz permits more computations being done by the digital computer (formerly STU and programmer) portion of the machine, reducing the number of DDA processor's general purpose computer modules (GPCM) from 24 to 8. The triple redundancy of the arithmetic modules has been eliminated in the programmer. This reduction in modules permits repackaging of the remaining modules into the vacated space of the processor (formerly CPU) and reducing the LRU count to two boxes for the PCRS computer. The two computer LRUs are the computer and the interface units and are shown in Figure 15A-10.



SIZE - 519 IN^3
WEIGHT - 19.8 LBS
POWER - 153
MTBF - 5851

INTERFACE UNIT



SIZE - 519 IN^3
WEIGHT - 18.8 LBS
POWER - 137
MTBF - 4369

COMPUTER UNIT

Figure 15A-10. PCRS Computer Central Complex

AIR DATA CONVERTER

I. GENERAL DESCRIPTION

The Air Data Converter (ADC) provides pitot static pressure and total temperature information to the CCC.

II. COMPONENT LOCATION

The ADC is located in the main avionics compartment.

III. MAJOR COMPONENT DESCRIPTION

A. The ADC electronics are contained in seven plug-in modules for ease of maintenance:

Static pressure
Dynamic pressure
Total temperature
Power supply, timing and reference
3-Transmitters, Digital Data (TDD's)

B. The complete ADC (figure 15B-1) weighs approximately 8 pounds and follows MIL-HDBK-217 derived MTBF of 4000 hours.

IV. SYSTEM OPERATION

A. There are no "operating procedures" for the ADC. Assuming ADC circuit breakers are set, the ADC turns ON when ship's power is applied.

B. The ADC converts inputs of static pressure (P_s), total pressure (P_t) and total temperature (T_t) into digital outputs of static pressure (P_s), dynamic pressure (Q_c) and total temperature (T_t) for use by the CCC. The CCC uses these inputs to compute true airspeed (TAS), mach number (M), air density ratio and other functions for the navigation and fire control systems.

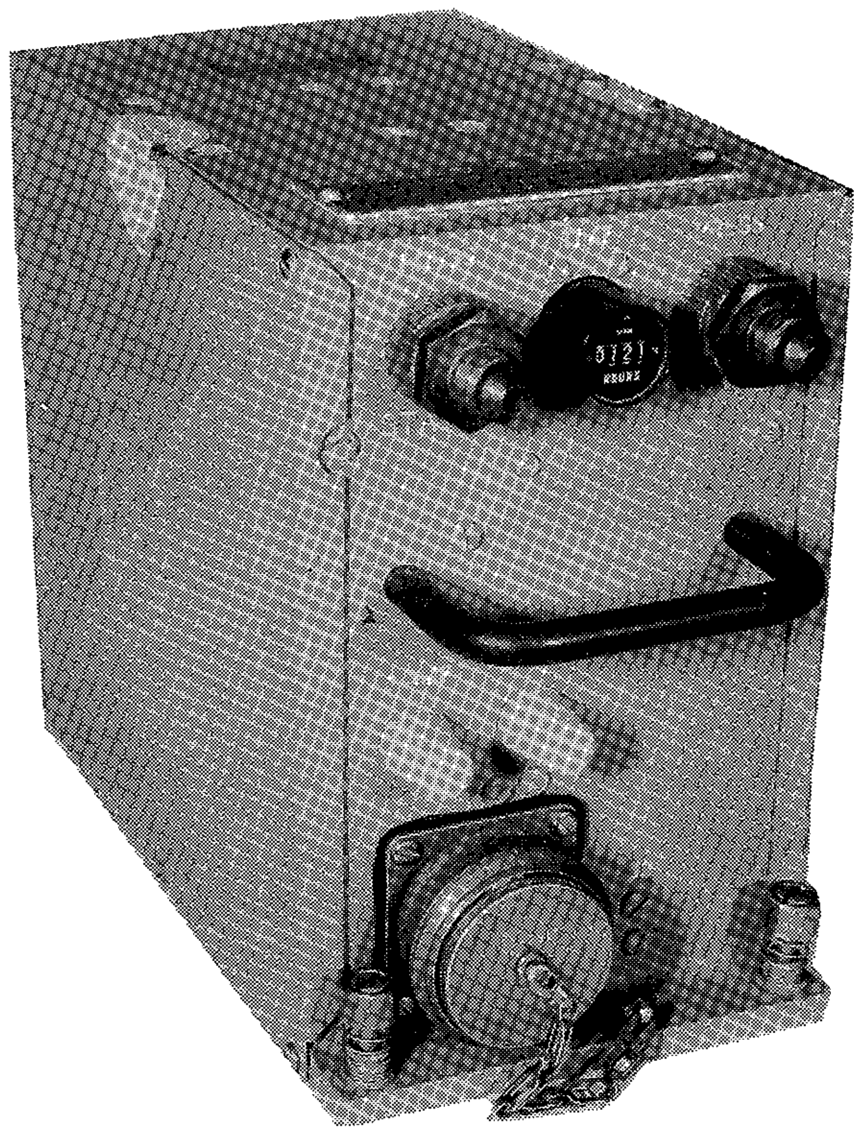


Figure 15B-1. Air Data Converter

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Pressure inputs to the ADC are obtained from two static pressure ports located on the underside of the aft fuselage section. Total pressure is obtained from a pitot tube on the right side of the ships nose. The total temperature probe is also located on the right side of the nose section.

Three TDD's, one for each output, transmits the digital data to the CCC in 20-bit words. Sixteen bits are reserved for data. One each bit is reserved for parity, validity, malfunction and test (malfunction and test bits are not used).

V. PCRS CONFIGURATION

The PCRS ADC will be functionally the same as the Baseline model. However, the requirement for 3-TDD's will be eliminated by a redesign of the ADC's output circuitry.

DIGITAL TO ANALOG CONVERTER

I. GENERAL DESCRIPTION

A. The digital to analog (D/A) converter converts digital data from the CCC to synchro format. This information is then used by the Bearing, Distance and Heading Indicators.

II. COMPONENTS AND LOCATION

Component	Quantity	Location
D/A Converter	1	Main Avionics Comp.

III. MAJOR COMPONENT DESCRIPTION

A. The D/A converter (see figure 15C-1) contains four plug-in modules:

1. Power supply
2. Logic
3. Control transformer
4. Torque receiver

The unit weighs approximately seven pounds and is contained within an outline 12 inches long, 4.25 inches high, and 5.75 inches wide.

IV. SYSTEM OPERATION

A. The D/A converter becomes operational when its circuit breaker is pushed in and ships power is applied. The four modules are described below.

1. Power supply - The power supply module converts 115V, 400 Hz single phase power to the DC voltages required for operation.

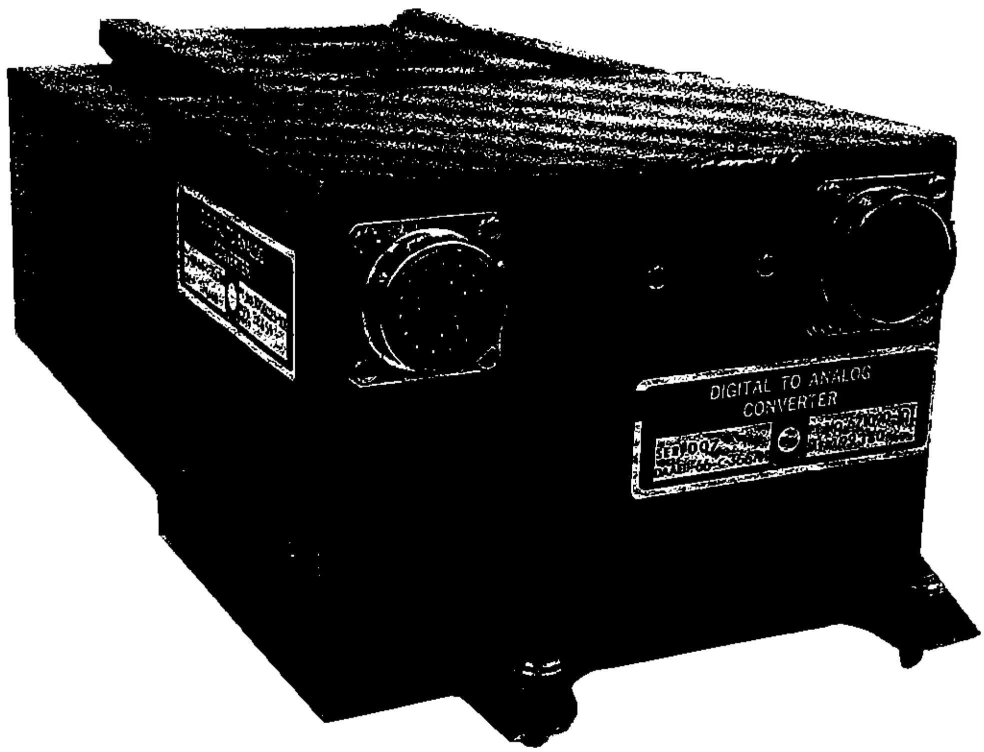


Figure 15C-1. Digital/Analog Converter

The Receiver, Digital Data (RDD), which receives the digital data from the CCC is housed within the power supply module. The power supply module is also the D/A main frame onto which the other three modules mount.

2. Logic module - The logic module provides the timing and logic necessary to:

   a. Decode the address of the received digital word.

   b. Gate and buffer the word to the proper conversion channel.

   c. Perform the tests necessary to determine operational capability.

3. Control Transformer. The control transformer module converts six 8-bit digital words into three 3-wire synchro outputs. The synchro outputs are used to drive the BDHI compass card and the two bearing needles.

4. Torque receiver - The torque receiver module converts two, 9-bit binary coded decimal (BCD) words into three 3-wire torque receiver outputs to drive the BDHI range indicators.

V. PCRC CONFIGURATION

A. The D/A converter has been deleted in production aircraft. The digital-to-analog conversion functions will be accomplished within the computer Input/Output (I/O) section.

SELF-CONTAINED NAVIGATION SYSTEM

I. GENERAL DESCRIPTION

The AH-56A self-contained navigation system utilizes a variety of sensors to measure accelerations, velocities, clearance altitude, heading and attitude. The system is capable of making these measurements accurately regardless of weather or visibility. Outputs of the system are processed to provide the pilot with continuous displays of present position in UTM coordinates, vehicle heading, attitude and altitude. Other information is made available for display when selected. Examples are range and bearing to a target or checkpoint and UTM coordinates of a target or checkpoint.

Primary navigation capability is provided by the Doppler Heading Attitude Reference System (DHARS). A number of backup and auxiliary equipments are also available to provide degraded-mode navigation capability in case of equipment failure due to malfunction or combat damage.

Of necessity, the navigation system interfaces with most of the other avionic equipment on the vehicle. Also, in discussing navigation system operation, reference must be made to a number of controls, indicators and displays.

II. COMPONENTS AND LOCATIONS

(See figures in Section 1.)

A. DHARS Components

1. Doppler Section

The Doppler section consists of a Receiver-Transmitter-Antenna (RTA) unit and a Signal Data Converter (SDC) unit. Both units are located in the aft avionics compartment.

2. HARS Section

The HARS (inertial) section consists of three units. A Vertical Reference Unit (VRU) is located in the ammo debris compartment. A Heading Reference Unit (HRU), mounted on the doppler RTA, and an Adapter-Compass Electronics (ACE) unit are both located in the aft avionics compartment.

B. Backup and Auxiliary Equipment

A Remote Compass Transmitter (flux valve) is located on the bottom of the rear fuselage. A Turn Rate Gyro (TRG) and a Rate Switching Gyro (RSG) are located in the main avionics compartment. A Standby Vertical Gyro (SVG) is located in the forward gyro compartment. A Navigation Junction box (Nav J Box) is located in the ammo debris compartment.

C. Controls, Indicators, and Displays

All controls, indicators and displays are located in the cockpit crew stations. See the system operation discussion which follows for details.

III. MAJOR COMPONENT DESCRIPTION

A. DHARS Components

1. Doppler Section

The Doppler section consists of the Receiver-Transmitter-Antenna (RTA) unit and the Signal Data Converter (SDC) unit (see figure 16A-1). The RTA transmits a 13.325-GHz signal in three depressed beams and receives the ground-reflected energy from which it extracts the Doppler information.

From this information the SDC develops velocity along the heading (V_H) of the vehicle, velocity across heading or drift (V_D), and velocity in the vertical direction (V_V). The SDC also compares the phase shift of the returned signal with respect to the transmitted signal to determine the absolute altitude of the AH-56A above the terrain.

The Signal Data Converter velocity outputs are in two forms, digital and analog. The digital information is routed to the Computer Central Complex (CCC) where it is used to compute navigation data and to produce weapons fire control voltages.

The Analog velocity outputs ($V_H V_D$ and V_V) are interconnected through the Navigation junction box to the pilot's Attitude Director Indicator when the hover (HOV) mode is selected. Doppler altitude is also an analog output and is directly interconnected through the Navigation Junction Box (figure 16A-2) to the two flight station Radar Altimeter Indicators (figure 16A-1).

2. Inertial Section

The inertial portion of the DHARS is aligned to true north and measures the vehicle acceleration in the true north-south (A_N) and in the true east-west (A_E) directions. It also measures acceleration along (A_H) and across (A_D) the vehicle's heading and the vertical acceleration (A_V) both positive and negative.

The Computer Central Complex (CCC) makes use of the true heading data and the vehicle accelerations and velocities data generated by the DHARS to compute grid heading and position information for display on the pilot's navigation instruments and displays. These include numeral readouts in kilometers of his position with respect to the UTM Grid map, the instantaneous velocity of his vehicle in heading, drift, and the vertical axis as well as the physical attitude (pitch and roll) of the vehicle with respect to the earth's surface.

The aircraft north and east position with respect to the UTM map appears on the Computer Control Panel No. 1 and No. 2 numerical readouts (figure 16A-3). The computed output is also connected directly to the Map Plotter in both flight stations (figure 16A-4). When a plotter is energized, it will trace a permanent record of the vehicle's ground track and present position on grid paper.

The inertial portion of the DHARS is packaged in three separate units, one of which - the Heading Reference Unit (figure 16A-5) is mounted on the Doppler Receiver-Transmitter-Antenna. The RTA is mounted on a servo driven gimballed platform and serves to keep both the RTA and the HRU level with respect to the earth's center of gravity. The HRU supplies true heading and the vehicle acceleration east and north data mentioned previously.

The Vertical Reference Unit (VRU) (figure 16A-5) is the primary source of vehicle pitch and roll data. This data is displayed on the "ball" of the pilot's Attitude Director Indicator (ADI) and is used to stabilize the Doppler antenna (RTA). The pitch signal is used to control the TOW missile after release. Pitch and roll data is also converted to digital form and used by the computer for deriving the stabilizing inputs to the Swiveling Gunner's Station.

The Vertical Reference Unit provides the Computer with vehicle heading, drift and vertical accelerations. The Computer uses the heading and drift acceleration data in its computation of gyro torquing terms. The acceleration data from all three axes is programmed into the computations for controlling the swiveling gunner's station.

The Adapter-Compass Electronics (ACE) unit shown in figure 16A-6 is the third inertial subsystem unit. It supplies DC and AC voltages to the HRU and VRU, receives and acts upon computer commands, and provides the interface electronics between the inertial system and the computer. This interface includes the conversion of attitude and heading analog data into digital form for the computer and reconversion of computer digital data to analog form for use by the inertial system.

B. Backup and Auxiliary Equipment (See figure 16A-7)

The remote compass transmitter (flux valve) supplies a magnetic heading signal to the DHARS for coarse alignment purposes and also serves as the major heading source for the system when both the VRU and the HRU units fail.

In case only the VRU fails, a Standby Gyro is automatically switched in to control the attitude ball on the Pilot's ADI. The Standby Gyro pitch and roll outputs are connected directly to the standby attitude indicators in both flight stations.

The Turn Rate Gyro provides a turn rate signal to the Attitude Director Indicator turn rate bar. Its output is also connected into the heading servo in the DHARS ACE unit in the "backup" failure mode to smooth the flux valve output.

All of the AH-56A navigation systems are interconnected to some extent through the Navigation Junction Box (figure 16A-2). The Pilot's and the Copilot/Gunner's MODE Select panels control relays in the navigation Junction Box which select the various outputs to be displayed on the flight instruments.

IV. SYSTEM OPERATION

A. Low altitude operation is necessary for survival in a hostile environment. An appreciable portion of each mission - and virtually all of the tactical part of the flight - will be flown nap-of-the-earth (i.e., as low as possible). This in turn places an upper limit on speed depending on the roughness of the terrain. Since the flight path will be such as to minimize exposure time, the pilot's view of the terrain will be limited, and consequently determination of present position will be largely dependent upon the accuracy of the navigation system. The condition is aggravated when the pilot flies over unfamiliar territory.

The velocity and position accuracy requirements, over the speed ranges expected in a typical mission, dictate the use of a doppler radar for velocity sensing. The long-term, low-speed accuracy of a doppler radar is far superior to that of an inertial platform. Doppler error is a function of speed, increasing as speed increases, while inertial error is time dependent.

B. For the following discussion, refer to figure 16A-8. Figures 16B-12 and 16B-13 show the BDHI and ADI.

The integrated Doppler-Inertial System combines the best characteristics of a Doppler velocity sensor, which has excellent long-term accuracy, with the inherent short-term accuracy characteristics of inertial systems.

In the Doppler section of the Doppler Heading Attitude Reference System, velocity is determined by measuring the amount of frequency "shift" due to relative motion between the helicopter and the surface of the earth. Since an aerial vehicle has freedom of movement in three dimensions (fore/aft, left/right, up/down), each of the three axes is sensed.

The Doppler radar section provides heading velocity, drift velocity, vertical velocity, and radar altitude outputs to the Computer Central Complex and to the Navigation Junction Box.

In addition to the Doppler velocity and altitude outputs to the Computer Central Complex, the velocity and altitude outputs can be displayed on the various instruments. Radar altitude is continuously displayed on the two Radar Altitude Indicators (RAI). Vertical velocity is displayed on the left index of the pilot's Attitude Director Indicator and heading axis velocity and drift axis velocity are displayed on the flight director bars of the pilot's Attitude Director Indicator when HOV STEER is selected on the pilot's Mode Select Panel. A digital display of the vector sum of heading axis velocity and drift axis velocity is available on the Computer Control Panel when GS NAVIGATION is selected. Doppler section operation is enabled by the Doppler switch on the pilot's Mode Select Panel.

Functional operation of the Doppler section is indicated by the NAV MODE annunciator light on the Computer Control Panel. Proper operation is indicated by a DOP annunciator. For short-term (less than 30 seconds) degradation or failure of the Doppler section, "memorized" information is used by the Computer Central Complex and a MEM annunciation appears. For long-term failure of the Doppler section, a secondary velocity source (true air speed calculated from Air Data Converter outputs) is used by the Computer Central Complex, and a TAS light annunciation appears on the Computer Control Panel.

Built-In Test Equipment (BITE) is incorporated in the Doppler section of the Doppler Heading Attitude Reference System. A continuous "end-to-end test and calibrate" cycles through the Doppler section. Any detected failure will set the BITE indicator on the Receiver-Transmitter-Antenna or on the Signal Data Converter, and will cause a fault annunciation on the Fault Location Aural Warning System (FLAWS) Equipment Status Panel in the copilot's station.

The Heading Attitude Reference System (HARS) section of the Doppler Heading Attitude Reference System (DHARS) is an inertial reference system. An inertial system measures the external force (acceleration) which tends to cause a change in the motion of a stabilized system.

All present inertial reference systems operate by measuring the external acceleration forces with respect to a "stable element." The "stable element" is physically stabilized in space coordinates by the use of gyroscopes. The gyroscopes maintain the "inertial platform" in proper alignment with true north and horizontal to the surface of the earth. Accelerometers mounted on the major axes of the aligned inertial platform measure the external acceleration forces which act upon the helicopter. Integrators within the Computer Central Complex use accelerations from the inertial section to derive velocity information, distance traveled information, and distance to checkpoint information. The Computer Central Complex also uses the inertial accelerations in various fire control calculations.

Since the "inertial platform" is space stabilized (aligned to true north and horizontal to the surface of the earth) the heading of the helicopter (with respect to true north) and the pitch and roll attitude of the helicopter (with respect to the horizontal) can be measured by displacement of the helicopter structure from the inertial reference. Heading information supplied by the inertial section is used in various navigation functions. Pitch and roll attitude information is used for coordinate system conversions, navigation calculations, and fire control functions. Heading, pitch, and roll are continuously displayed on the instruments.

The critical requirement for inertial reference systems is alignment of the inertial platform to true north and horizontal. Normally, alignment of the inertial platform is accomplished in two states - "leveling" of the platform ensures proper orientation of the inertial reference to the horizontal, and "gyrocompassing" of the platform ensures proper alignment of the inertial reference to the north reference. Since proper alignment of the inertial platform is the critical requirement, several corrections to the platform are made by the Computer Central Complex. Corrections made by the Computer Central Complex include "gyro bias" which compensates for any drift of the gyroscope due to manufacturing imperfections, compensation for "earth rotation rate," a "transport rate" compensation to cancel helicopter movement on the earth, a "coriolis" correction, and a correction to compensate the inertial system for the pear-shape of the earth.

Computer controlled alignment of the inertial section is initiated on the Computer Control Panel in the pilot's right console, and may be accomplished by normal "gyrocompass" technique, by using an optically determined heading reference, or by using a heading reference stored within the Computer Central Complex.

The Heading Attitude Reference System (HARS) section of the Doppler Heading Attitude Reference System (DHARS) provides outputs of heading, pitch, roll and five axis (north, east, heading, drift, and vertical) accelerations.

In addition to the heading, pitch, roll, and acceleration outputs provided to the Computer Central Complex, the heading and attitude outputs of the Heading Attitude Reference system are displayed on the instruments. Heading information is continuously displayed on the pilot's and copilot's Bearing Distance Heading Indicators. Pitch and Roll information is displayed on the pilot's Attitude Director Indicator ball.

Proper alignment of the Heading Attitude Reference System is critical to navigation, attitude, and fire control accuracy. The Heading Attitude Reference System is enabled only when the Computer Central Complex is operational. The GYRO COMP switch on the pilot's Mode Select Panel must be selected to the GRD position for proper alignment of the Heading Attitude Reference System. Computer controlled alignment is initiated using the ALIGN switches on the Computer Control Panel.

Normal "gyrocompass" alignment of the Heading Attitude Reference System is initiated when the INITIAL ALIGN switch on the Computer Panel annunciates GYRO.

Alignment of the Heading Attitude Reference System can also be accomplished using a heading reference "stored" in the memory section of the Computer Central Complex. Depression of the STOR INITIAL ALIGN, and either VEH (vehicle) or OPT (swiveling gunner's station optical sightline) on the REF HD switch, allows elimination of the "gyrocompass" phase of normal alignment and reduces alignment time.

Functional operation of the Heading Attitude Reference System is indicated by a SLAVE annunciation on the NAV MODE switch of the

Computer Control Panel. In the event of Heading Reference Unit failure, a "back-up" mode of operation is initiated. In the "back-up" mode, the Heading Attitude Reference System uses magnetic heading information from the Remote Compass Transmitter (flux valve) as the heading source. In "back-up," the NAV MODE annunciator is switched to COMP and magnetic heading information, smoothed by "turn rate" from the Turn Rate Gyro, is supplied to the Bearing Distance Heading Indicator compass card.

In the event of Vertical Reference Unit failure, a "degraded gyro" mode of operation is initiated. In the "degraded gyro" mode, the Heading Attitude Reference System uses secondary pitch and roll from the Standby Vertical Gyro (SVG) as the attitude information source. In "degraded gyro," the NAV MODE annunciator is switched to COMP, and standby pitch and roll is supplied to the pilot's Attitude Director Indicator ball.

In the event of Adapter Compass Electronics failure, the entire Heading Attitude Reference System is shut down. Standby pitch and roll information from the Standby Vertical Gyro drives the Standby Attitude Indicators (SAI) in each crew station. Heading information is supplied by a Standby "whiskey" Compass in each station. Radar altitude and HOV velocities from the Doppler section remain functional.

Built-In Test Equipment (BITE) is incorporated within the Heading Attitude Reference System section. Individual BITE indicators monitor the condition of the Heading Reference Unit, the Vertical Reference Unit, and the Adapter Compass Electronics.

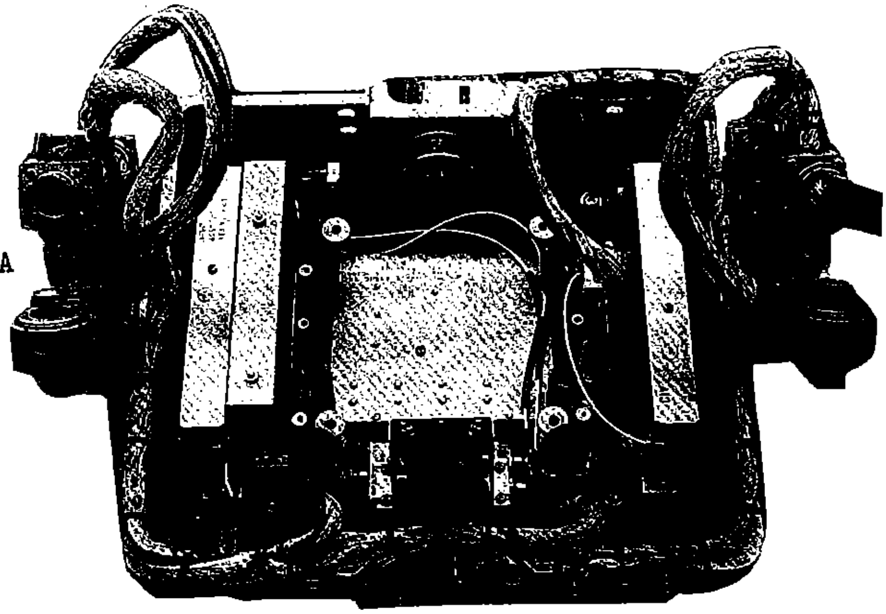
V. PCRS CONFIGURATION

(See figures 16A-9 and 16A-10.)

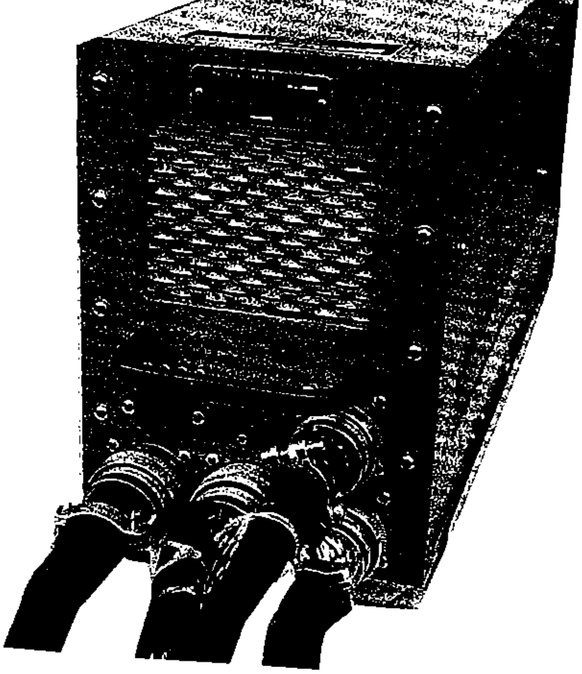
The simplified DHARS derives from, and is functionally similar to, the baseline system. Numerous changes have been incorporated to reduce cost and enhance reliability without compromising performance. Most of the changes are of an evolutionary nature, taking advantage of components and techniques not available when the original system was designed. Some are simplifications resulting from the elimination of functions no longer deemed necessary to the system, or by electing to perform the functions elsewhere in a more economical manner. The change providing the largest cost saving, is elimination of the VRU and its associated circuitry. Attitude information will be provided by a standby quality vertical gyro (e.g., AN/ASN-76 AHRS). The pitch and roll angles supplied by this gyro are accurate to approximately $\pm 1^{\circ}$. This is adequate for stabilization of the RTA and for driving the cockpit instruments, but will not suit the requirements of the fire control program. For this purpose, verticality no worse than $\pm 0.15^{\circ}$, 1σ, is required. Methods exist for correcting the vertical error in the computer by using the differences between doppler and inertially derived velocities.

Repackaging and simplification of design has led to adoption of a three-LRU system (RTA, HRU and SDC) all of which will be mounted in the aft avionics compartment. The AHRS platform will be mounted in the location of the current VRU and its electronics box will be installed in the aft electronics compartment. The Standby Vertical gyro is retained for backup. Backup heading is provided by the magnetic heading output of the AHRS. Incorporation of this system will lead to a cost savings of approximately 30 percent over the baseline system.

RECEIVER-TRANSMITTER-ANTENNA

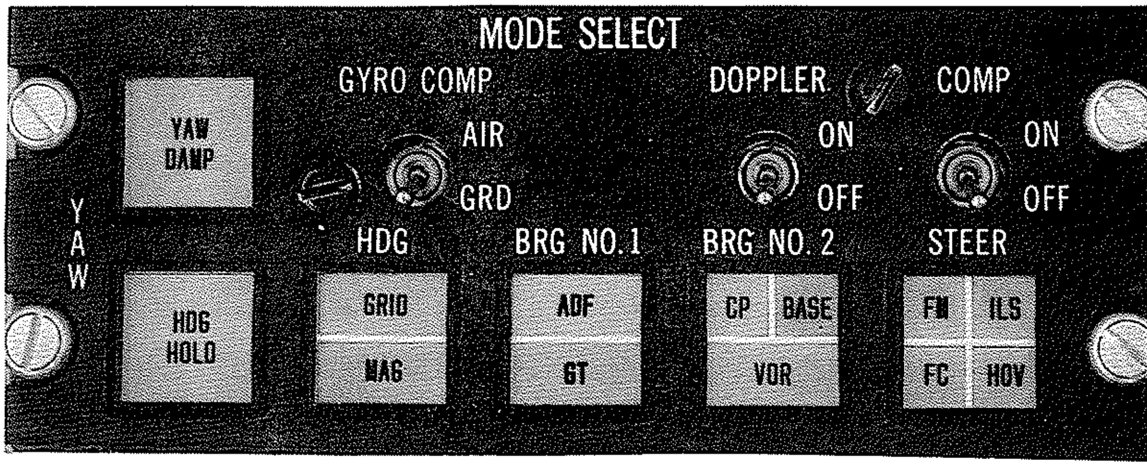


SIGNAL DATA CONVERTER

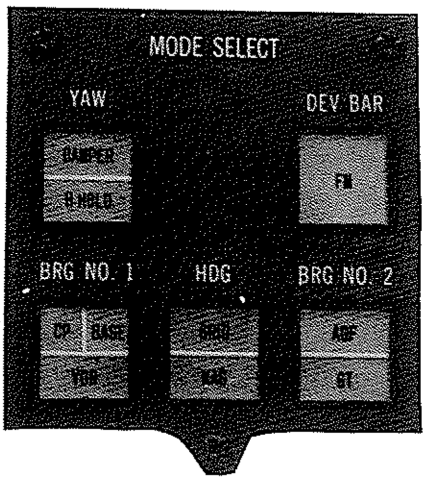


RADAR ALTITUDE INDICATOR

Figure 16A-1



PILOTS MODE SELECT PANEL



COPILOTS MODE SELECT PANEL

NAVIGATION JUNCTION BOX

Figure 16A-2

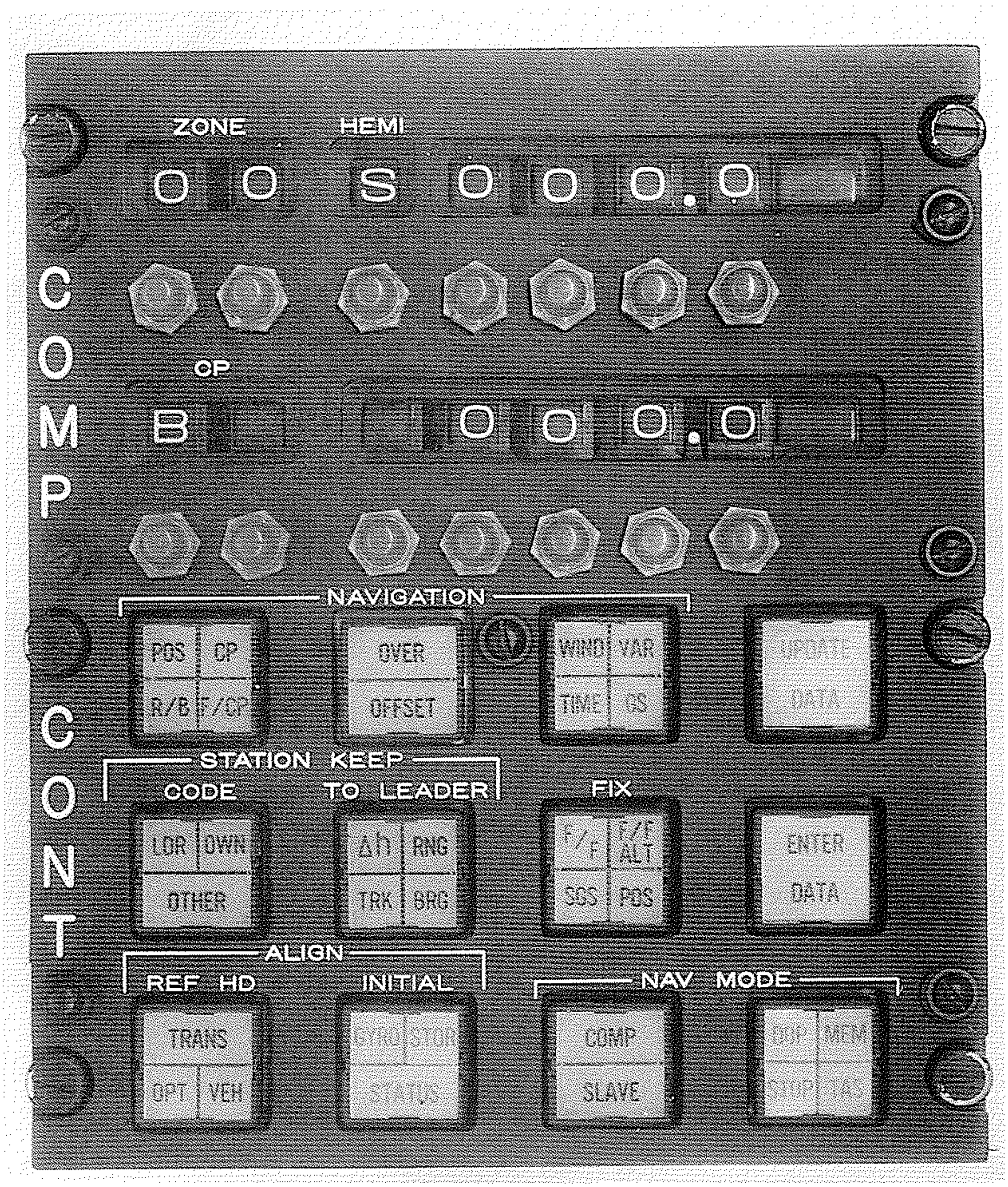


Figure 16A-3. Computer Control Panel

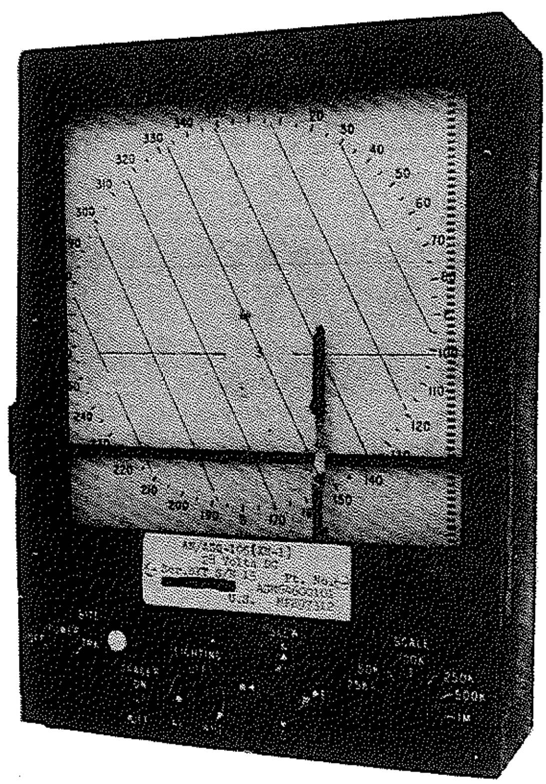
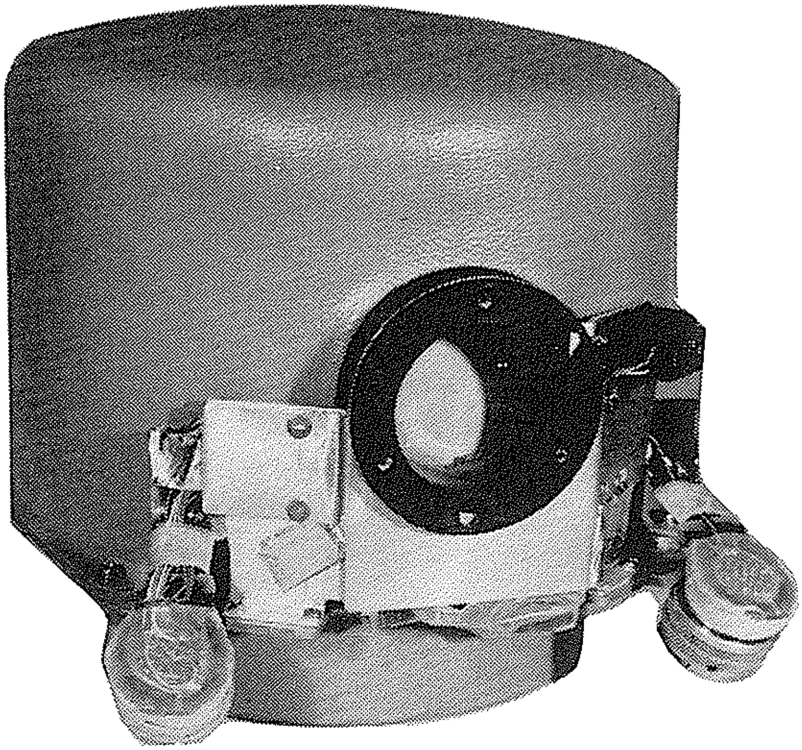
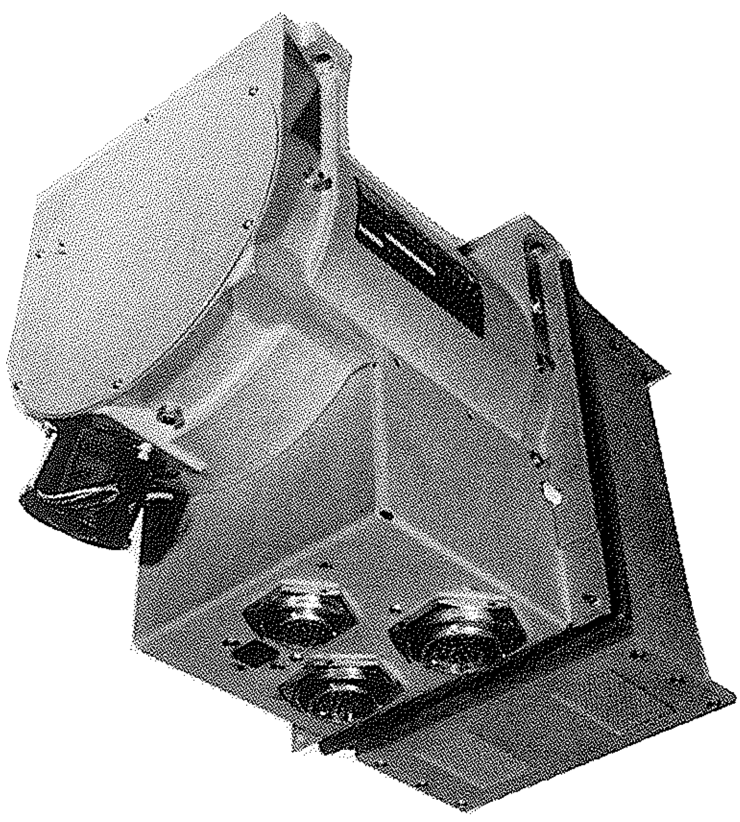


Figure 16A-4. Map Plotter



HEADING REFERENCE UNIT



VERTICAL REFERENCE UNIT

Figure 16A-5

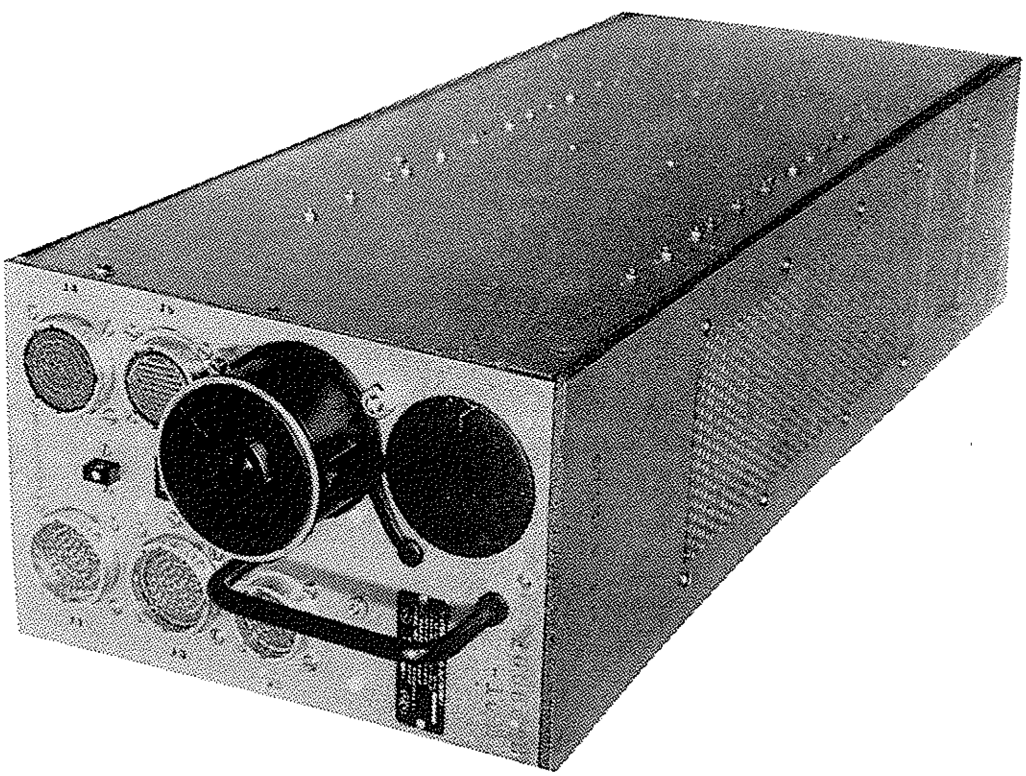
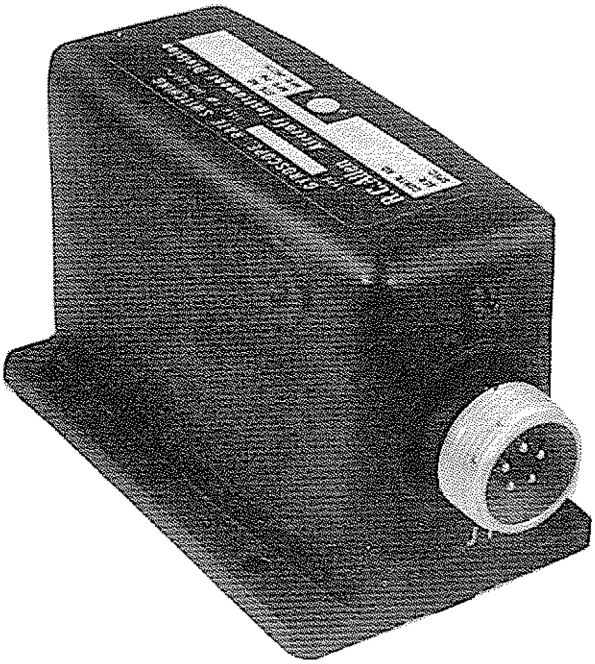


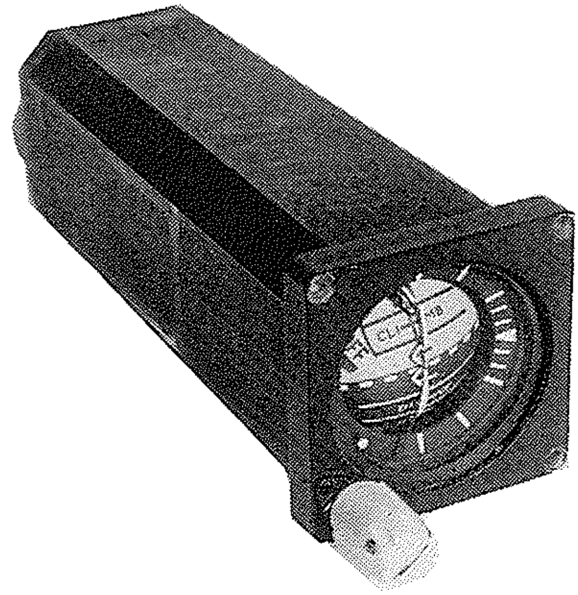
Figure 16A-6. Adapter, Compass Electronics



RATE SWITCHING GYRO

TURN (YAW) RATE GYRO

STANDBY VERTICAL GYRO



STANDBY ATTITUDE INDICATOR

Figure 16A-7. Standby Equipment

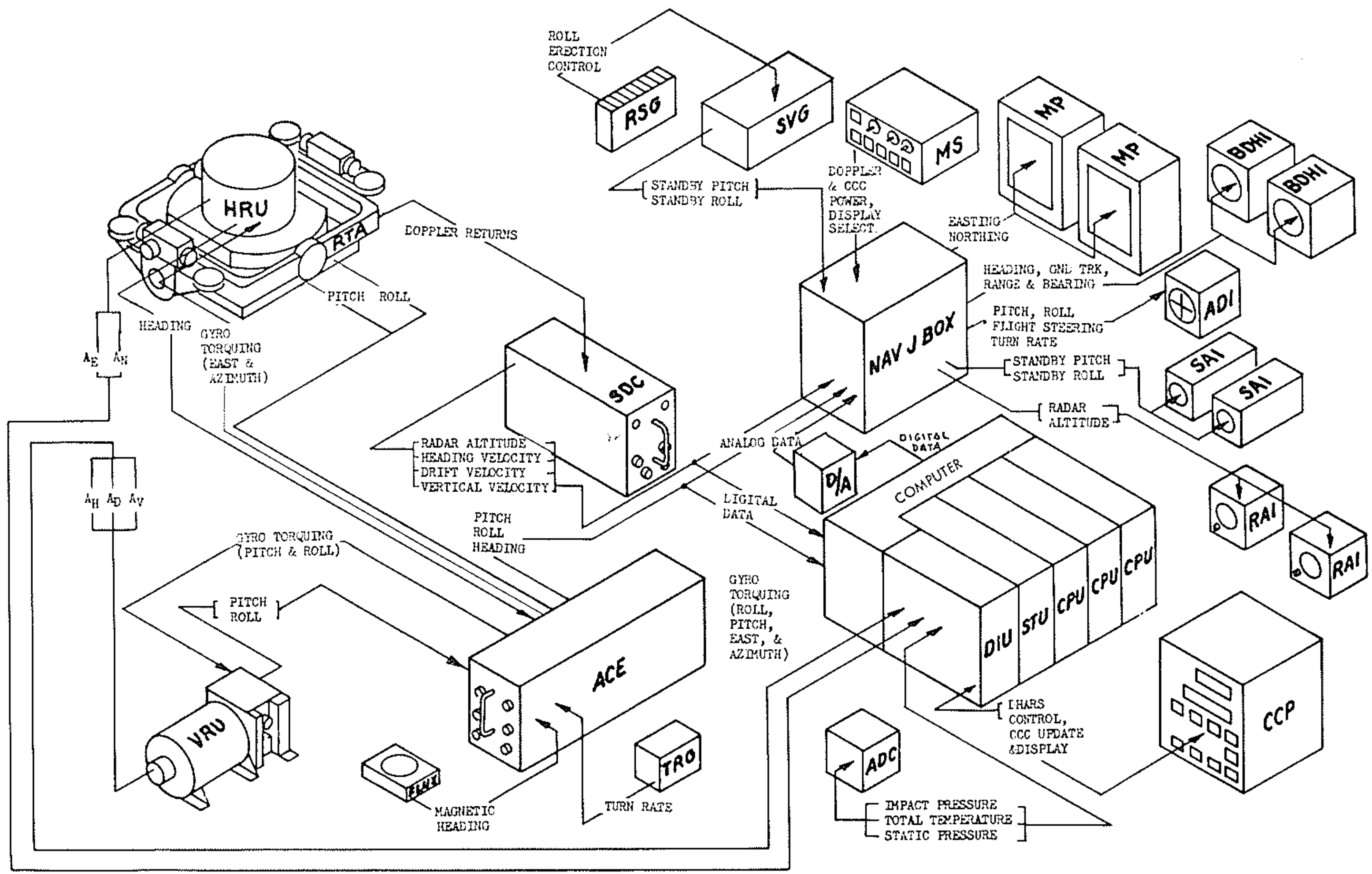


Figure 16A-8. Navigation System Block Diagram

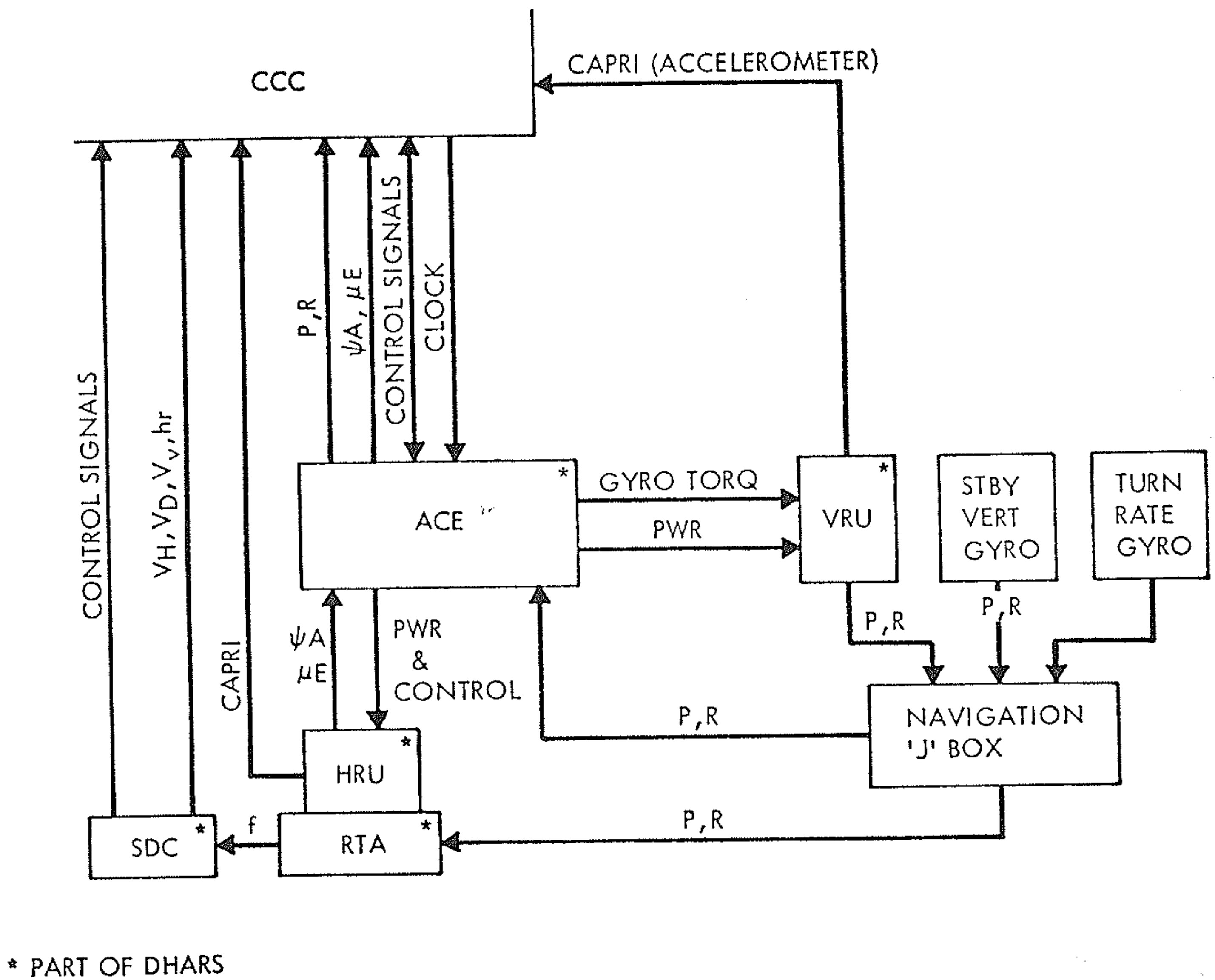


Figure 16A-9. Baseline DHARS Block Diagram

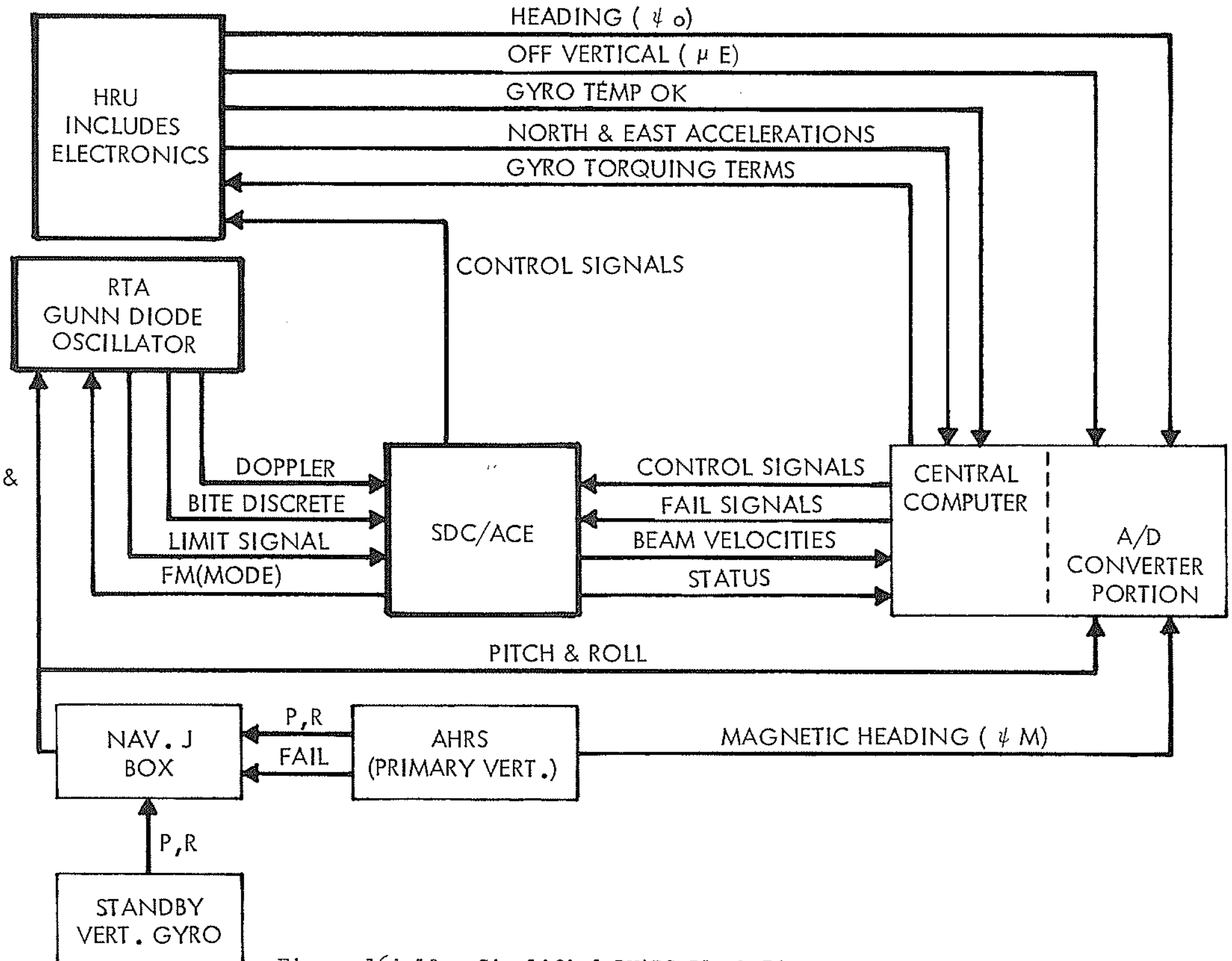


Figure 16A-10. Simplified DHARS Block Diagram

RADIO NAVIGATION SYSTEM

I. GENERAL DESCRIPTION

The Radio Navigation System installed on the AH-56A provides the capability to determine helicopter position (range and bearing to or from a selected ground beacon transmitter) and provides an Instrument Landing System (ILS) for the helicopter.

An ADF Receiver provides radio homing to a selected transmitter, or navigation position "fix" information by selection of multiple stations.

Each of the displays for the radio navigation equipments is available on the pilot's and copilot's Bearing Distance Heading Indicators (BDHI) and/or on the pilot's Attitude Director Indicator (ADI) as selected on the respective Mode Select Panels. (See Figures 16B-12 & 16B-13)

Each of the control panels for the radio navigation equipments is installed in the pilot's station, allowing only the pilot to make channel selection and operating mode selection. The copilot can select the radio navigation display for his Bearing Distance Heading Indicator.

Built In Test Equipment (BITE) is not provided in any of the radio navigation system equipment, however, a "self-test" capability is provided on several of the equipments.

II. COMPONENTS AND LOCATIONS

Component	Quantity	Location
A. AN/ARN-82 VOR/LOC Receiver	1	Aft avionics compartment
1. Nav-Comm Control Panel	1	Pilots left console
2. VOR/LOC Antenna	2	Either side of ventral fin
B. AVQ-70 DME Interrogator	1	Aft avionics compartment
1. DME Control Panel	1	Pilots right console

Component	Quantity	Location
2. DME Antenna	1	Underside of aft fuselage section
C. AN/ARN-58 GS/MB Receiver	1	Aft avionics compartment
1. Marker Beacon Control Panel	2	Right side of each instrument panel
2. GS Antenna	1	On nose of aircraft
3. MB Antenna	1	Underside of forward fuselage section
D. AN/ARN-89 ADF Receiver	1	Main avionics compartment
1. ADF Control Panel	1	Pilot's right console
2. ADF Sense Antenna	1	Belly turret fairing
3. ADF Loop Antenna	1	Lower mid-fuselage section

III. MAJOR COMPONENTS DESCRIPTION

A. AN/ARN-82, VOR/LOC Receiver (Figure 16B-2)

The very high frequency omni-range localizer (VOR/LOC) receiver is standard AN/ARN-82 military equipment. The VOR section of the VOR/LOC receiver provides heading "to or from" a selected ground VOR beacon, deviation "left or right" of a selected course, and a VOR station identification tone. The LOC section of the AN/ARN-82 provides localizer "left or right" steering commands to enable helicopter alignment with the runway for instrument landing system approach.

A VOR ground beacon transmitter, operating between 108 megahertz and 118 megahertz, transmits two separate antenna patterns. One of the signals, a reference phase which is frequency modulated, is non-directional and has a constant phase throughout 360° of azimuth. A second signal, the variable phase which is amplitude modulated, is directional in azimuth and is rotated about the reference phase such that the two signals are aligned at magnetic north and become increasingly out of alignment in a clockwise direction from magnetic north. Helicopter heading to or from the VOR beacon is determined by measurement of the misalignment of the two VOR transmitted signals.

VOR heading information is displayed on the Bearing Distance Heading Indicator, when VOR is selected on the Mode Select Panel. The pilot can select a desired course using the CRS SET knob on the Bearing Distance Heading Indicator; offset from the desired course will be displayed on the offset section of the course pointer. A "to/from" flag indicates whether a heading "to" the station or "from" the station is indicated by the #2 bearing pointer. When VOR mode has been selected on the Mode Select Panel, the compass card on the Bearing Distance Heading Indicator displays a computed magnetic heading.

A LOC ground beacon transmitter, operating at "odd tenth" megahertz channels between 108 megahertz and 112 megahertz, transmits two signals which indicate helicopter alignment with the runway. One of the signals, which is received when the helicopter is left of the proper approach alignment, yields a "steer right" flight director command.

LOC "left or right" steering commands are displayed on the flight director needle of the pilot's Attitude Director Indicator when ILS STEER is selected on the pilot's Mode Select Panel. Glide slope information from the AN/ARN-58, GS/MB Receiver supplements the localizer steering commands in the instrument landing system mode.

Major components of the AN/ARN-82, VOR/LOC system installed in the helicopter are indicated in the following list:

Built In Test Equipment (BITE) is not installed in the AN/ARN-82, VOR/LOC system, however, selection of the TEST position on the NAV-COMM Control Panel provides a limited "self-test". The TEST position breaks squelch in the VOR/LOC receiver, causes rotation of the #2 bearing pointer to zero degrees, and displays a "from" flag with the CRS SET on the Bearing Distance Heading Indicator set to zero degrees.

Warning flags are displayed on cockpit instruments in the event that the received signal strength is not sufficient to provide reliable data. The VOR/LOC receiver system can be used as a communications receiver for frequencies above 118 megahertz and below 127 megahertz.

1. NAV-COMM Control Panel (Figure 16B-1)

   Provides operating power (PWR) to the VOR/LOC and GS/MB receivers, controls channel selection for VOR, LOC and GS operation, and initiates AN/ARN-82 "self-test".

2. Two VOR/LOC receiving antennae, (Figure 16B-3)

B. AVQ-70, DME Interrogator (Figure 16B-5)

The distance measuring equipment (DME) interrogator, provides radar range to any one of 126 selected VOR/DME or VORTAC ground stations. The distance measuring equipment operates in the "L" band portion of the frequency spectrum, however, the distance measuring equipment is tuned in the same numerical reference as is the VOR/LOC system.

The AVQ-70 transmits an interrogation "pulse pair" to the ground VOR/DME or VORTAC beacon. The ground beacon, after a fixed time delay, transmits a range reply to the airborne DME system. The AVQ-70 measures the elapsed time between transmission of the range interrogation and reception of the range reply and translates elapsed time to "distance" by radar ranging techniques. DME "distance" is displayed on the miles counter of the Bearing Distance Heading Indicators when VOR is selected on the associated Mode Select Panel.

Maximum range of the DME system is 75 nautical miles in the NORM mode and approximately 197 miles in the SRCH O'RIDE mode. Depression of the SELF TEST on the DME Control Panel causes a range

Figure 16B-1
Nav-Comm Control Panel

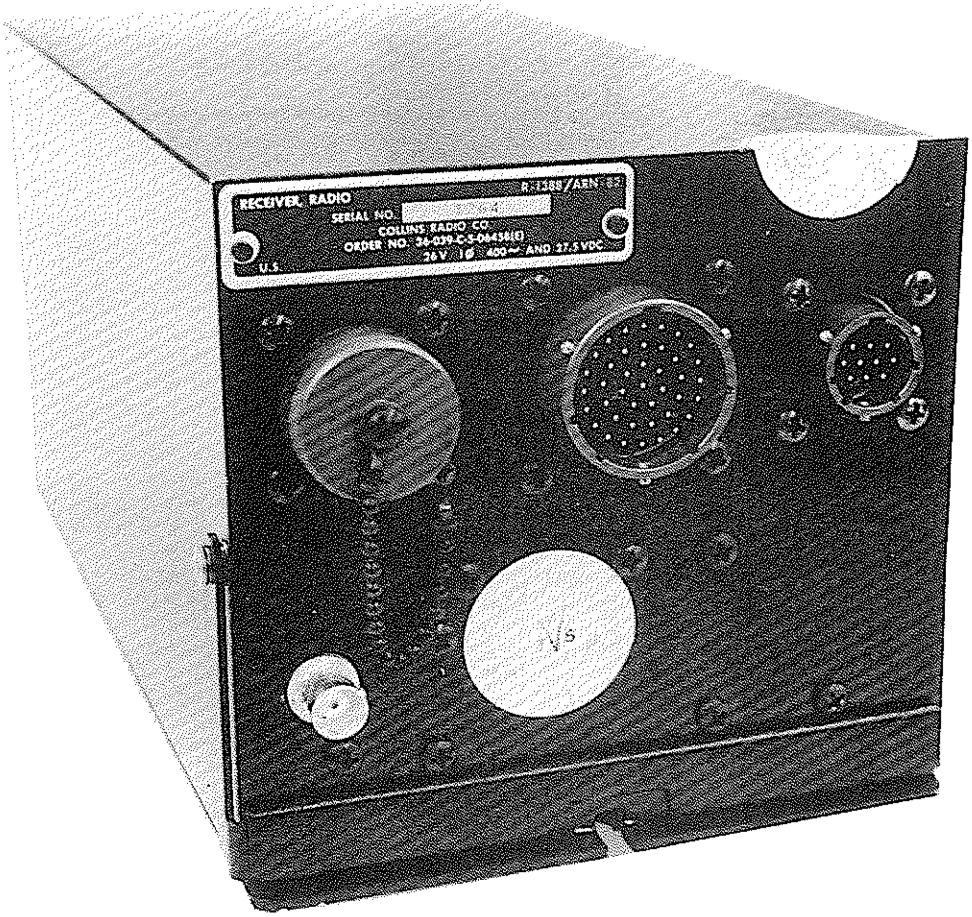


Figure 16B-2
AN/ARN-82 VOR/LOC Receiver

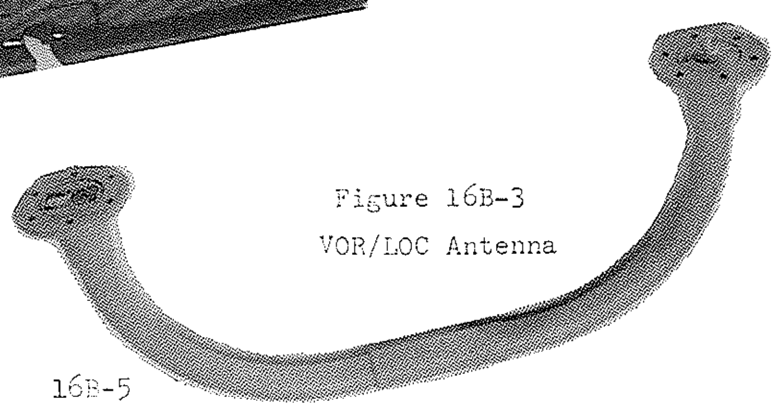


Figure 16B-3
VOR/LOC Antenna

indication of 94.4 nautical miles in the MILES counter if the DME is operating properly. BITE is not included.

Major components of the AVQ-70, DME system are indicated in the following list:

1. DME Control Panel (Figure 16B-4) provides DME channel selection in VOR numerical reference, mode selection, power control, and self-test initiate.

2. DME Antenna

C. AN/ARN-58, GS/MB Receiver (Figure 16B-6)

The glide slope/marker beacon receiver provides glide slope "up or down" steering commands and marker beacon "distance from end of runway" information for instrument landing system approach. Operating power for the AN/ARN-58 is provided through the NAV-COMM Control Panel (Figure 16B-1) in the pilot's left console. Channel selection for the GS section of the GS/MB receiver is controlled by the LOC channel selection on the NAV-COMM Control Panel.

The solid-state glide slope section of the AN/ARN-58 receiver operates on any one of 20 channels between 329.3 megahertz and 335.0 megahertz. The ground GS transmitter beacon generates two separate signals which indicate helicopter alignment with the instrument landing system glide path. One of the signals, which is received when the helicopter is above the proper glide path alignment, yields a "steer down" flight director command. The second signal, which is received when the helicopter is below the proper glide path alignment, yields a "steer up" flight director command.

Glide slope "up or down" steering commands are displayed on the flight director needle of the pilot's Attitude Director Indicator when ILS STEER is selected on the pilot's Mode Select Panel. Localizer information from the AN/ARN-82, VOR/LOC Receiver supplements the glide slope steering commands in the instrument landing system mode.

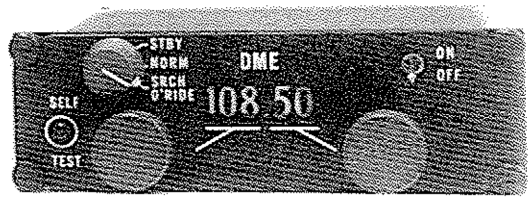


Figure 16B-4
DME Control Panel

Figure 16B-5
AVQ-70 DME Interrogator

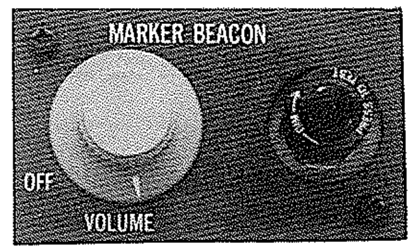


Figure 16B-7

Marker Beacon Control Panel

Figure 16B-6

AN/ARN-58 GS/MB Receiver

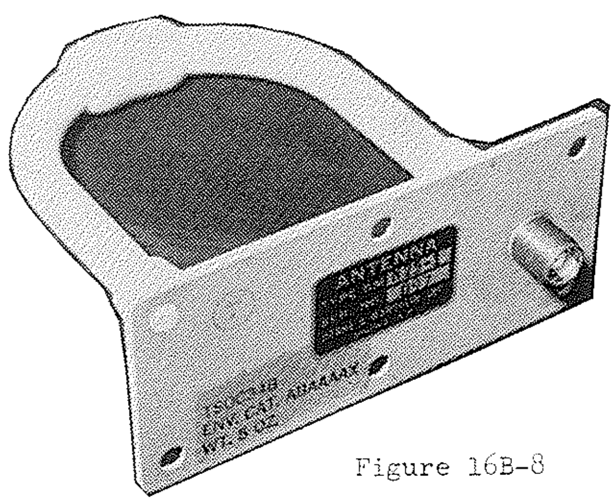


Figure 16B-8

Glide Slope Antenna

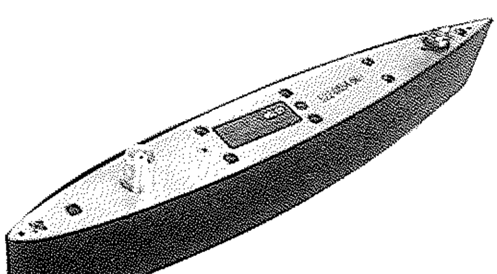


Figure 16B-9

Marker Beacon Antenna

The marker beacon section of the AN/ARN-58 receiver operates on a single fixed channel at 75 megahertz, and is a superhetrodyne receiver which provides lamp flasher and tone indications when over a ground beacon. Marker beacon lamp flasher indications are displayed on each of the two Marker Beacon Control Panels, and tones may be monitored through the ICS System.

Built In Test Equipment (BITE) is not included in the AN/ARN-58 System, nor does the system have "self-test" features.

Major components of the GS/MB Receiver system installed in the helicopter are indicated in the following list:

- Two Marker Beacon Control Panels, (Figure 16B-7) one in the right side of each instrument panel, are installed.
- GS Antenna (Figure 16B-8) is located at the nose of the helicopter.
- MB Antenna (Figure 16B-9) is located at the underside of the forward fuselage section.

D. AN/ARN-89, ADF Receiver

The automatic direction finder (Figure 16B-10) provides relative bearing from the helicopter to a selected ground radio transmitter, and is used for homing and/or navigation position "fix" purposes.

Direction finders make use of the sensitivity characteristics of two types of antenna systems to provide bearing measurement. The "sense" antenna has nondirectional receiving pattern characteristics, that is, the sense antenna receives energy equally well in all directions. The "loop" antenna is bidirectional, that is, the loop antenna receives radiant energy well in two directions and poorly in any other direction. When the two antenna patterns are combined, the resultant antenna pattern is a unidirectional "cardioid" pattern which receives radiant energy poorly (null) in one direction and fairly well in other directions. The energy received by the automatic direction finder is a function of antenna position and

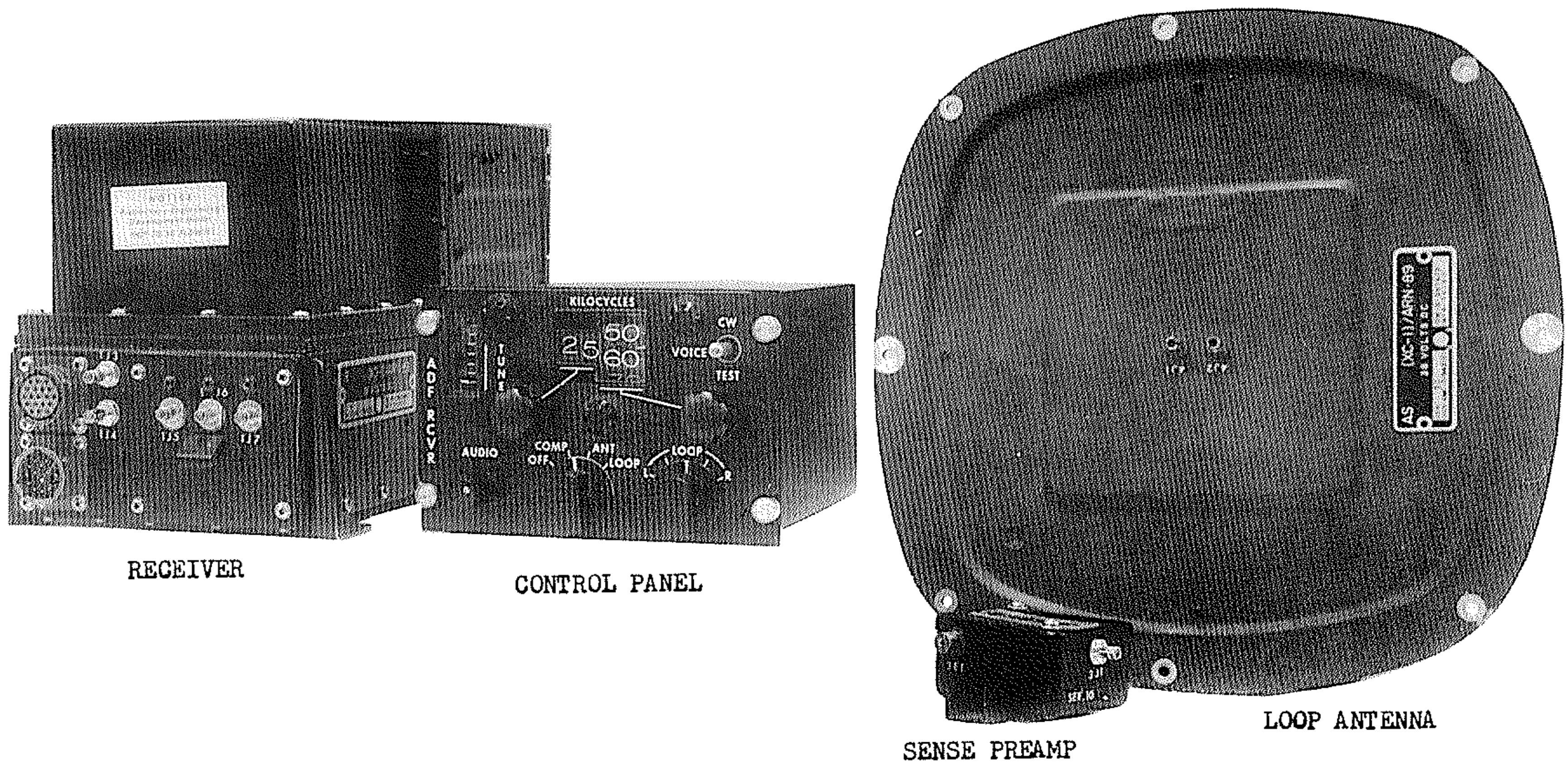


Figure 16B-10
AN/ARN-89 ADF System Components

transmitter direction. When the antenna system is rotated to alignment with the direction of the transmitter, a minimum radiant energy level is received. Thus, by positioning the antenna system to receive a minimum energy level, relative bearing of the helicopter to the radio transmitter can be measured within approximately 3° of azimuth.

Major components of the automatic direction finder equipment installed in the AH-56A are indicated in the following list:

1. ADF RCVR Control Panel

   Provides mode selection, channel selection between 100 kilohertz and 3,000 kilohertz, manual antenna control, and contains four plug-in modules for the receiver system.

2. ADF Receiver (Remote Unit)

   Contains four plug-in modules which process the received radiant energy to provide relative bearing.

3. ADF Sense Antenna

   Is the nondirectional antenna system which feeds an ADF Sense Antenna Preamplifier installed in the belly turret fairing through a lightning arrester.

4. ADF Loop Antenna

   Is the bidirectional antenna system which combines with the sense antenna to provide the direction finder capability.

IV. SYSTEM OPERATION

A. Integrated Radio Navigation Approach and Instrument Landing System

A complete radio navigation approach capability (Figure 16B-14) is provided by the combination of the AN/ARN-82 (VOR/LOC Receiver), the AVQ-70 (DME Interrogator), and the AN/ARN-58 (GS/MB Receiver).

Assume that the helicopter is several miles from a VOR station, and the pilot has been told to approach the station on a specified magnetic heading. The pilot selects the frequency of the VOR station on the NAV-COMM Panel and on the DME Panel, and selects VOR on the Mode Select Panel. The compass card on the Bearing Distance Heading Indicator displays magnetic heading of the helicopter, the #2 bearing pointer indicates the relative heading to or from the VOR station, and the MILES counter indicates distance to the VOR station. The pilot then selects the course which he has been told to fly into the VOR station using the CRS SET knob on the Bearing Distance Heading Indicator, and the offset section of the course pointer indicates any deviation from the desired course. By steering the helicopter to achieve alignment of the offset portion of the course pointer with the fiducial marker on the Bearing Distance Heading Indicator, the pilot is guaranteed that the helicopter is flying the heading which he has been instructed to fly.

When the helicopter is within range of the Instrument Landing System, normally 10 to 15 nautical miles from the runway, the pilot can tune the NAV-COMM Control Panel to a LOC channel and depress ILS STEER on the Mode Select Panel. Glide slope steering commands and localizer steering commands are displayed on the flight director bars of the pilot's Attitude Director Indicator. By steering the helicopter to maintain the flight director bars in the center of the Attitude Director Indicator, the pilot is ensured that the approach is correct. Marker beacon lamp flasher and tone signals notify the pilot of the "distance to touchdown" at the outer marker and middle marker checkpoints.

Three modes of operation are available in the AN/ARN-89, automatic direction finder system, and are selected on the ADF RCVR Control Panel (Figure 16B-11) in the pilot's right console.

- COMP: in the compass mode, both the sense antenna and the loop antenna are used to provide bearing from the helicopter to the selected radio transmitter. ADF bearing is displayed on the #1 bearing pointer of the Bearing Distance Heading Indicator when ADF is selected on the mode select panel.

Figure 16B-11. ADF Receiver Control Panel

- ANT: in the antenna mode, only the sense antenna is used to provide the functions of a voice or continuous wave (CW) radio receiver.

- LOOP: in the loop mode, only the loop antenna is used for manual direction finding. Manual tuning of the LOOP control to achieve maximum upward deflection of the TUNE needle provides relative bearing. Manual direction finder bearing is also displayed on the #1 pointer of the Bearing Indicator when ADF is selected on the Mode Select Panel.

Built In Test Equipment (BITE) is not provided in the AN/ARN-89, ADF Receiver System. ADF "self-test" is accomplished in COMP mode after achieving relative bearing to a selected station. Depression of the TEST switch on the ADF RCVR Control Panel will cause 180-degree rotation of the #1 bearing pointer.

V. PCRS CONFIGURATION

A. The AVQ-70, DME has been deleted. The pilot is provided the same functions by setting the computer control panel check points (CP) to VOR station coordinates. Bearing and distance are read on the pilot's and copilot's BDHI when CP is selected on either Mode Selector. Deletion of the DME provided $7,714,000 investment cost savings.

B. The AN/ARN-58 GS/MB has been deleted, however, complete provisions have been retained. This system is not required for the tractical environment and infrequent need for the civil environment does not warrant its permanent installation.

Figure 16B-12. Bearing, Distance, Heading Indicator

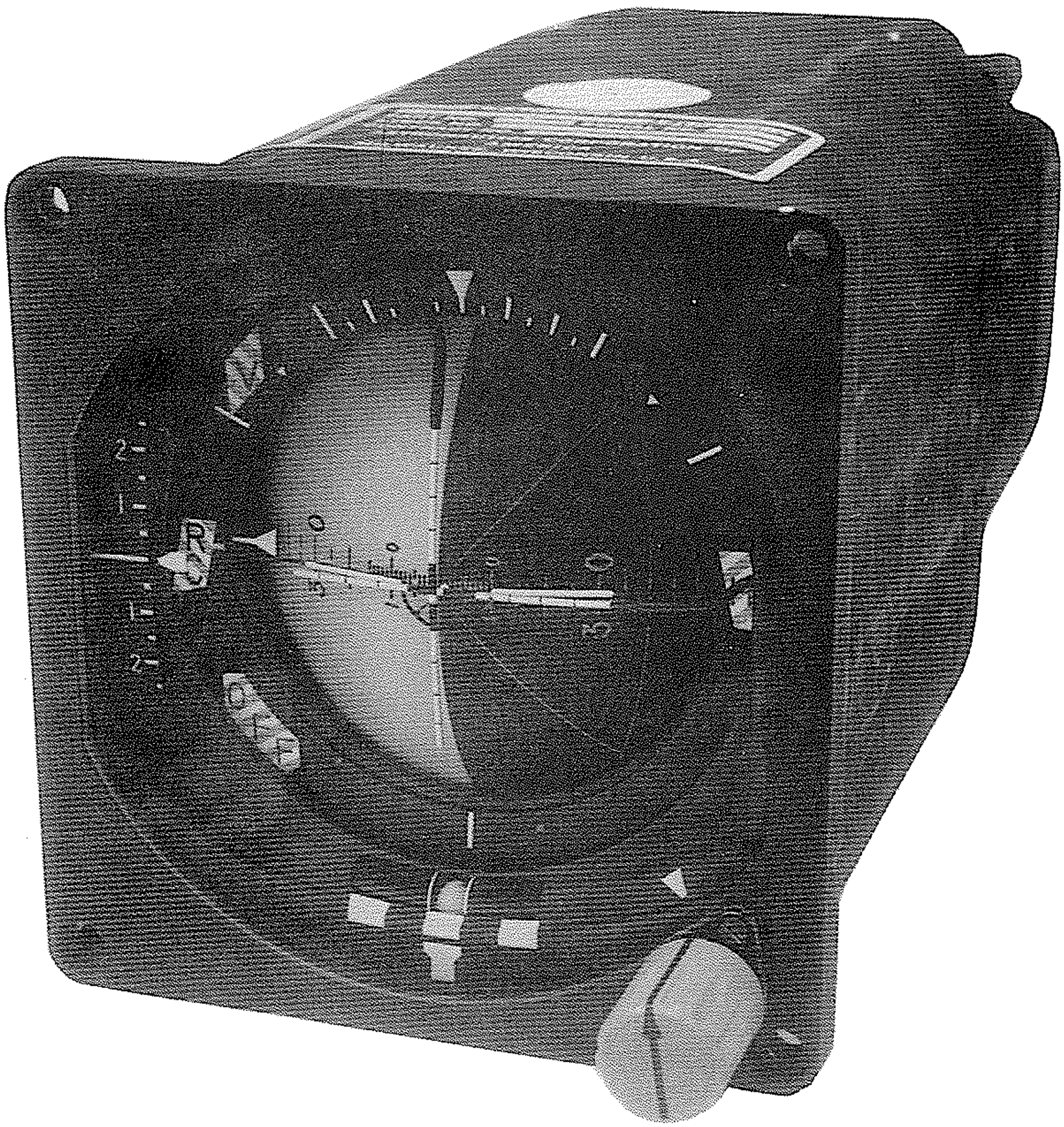


Figure 16B-13. Attitude Director Indicator

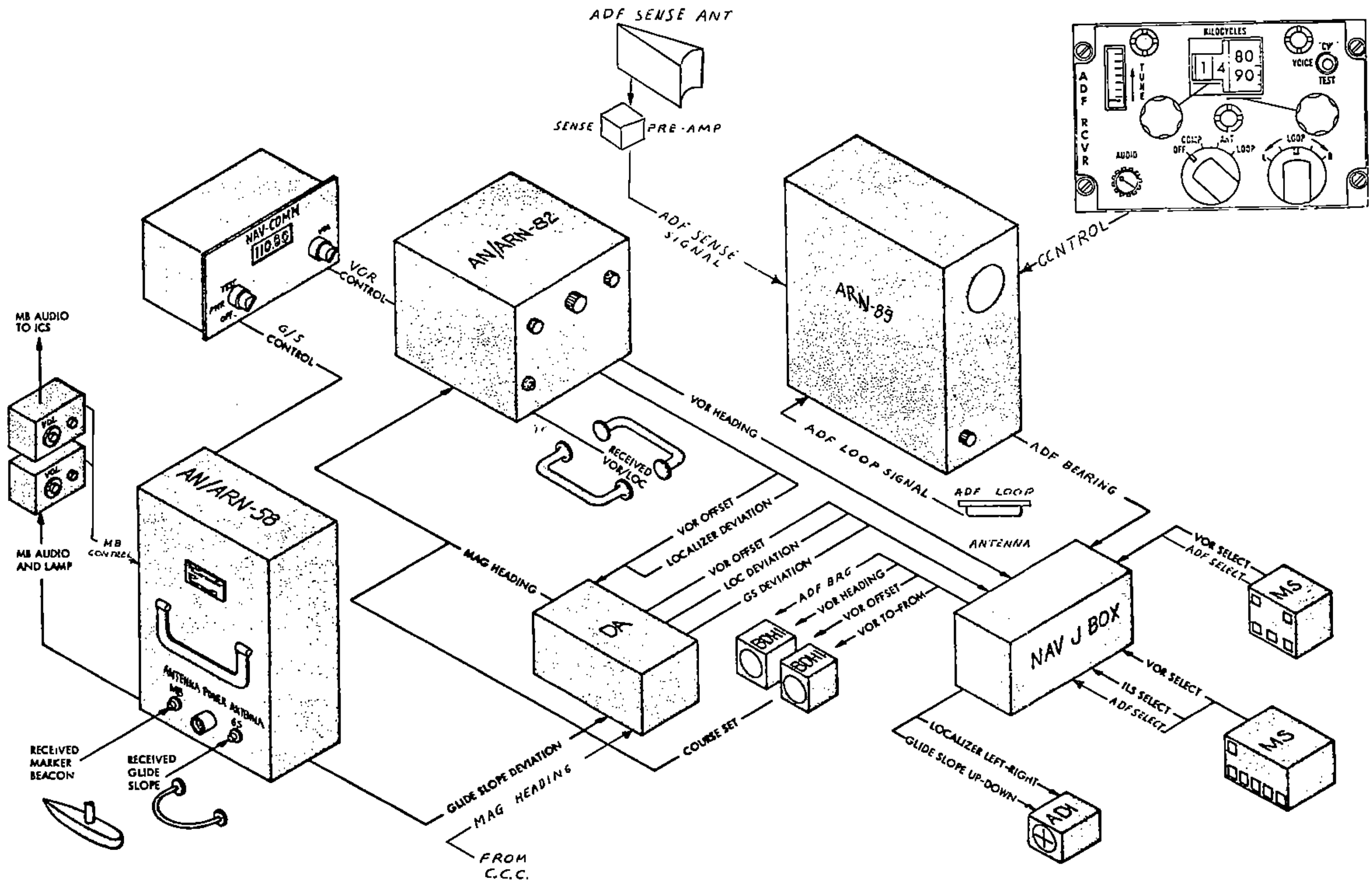


Figure 16B-14. Radio Navigation System Interface Block Diagram

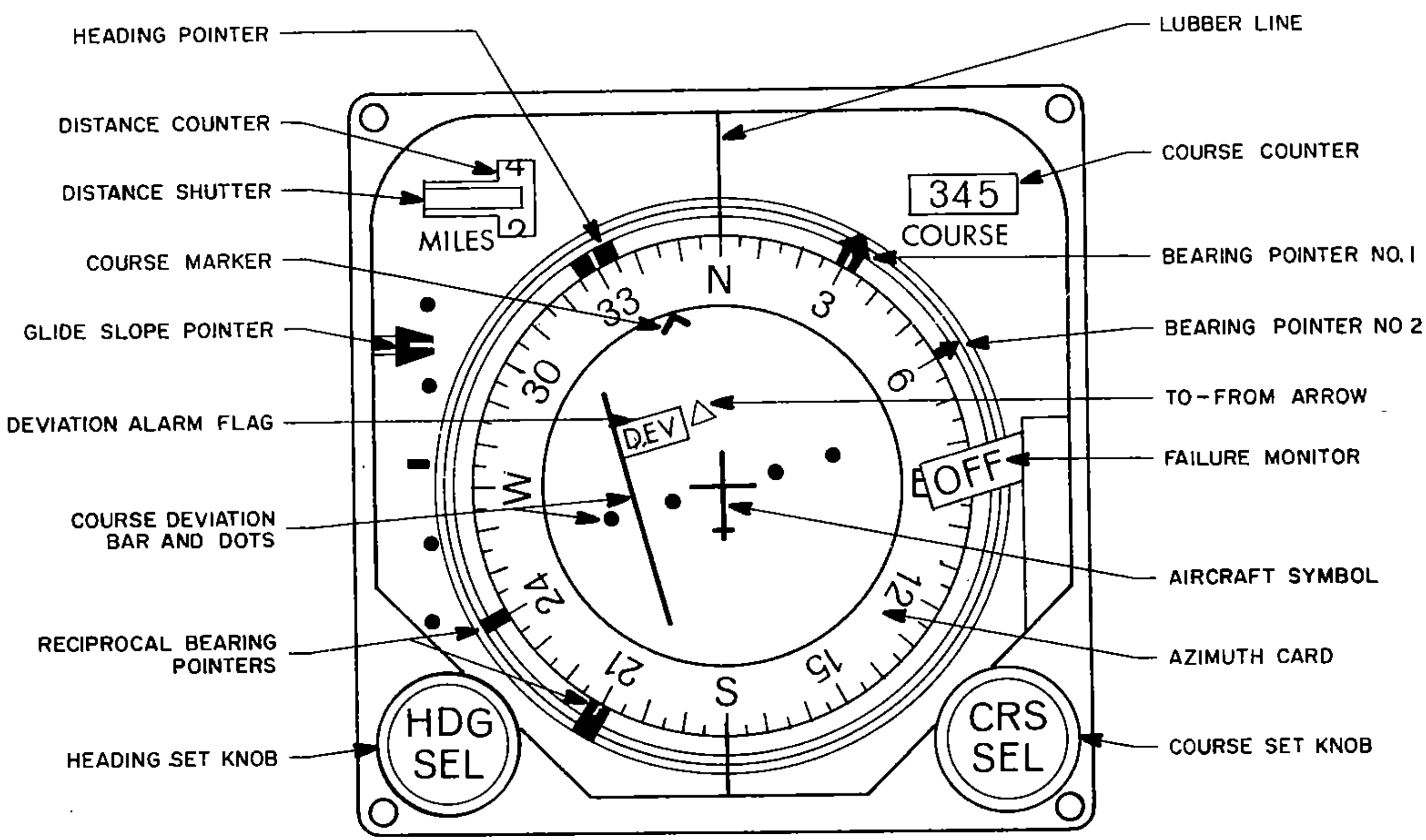


Figure 16B-15. PCRS 4 Inch HSI Pilot

I. GENERAL DESCRIPTION

The copilot/gunner has the primary responsibility for weapon delivery in the AH-56A weapon system. In order to provide the copilot/gunner with maximum assistance in performing his assigned task, a swiveling gunner's station (SGS) is located in the forward cockpit and is the major element of the fire control system.

The basic interrelationship between the overall weapon system is depicted in Figure 17-1. Inputs from the gunner's line of sight (LOS) and target range are provided to the computer from the swiveling gunner's station as the gunner accurately tracks the target. Additional fire control parameters such as vehicle velocities, attitude and air data are transmitted from the sensing systems to the computer. The gunner's line of sight position data is transmitted directly to the turret servos permitting the gun line to be servoed to the gunner's LOS. Within the computer the required equations are solved for train and elevation lead angles. These differential gun line commands are processed to the turrets and displace the gun line from the line of sight in order to produce target hits.

Provisions for proper controls, indications and interlocks to aid in coordinated weapon launching operations is a feature of the AH-56A Fire Control System.

In the event either crew member is disabled, it is necessary that the weapon system survive and is usable under control of the remaining member.

If the pilot is disabled, it will be necessary for the copilot/gunner to fly the aircraft. A feature of the gunner's station is that it includes the necessary controls to permit rapid transition from a gunnery role to a piloting role.

If the copilot/gunner is disabled, the pilot can retain control over weapon delivery functions.

II. COMPONENTS AND LOCATION

Name of Component	Number Per Aircraft	Location in Aircraft
A. Swiveling Gunners Station (SGS)	1	Located in front Cockpit
1. SGS Turntable	1	Lower end of SGS
2. Turntable Drive System	1	Left rear corner of SGS
3. Foot Safety Heelplates	1	Mounted on turntable
4. Periscope Sight Assembly	1	Mounted on turntable
5. SGS sighthead assembly	1	Mounted on bottom of turntable
6. Sighthead mirror	1	Mounted behind sight-head window
7. Sighthead Stabilizer Electronic Module	1	Located in sighthead assembly
8. SGS left hand grip	1	On left side of peri-scope sight assembly pedestal
9. SGS Right hand grip	1	On right side of peri-scope sight assembly pedestal
B. Copilot/Gunner's Weapons Control Panel	1	On upper periscope sight pedestal
C. Copilot/Gunner's Direct Sight	1	Mounted on periscope column
D. Laser Ranging Subsystem	1	Mounted on SGS (except power supply
1. Laser Tranceiver	1	Mounted on left side of sighthead assembly
2. Laser pulse-Forming Network	1	Mounted on turntable
3. Range Computer	1	Mounted on turntable
4. Temperature Control Unit	1	On turntable floor
5. Laser Power Supply Unit	1	In fire control avionics compartment

II. COMPONENTS AND LOCATION (Cont'd)

Name of Component	Number Per Aircraft	Location in Aircraft
E. Weapons Control Unit	1	In fire control avionics compartment
F. Night Vision System	1	On sighthead
G. TOW Missile system	1	On the sighthead

III. COMPONENT DESCRIPTION

A. SWIVELING GUNNER'S STATION (SGS)

The swiveling gunner's station consists of several line replaceable units (LRUs) which comprise the stabilized periscope sighting system, the laser/manual ranging system, the direct sight for target acquisition and heads-up gun firing, and the weapon selection and control system. Additionally, provisions are included for installation of the TOW missile tracker and error detector as well as for the night vision system sensor. Figure 17-5 shows the SGS functional block diagram.

Figure 17-6 lists the LRU's which comprise the swiveling gunner's station.

Major subsystems of the SGS are as follows:

1. SGS Truntable - The SGS turntable and fixed support is the basic structural element of the swiveling gunner's station. The turntable is supported by balls contained in a circular track around the perimeter of the turntable and the fixed portion of the sighting station support.

2. Turntable Drive System - The turntable drive system is located at the left rear corner of the SGS fixed station. The system consists of one drive motor with a tachometer generator, brake, gear train, a clutch, a coupling, and the necessary electronics to control the motor, brake, and clutch. The turntable drive motor can rotate the turntable at a maximum rate of 100 degrees per second.

3. Foot Safety Heelplates - Hinged right and left foot safety heelplates are mounted on the turntable. The copilot/gunner cannot rotate the turntable electrically unless his feet are on the heelplates, the turntable mechanical handcrank clutch is disengaged and the cyclic stick and collective levers are decoupled and stowed.

4. Periscope Sight Assembly - The periscope sight assembly is mounted on and rotates with the turntable. Internally, the periscope sight assembly contains the optical elements which provide the optical path from the target through the sighting head assembly to the copilot/gunner's eye. The periscope optics are designed to provide 3 power and 12 power levels of magnification. The upper periscope assembly pedestal contains a range and status indicator, missile hangfire indicator light and optics for night vision indicator installation.

5. SGS Sighthead Assembly - The sighthead assembly is mounted on the bottom of the turntable and extending through the lower fuselage skin. The entire structure rotates ± 200 degrees with the SGS turntable. The assembly consists of the following:

6. Sighthead Mirror - The sighthead mirror consists of a metal plate covered with highly polished glazed beryllium. The mirror is mounted on the train gimbal assembly behind the sighthead window. The mirror and train gimbal rotate the ± 200 degrees along with the sightline stabilizer for coarse train sightline adjustment. For fine train sightline stabilization, the mirror and train gimbal is allowed to rotate an additional ± 2.5 degrees. The mirror can rotate from 20 degrees above to 70 degrees below the ADL.

7. Sightline Stabilizer Electronic Modules - The sightline stabilizer electronic modules provide the necessary electronic functions for stabilizing the sightline mirror train and elevation servo motors. The modules also furnish necessary information to the computer central complex for the computation of

ballistics correction used to position the nose and belly turrets. Figure 17-4 shows how the basic fire control system in terms of the computer operation.

8. SGS left Hand Grip - The SGS left hand grip is located on the left side of the periscope sight assembly pedestal. The grip contains a trigger for weapon firing, four thumb-operated switches which control magnification, manual ranging, laser ranging and the radio/intercom.

9. SGS Right Hand Grip - The SGS right hand grip is located on the right side of the periscope sight assembly pedestal. The grip contains three pushbutton switches for selecting direct/stabilized mode, seat center, and pilot target designate functions. In addition, a thumb operated tracking transducer is provided for precise target tracking.

B. Copilot/Gunner's Weapons Control Panel - The copilot/gunner's weapons control panel is located on the upper periscope sight pedestal. The panel provides the copilot/gunner with the controls and indicators necessary to operate the fire control system and to select and arm all the weapons.

C. Copilot/Gunner's Direct Sight - The copilot/gunner's direct sight consists of a combining glass with a collimated reticle. The sight may be operated in either a stabilized or direct mode in elevation. It is attached to the periscope column in front of the copilot/gunner for head-up viewing. The line-of-sight is controlled by the thumb tracking transducer. This sight can be used to aim all flexible weapons.

D. Laser Ranging Subsystem - The function of the laser ranging subsystem is to provide a range input to the CCC. The range input is used to compute ballistic correction and linear motion compensation data. All units of the laser ranging subsystem except the laser power supply are located on the SGS.

The range finder consists of a neodymium laser incorporated into the copilot/gunner's swiveling sight. This laser range finder incorporates a pulse rate of 10 per second, which is utilized to develop range data for linear motion compensation computations ballistics and target motion. The major units of the laser ranging subsystem are as follows:

1. Laser Transceiver - The laser transceiver is mounted on the left side of the sightline stabilizer optical bench (sighthead assembly). The unit consists of a laser optics cell, laser transmitter, and laser receiver. The transmitter emits short pulses of coherent light to a target visually sighted through the periscope or direct sight. These pulses are reflected back from the target to the receiver. A silicon avalanche diode in the receiver senses the reflected light pulses. These electrical pulses stop a time-measuring device in the range computer that started when the initial laser pulses were emitted. The elapsed time interval is then converted to target range by the computer central complex and then displayed in meters on the range indicator in the periscope.

2. Laser pulse-Forming Network - The pulse-forming network is basically a capacitive circuit capable of being charged with electrical energy. When laser ranging is selected and the range insert switch on the SGS left grip is pressed, a short-duration energy pulse with characteristics of extremely high voltage and current is discharged rapidly into the laser flash-tube circuit.

3. Range Computer - The range computer generates incremental time information based on electrical pulses from the laser receiver or stadiametric ranging as selected by the SGS left grip circle select switch and magnification select switch. This information is transmitted to the computer central complex where it is converted into equivalent range information, then transmitted to the range indicator in the periscope.

4. Temperature Control Unit - The temperature control unit is located on the turntable floor under the copilot/gunner's seat. The unit contains a thermoelectric unit to cool the laser liquid coolant and an air-to-air heat exchanger to cool the air in the sealed sighthead. The unit also contains a coolant servicing filler cap, a coolant filter and deionizer assembly.

5. Laser Power Supply Unit - The laser power supply unit is located in the fire control avionics compartment below the pilot. The unit provides high voltage to charge the laser pulse forming network. It also supplies high voltage trigger pulses to actuate the laser transmitter. Before each trigger pulse it supplies a reset pulse to the range computer. This pulse resets the time measurement counter used in calculating range.

E. Weapons Control Unit - The weapons control unit (WCU) functions as an interface between the CCC and the armament systems. The WCU is mounted in the fire control avionics compartment. The WCU is essentially a two-way interpreter. The CCC can digest digital information, while the various weapons systems are analog oriented, so the WCU contains the necessary analog-to-digital and digital-to-analog converters that allow the CCC and the armament systems to communicate with each other. In addition, the WCU contains weapons select logic to ensure acceptable weapons combinations, and fire interrupt logic to disable weapon firing under certain conditions.

F. Night Vision System (NVS) - The SGS contains units of the Night Vision System. The NVS sensor is mounted in the sighthead and an IR window is mounted on the sighthead pod in front of the NVS sensor. The infrared sensor line of sight is stabilized and boresighted to the copilot/gunner's line of sight. The components for the system such as the power supply unit, signal processing unit, compressor heat exchange unit, cathode ray tube controls and display units are installed on the swiveling gunner's station's turntable and periscope pedestal.

G. TOW Missile System - Some of the TOW Control Equipment (TCE) is installed in the SGS's sighthead. The TOW missile infrared tracker line of sight is stabilized and boresighted to the copilot/gunner's line of sight. The TOW error detector is also installed in the sighthead, near the infrared tracker.

IV. SYSTEM OPERATION

A. Weapon Control - Independent weapon controls are provided for the pilot and copilot/gunner. The pilot and copilot/gunner can select and fire separate weapons at the same time. Each crewman has controls and indicators to permit selection and firing of either of the guns, the pilot has controls for firing rockets and selective and emergency jettison of external stores. The copilot/gunner has controls for selection and firing of the TOW missile and emergency jettison capability of external stores.

Pilot's Controls - The pilot's controls consist of:

1. Sight controls for helmet and direct sight
2. External stores jettison controls (selective and emergency)
3. Weapon selection switches
4. Master ON-OFF armament switch
5. Weapon firing switch (trigger)
6. Ammunition rounds counters
7. Fire interrupt indication

A trigger on the pilot's cyclic control stick fires all properly selected weapons. A firing interrupter button on the pilot's control stick interrupts all firing sequences regardless of which crewman is firing. Release of the control for a particular flexible weapon, immediately returns the unselected weapon to the stow position. Firgure 17-7 shows the pilot's weapon control panel configuration.

Copilot/Gunner's Controls - The copilot/gunner's controls consist of:

1. Sight tracking controller thumb operated transducer
2. External stores emergency jettison controls
3. Weapons selection switches
4. Master ON-OFF armament switch
5. Weapons firing switch (trigger)
6. Ammunition rounds counters
7. Power ON switch
8. Laser/manual ranging controls
9. Fire interrupt indication
10. Night vision system controls

The copilot/gunner's controls permit operation and control of guns or missile. Release of the control for a particular flexible weapon, immediately returns the unselected weapon to the stow position. Selection of an alternate flexible weapon is possible without having to return the SGS to the neutral position. Figure 17-8 shows the copilots/gunner's weapon control panel configuration.

PILOT HELMET SIGHT

The helmet sight system provides pilot control of the gun turrets. In addition, the pilot can cause the swiveling gunner's station to be slewed to his line of sight in order to designate a target to the copilot gunner.

Components of the helmet sight system, and their general location in the aircraft, are as shown in Figure 17-9.

In operation, the system accurately measures the train and elevation angular position of the pilot's helmet. This data is sent to the central computer where ballistics corrections for the selected weapon(s) are added. The resultant total gunline command is then sent to the selected weapon(s) via the SGS weapon control unit.

The helmet sight and sensor assembly (HSSA), which is mounted on the standard Army flight helmet, contains an adjustable sight piece that projects a collimated reticle image into the pilot's right eye. The reticle thus appears to be in the plane, normal to his line of sight, containing the target. A knob on the HSSA (item 5 of Figure 17-9) permits lateral and vertical adjustment of the sightpiece to accommodate pilot differences. When not in use the sightpiece may be stowed out of sight in the HSSA by operation of another knob (item 1 of Figure 17-9) on the HSSA. Reticle brightness is adjusted by a control on the pilot's weapon panel. A backup reticle lamp is provided in the HSSA in case of primary lamp failure. This lamp is controlled by the same primary reticle lamp brightness control.

The reticle pattern seen by the pilot is identical to that of the pilot direct sight (see Figure 17-11). Prior to take off the pilot electrically boresights his helmet sight by aligning the reticle images of his two sights and operating a momentary switch on his weapons panel.

Light source assemblies (LSAs) are mounted to each side of, and slightly behind, the pilot generate rotating beams of infrared light (see items 3 of Figure 17-9). These beams sweep across the helmet striking photodiode sensors located in the HSSA. When struck, these sensors (2 on each side) generate pulses which are amplified and sent to the sensor electronics assembly (SEA) in the fire control compartment located beneath the pilot (see item 6 of Figure 17-9). In the SEA the sensor pulses are used to control counters in conjunction with reference pulses from the LSAs. These counters develop numbers which are a function of the HSSA angular position in LSA coordinates. The angles are transmitted digitally to the computer central complex where they are converted to helmet angles in aircraft coordinates (train and elevation).

Except for an electrical cable, there is no mechanical connection to the helmet. In the interest of safety, the cable has a quick-disconnect connector that pulls apart if more than a ten pound pull is applied to the cable.

Angular coverage of the system is ± 90 degrees in train and from +20 to -30 degrees in elevation. As the pilot turns his head from one side of the aircraft to the other, the computer causes the system to automatically switch to the LSA and HSSA photodiode set on that side. This simplifies the system because the data from each side is processed sequentially rather than in parallel.

Every component of the system except the HSSA has an elapsed running time indicator and a built-in-test equipment indicator, or BITE indicator. A BITE indicator for the HSSA is located on the front of the SEA. This approach is used to minimize the weight added to the helmet. Total system weight is less than 10 ounces.

Figure 17-10 is a block diagram of the pilot helmet sight system.

Pilot Direct Sight

The pilot direct sight is used for firing rockets and guns in a forward-only mode. When the pilot or copilot/gunner selects rocket position on the stores control panel, a reticle position command signal is sent from the computer central complex to position the direct sight reticle image. This signal is a function of target range, aircraft velocity, and rocket ballistics. The reticle may be moved up to 100 milliradians in elevation (depression) in response to this command.

When the pilot wishes to fire either or both of the turreted weapons with the direct sight, the reticle is electrically caged to the zero degree train and zero degree elevation position. In addition, the reticle may be mechanically caged by operating a handle on the front face of the sight. (See Figure 17-11).

The sight is located directly in front of the pilot on top of his instrument panel. The reticle image is collimated by a 3.5 inch diameter lens using four air spaced elements. This image is then reflected off a combining glass into the pilot's line of vision. Because the image has been collimated, the reticle appears to be at infinity or, practically, in the plane of the target. The reticle pattern seen by the pilot is as shown in Figure 17-11.

Reticle brightness is adjusted by a control located on the front of the sight. A backup reticle lamp filament is activated by pulling out on the brightness control in the event the primary filament fails. The lamp is readily accessible in a holder located on the left side of the sight body.

A test switch located on the sight front panel permits an end-to-end check of the sight's servo system. Pressing the test switch causes the reticle to depress through an angle of 25 milliradians, or one radius of the larger circle of the reticle pattern.

An inclinometer, or ball-bank indicator, on the front of the sight aids the pilot in maintaining the wings level, zero sideslip, altitude necessary for accurate 2.75 inch rocket firing.

If the pilot selects rockets with the sight mechanically caged, a lamp on the control panel is illuminated. As previously discussed, the sight must be uncaged in order for the reticle to follow the computed rocket lead angle command.

A quick disconnect latch, one each side of the sight, permits the sight to be readily removed from its base, or mount. Replacement of the sight does not require the sight to be reboresighted as long as the sight base had not been disturbed.

Movement of the reticle image in the rocket mode of operation is accomplished by moving a mirror within the sight forebody. A block diagram of the pilot direct sight is shown in Figure 17-12.

STORES CONTROL SYSTEM

The stores control system equipment is composed of a pilot's control panel, a programmer, and four control directors. Figure 17-13 is a simplified block diagram of the system. Figure 17-14 is the stores control cockpit control panel. A brief description of the equipment follows:

1. Pilots Control Panel - The pilot's control panel is installed in the pilot's cockpit immediately below the main instrument panel and to the right of the SGS weapons control panel. Figure 17-14 shows the control panel and the general arrangement of the controls. Displays, in addition to those indicating selection when the panel is in an active state, are as follows:

    a. Weapon Type Loaded - Displayed on the individual, illuminated station select switch when activated:

- Impact fused rockets - RDT legend illuminated
- Smoke rockets - SMK RDT legend illuminated
- VT fused rockets - VT RDT legend illuminated
- Stores or tanks - ST legend illuminated

b. Interlocks when in effect - Displayed on the reset illuminated switch:

- Reset, whenever firing philosophy is violated - R legend illuminated
- A fault is in the system - NO GO legend illuminated

c. System in Test - Displayed on reset switch - T legend illuminated

d. Selected station is empty - Displayed on reset switch - E legend illuminated.

The control panel assembly, in addition to the selection and display functions previously described, contains the interlock circuitry.

2. Programmer

The programmer is installed in the fwd fire control/avionics bay. The following functions are implemented by the programmer:

a. Weapon inventory and control function

b. Station priority transfer logic function

c. Firing or release interval and pulse forming output function

The programmer has no direct operating controls. It does have a test initiate switch to ready the system for ground checkout. Additionally, it contains the central fault indicators for the system.

3. Control Director - There are four control directors, one installed in each wing pylon. These distribute the release and firing pulses within each pylon station under commands from the programmer.

The system, additionally, has the following features which safeguard the vehicle in the air, when the system is selected for use, and on the ground during reloading:

1. Interlocks - Interlocks are provided which interrupt all firing and release outputs from the system whenever a selected delivery option is in conflict with air vehicle safety.

2. Automatic Station Select - Automatic station selection is provided to compensate for lateral imbalance whenever the lateral imbalance envelope of the vehicle would be exceeded by the selected delivery option.

3. Gun Turret Fire Interrupt - The system generates a signal, whenever the trigger or store release button is actuated, which serves to interrupt the firing of any gun turret being used. This firing interruption is continuously maintained from the initiation of the store control system firing cycle until a period of 300 ms elapses after the trigger or store release button is released.

4. Ground Safety during Arming - Provisions for ground safety during arming is achieved by the following means:

    1. A voltage monitor is provided, visible to the armorer during loading, that will indicate dangerous stray voltages when present.

    2. A special flagged safety pin is provided for use during loading. When inserted into its receptacle in the control director, the pin interrupts all firing circuits at the control director, homes the stepper switches and resets the store registration switch to OFF.

V. PCRS CONFIGURATION

Differences between the developed configuration of the fire control system and the system described herein are minor and are as follows:

1. Simplification of the gunner's station

2. Refinement in the computer programming

3. Lead sulphide helmet sight detectors in lieu of silicon.

These differences do not constitute any technical risk. The change to the lead sulphide helmet sight sensor eliminates a sunshine problem present with the developmental system and has been proven on other helmet sight programs.

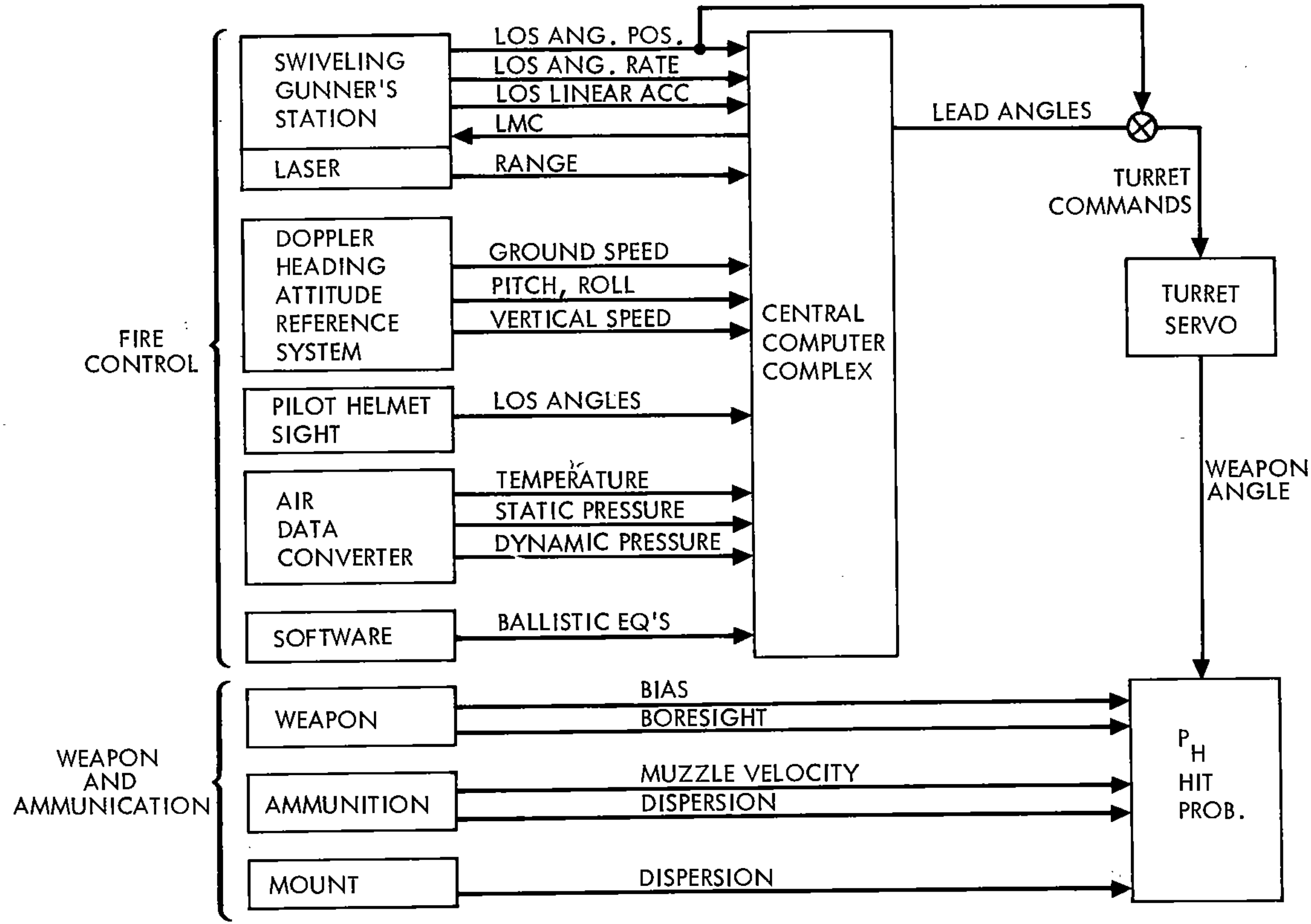


Figure 17-1. Fire Control Signal Flow - Gunnery

Figure 17-2. AH-56A Sighting Station

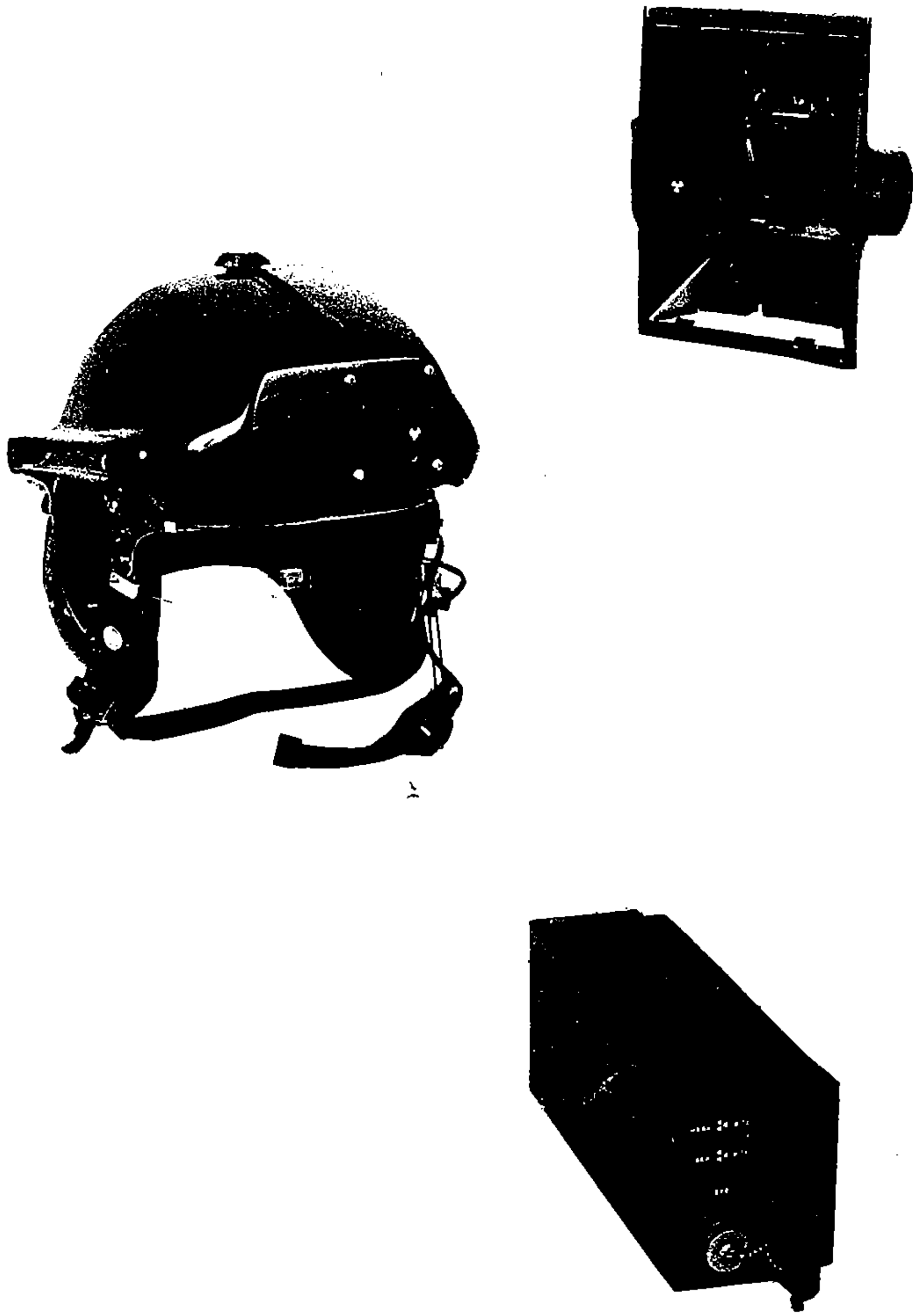


Figure 17-3. Helmet Sight Components

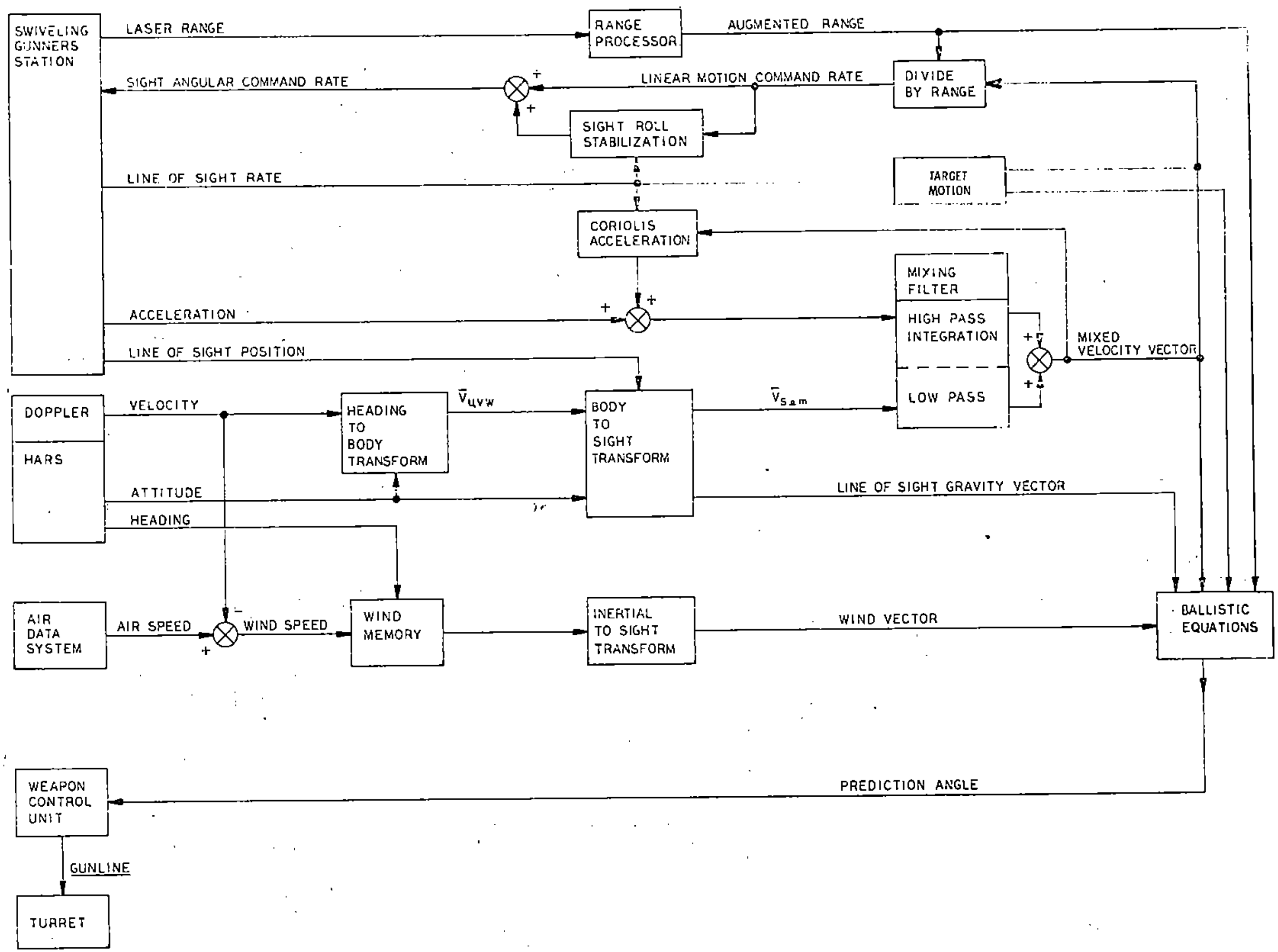


Figure 17-4. Fire Control Signal Processing

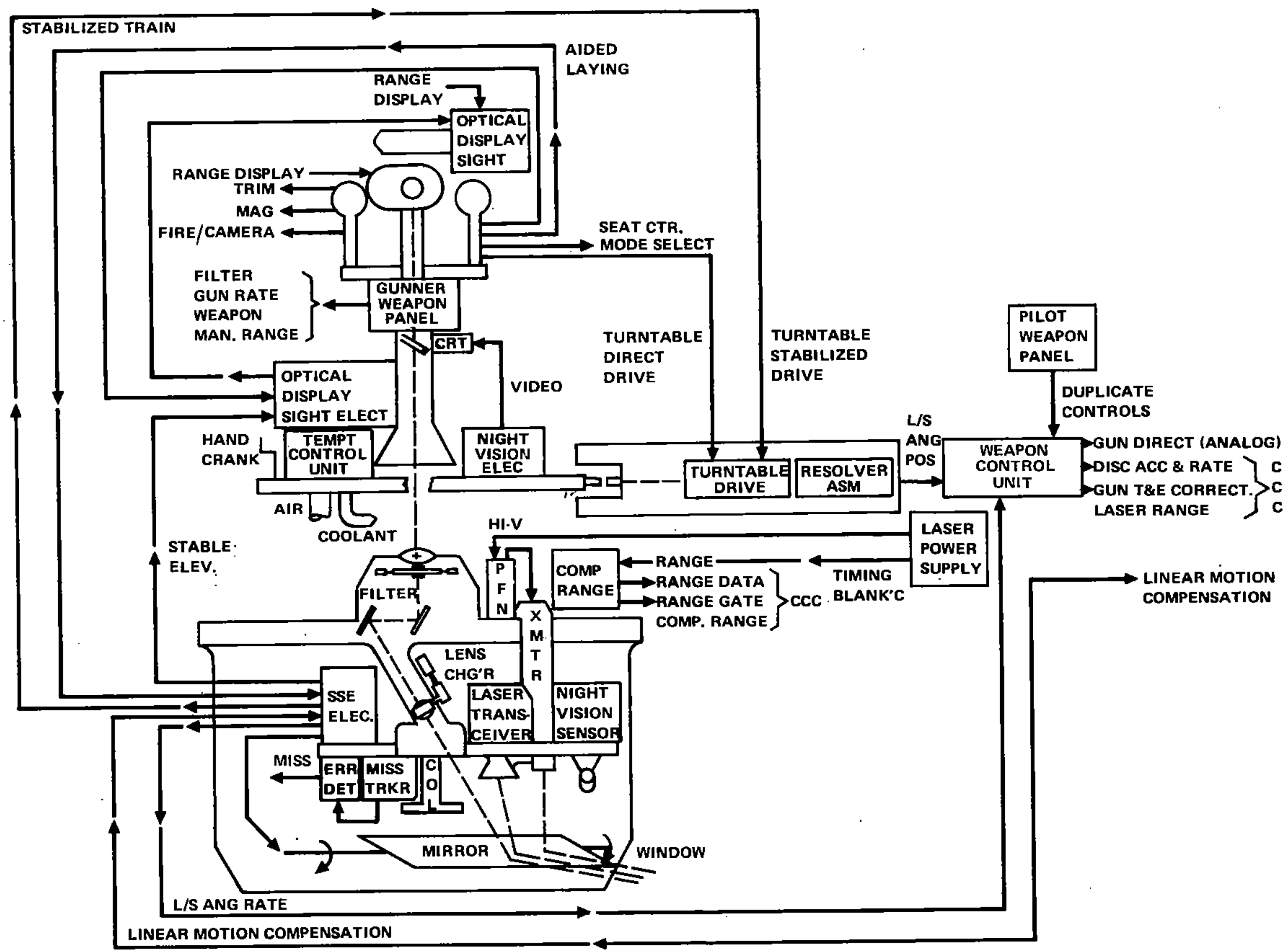


Figure 17-5. SGS Functional Block Diagram

- SIGHT, OPTICAL DISPLAY
- GRIP ASSY. CONTROLLER RH
- GRIP ASSY. CONTROLLER LH
- PANEL, WEAPONS - PILOT
- PANEL, WEAPONS - GUNNER
- CONTROL UNIT, WEAPONS
- POWER SUPPLY, LASER
- TRANSMITTER, LASER

- CONTROL UNIT, TEMPERATURE
- COMPUTER, RANGE
- NETWORK, PULSE FORMING
- SIGHTING HEAD ASSEMBLY
- DRIVE, TURNTABLE
- SUPPORT SIGHTING STATION, AIRCRAFT WEAPONS
- PERISCOPE SUBASSEMBLY - UPPER
- CONTROL, OPTICAL DISPLAY SIGHT

Figure 17-6. Swiveling Gunners Station Line Replaceable Units

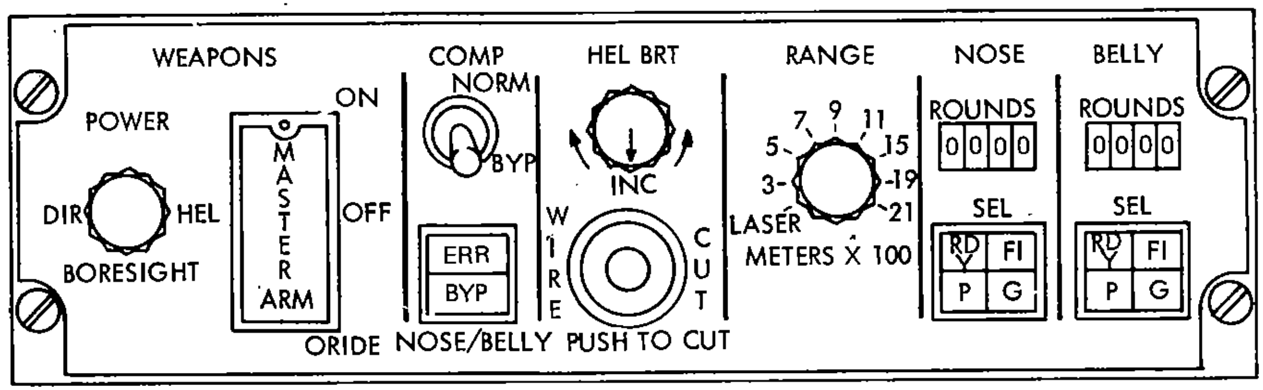


Figure 17-7. Pilot's Weapon Control Panel

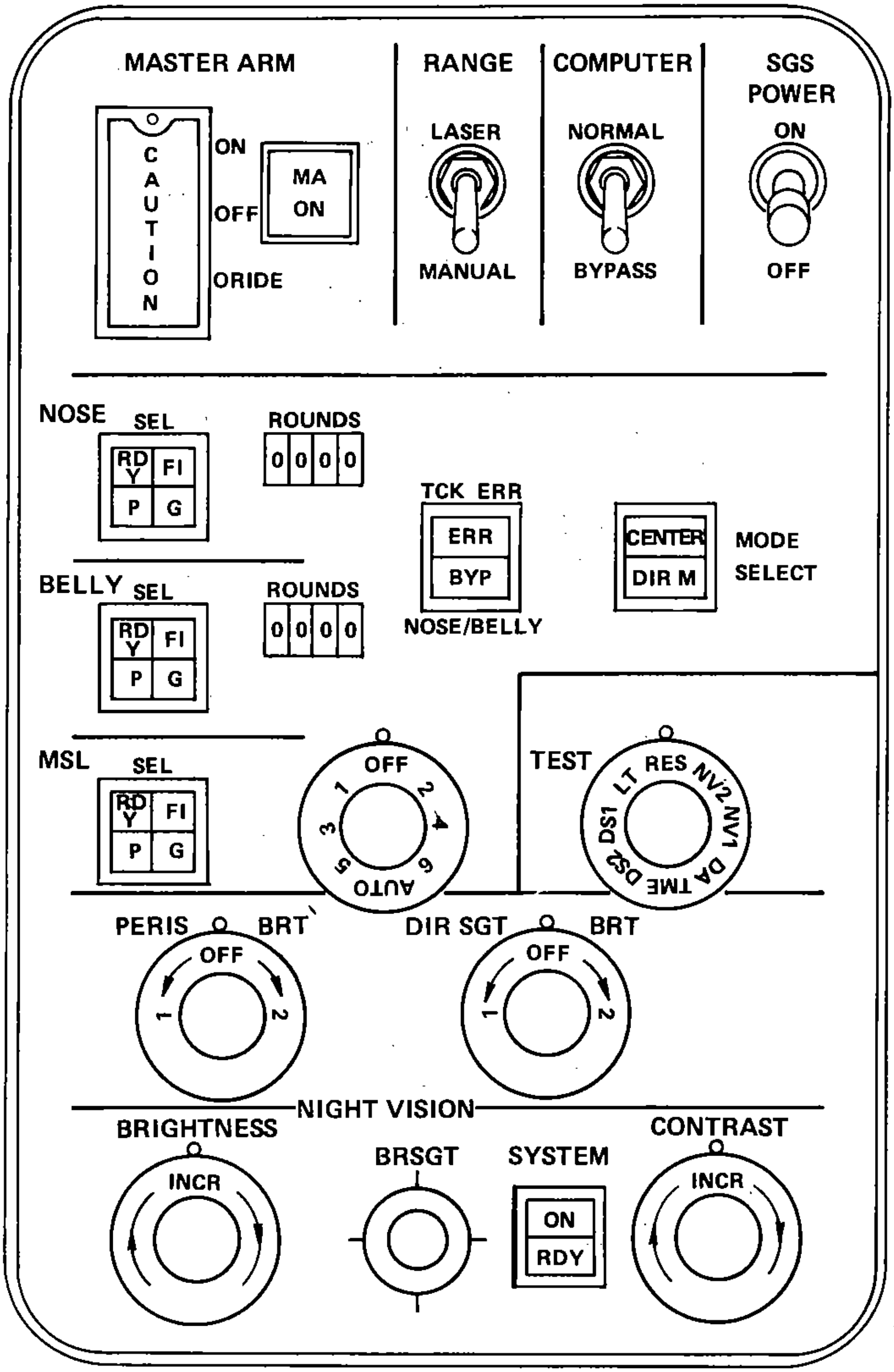
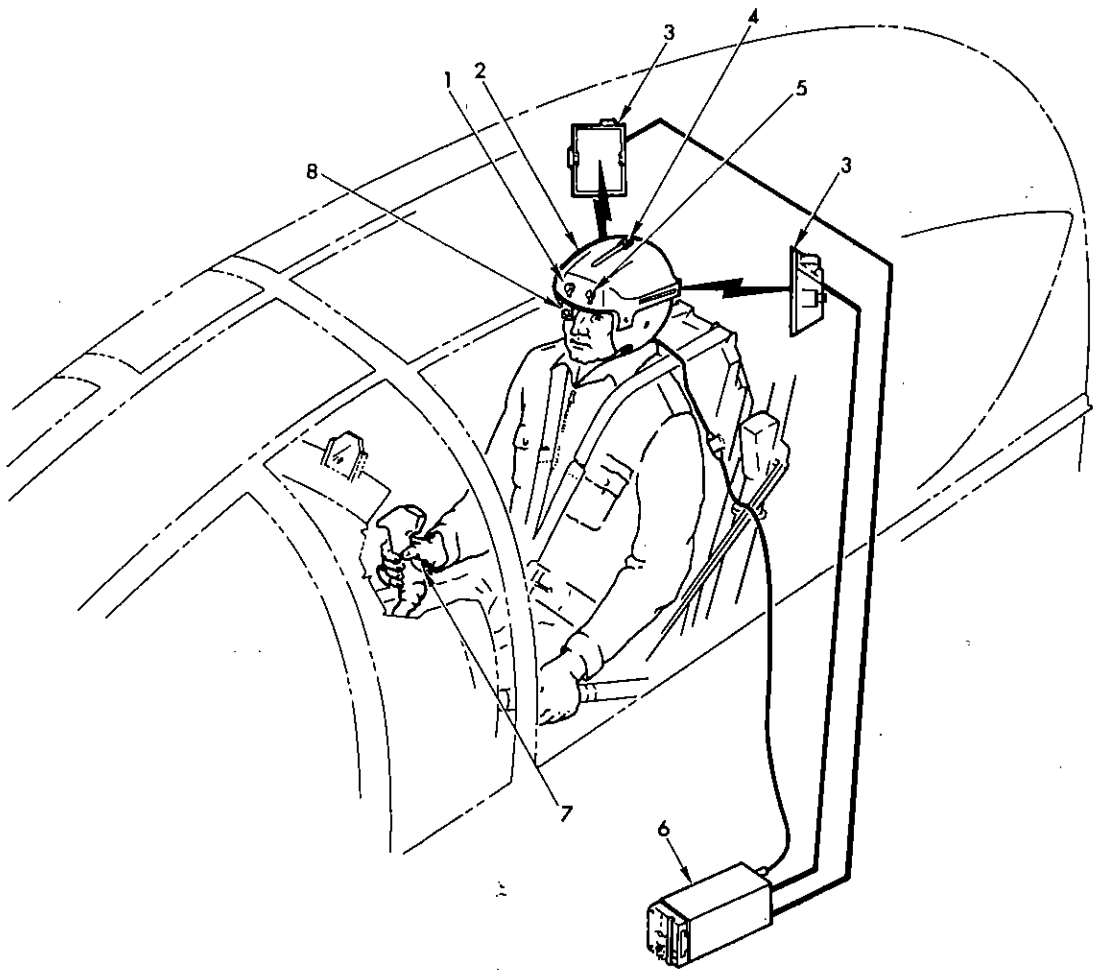


Figure 17-8. Gunner's Weapon Control Panel



1 Reticle sight stow knob
2 Helmet sight and sensor assembly
3 Light source assemblies
4 Helmet visor adjustment knob
5 Reticle sight adjustment knob
6 Sensor electronics assembly
7 Trigger switch
8 Reticle sight

Figure 17-9. Helmet Sight System Components

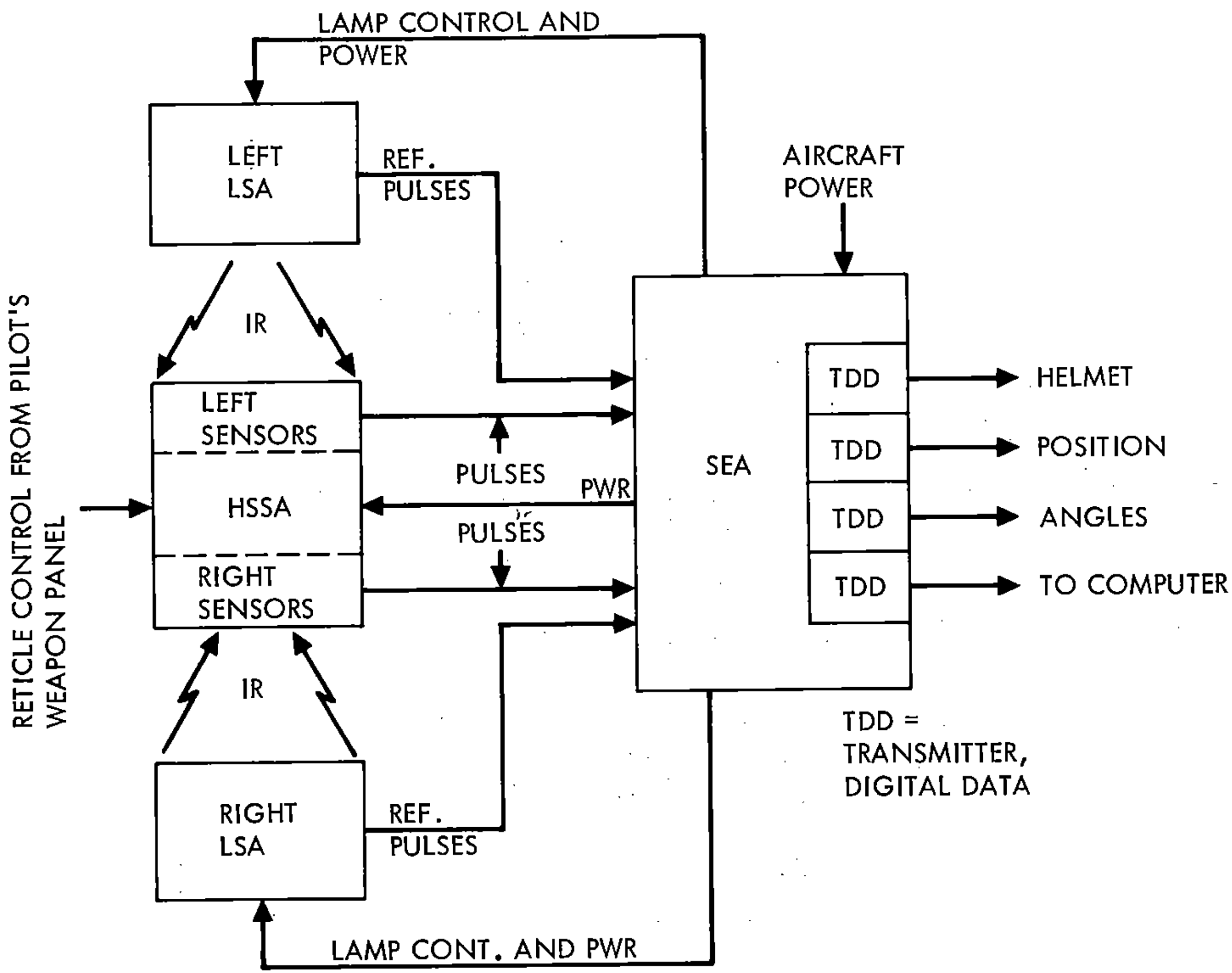


Figure 17-10. Pilot Helmet Sight Block Diagram.

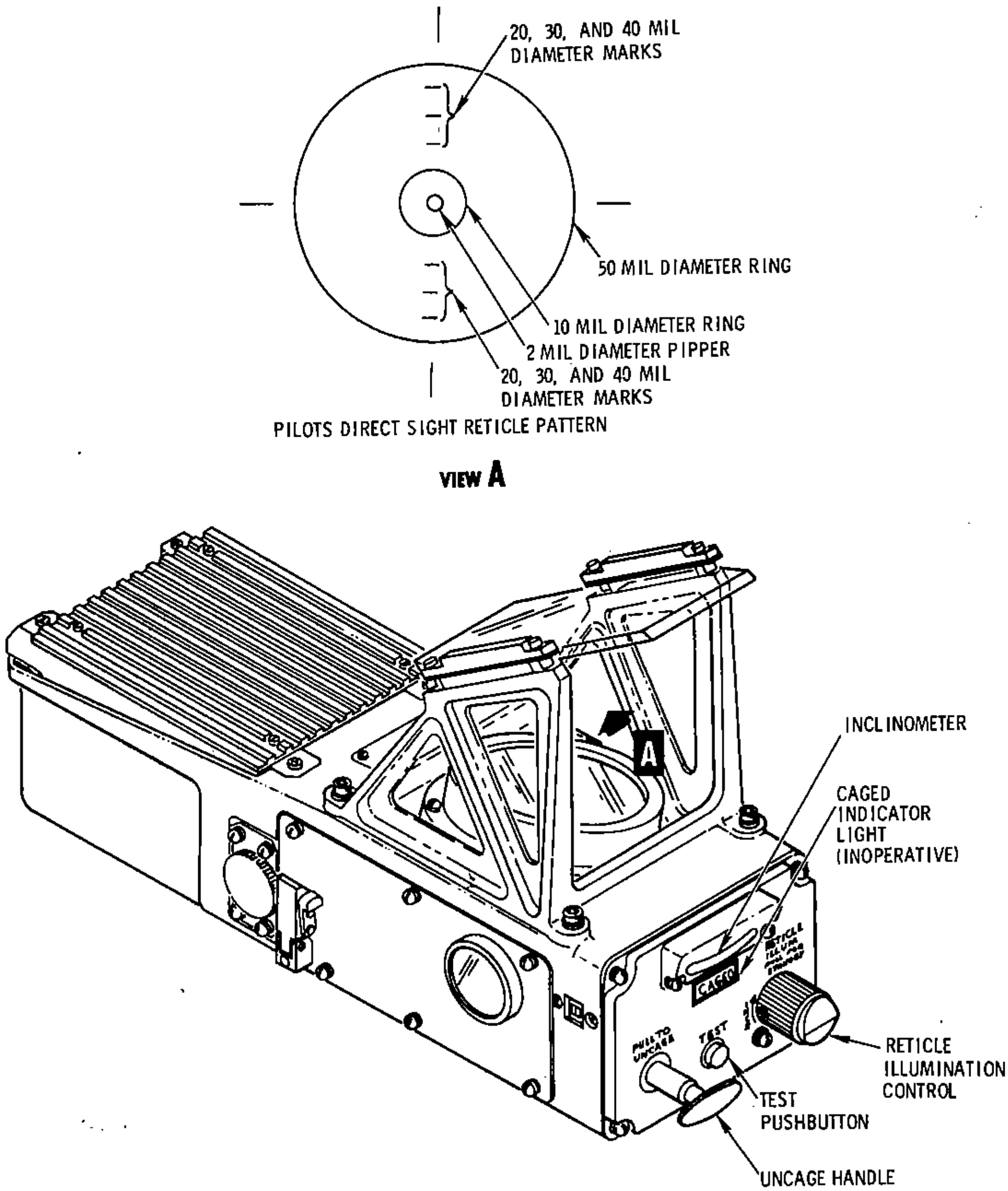


Figure 17-11. Pilot's Direct Sight

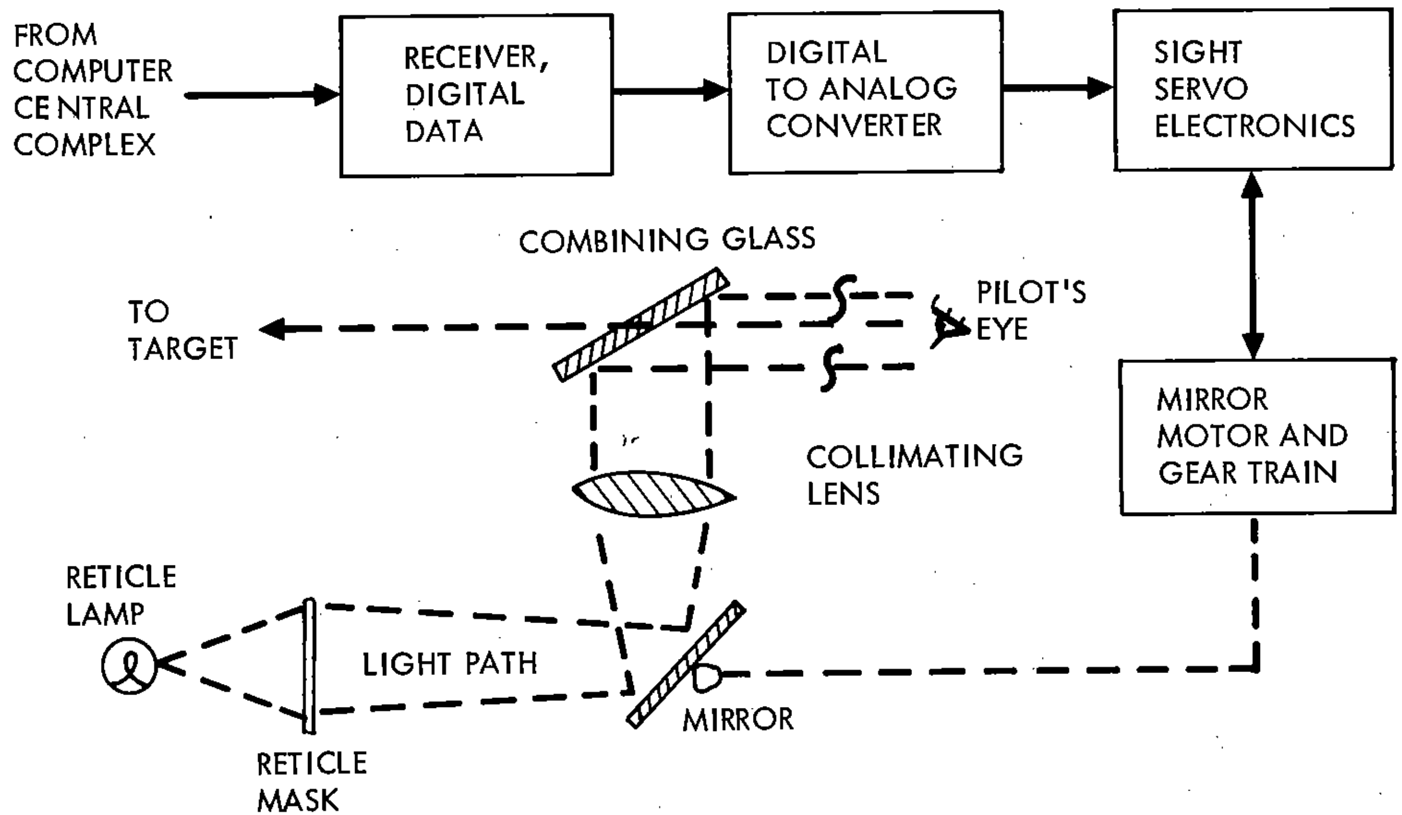


Figure 17-12. Pilot's Direct Sight Block Diagram

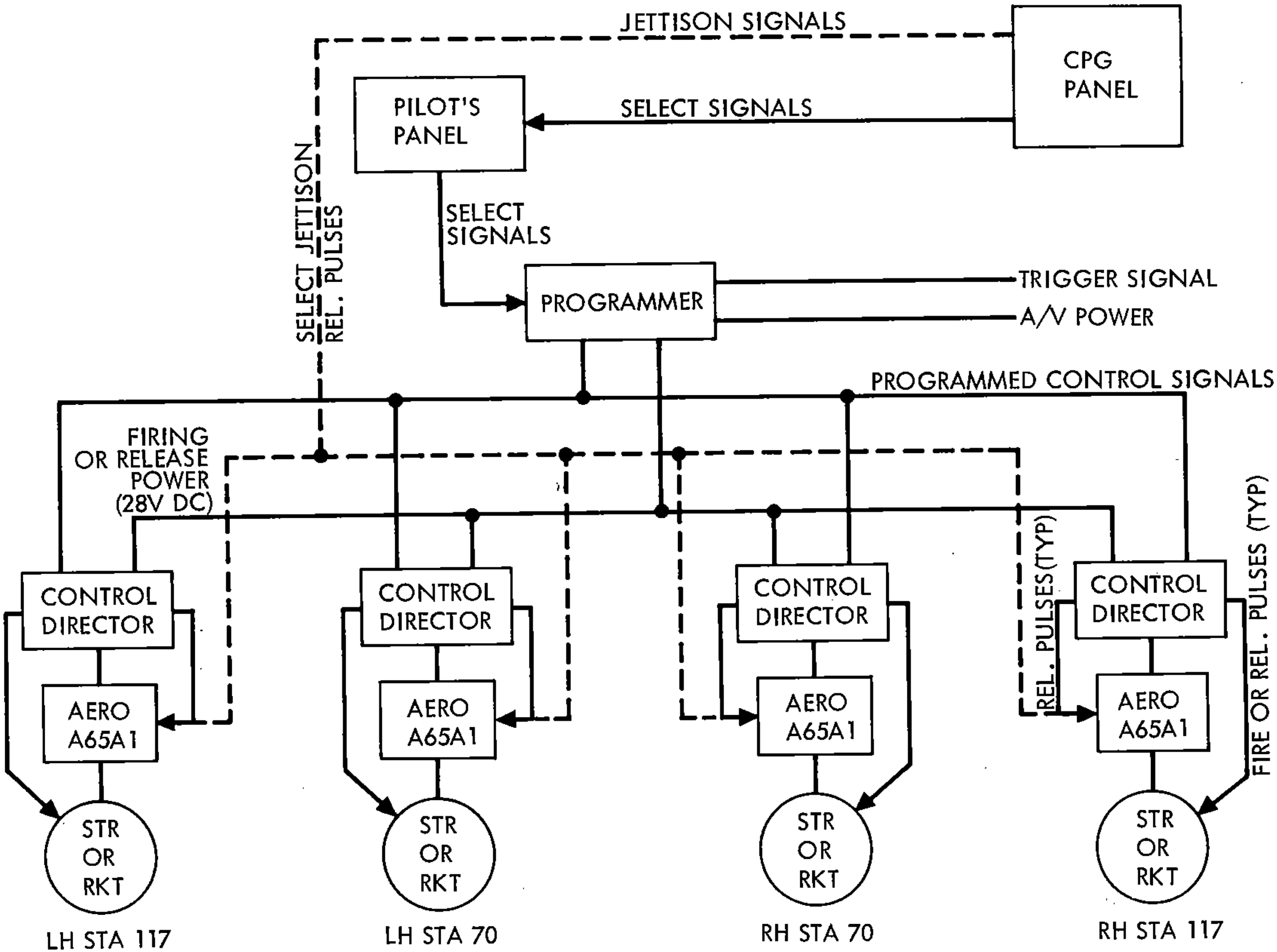


Figure 17-13. Block Diagram - Store Control Subsystem

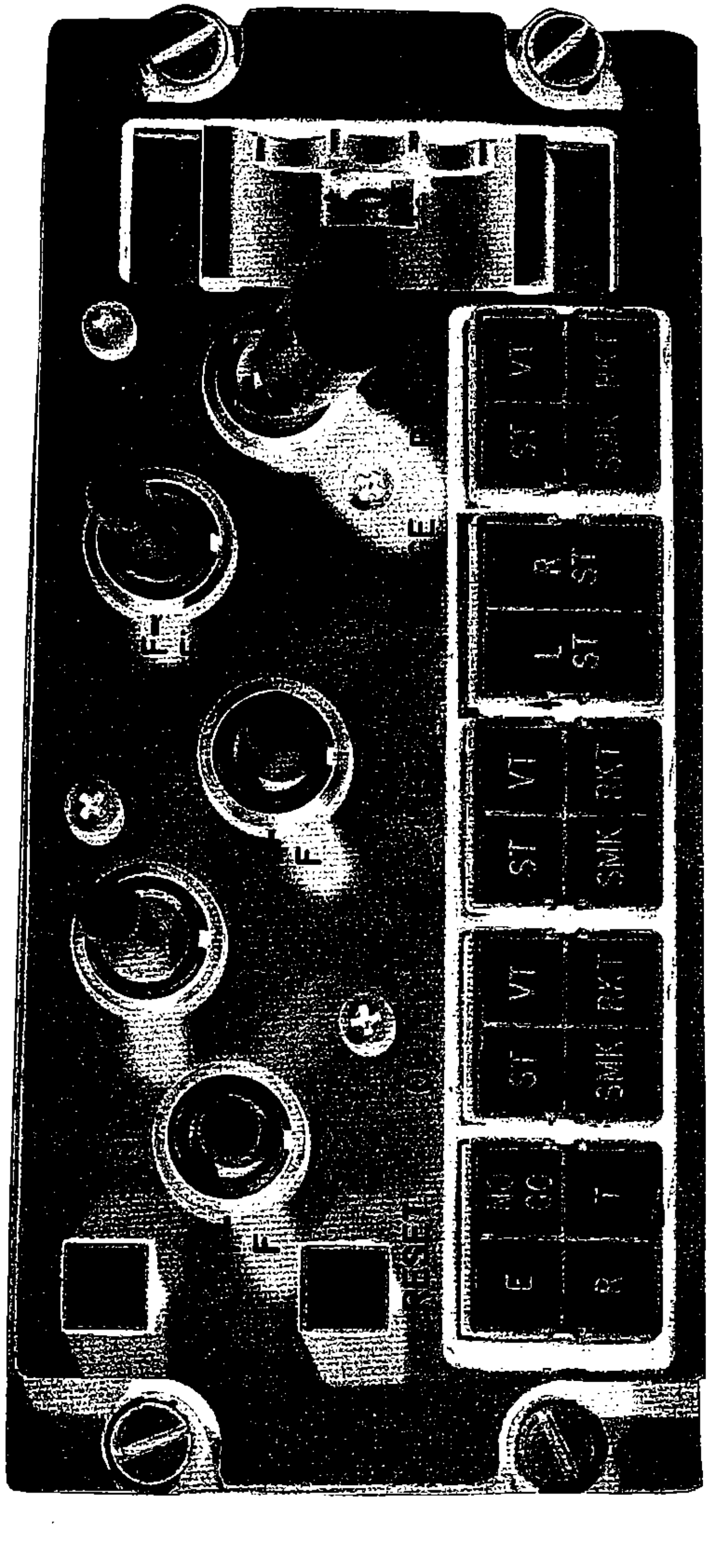


Figure 17-14

NIGHT VISION SYSTEM

I. GENERAL DESCRIPTION

A. Background

Battlefield surveillance and weapon direction historically has been performed optically with human eyes. The eye, though a marvelous instrument, suffers greatly in performance when the scene illumination drops to below a one footcandle level. This illumination level occurs at twilight when the color perception of the human eye disappears.

Near the turn of this century when accurate artillery and automatic weapons became standard equipment in armed forces, being visually observed by the enemy became highly dangerous because of the fire power he could bring to bear on a known target location. Eventually, almost all supply, evacuation, redeployment, reinforcement and staging activities were performed at night when men and material concentrations in the open could not be observed. The time worn phrase "under cover of darkness" assumed full meaning for about five decades. Until the 1940's very little work was successfully done in attempting to penetrate the "cover of darkness." Incandescent illumination with flares was the primary means used to penetrate darkness. The discovery of the S-1 high quantum efficiency photoemissive surface just prior to WW II gave rise to a family of devices. The most familiar was the "Snooper Scope." These early devices had comparatively low sensitivity, short ranges and frequently were used in conjunction with IR searchlights, particularly on tanks, to improve range capability. Searchlights, of course, are highly vulnerable if the enemy has a device to look for IR radiation.

In the late fifties the S-20 multi-alkali photoemissive material was discovered. In conjunction with concurrent development of fiber optics a new family of devices came into being -- multistage Image Intensifiers (I^2) as well as Low Light Level TV (L^3TV).

The I^2 devices are usually three stage light amplifiers that operate by moonlight, starlight or sky glow in the visible region of the electromagnetic spectrum. The L^3TV differs primarily in that the image is presented to the observer in raster form. These highly sensitive multistage devices are overload sensitive (gunflashes, headlights, etc. tend to cause severe display smear) and dependence upon ambient light limits their scope of operational utility. In the mid 1960's the far infrared devices came into prominence for vehicle borne night vision device applications. The major factor in fruition of these devices was the development of integrated semi-conductor processing which made the size and weight of IR systems reasonable. Performance in the far infrared spectral region for these devices is excellent.

B. Cheyenne Night Vision Sight (AN/ANS-25)

Lockheed considered the previously mentioned approaches and selected the far IR real time thermal imaging device as the most suitable for the Cheyenne helicopter. The device, originally called PINE now generally referred to as NVS is fully integrated with the swivelling gunner's station of the Cheyenne as shown in Figure 18-1. It enables the copilot/gunner to perform all target acquisition and fire control functions at night, as well as in certain types of weather and camouflage conditions. Provisions are also incorporated to display the image to the pilot. An independent display is required in the pilot's cockpit for that purpose.

The Cheyenne NVS or AN/AAS-25 Night Vision System is an advanced Forward Looking Infrared (FLIR) System with multi-element detector array that converts infrared (IR) radiation to a TV format for display on a one-inch Cathode Ray Tube (CRT). This CRT is viewed through the Swivelling Gunner's Station (SGS) optical periscope when Night Vision is selected. All NVS Controls are part of the SGS.

The NVS was designed to mount on the SGS with the IR Sensor located in the lower sighthead. Built-In Test (BIT) circuitry is included which permits fault isolation to the Line Replaceable Unit (LRU).

II. COMPONENTS AND LOCATIONS (See Figures 18-2 and 18-3)

Component	Quantity	Location
Infrared Sensor Unit	1	Mounted on lower section of SGS
Signal Data Converter	1	Mounted on aft section of SGS turntable drive assy
Indicator	1	Mounted adjacent to SGS periscope
Compressor	1	Mounted on RH side of SGS turntable drive assy
Electronic Power Supply	1	Mounted on LH side of SGS turntable drive assy

III. MAJOR COMPONENT DESCRIPTION

A. Infrared Sensor Unit

The sensor optically scans the target area and collects IR radiation in the 8- to 11.5-micron spectral region. The IR radiation is focused on a multi-element detector array. The detector array is mounted inside a permanently sealed dewar, and is maintained at an operating temperature of $25^{\circ}K$ by a refrigeration unit mounted on the back of the dewar. The detectors convert the IR energy into video information in the form of electrical signals. These signals are connected to the signal processor unit. The functional block diagram is shown in Figure 18-4.

The optomechanical folding layout is shown in Figure 18-5. This layout depicts the eight major subassemblies listed below. These assemblies are designed in modular form in order to facilitate ease of assembly, checkout, and maintenance.

1. Telescope assembly

2. Field of view actuator

3. Horizontal scanner

4. Vertical interlace mirror assembly

5. Detective assembly

6. Vertical boresight and synchronizing generator

7. Narcissus (NARC) source

8. Automatic responsivity control (ARC) source

A wide field-of-view is provided by the insertion of a telephoto lens cell immediately behind the narrow field-of-view objective lens. The wide field-of-view optics are mounted in a casting that rotates about an axis 90 degrees to the optical axis. When the narrow-field-of-view is in use, the wide-field-of-view lens assembly is rotated so that all lens elements and the housing are outside the optical path of the narrow-field-of-view system. The rotary motion is provided by an electromagnetic rotary activation through a self-locking mechanism.

Both fields-of-view are fixed focus and adjusted to focus on a common spherical focal surface. The curvature of the detector array approximates the curvature of the focal surface.

Three plane mirrors are positioned in the optical system to fold the optical path into the minimum space available. The first mirror, located at the center of focal surface curvature, is used to produce the azimuth scan, the second is used for boresight adjustment and the third produces the scan interface.

A light source illuminating a photocell by reflection from the azimuth scan mirror provides a sync pulse at the neutral mirror position. This pulse is used to generate the display horizontal sweep.

Responsivity of each detector element in the detector array varies enough to require normalization of the array. To equalize the responsivity, an automatic responsivity (ARC) circuit is incorporated into the NVS, thus eliminating the need for re-adjusting the multi-channel detector bias resistors whenever the sensor or the SDC is replaced in a system.

The interlace mirror is driven by a magnetic actuator that positions the optical image vertically to four different positions on the detector array. Positioning of the mirror occurs during the horizontal scan mirror retrace time. The different vertical positions overlap each other to permit optical overscanning which minimizes the horizontal line structure on the display.

A germanium window, which mounts on the outside of the SGS pod, permits the Sensor to "see" the IR scene while protecting it from exterior environmental elements. The window is coated with high hardness antireflective coatings to reduce reflections and is designed to withstand gunblast forces up to 20 psig. Window configuration is shown in Figure 18-6.

B. Signal Data Converter

The SDC amplifies, filters, and multiplexes into a single channel the signals generated by the multi-element detector array. In addition, the SDC Signal provides the logic and control functions for these operations as well as the synchronization and control of the scan mechanisms and display.

A crystal-controlled clock signal and logic circuitry control the multiplexing so that each sensor detector channel is sampled in an orderly sequence. The sensor detector channels are sequenced serially so that detector channels 1, 5, 9, etc., are combined in order in the first intermediate channel, channels 2, 6, 10, etc., are combined in the second, etc. The other intermediate channels are sampled sequentially in a high level sequence so that successive samples follow the sensor detector signal channels in numerical order 1, 2, 3, 4, etc., from top to bottom. This permits the display CRT beam to be moved from top to bottom in a single constant rate vertical sweep for each complete sampling cycle. A high speed blanking signal is also generated by the multiplexer logic to blank the CRT beam between samples. Other functions carried out in the signal processor include the synchronization and control of the scan devices in the sensor, and the generation of the deflection signals for the display. The unit configuration for the signal data converter is shown on Figure 18-7, and the functional diagram in Figures 18-8 and 18-9.

C. Indicator or Display Unit

The display unit is a one-inch cathode ray tube (CRT) that presents a real time visual presentation of the IR sight information to the operator. The CRT is viewed through the SGS sight.

The IR image is presented on the display in a raster format. The raster is synchronized horizontally to the mechanical mirror scan in the Sensor, and vertically to the electronic scan generated by multiplexing. As both vertical and horizontal scans are linear with time and are unidirectional, there is no need for driven sweep. Only one synchronization pulse input to the indicator is required for each scan. Figures 18-10 and 18-11 illustrate the dimensions and construction of the unit. Figure 18-12 shows the parameters of the displayed image. Interlace (square wave), video, self test and power inputs comprise the remaining indicator inputs.

D. NVS Compressor

Cooling of the multi-element detector array is provided by a compressor and a refrigerator that mounts on the IR Sensor. Helium gas at approximately 300 psig is supplied to the refrigerator by the compressor via the helium lines.

A salient feature of the compressor is its low moving mass and low operational speed of 3 rps which greatly reduces the vibration induced by this device and brings about a longer operating life. The use of a two-speed cryoengine which operates at double the normal operation speed for the first 15 minutes after turn on, yields a cool-down time of less than 20 minutes. A functional diagram is shown in Figure 18-13.

E. NVS Electronic Power Supply

All NVS system power (with the exception of the aircraft +28 volts dc, and 115V, 3 Ø, 400 Hz) is provided by the Electronic Power Supply. The dc voltages generated consist of -25 vdc, +25 vdc, -15 vdc, +15 vdc and +5 vdc.

The EPS provides all power required for operation of the NVS Sensor and associate electronics. It operates directly from standard MIL-STD-704 400-Hz aircraft power to generate the above filtered and regulated dc power. It also provides 360-Hz power to drive the azimuth scan mirror since the frequency stability of the standard 400-Hz aircraft power is not adequate for this purpose. The unit is mounted just outboard of the gunners left foot as shown in Figure 18-3. The dimensional outline is shown in Figure 18-14 and the EPS interface with the Cheyenne is shown in Figure 18-15.

IV. SYSTEM OPERATION

A. Normal Operation

IR scene energy is passed through the IR Window and reflected to the Sensor by an SGS pointing mirror. The IR Sensor is a two field-of-view (FOV) device that collects IR energy emitted from the scene. The scene is scanned horizontally while the detector array is sampled vertically. Electrical signals generated by a multi-element detector array are sent to a Signal Data Converter (SDC) for processing.

The Signal Data Converter (SDC) amplifies the multi-element detector array's electrical signals and multiplexes them to a single line of video. The SDC also provides timing circuits for synchronization of IR Sensor and IR Display Automatic Responsitivity Control (ARC) to normalize the gain of the multi-element detector array, contrast control circuitry, and crosshair injection logic. The single line of video, along with the synchronization signal, is sent to the IR Indicator. The video information is used to drive the indicator's one-inch CRT. A mirror on the SGS is positioned by a lever which transfers the image on the one-inch CRT through the SGS periscope to the eyepiece.

Graphically, the function of the system is illustrated in Figure 18-16. The functional LRU interrelationships and signal flow are illustrated in Figure 18-17.

There are six functional controls for the NVS that are incorporated into the SGS. These are:

- On-Off Switch on GWP
- Brightness Control on GWP
- Contrast Control on GWP
- BITE control on GWP

- Boresight Control on GWP
- FOV Select Switch on Gunner's Left Handgrip.

Functionally, the controls give the copilot gunner the ability to exercise the NVS as follows:

Control	Function
Brightness, Contrast	Provides a means for the CP/G to adjust the picture for optimum image quality
NVS System ON	Enables CP/G to turn NVS System On
Built-In-Test	A two position switch permits CP/G to fault isolate to a Line Replaceable Unit (LRU).
FOV-Wide/Narrow	Enables CP/G to select either wide or narrow viewfield
Boresight	Enables NVS alignment with other SGS sensors

The normal operation of the NVS involves the following steps:

1. Check Cryogenic Refrigerator helium pressure -- it must be above 200 psig.
2. Turn SGS seat power on.
3. Turn NVS on by depressing the alternate "ON-OFF" pushbutton on the GWP.
4. Image should appear in the SGS eyepiece after 15 minutes.

5. Adjust image for best resolution using the brightness, contrast, and SGS diopter controls.

6. Using distant target (1000 m or more from the ship) superimpose day and night reticles on the target using boresight switch located on the GWP. When reticles are superimposed the NVS is ready for use.

B. Redundant or Emergency Operation

No emergency or redundant operations are possible with the NVS.

C. Fault Indication

There are three malfunction indications in the NVS.

1. Insufficient pressure in the helium reservoir.

2. Power supply "BITE" flag tripped.

3. Image discrepancy. This is by far the most important indication of all. If the image is clear and sharp, the system is working well. Conversely, impending failures generally will first evidence themselves by an image anomaly.

V. SUPPLEMENT DESCRIPTION

Subsequent to delivery at Lockheed the NVS has been tested in laboratory and in flight. No system failure has occurred and the NVS system has met or surpassed all QMR requirements, and has demonstrated ability to meet requisite QMR parameters.

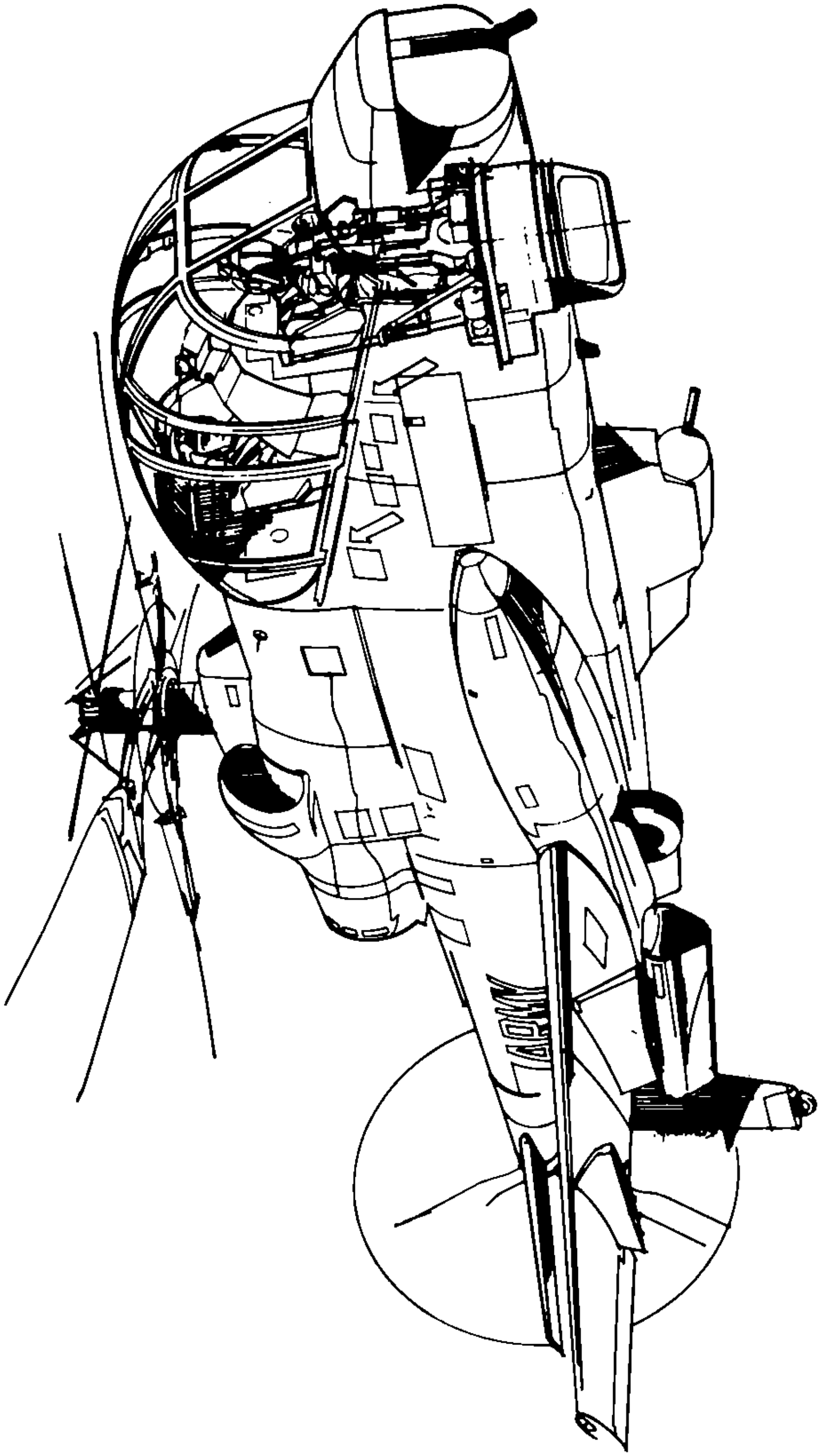


Figure 18-1

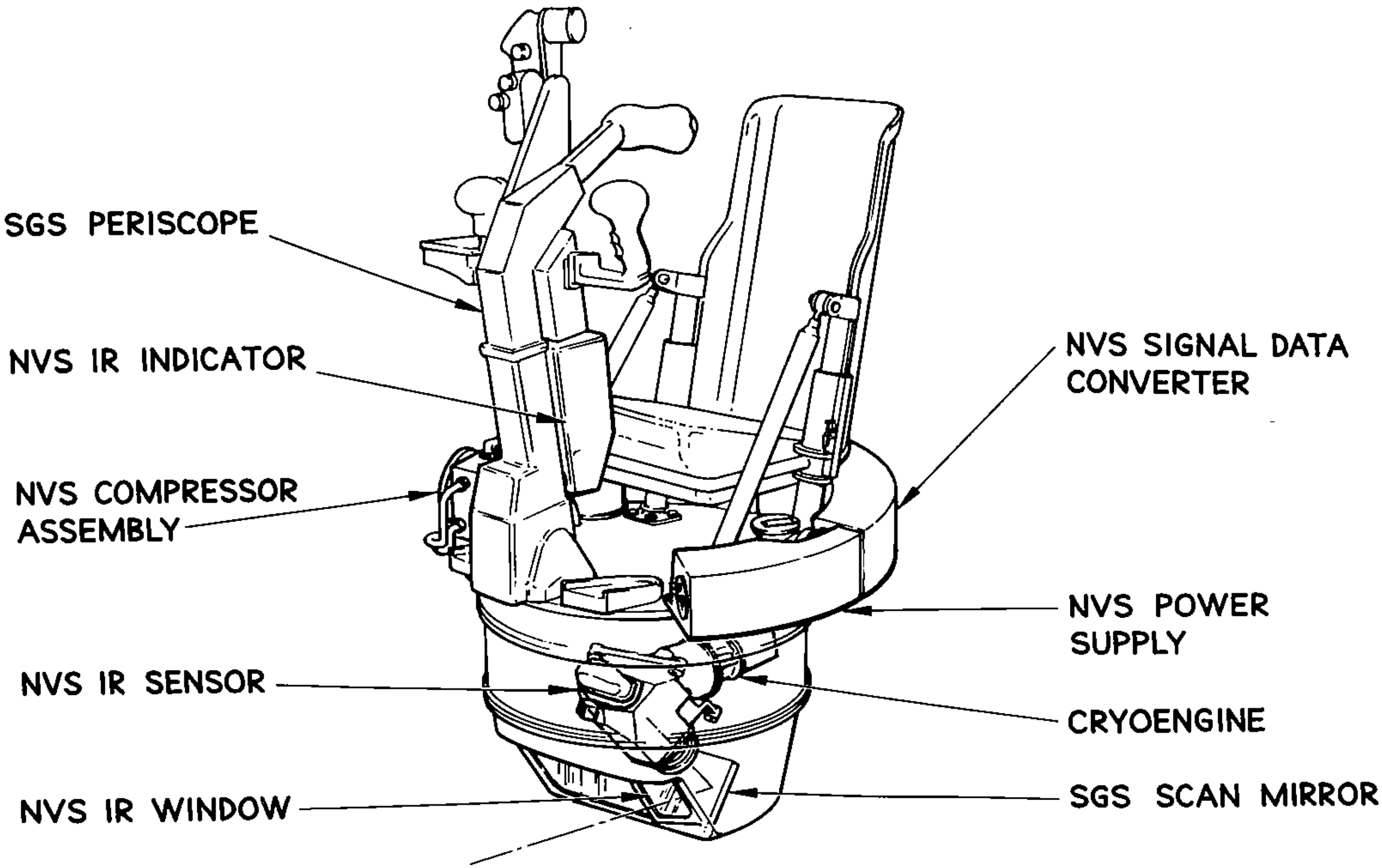


Figure 18-2. NVS Installation in Cheyenne Swiveling Gunner's Station

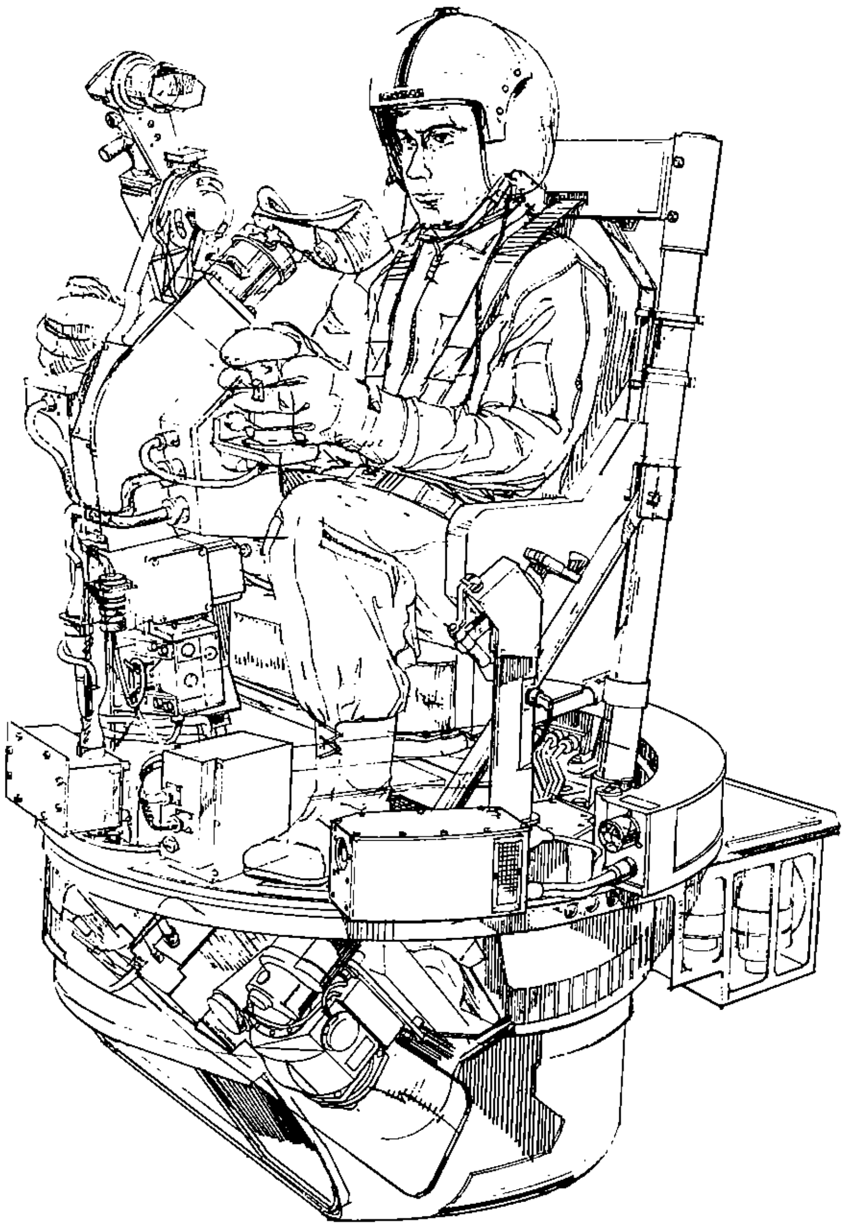


Figure 18-3. Passive IR Night Vision System

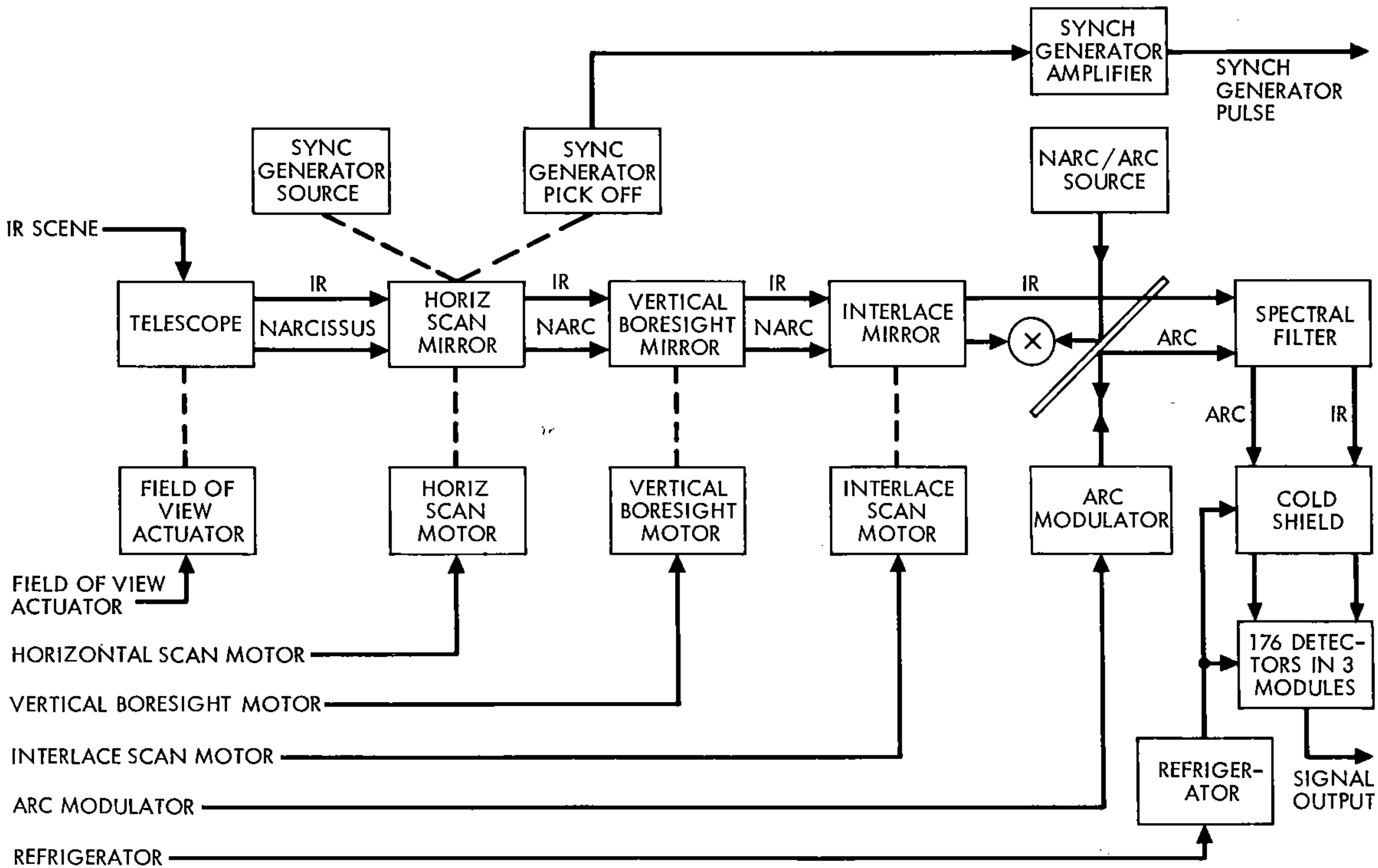


Figure 18-4. Sensor Block Diagram

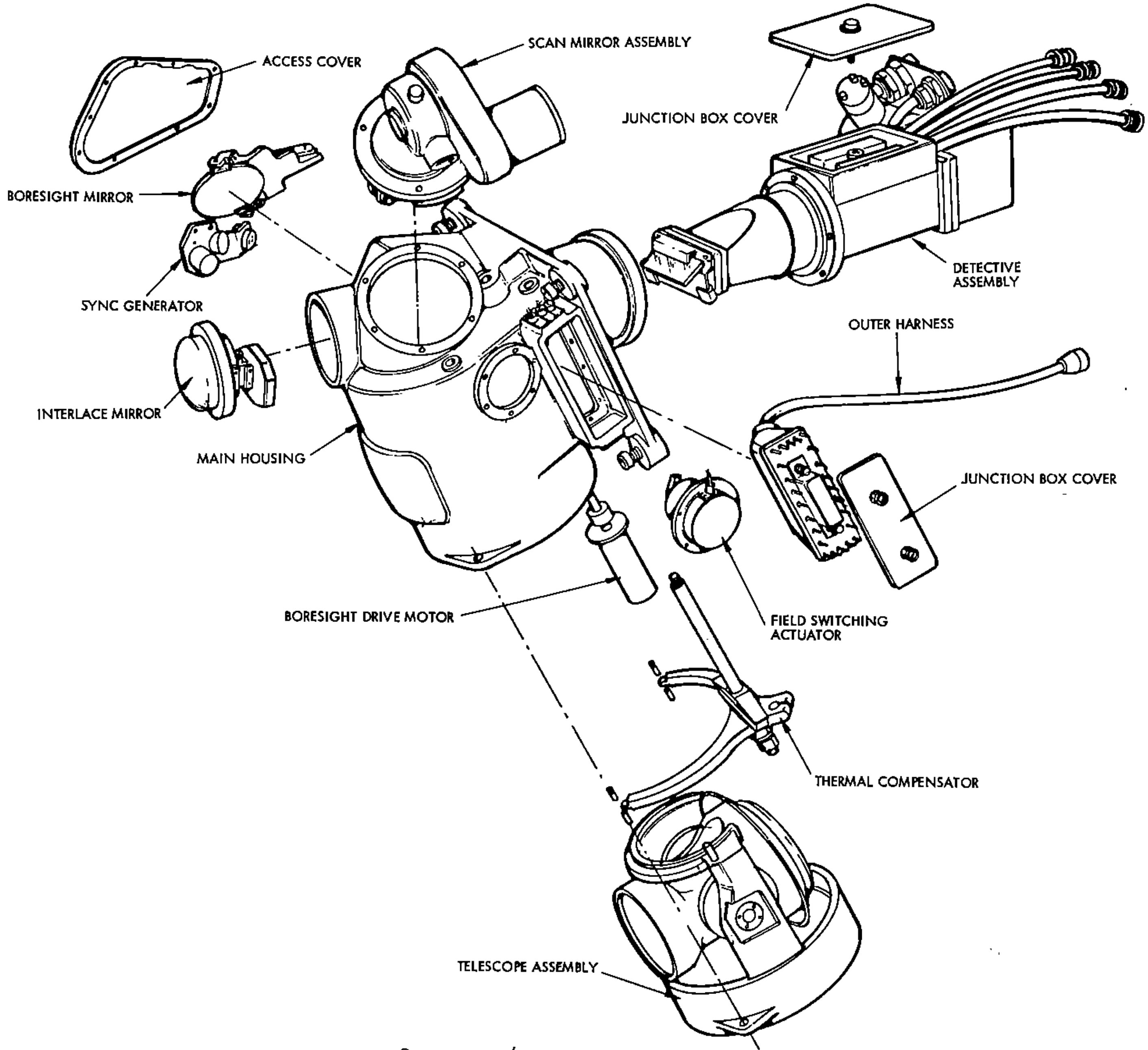


Figure 18-5. AN/AAS-25 Sensor

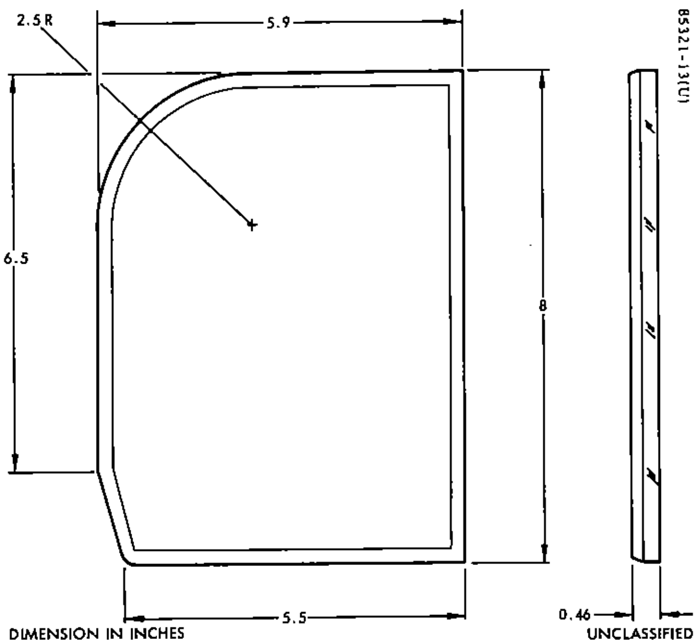
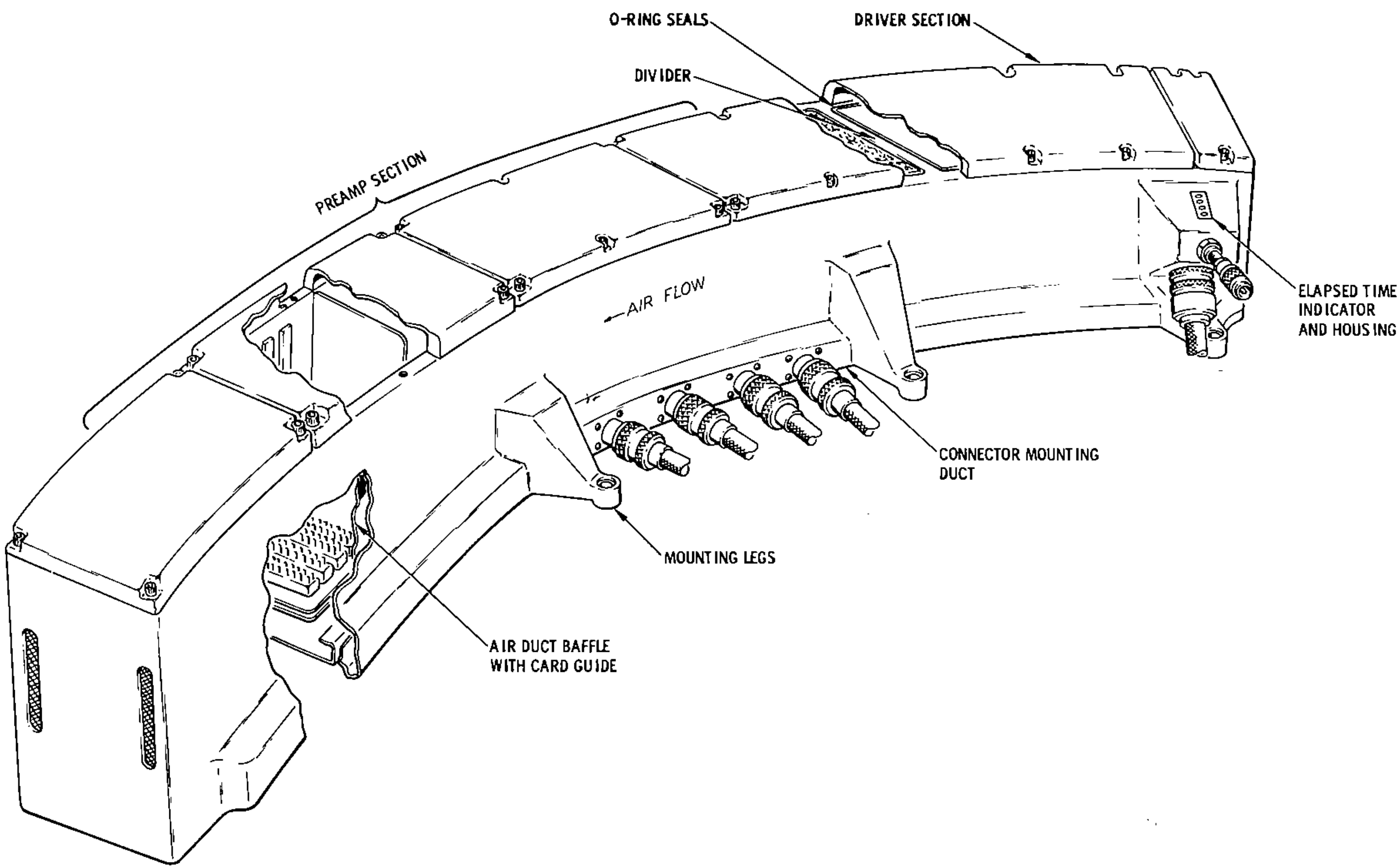


Figure 18-6. IR Window



AN/AAS-25 ( ) Signal Data Converter (Preproduction Design)

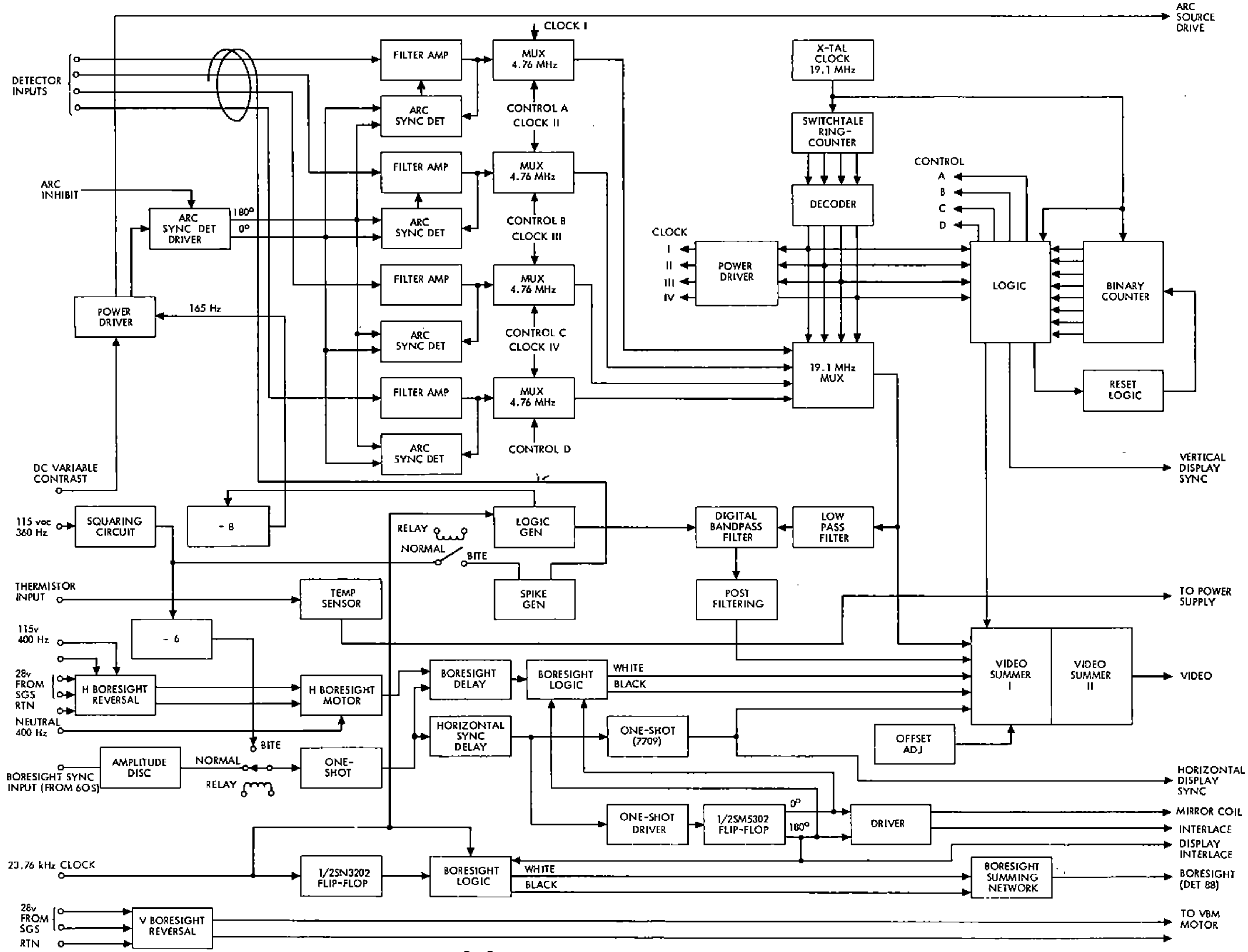


Figure 18-8. NVS Signal Data Converter

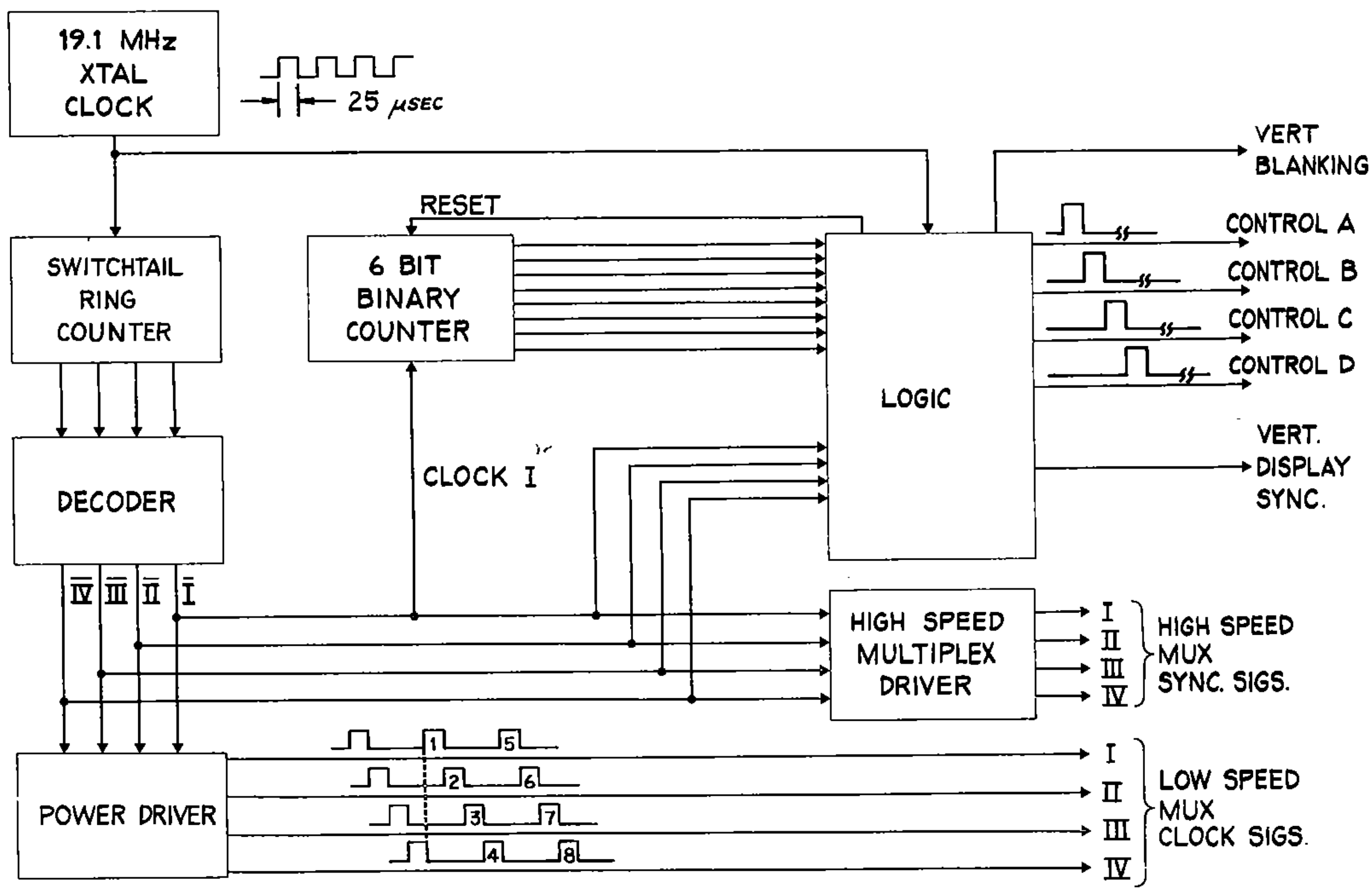


Figure 18-9. Logic

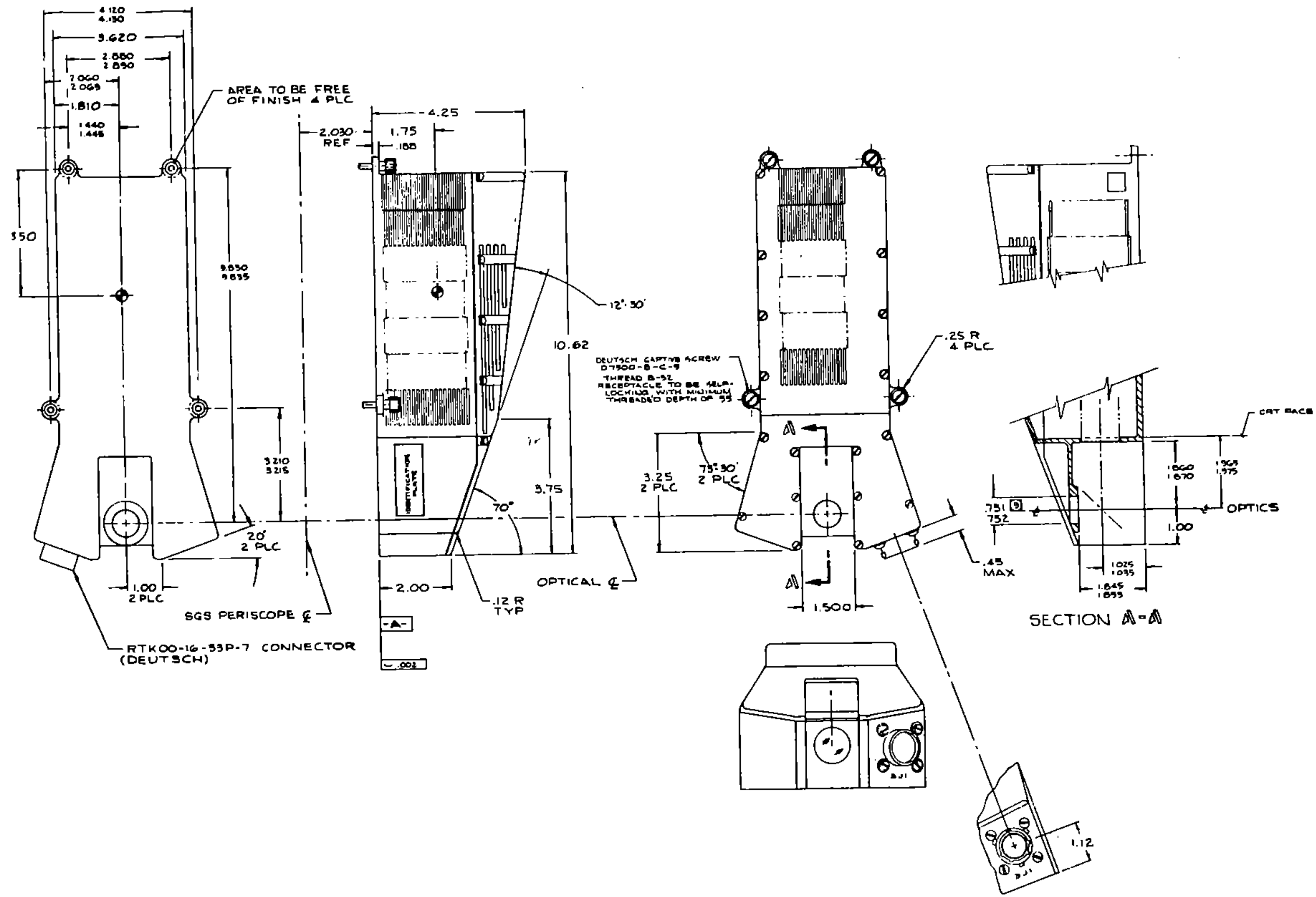


Figure 18-10. Installation Control - Display

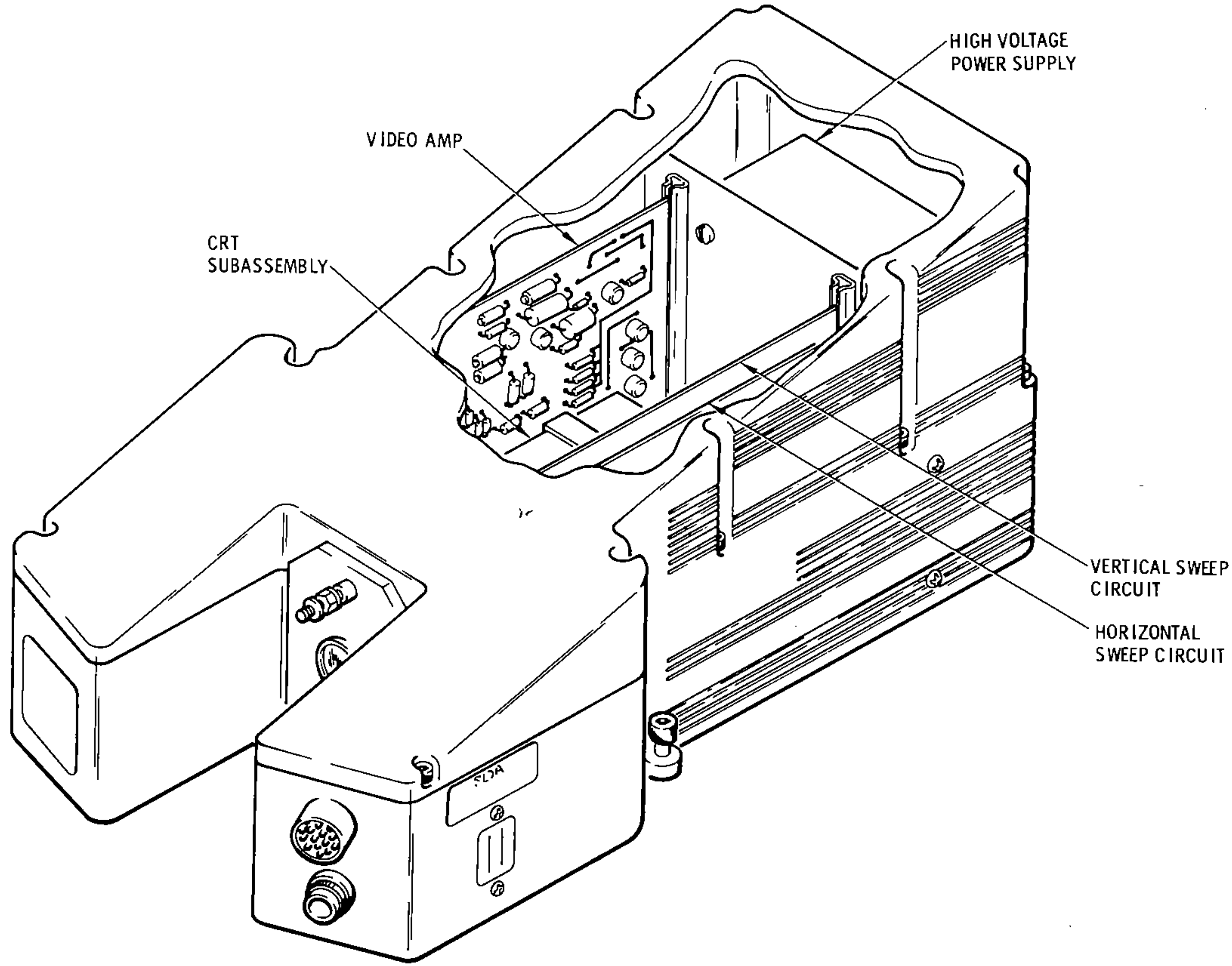
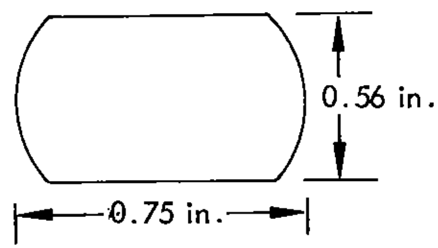


Figure 18-11. AN/AAS-25 ( ) IR Image Indicator

1. PROVIDE IMAGE OF IR SIGHT INFORMATION FOR VIEWING THROUGH SGS PERISCOPE SYSTEM

- IMAGE SIZE -



- BRIGHTNESS - $\geq$ 30 ft - lamberts
- COLOR - GREEN (WHITE-GOAL)
- RESOLUTION - 800 LINES/in AT 30 ft - lamberts
- DYNAMIC RANGE - $\geq$ 10 SHADES OF GREY
- LINEARITY - $\pm$ 10 PERCENT FROM MEDIAN SWEEP SLOPE
- DISTORTION - NONE APPARENT

Figure 18-12. Display - Functional Parameters

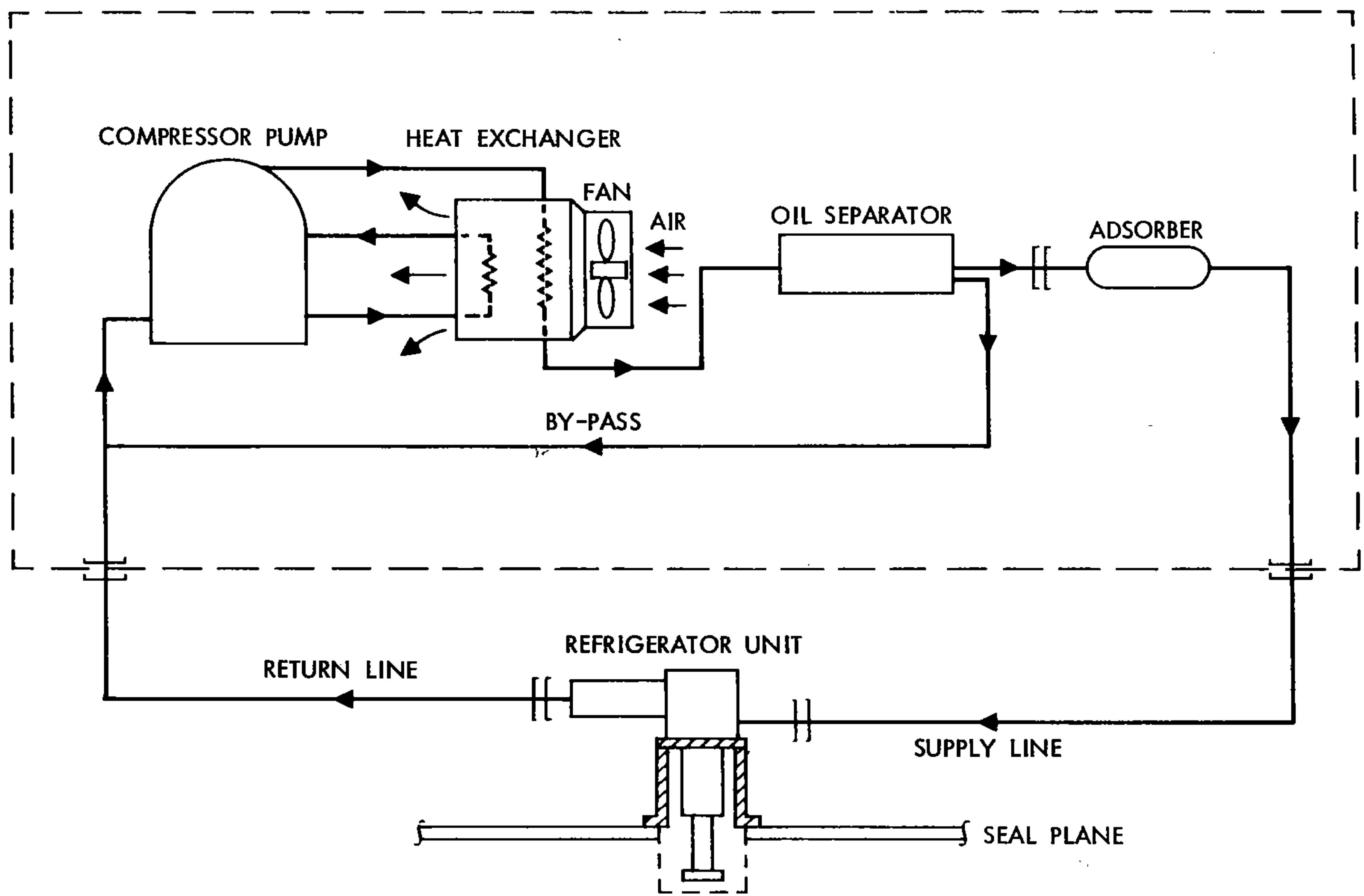
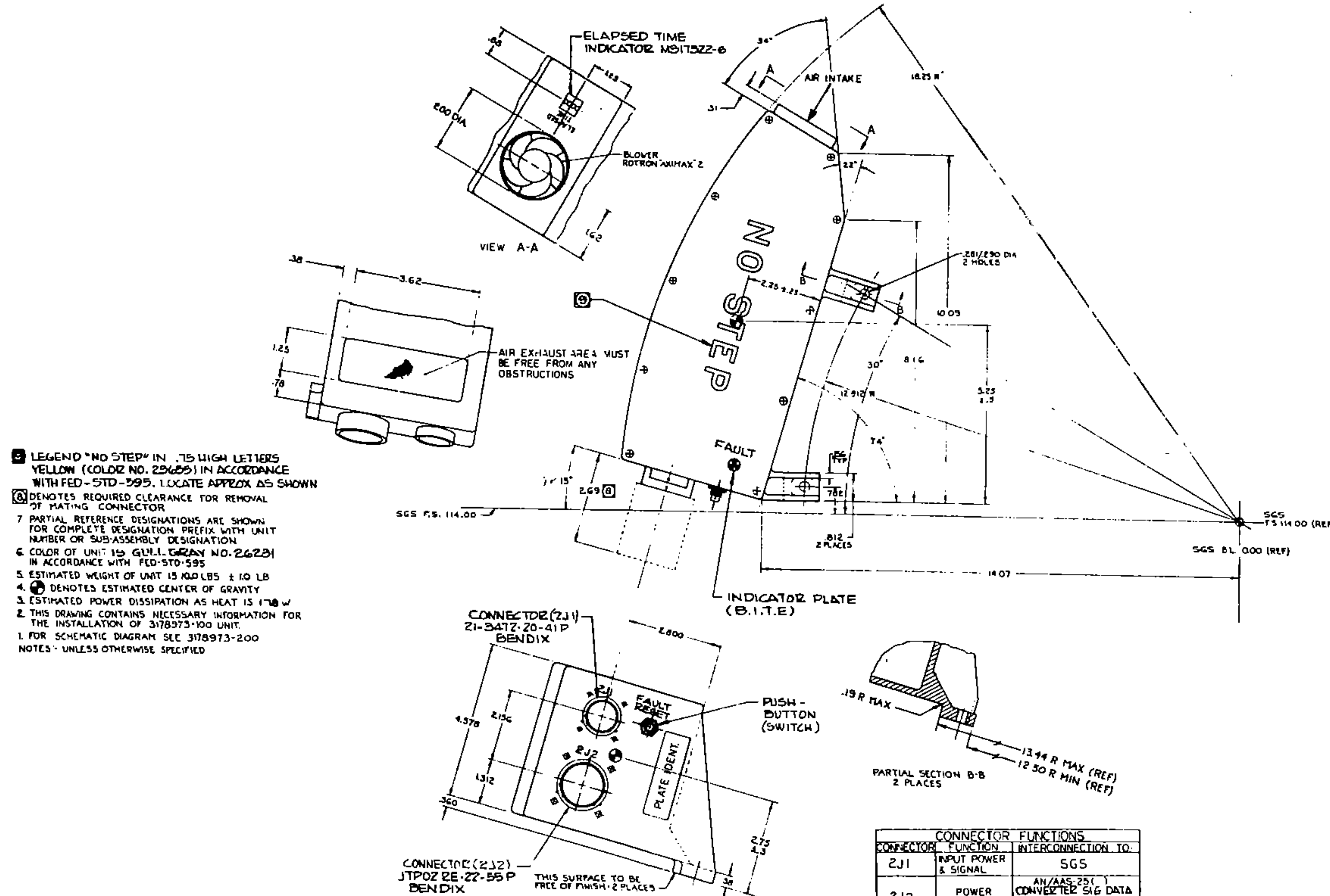


Figure 18-13. NVS Cryogenic Refrigerator Functional Diagram



CONNECTOR FUNCTIONS		
CONNECTOR	FUNCTION	INTERCONNECTION TO
2J1	INPUT POWER & SIGNAL	SGS
2J2	POWER & SIGNAL	AN/AAS-25( ) CONVERTER SIG DATA HAC P/N 3178972-100 INDICATOR I.R. IMAGE HAC P/N 3178977-100

Figure 18-14. Installation Control - Power Supply

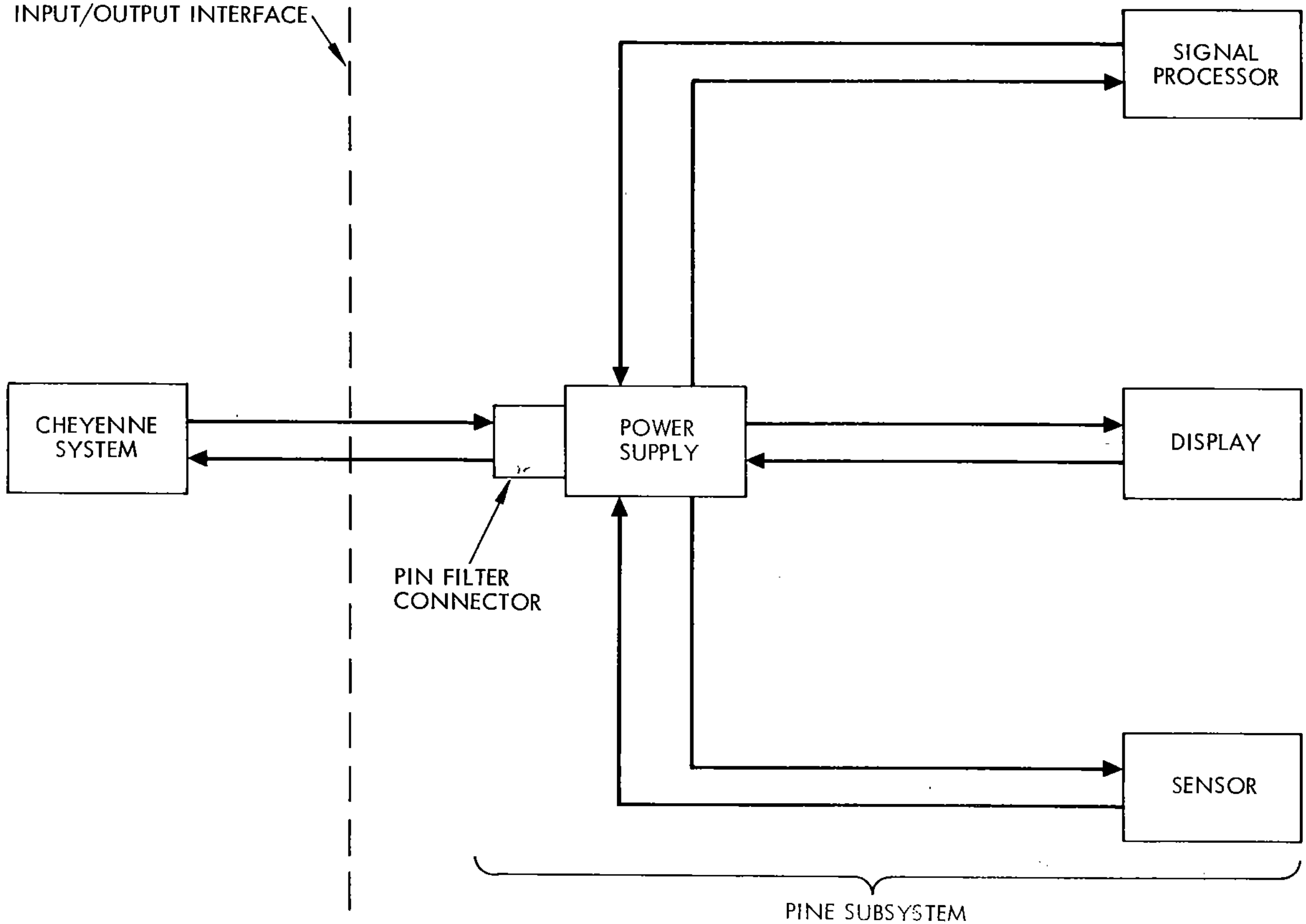


Figure 18-15. Cheyenne/Night Vision Sight Interface

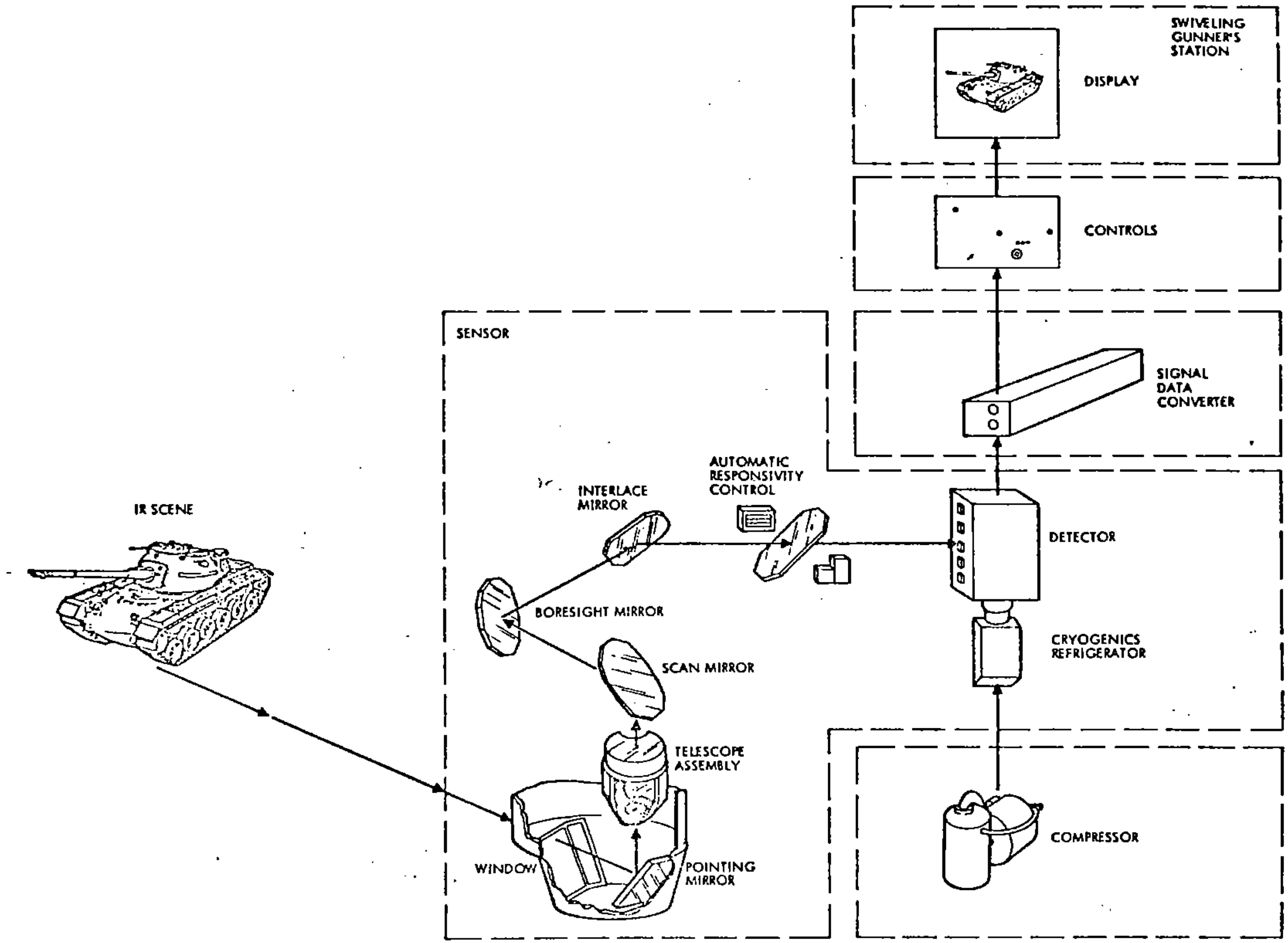


Figure 18-16. NVS Functional Diagram

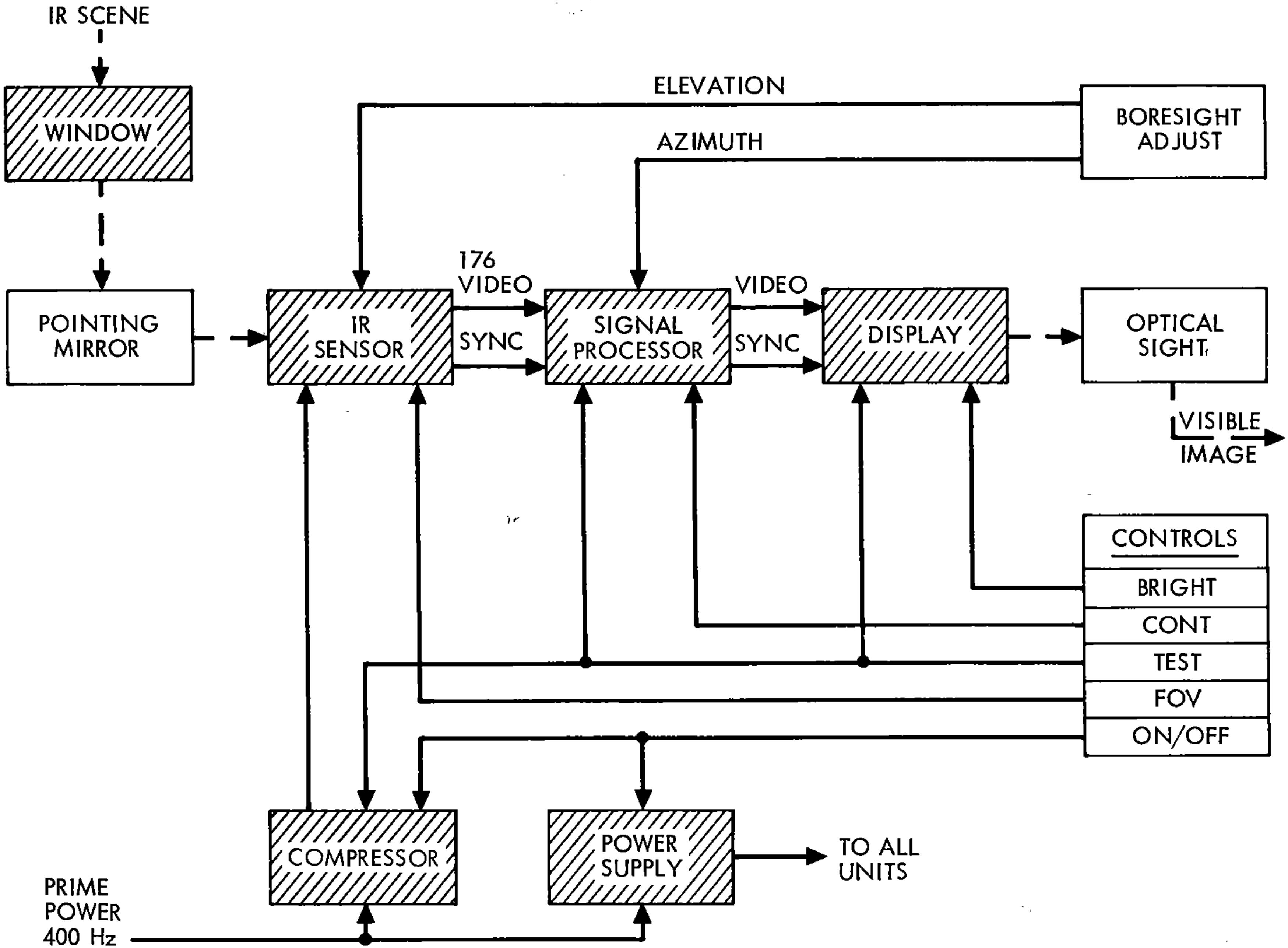


Figure 18-17. System Functional Block Diagram

TOW SYSTEM

I. GENERAL DESCRIPTION

A. TOW is an acronym derived from Tube launched, Optically tracked, Wire guided missile system. The TOW system consists of three main elements:

1. TOW Control Equipment (TCE) which is the optical-electronic system that generates the signals necessary to launch and guide the missile.

2. TOW missile consisting of:

   a. Missile

   b. Shipping container/launch tube

3. TOW three round pod

NOTE

Items 2 and 3 are described in Section VI.

II. COMPONENTS AND LOCATIONS

Component	Quantity	Location
Infrared Tracker	1	Mounted on SGS
Error Detector	1	Mounted on SGS
Signal Processing Amplifier	1	Aft Avionics Bay
Missile Command Amplifier	1	Aft Avionics Bay
Electronic Power Supply	1	Aft Avionics Bay
Remote Pylon Control	2	Leading edge of each wing next to wing pylon
Remote Armament Control	2	Located in each TOW pod

III. MAJOR COMPONENT DESCRIPTION

A. TOW Control Equipment (TCE)

Figure 19-1 is a simplified block diagram of the TOW system and it depicts the interrelationships of the TCE and their interface with the air vehicle and their associated equipment. A description of the TCE component units follows:

1. Infrared Tracker (IRT)

The IRT is hard mounted to the optical bench of the SGS. It is aligned or boresighted to the SGS LOS. This boresight is accomplished initially on the ground by means of an optical collimator. It can be updated in flight by means of an internal collimator previously referenced to the optical boresight achieved on the ground.

The IRT receives the modulated infrared signal from the missile I-R source and generates elevation and azimuth signals proportional to the angular separation between the gunner's LOS and the line of sight to the missile source. These signals are then sent to the Error Detector for further processing. The IRT has three fields of view which are progressively switched from wide to medium field of view at 1.8 seconds and to the narrow field of view at 2.79 seconds after missile first motion. The wide (12°) and the medium (3°) fields of view are optimized to contain the worst case capture phase trajectory transients resulting from launching conditions. The narrow field of view (0.5°) is optimized to provide sufficient sensitivity to detect and guide the missile accurately at its maximum operational range. Uninterrupted guidance of the missile from its capture in the field of view to target impact dictates that the missile must never leave the field of view of the IRT.

2. Error Detector (ED)

The ED processes the elevation and azimuth angular error signals received from the IRT to derive elevation and azimuth differential error voltages that are proportional to the actual physical separation of the missile from the SGS LOS. These error signals are then transmitted to the Signal Processing Amplifier (SPA) for further processing.

3. Signal Processing Amplifier (SPA)

The SPA is a multipurpose unit, for the most part serving as an interface between the raw data sources and the rest of the TOW system. Its primary function includes computing the missile guidance command utilizing the Error Detector signals, vehicle airspeed, line-of-sight rates, line-of-sight position angles and vehicle pitch and roll angles. Additionally, the SPA functions include the generation of status signals (missile fire enable, missile present, hangfire and maneuver), missile wire cut signal, missile selection provisions (manual and automatic) and BITE logic.

The manual and automatic missile select provisions requires further discussion. Missile selection is initiated by the CPG from his armament panel on the SGS. He has the capability of selecting any one of twelve missiles or to select the automatic missile firing mode. In the case of manual selection, the CPG selection signal is sent to the SPA which in turn interrogates the particular missile's location (pod, pylon) and if there is a missile present a signal is returned to SPA which in turn sends a missile present signal to the CPG armament control panel. In the automatic mode the basic procedure is the same except that missile selection is done automatically on a priority basis.

4. Missile Command Amplifier (MCA)

The missile command amplifier (MCA) contains the necessary circuitry to perform the following functions:

a. It processes the SPA guidance commands, the missile coordinate system resolved angular LOS rates, the gravity bias command and the open loop steering functions, as applicable, into the FM multiplexed guidance signal which is transmitted to the missile via the wire command link.

b. Provides timing and buffering functions to serve as a system clock.

c. Provides current to the fire, prefire, and wire cut squibs.

d. Performs BITE testing.

NOTE

The gravity bias program compensates for the effect of gravity during the missile flight.

5. Electronic Power Supply (EPS)

The EPS generates the DC and AC voltages necessary for operation of the TCE.

6. Remote Pylon Control (RPC)

There are two RPCs required; one L.H. and one R.H. They serve as switching junction boxes for routing MCA generated signals to the proper pylon station on command from the SPA selection signals.

7. Remote Armament Control (RAC)

The RAC serves as a switching junction box within the TOW pod for routing the prefire, fire, wire cut and guidance signals to the selected missile to be fired. It further functions to provide prefire, fire and wire cut signal lines isolation for the unselected missiles in the pod.

IV. TOW SYSTEM OPERATION

The following launch and guidance sequence is presented in order to provide an understanding of the TOW system operation. For a typical flight:

A. The preparation sequence prior to takeoff is: the CPG shall turn on the SGS and select the TOW missile system on his weapons panel. This will initiate the automatic TOW built in test (BIT) and will also automatically boresight the IRT to the SGS line of sight. A TOW "ready" condition will be indicated (within 85 seconds) on the gunner's weapons panel by illumination of the PRS (present) light. The BIT verifies the operational readiness of the following functions:

   1. Boresight servo electronics
   2. ED narrow field of view electronics
   3. Rate resolution
   4. Pitch and yaw open loop commands and angular constraints
   5. Error resolution and roll constraint logic
   6. Pitch and yaw error, rate and self-balance channels
   7. PItch and yaw VCOs
   8. G-bias generator
   9. Wire drivers

   The IRT is automatically boresighted to the SGS line of sight by electro/mechanically aligning it to a collimated IR source originating in the periscope optics and injected into the IRT narrow field of view (see Figure 19-2).

   After taking off and finding the target, the gunner will begin tracking the target and the pilot will select the fire control (FC) position on his mode select panel. Relative SGS azimuth and elevation LOS angles from the aircraft ADL to the target are now displayed on the pilot's ADI. The CPG will then select the missile he desires to fire. He may instead select the automatic mode of fire. The CPG and pilot will place their master arm switches in the "ON" position. While approaching the target, the CPG will track the target and the pilot will align the aircraft to the SGS line of sight. To align the aircraft, the pilot must fly to the needles on the ADI. When the aircraft is aligned to within ±2 degrees of the SGS line of sight and the vertical roll angle is less than ±5 degrees and the SGS elevation and azimuth LOS rates are less than ±30 milliradians per second, a fire enable light will illuminate. This light is located within the periscope optics and will inform the gunner that conditions have been met to provide optimum capture and hit probability. The gunner will actuate his trigger to start the TOW programmer. There is a 1.5-second delay prior to the launch motor firing. During this time several things occur within the TOW system. The missile discriminator circuits for pitch and yaw channels are self-balanced, the IR tracker is in the wide field of view, the missile gyro is spun up, the missile thermal batteries are activated, and the delta rho computer is enabled. After the 1.5-second delay, the fire command is sent to the launch motor which burns for 0.035 second. After launch, the missile is initially guided by the open loop steering commands since the IR source is not yet visible to the IR tracker. The missile is steered via commands from the missile command amplifier. The signal processing amplifier provides information to the missile command amplifier for this steering phase. These units working together utilize information of the pod position from which the missile is fired, aircraft airspeed, aircraft roll attitude, line-of-sight position, and line-of-sight rate, in order to compute the commands necessary for the missile to fly into the IR tracker field of view. 0.29 seconds after launch, the missile flight motor is ignited and burns for approximately 1 second. Missile capture occurs approximately 0.45 second after launch. Closed-loop steering is initiated shortly thereafter.

Approximately 1.8 seconds after launch, the field of view of the IR tracker is switched from the wide field of view to the medium field of view. Approximately 2.8 seconds after launch the IR tracker is switched to the narrow field of view. Figure 19-3 shows the capture geometry. The missile is guided by closed-loop steering after the missile beacon on the tail of the missile is detected by the IR tracker. The IR tracker raw error signal is sent to the error detector which processes the information and sends it to the signal processing amplifier. This amplifier, utilizing this error signal and outside sources of information converts this raw error into missile coordinates. These coordinate signals are sent to the missile command amplifier which sums these signals with line-of-sight rates and a "G" bias signal. This signal is limited to prevent the missile from flying out of the line of sight of the IR tracker at short ranges. The pitch and yaw commands are then multiplexed and sent, via the two wires to the missile, steering it in the proper direction to bring the missile to the LOS of the SGS. At the time of impact, guidance is lost and this loss initiates a "wire cut" signal which is sent to the empty launch tube to ignite the "wire cut" squibs. At approximately 18.5 seconds after launch a programmer reset signal is sent to the SPA. Resetting the programmer permits a second missile to be launched. In addition to the above sequence of events, the pilot may maneuver the aircraft immediately after the missile leaves the launch tube. This will permit him to take evasive action in case of return fire.

B. Emergency/Redundant Operations

The TOW system contains no provisions for emergency or redundant operations with the exception that if the automatic missile select mode fails there is a backup manual select mode.

C. Fault Indications

The TOW system performs an automatic BIT each time the system is selected. A "GO" condition will be indicated by illumination of the "PRS" light on the gunner weapons panel within 85 seconds from missile system selection. Failure of this light to illuminate within this time indicates a "NO GO" condition. The "NO GO" condition will be indicated on the FLAWS panel as a "MISSILE" system failure.

V. PCRS CONFIGURATION

The TOW control equipment has been relocated to the main avionics bay. This resulted in a savings of 26 pounds. The space was made available by the elimination of other avionics equipment.

VI. SUPPLEMENTAL DESCRIPTION

A. TOW Missile

The TOW missile is depicted in Figure 19-4. The major components are described below.

1. Airframe

The airframe consists of three aluminum sections joined together, the two most rearward sections being cylindrical in shape while the forward section is a truncated cone transitioning from the rear sections to mate with the warhead assembly. Suitable structures are provided internally to attach the wings, control surfaces and to mount the required internal assemblies. There are four wings, in cruciform arrangement, attached to the airframe at approximately the midpoint of the missile. These wings are folded flush into the airframe during the time the missile is in the launch tube. When the missile exits the tube at launch, the wings are spring loaded out to the in-flight position. The wings are not movable in flight. Aerodynamic missile control is exercised by four movable control surfaces, also in cruciform arrangement but rotated 45 degrees relative to the wing positions, which are mounted at the rear of the missile. These surfaces are also folded flush with the airframe while the missile is in the launch tube. The inboard extremeties of the control surfaces are attached to the control surface actuators which furnish the motive power for their movement.

2. Control Surface Actuators

The four individual control surface actuators are the motive power source for moving the control surfaces. The control surfaces are operated in the "bang-bang" mode with differential control being accomplished by modulating the dwell time associated with each end of the control surface movement under missile gyro and TCE command signal inputs. In this manner, the flight attitude of the missile in pitch, roll and yaw is controlled. The actuators are pneumatic pistons powered by high pressure helium gas. The gas is contained in a pressure vessel which is opened by an explosively actuated cutter at missile launch. By means of an aluminum manifold the gas flows first to a pressure reducing valve and thence to the solenoid operated control valve for each actuator. The control valve meters the flow of gas to the appropriate side of the piston in response to the command signal inputs.

3. Warhead Assembly

The warhead assembly consists of the basic shaped charge warhead and the safeing and arming assembly. The warhead assembly forms the nose section of the TOW missile and is attached to the truncated cone section of the airframe. The safeing and arming mechanism controls the enabling of the fuse for the warhead. It is designed to render the warhead safe for transport and ground handling. At launch the system assures that adequate separation from the vehicle is present before arming the warhead.

4. Launch Motor

The launch motor is a high impulse solid propellant, single nozzle rocket motor which is mounted in the aft portion of the airframe. Its purpose is twofold; first, to provide sufficient launch velocity to enable the missile to be aerodynamically controllable up to the time of flight motor ignition and secondly, to achieve burnout before the missile exits from the launch tube so as to prevent rocket blast impingement on the vehicle. The launch motor is ignited by electrical squibs on system command.

5. Flight Motor

The flight motor ignites after the missile has cleared the aircraft and accelerates the missile to its maximum coast velocity. It is a solid propellant rocket motor mounted in the midsection of the missile. This necessitates the canting of the two exhaust nozzles so as to exhaust on either side of the missile. Two nozzles are required to prevent asymmetric thrust.

6. Wire Command Link

The wire command link is a two-wire system of single strand high strength steel wire with the outgoing ends anchored to the missile launch tube. At missile launch this wire is payed out from the missile as required from two small diameter spools mounted in the aft end of the missile. The wire command link provides the patch for multiplexed pitch and yaw commands from the TCE to the launcher.

7. Gyro

The gyro is a single two-axis displacement gyro used to roll stabilize the missile and to reduce the effects of crosswinds on missile accuracy at short range. Electrical pickoffs provide requisite signals to the missile electronics. It is mounted immediately aft of the flight motor and is driven by compressed gas stored in a pressure vessel adjacent to it. It is activated at the time the fire signal is received by the missile.

8. Missile Electronics Assembly and Power Supply

The missile electronics assembly, together with the thermal batteries comprising the power supply, perform the necessary "on missile" electronic functions during missile flight. The missile electronics assembly is a torus shaped package installed immediately aft of the warhead. The three thermal batteries, which are initiated on the fire signal, are mounted adjacent to the gyro. The missile electronic assembly receives missile pitch and yaw steering signals from the TCE through the wire command link. It also receives missile roll and yaw signals from the missile gyro. These signals are processed and the necessary voltage signals are applied to the solenoid of the control valve of the control surface actuators to achieve missile compliance with the commands.

9. I-R Beacon Assembly

The I-R beacon assembly consists of an I-R source plus its associated motor driven modulator disc. It is mounted in the rear of the missile so that its modulated I-R emissions can be detected by the aircraft mounted (SGS) I-R detector.

B. Shipping Container and Launch Tube

The combination launch tube and shipping container stays with the missile from the time of manufacture to the time of launch. The container acts as a launcher tube when it is installed in the pod mounted on the wing pylon on the AH-56. The launch container provides the protective enclosure for the missile while the container has been removed from the overpack for issue and while the container has been installed in the launcher and not yet fired. The container is fabricated from fiberglass and is cylindrical. It contains electrical connection points to attach the "on aircraft" equipment to the missile. It is sealed to prevent dust and moisture from damaging the missile. The container also contains a 100"g" - 20"g" holdback assembly. Its function is to lock the missile longitudinally in position in the launcher tube, in 100"g" position during shipment. When installed in the launcher the assembly is mechanically depressed and held in the 20"g" position for launch.

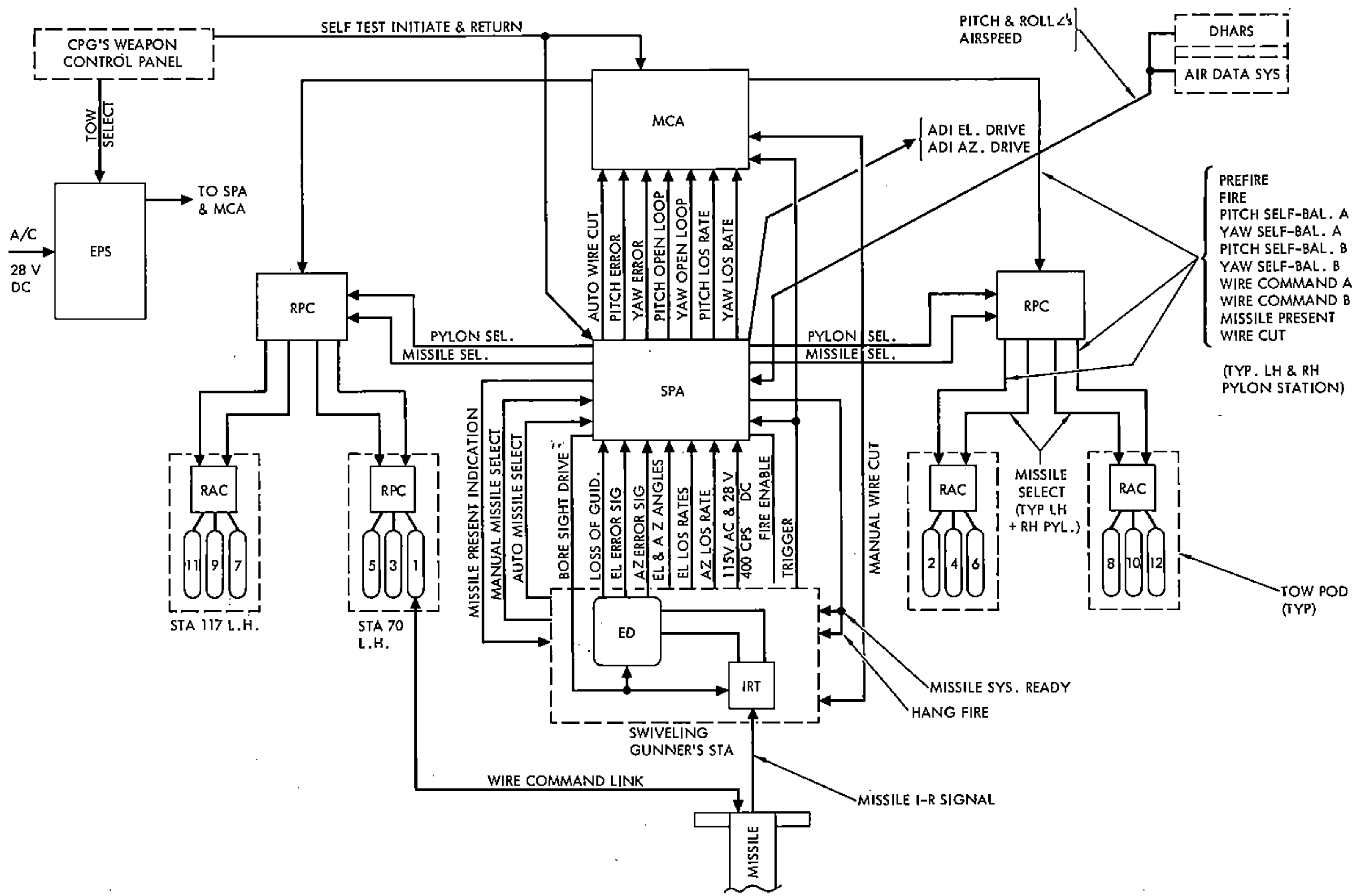


Figure 19-1. TOW Simplified Block Diagram

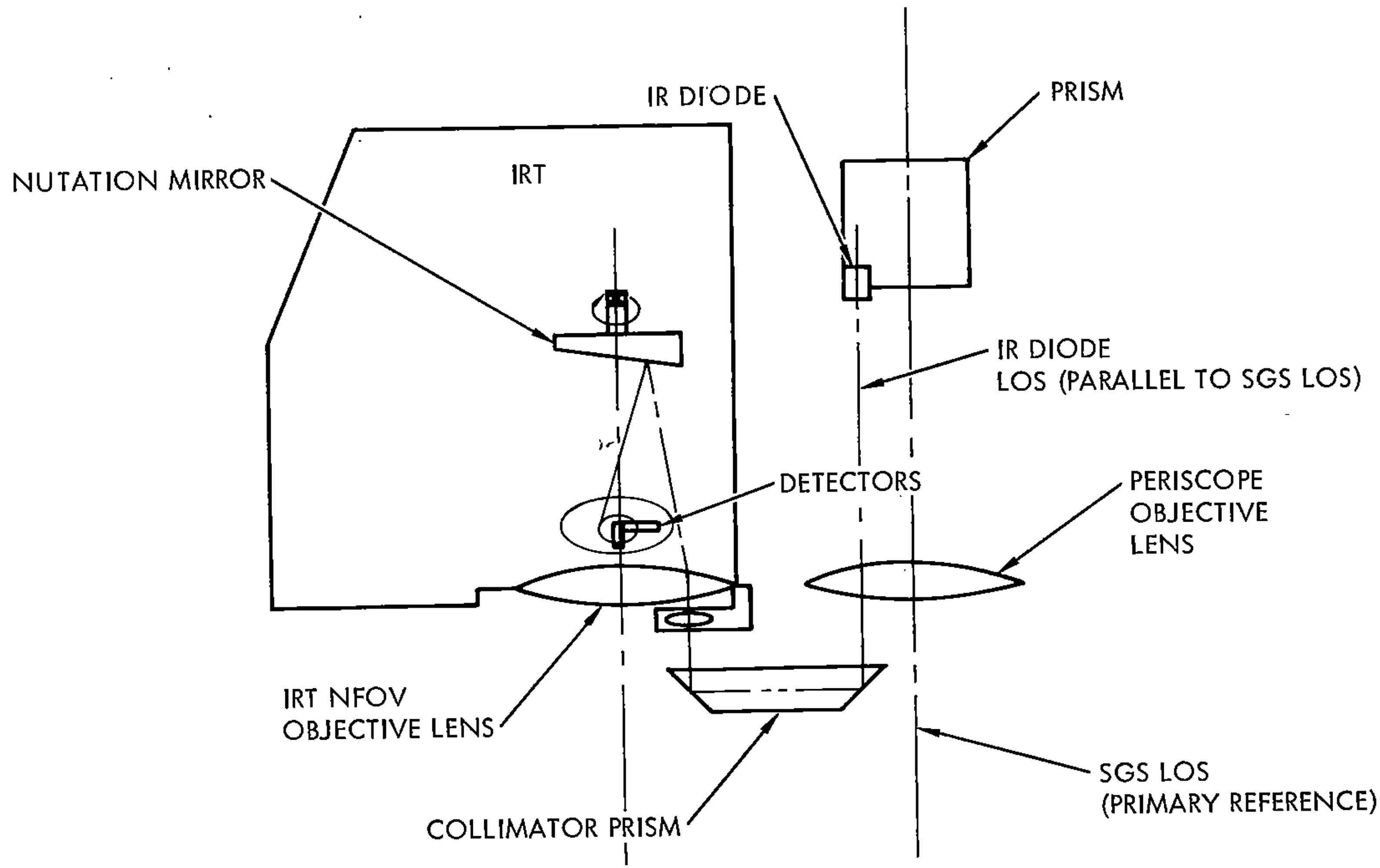


Figure 19-2. IRT Boresight

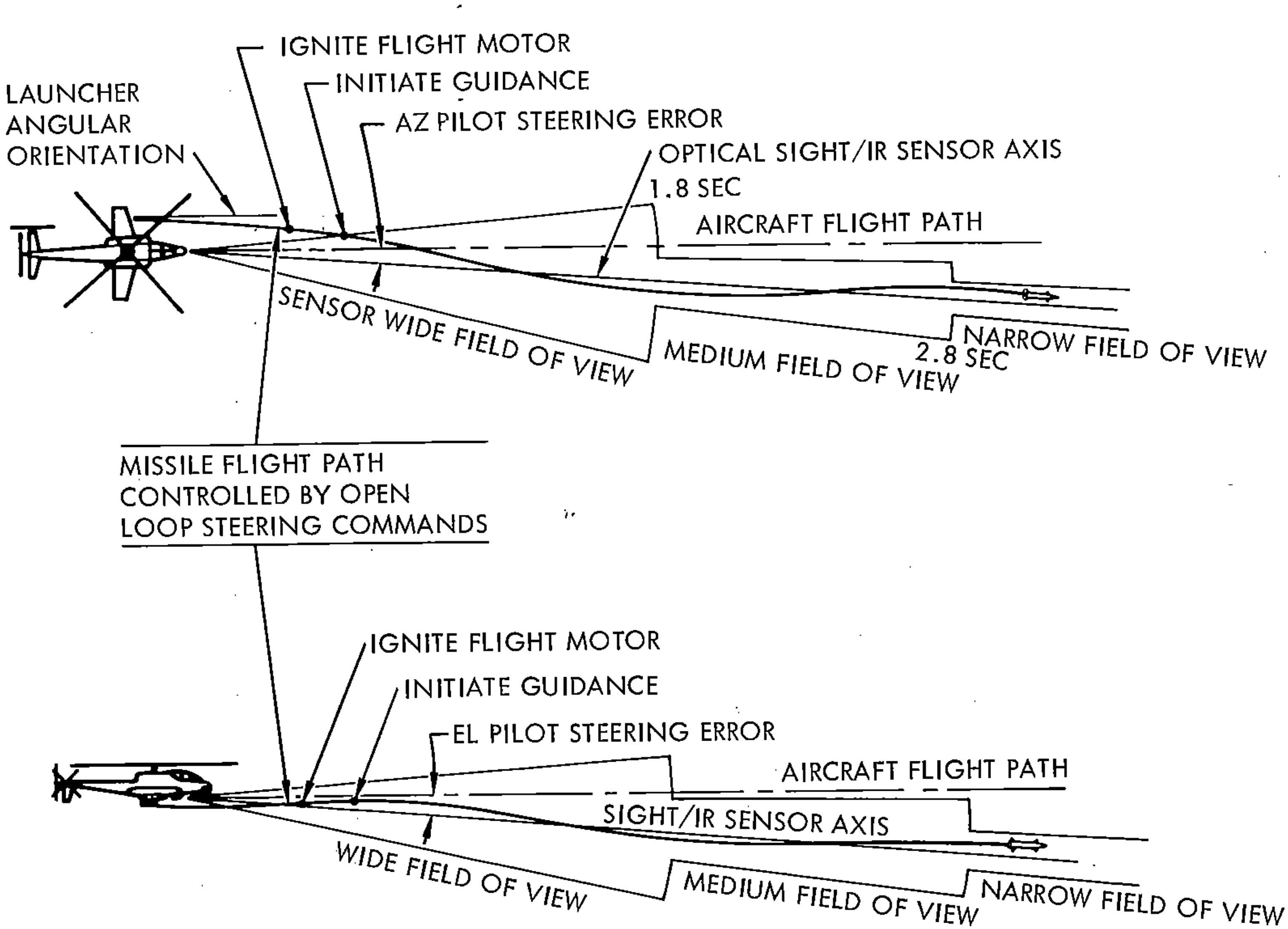


Figure 19-3. Capture Geometry (Not Scaled)

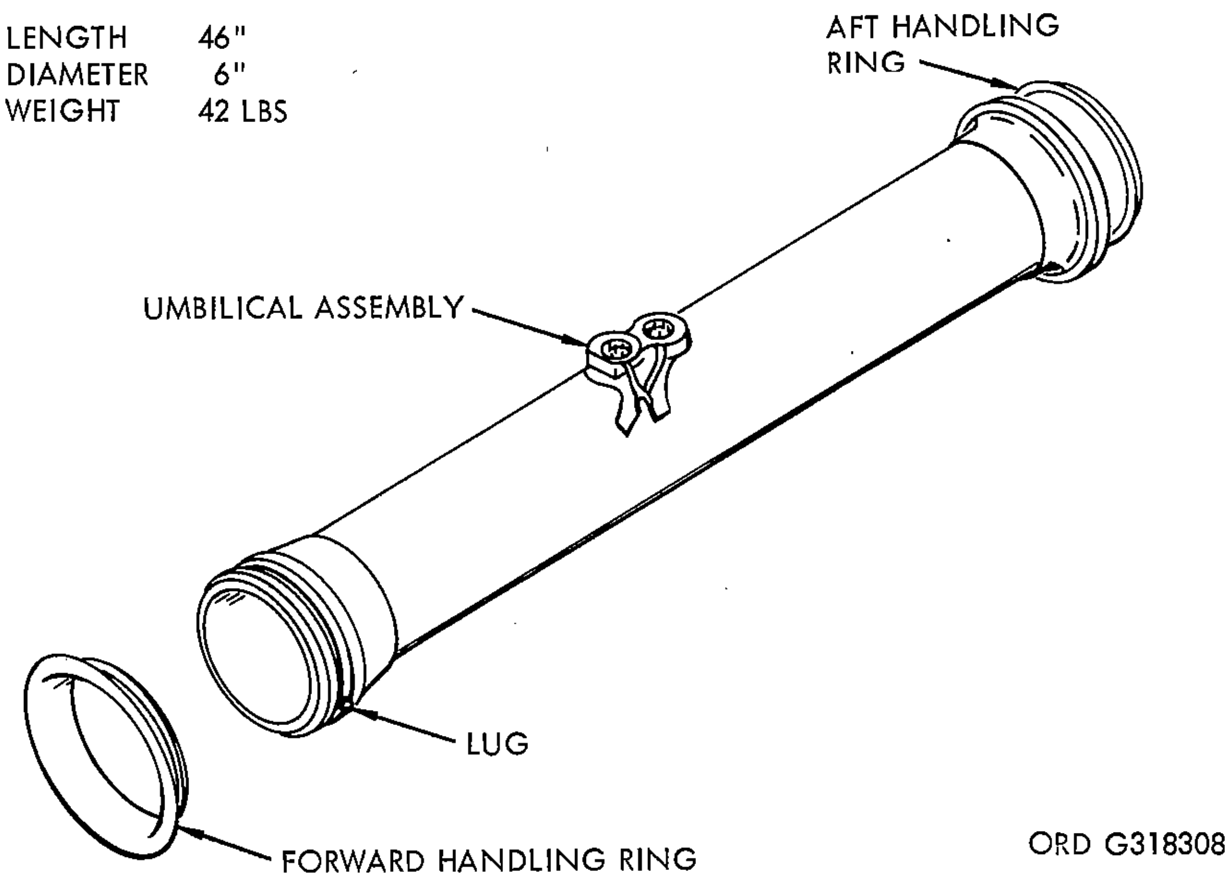


Figure 19-4. Launch Container

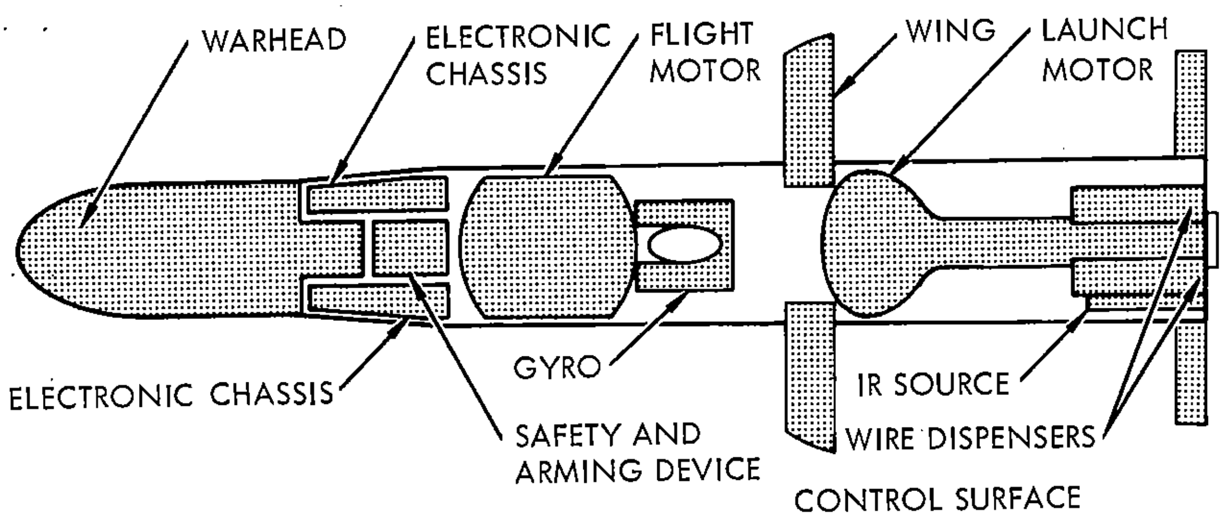


Figure 19-5. TOW Missile

ARMAMENT SYSTEMS

I. INTRODUCTION

A. The function of the integrated Armament and Fire Control systems of the AH-56A is to enable accurate delivery of the internal and external armament stores carried by the helicopter. There are four armament systems which provide three types of coverage which are described in the following sections:

1. Suppressive Fire Systems. (Nose Turret Systems).
   Section 20A - XM-51 40mm grenade launcher system
   Section 20B - XM-53 7.62mm machine gun

2. Light Point Fire System
   Section 20C - XM-52 30mm automatic gun

3. Heavy Point Fire System
   Section 20D External Stores System

LOCKHEED
CALIFORNIA COMPANY

XM-51 40 mm GRENADE LAUNCHER

I. GENERAL DESCRIPTION

The XM-51 (40 mm Grenade Launcher) armament subsystem is an electrically operated gun system that provides wide angular coverage and rapid fire. When installed on AH-56A aircraft this subsystem is used primarily as an offensive weapon against area targets. Operation of the XM-51 system is completely automatic. Either the pilot or the copilot/gunner can sight and fire the gun by actuating the proper control switches on their Weapons Panels and squeezing the trigger.

II. COMPONENT LOCATION: (See figure 20A-1)

Name of Component	Number per Aircraft	Location in Aircraft
A. Turret Assembly	1	Forward lower side of aircraft nose
B. XM-129 Grenade Launcher	1	Mounted in turret elevation gimbal
C. Ammunition Storage Drum	1	In forward section of ammo bay, accessible through door in bottom of fuselage.
D. Drive System	1	Various parts of system
1. Ammunition Feed System-	1	Between turret and ammunition storage drum along right hand side of fuselage
E. Turret Control Module	1	Mounted on rack in aft end of debris bay

1 40-MM GRENADE LAUNCHER
2 FORWARD RIGID AMMUNITION CHUTE
3 MAGAZINE ASSEMBLY
4 AFT FLEXIBLE DRIVE SHAFT
5 AFT FLEXIBLE AMMUNITION CHUTE
6 CONTROL MODULE
7 AIRCRAFT FIXED DRIVE SHAFT
8 AIRCRAFT FIXED AMMUNITION CHUTE
9 CARTRIDGE EJECTOR
10 XM-51 NOSE TURRET ASSEMBLY
11 FORWARD FLEXIBLE AMMUNITION CHUTE
12 VERTICAL RIGID AMMUNITION CHUTE
13 CARTRIDGE DRIVE ASSEMBLY
14 MODE TRANSFER ASSEMBLY
15 FORWARD FLEXIBLE DRIVE SHAFT

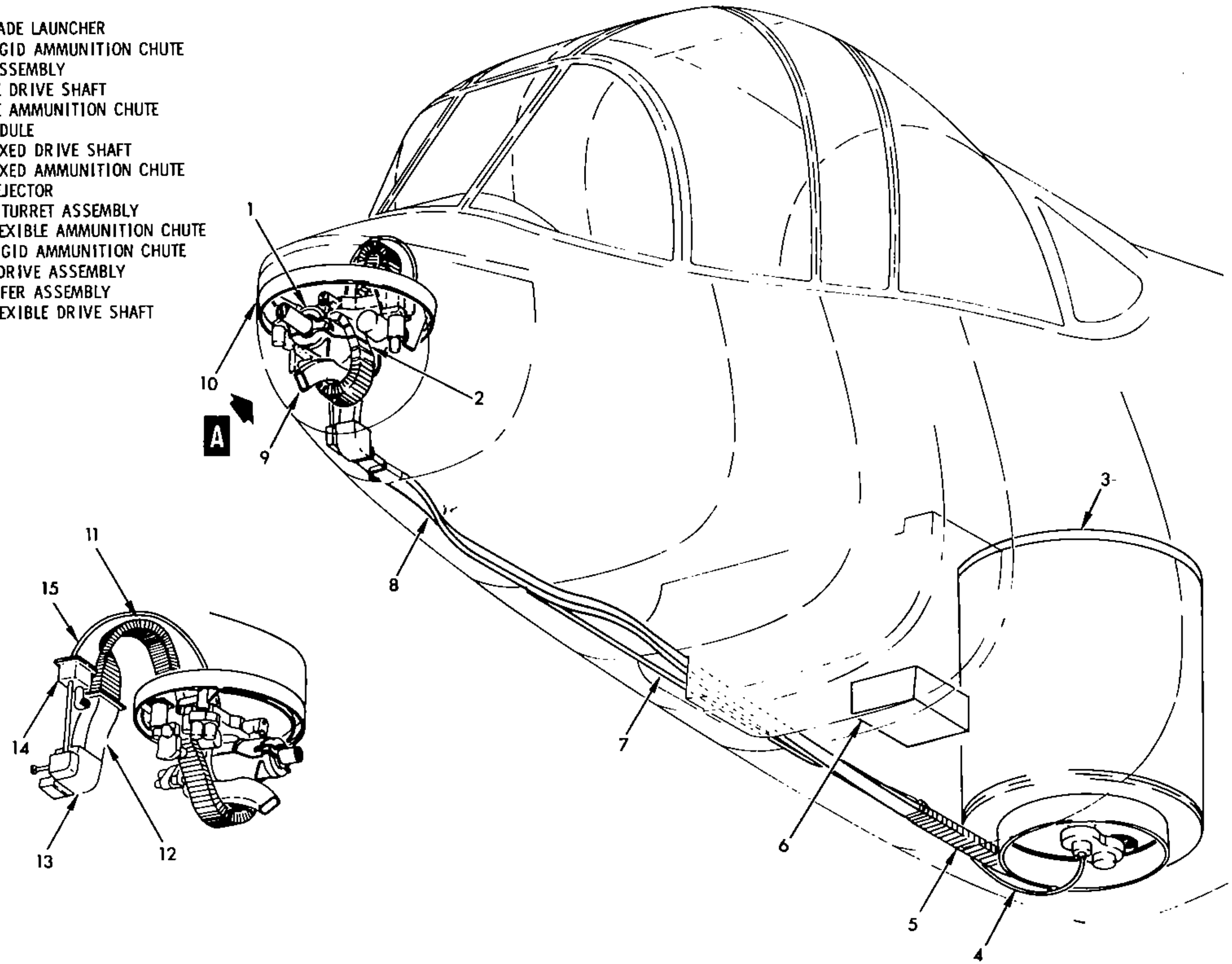


Figure 20A-1. XM-51 System Component Location

III. MAJOR COMPONENT DESCRIPTION

A. Turret Assembly (See figure 20A-2)

The nose turret is located just forward and below the copilot/ gunner's station and is housed inside an aerodynamically designed turret fairing. The turret assembly consists of a turret fairing, gun fairing, XM-129 gun (40 mm grenade launcher), azimuth and elevation gimbals, azimuth and elevation drive motors, stow locks, and instrument drive assemblies. The turret is capable of 100 degrees left and right rotation from the stowed position, and 18 degrees elevation and 70 degrees depression from the stowed position.

B. XM-129 Gun - 40 mm Grenade Launcher (See figure 20A-3)

The 40 mm grenade launcher is an externally powered, rapid firing, air-cooled weapon designed to launch anti-personnel fragmentation type projectiles. It is driven through a series of rigid and flexible drive shafts, from an electric motor. The motor drives the weapon through the entire operational cycle of feeding, chambering, cocking, locking, firing, unlocking, extraction and ejection. The weapon is of the fixed breech reciprocating barrel design which is driven by a circular drum cam concentric to the barrel. The ammunition is belt-fed and percussion fired.

C. Ammunition Storage Magazine (See figure 20A-4):

The magazine drum assembly is a cylindrical structure that measures approximately 30 inches in diameter and 37 inches high and is mounted vertically in the magazine bay. Access is through two hinged access doors in the bottom of the fuselage. The magazine holds 680 rounds of linked 40 mm ammunition. The rounds are stored in a continuous belt within the magazine drum.

D. XM-129 Gun Drive and Ammunition Feed Mechanism (See figure 20A-5)

The XM-129 gun and the ammunition feed system are operated by an electric motor powered drive mechanism, which is installed in the

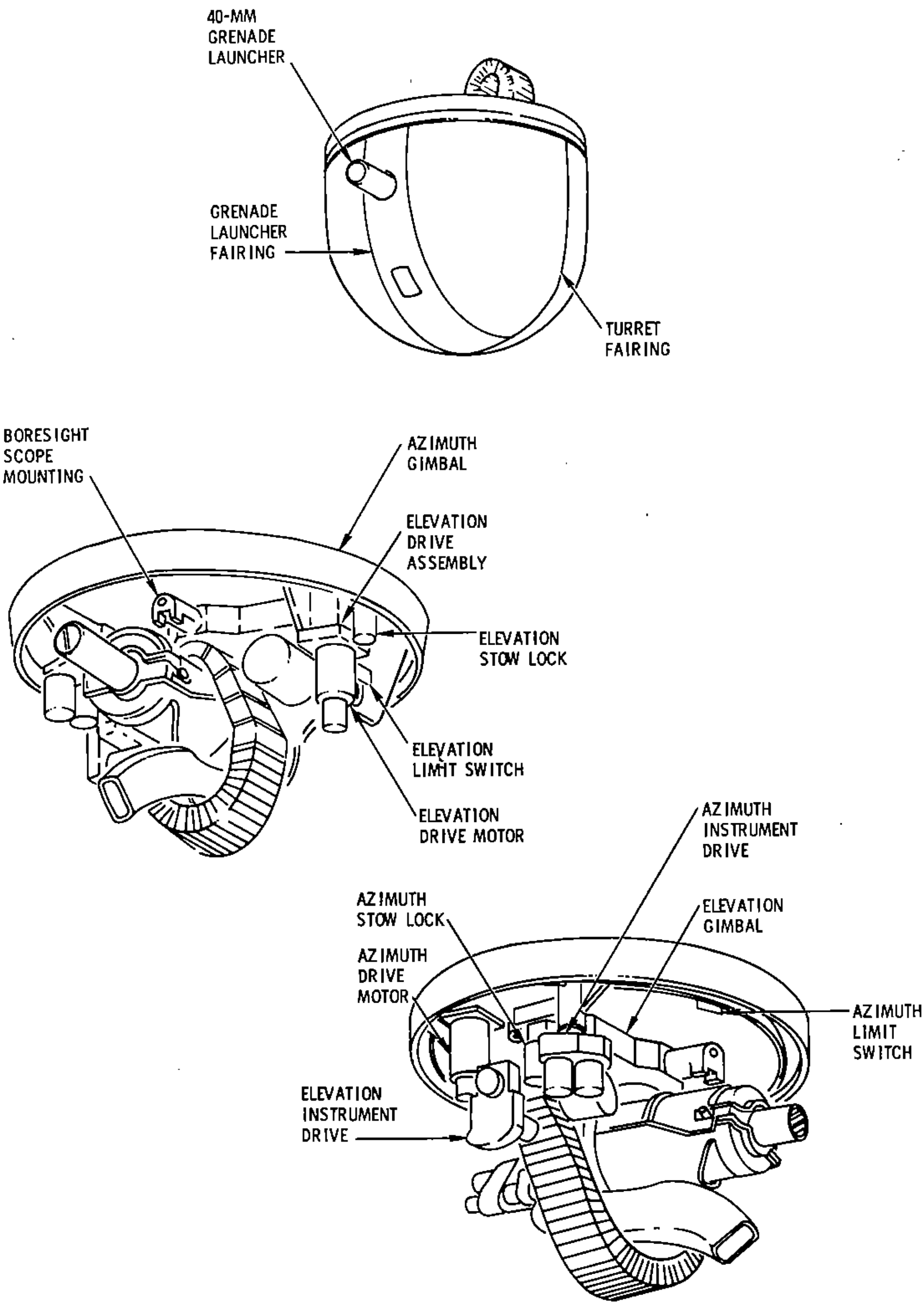


Figure 20A-2. XM-51 Turret Assembly

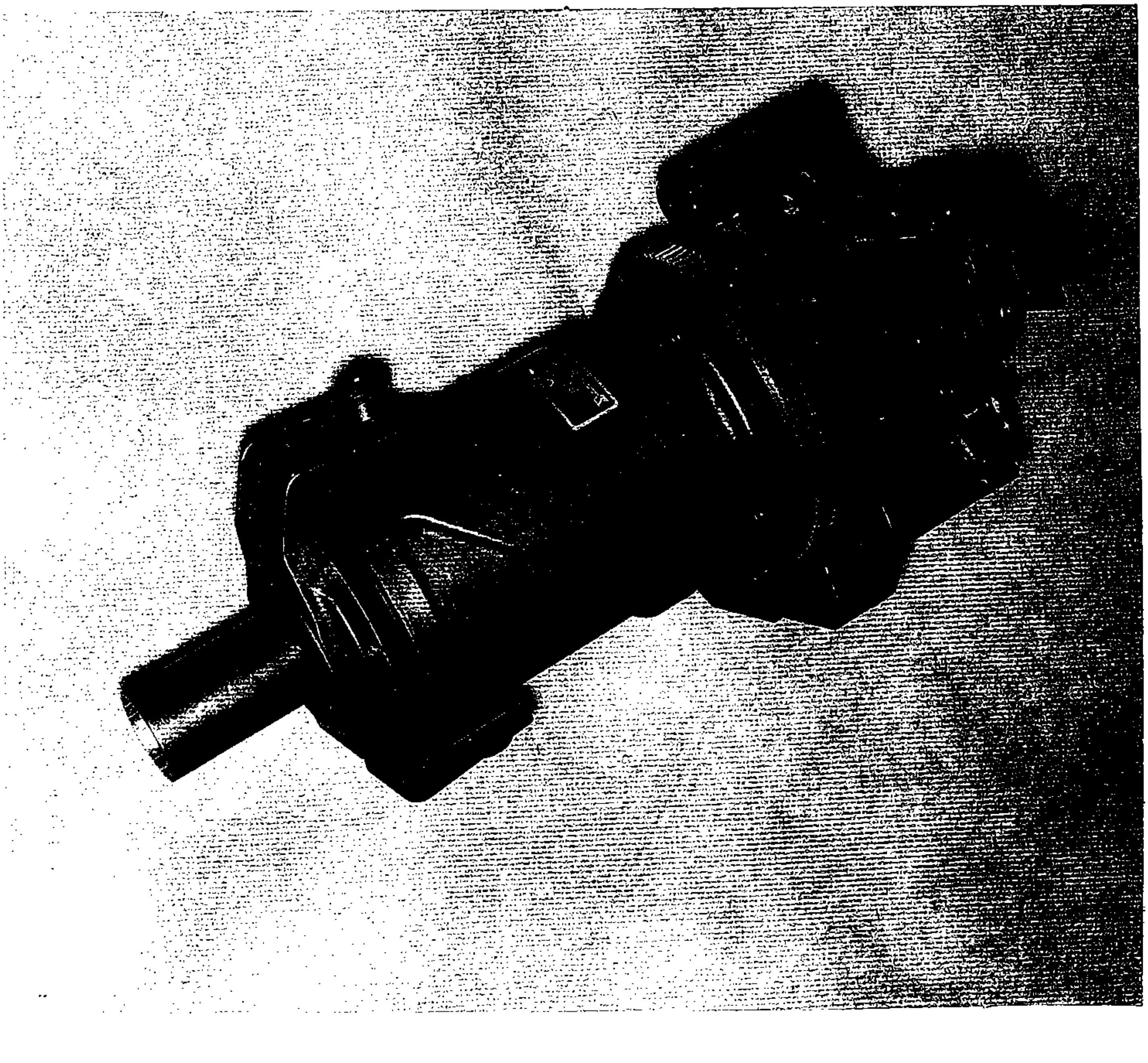


Figure 20A-3. XM-129 Gun (40 mm Grenade Launcher)

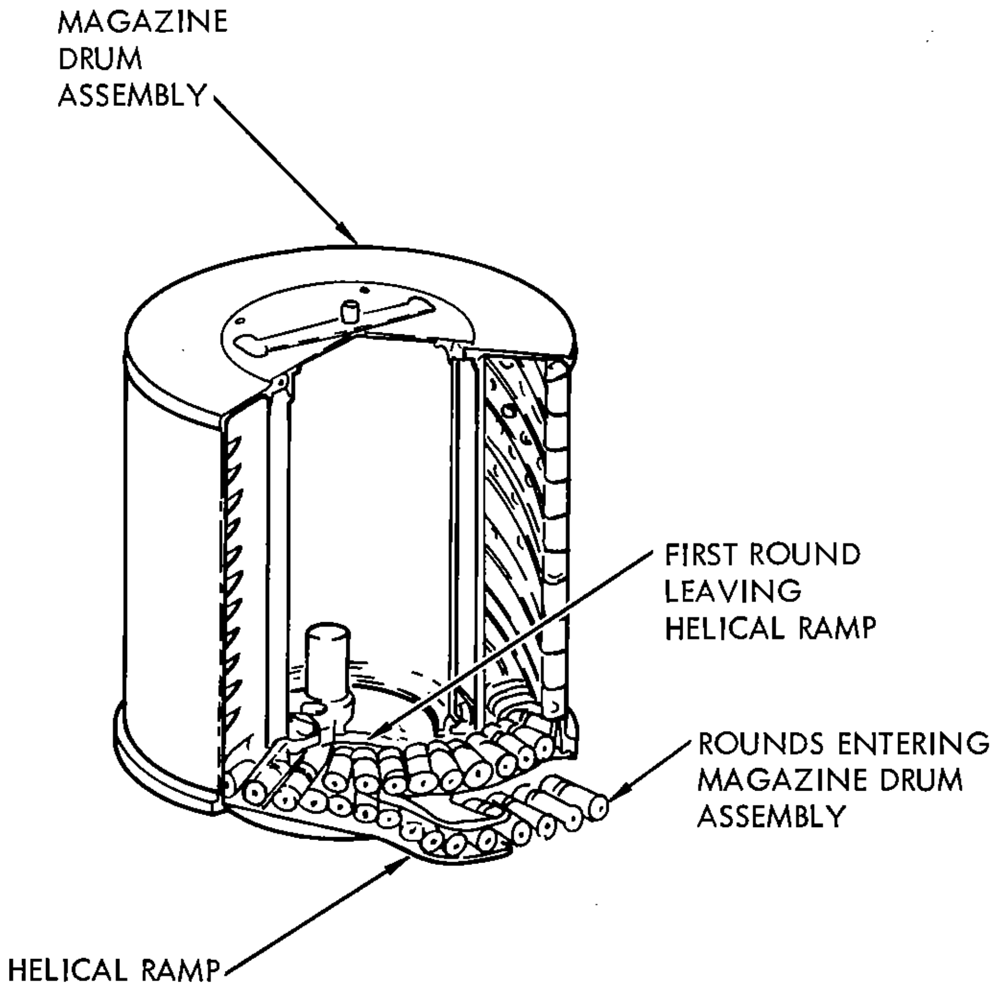
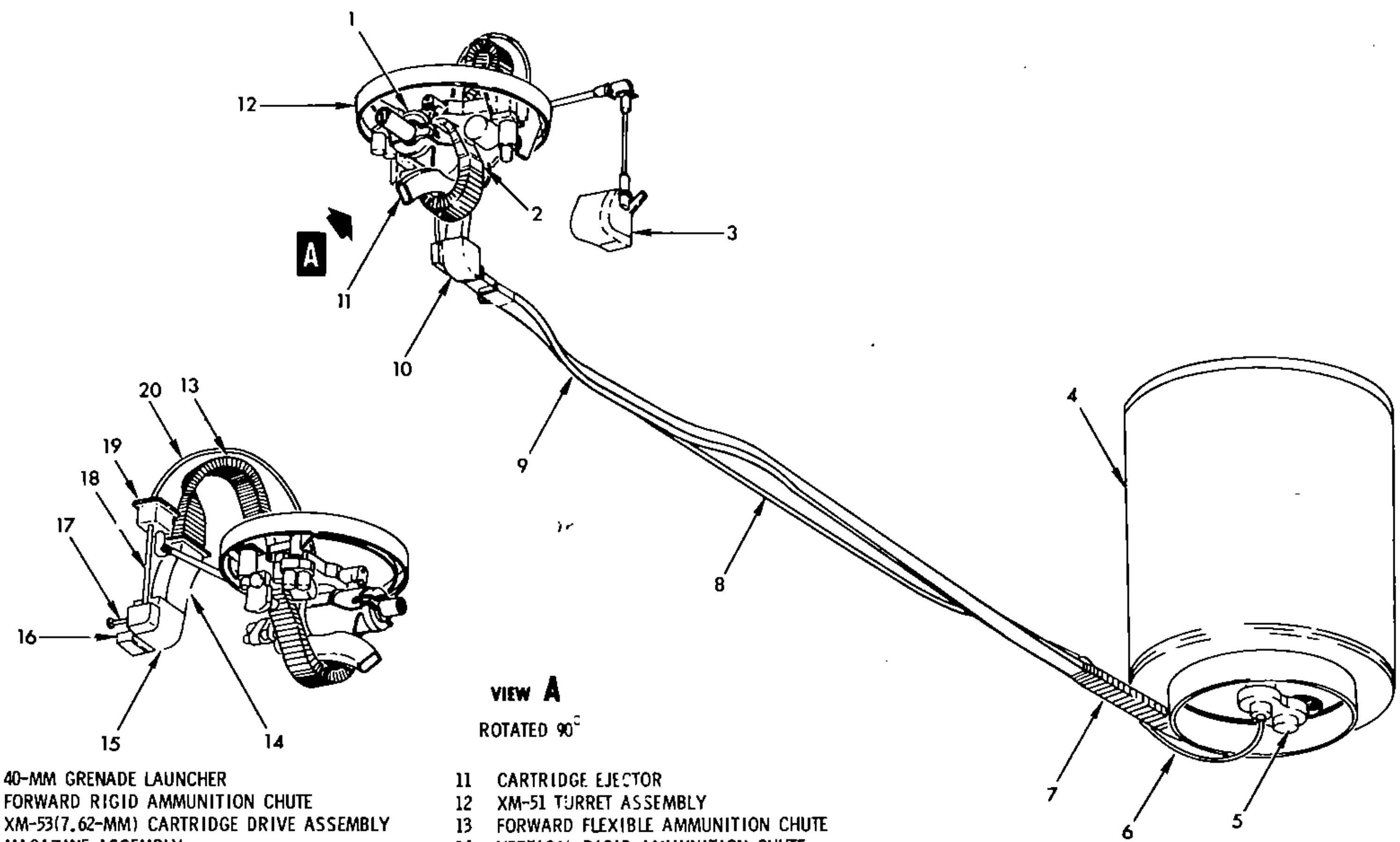


Figure 20A-4. Ammunition Storage Magazine



1 40-MM GRENADE LAUNCHER
2 FORWARD RIGID AMMUNITION CHUTE
3 XM-53(7.62-MM) CARTRIDGE DRIVE ASSEMBLY
4 MAGAZINE ASSEMBLY
5 DRIVE MECHANISM
6 AFT FLEXIBLE DRIVE SHAFT
7 AFT FLEXIBLE AMMUNITION CHUTE
8 AIRCRAFT FIXED DRIVE SHAFT
9 AIRCRAFT FIXED AMMUNITION CHUTE
10 XM-51 (40-MM) CARTRIDGE DRIVE ASSEMBLY
11 CARTRIDGE EJECTOR
12 XM-51 TURRET ASSEMBLY
13 FORWARD FLEXIBLE AMMUNITION CHUTE
14 VERTICAL RIGID AMMUNITION CHUTE
15 XM-51 (40-MM) CARTRIDGE DRIVE ASSEMBLY
16 AMMUNITION CHUTE ADAPTER
17 HORIZONTAL RIGID DRIVE SHAFT
18 VERTICAL RIGID DRIVE SHAFT
19 MODE TRANSFER ASSEMBLY
20 FORWARD FLEXIBLE DRIVE SHAFT

Figure 20A-5. XM-129 Gun Drive and Ammunition Feed Mechanism bottom of the magazine assembly. The drive mechanism operates the magazine assembly. The drive mechanism operates the magazine, ammunition feed system, and the gun.

The 40 mm grenades are link connected and fed to the gun by a combination of flexible and fixed chuting.

E. Cartridge Drive Assembly (See figure 20A-6):

The cartridge drive assembly is installed to provide a power boost to the approximately twelve foot run of linked cartridges.

F. Turret Control Module (See figure 20A-7)

The turret control module contains the necessary electronics and control circuits to operate the XM-51 grenade launcher system. The control module is mounted on a shelf in the aft section of the debris bay.

The control module contains the following major circuit components: power converter, preamplifiers, power amplifiers, fire interrupt logic circuit, and necessary BITE (Bilt-In Test Equipment) circuits.

IV. SYSTEM OPERATION

The XM-51 Suppressive Fire System is capable of firing up to 780 rounds of 40 mm anti-personnel fragmentation grenades. Rate of fire of the XM-51 system is 350 shots per minute. The 40 mm grenades can be fired in a field of ±100 degree azimuth, +18 degree to -70 degree elevation.

When the 40 mm grenades are being fed out of the drum, the linked grenades are carried through the ammunition chutes to the cartridge drive assembly and through fixed and flexible chutes into the grenade launcher. Mechanical power to drive the Cartridge Drive Assembly is provided by the ammo drum drive mechanism through fixed and flexible drive shafts to the Mode Transfer Assembly and by a flexible drive shaft from the Mode Transfer Assembly to the weapon.

Linked 40 mm grenades enter the weapon through a feed tray assembly from the flexible ammunition chute. Rotation of the ring gear on the launcher causes the barrel to move fore and aft (reciprocate). The linked round is

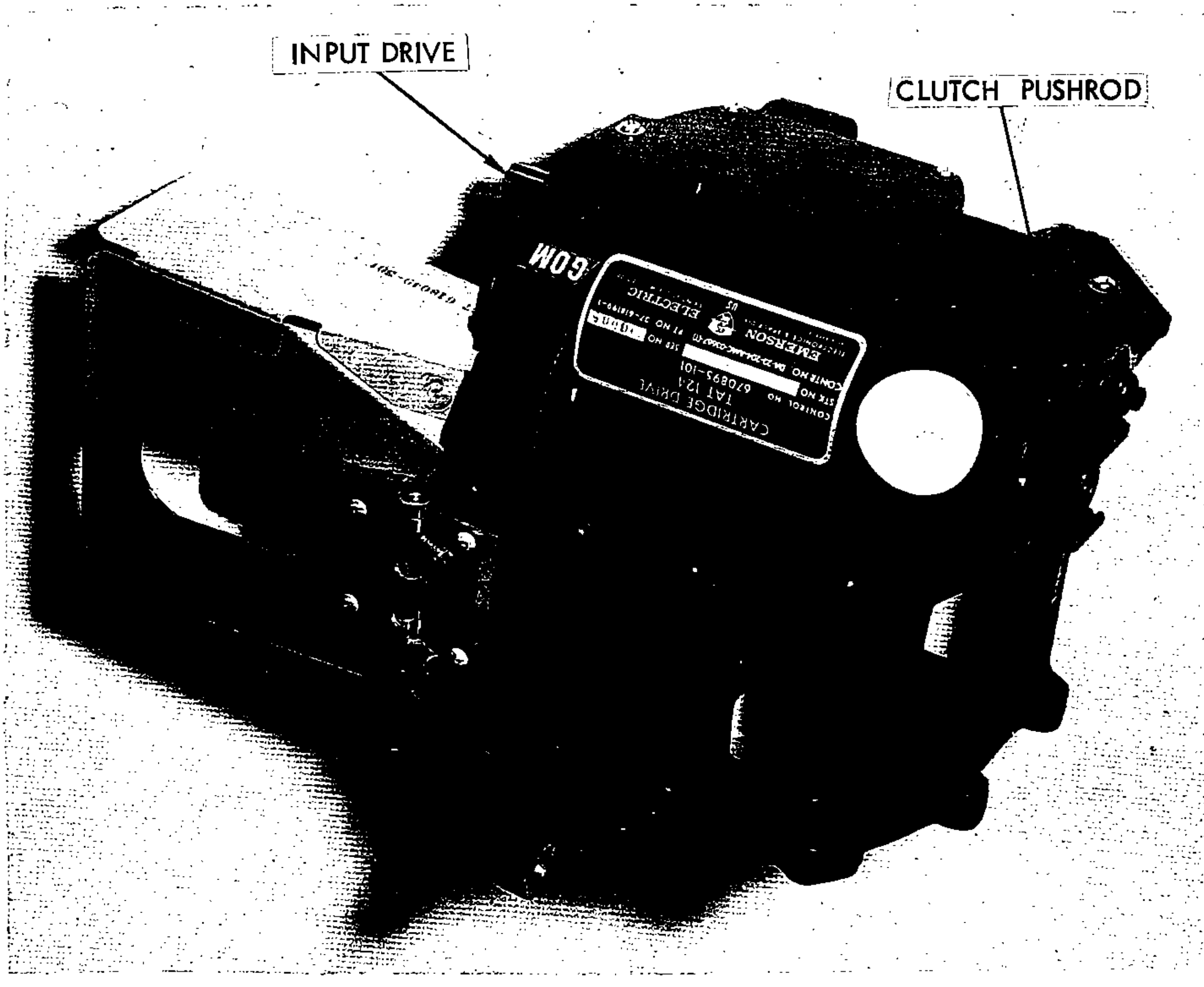


Figure 20A-6. Cartridge Drive Assembly

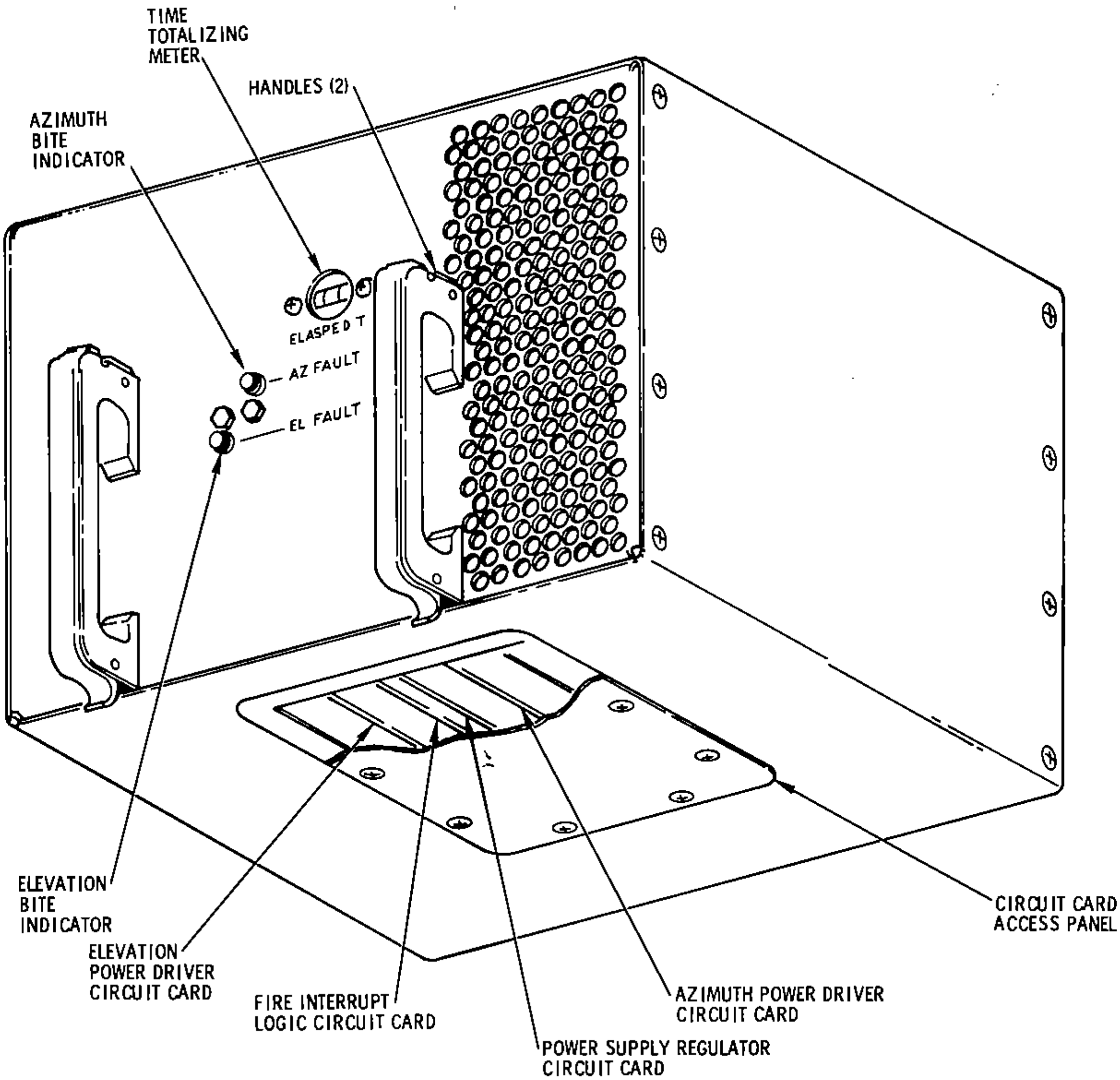


Figure 20A-7. Turret Control Module

chambered at the forward position of barrel travel. As the barrel reciprocates aft, the link is forced aft on the grenade case. At the aft position of barrel travel, the firing pin gear is released, causing the grenade to fire. As the fragmentation projectile exits the barrel of the launcher, mechanical ejection of the spent grenade case forces the case and link overboard.

The direction of weapon firing is determined by the position of the nose turret, since the weapon is physically mounted in the azimuth and elevation gimbals of the nose turret. The grenade launcher turret position is controlled by the commanded weaponline input to the Nose Turret Control Module. The turret drive motors, in both azimuth and elevation, are driven by commands from the Nose Turret Control Module to position the azimuth and elevation gimbals of the turret to the commanded position. Instrument drive assemblies provide turret position information to the turret control module. Two stow locks, one for azimuth and one for elevation, mechanically hold the turret in a "dead-ahead" position when the turret is not in use.

When the turret is cleared or shut down, the turret remains energized for a period of 5 seconds to allow the turret to return to "zero-zero" electrical stow. A ground safety pin can be inserted in the turret to prevent electrical movement of the turret or electrical operation of the weapon.

CAUTION: THE WEAPON CAN BE FIRED BY MECHANICAL ROTATION OF THE BARREL; THE WEAPON "SLEWS" RAPIDLY.

V. PCRS CONFIGURATION

The XM-51, 40 mm gun feed system chute is rerouted in the turret to increase the mean-rounds-to-stoppage (MRTS) of the system. The rerouting was made possible by the deletion of space requirements for the XM-53 gun system and raising the cartridge drive unit.

XM-53 7.62 mm MACHINE GUN

I. GENERAL DESCRIPTION

The XM-53 armament subsystem is an electrically operated, percussion fired, weapon system that provides wide angular coverage and rapid fire. As installed on the AH-56A aircraft, this weapon system is used primarily against area targets as an offensive weapon. Operation of the gun is completely automatic. Either the pilot or CPG can fire the gun by selection of the proper switches on their armament control panel and squeezing the trigger.

II. COMPONENTS AND LOCATION (See Figure 20B-1)

Name of Component	Number per Aircraft	Location in Aircraft
Ammunition Storage Drum	1	In forward section of ammo bay, accessible through door in bottom of fuselage.
1. Drive System	1	Between gun and ammunition storage drum.
2. 7.62 Cartridge Drive Assembly	1	Lower left, aft side of turret bulkhead.
Turret Control Module	1	Mounted on rack in aft end of debris bay.
Turret Assembly	1	Forward, lower side of aircraft nose.
XM-134 Gun (7.62 mm)	1	Mounted in turret elevation gimbal.
Ammunition Feed System	1	Between gun and ammunition storage drum.

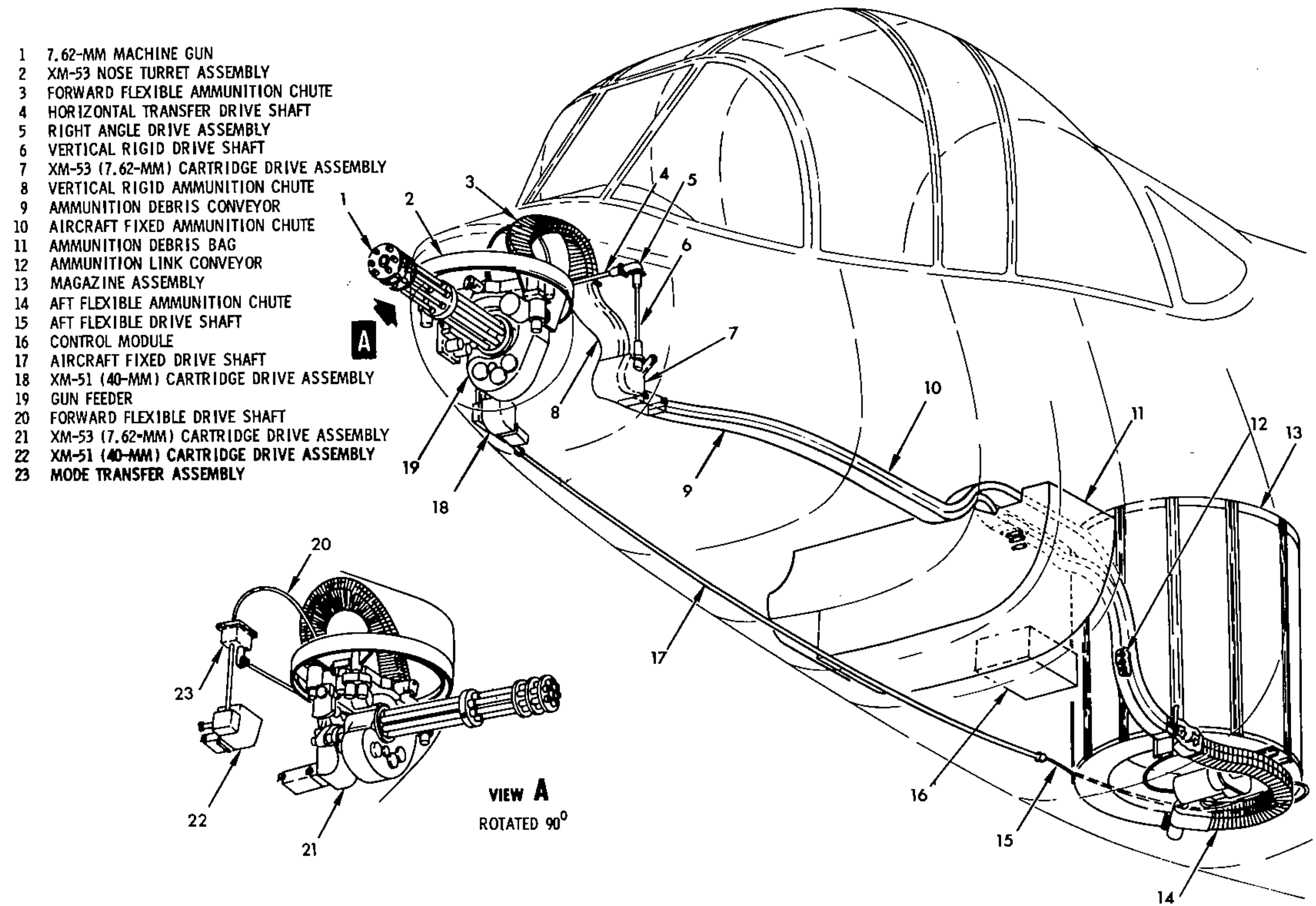


Figure 20B-1. XM-53 Turret System Component Location

III. MAJOR COMPONENT DESCRIPTION

A. Ammunition Drum

Installed in the forward portion of the ammo drum compartment, stores up to 11,720 rounds of 7.62 mm ammunition. The ammunition drum is driven by a four speed drive mechanism at the bottom of the drum. The drive mechanism also drives the minigun through a drive shaft. When the four speed drive mechanism is operated in the "download" direction, unlinked 7.62 mm rounds are transported from the drum to the minigun through a conveyor belt system installed along the left side of the fuselage. Spent cases are transported from the minigun to the ammo debris bay for stowage.

B. XM-53 Control Module

Shelf mounted in the ammo debris bay, receives a weaponline command from the Fire Control System and positions the nose turret in azimuth and elevation to correspond with the command. The XM-53 Control Module also contains firing logic and fire interrupt logic.

C. Nose Turret

Is mounted to the nose turret deck by four bolts, is positioned by electrical weaponline command signals from the Nose Turret Control Module. The nose turret can be positioned $\pm 110^{\circ}$ in azimuth, and the weapon installed in the nose turret can be positioned $+18^{\circ}$ to -70° in elevation.

D. 7.62 mm Minigun

Installed in the nose turret, is a six-barrel percussion fired weapon. The minigun receives unlinked 7.62 mm rounds from the conveyor system, percussion fires the rounds, and returns the spent round casing to the conveyor system. The conveyor system deposits the spent case in the debris bay. The pointing line of the minigun is determined by the position of the nose turret in which it is mounted. A "safing sector" on the minigun allows the weapon to be rendered safe.

IV. SYSTEM OPERATION

The XM-53 Alternate Suppressive Fire System is capable of firing up to 11,720 rounds of standard 7.62 mm ammunition at any one of four rates of fire -- 750 shots per minute (spm), 1500 spm, 3000 spm, or 6000 spm. Selection of the rate of firing is provided on the Pilot's Weapons Panel and on the copilot/gunner's Main Weapons panel. The minigun will operate at the selected rate of fire when the trigger on the cyclic control or on the swivelling gunner's station right grip is operated to the first trigger detent. Depressing the trigger to the second detent causes the minigun to operate at 6000 spm. The minigun can be operated in a field of $\pm 110^{\circ}$ azimuth, $+18^{\circ}$ to -70° elevation.

V. PCRS CONFIGURATION

The XM-53 (7.62 mm) machine gun system is deleted. The suppressive fire function of this system is transferred to the XM-51 and XM-52 gun systems remaining.

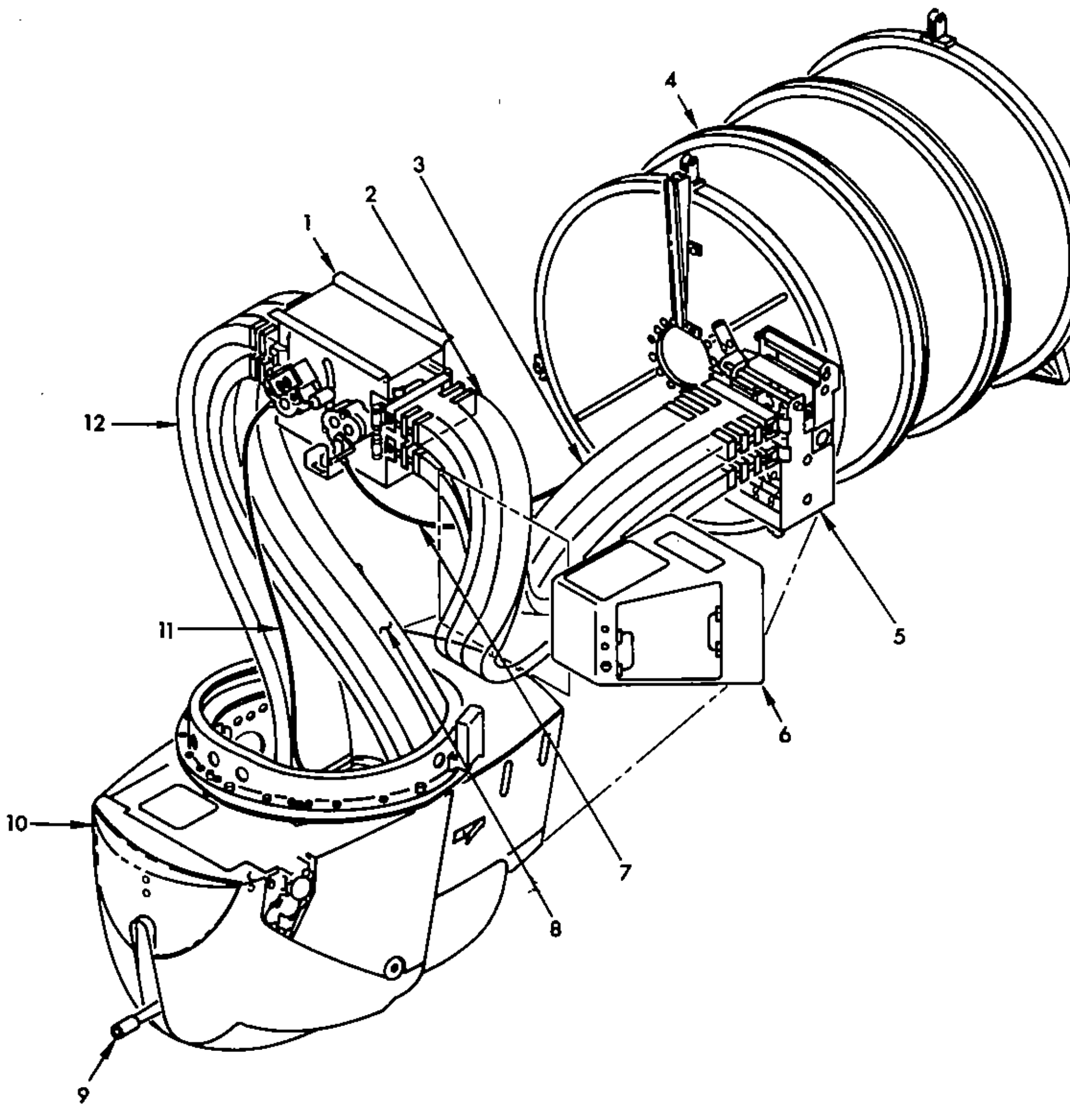
XM-52 30 mm AUTOMATIC GUN

I. GENERAL DESCRIPTION

The XM-52 System is capable of firing up to 2,010 rounds of 30 mm shaped charge fragmentation ammunition at a rate of fire of approximately 425 shots per minute. The 30 mm automatic gun can be operated in a field ±200 degrees azimuth (greater than a full circle), +21 degrees to -60 degrees elevation. A profile cam in the turret precludes the possibility of firing the belly weapon into any portion of the helicopter fuselage.

II. COMPONENTS AND LOCATION (See Figure 20C-1)

Name of Component	Number per Aircraft	Location in Aircraft
Turret Assembly	1	Mounted to belly of aircraft, below and just aft of the pilot's compartment
XM-140 Gun	1	Mounted in turret elevation gimbal
Ammunition Storage Magazine	1	In aft section of ammunition bay, accessible through door in bottom of fuselage
Ammunition Feed System	1	Between ammunition storage magazine and turret
Turret Control Module	1	Mounted in rack in forward section of left sponson, accessible through panel in sponsons leading edge



1 RESERVOIR ASSEMBLY
2 AFT FEED AMMUNITION CHUTE
3 AFT RETURN AMMUNITION CHUTE
4 MAGAZINE ASSEMBLY
5 DELINKER-RELOADER
6 CONTROL MODULE
7 AFT FLEXIBLE DRIVE SHAFT
8 FORWARD RETURN AMMUNITION CHUTE
9 30-MM AUTOMATIC GUN
10 XM52 BELLY TURRET ASSEMBLY
11 FORWARD FLEXIBLE DRIVE SHAFT
12 FORWARD FEED AMMUNITION CHUTE

Figure 20C-1. XM-52 Turret System Component Location

III. MAJOR COMPONENT DESCRIPTION

A. Belly Turret

Is suspended from the underside of the fuselage by a ring bearing, is positioned by electrical weaponline command signals from the Belly Turret Control Module. (See Figure 20C-2.)

B. 30 mm Automatic Gun

Installed in the belly turret, is a reciprocating barrel electrically driven weapon. The automatic gun receives rounds from the conveyor which transports the rounds from the reservoir to the weapon, chambers and fires the round, and ejects the spent 30 mm case. The pointing line of the automatic gun is determined by the position of the belly turret in which it is mounted. (See Figure 20C-3)

C. Ammunition Storage Magazine

Installed in the aft portion of the ammo drum compartment, stores up to 2,010 rounds of 30 mm shaped charge fragmentation ammunition. The Ammunition Drum is driven by a drive mechanism at the aft portion of the drum. When the drive mechanism is operated in the "download" direction, unlinked 30 mm ammunition is transported from the drum to the reservoir through a conveyor system. (See Figure 20C-4.)

D. Ammunition Feed System

Includes feed chute assemblies, a cartridge ejector assembly, and a reservoir assembly. The reservoir, which is installed at the forward portion of the ammo drum compartment, receives ammunition from the drum conveyor and holds the 30 mm rounds until the rounds are required by the weapon. Two drive mechanisms (one on the drum and one on the weapon) are used in the XM-52 system. The reservoir prevents misfeeding of the weapon by ensuring that rounds are fed to the weapon properly timed with the weapon drive mechanism. Two flexible shafts, one from the drum and one from the weapon, provide mechanical timing inputs to the reservoir.

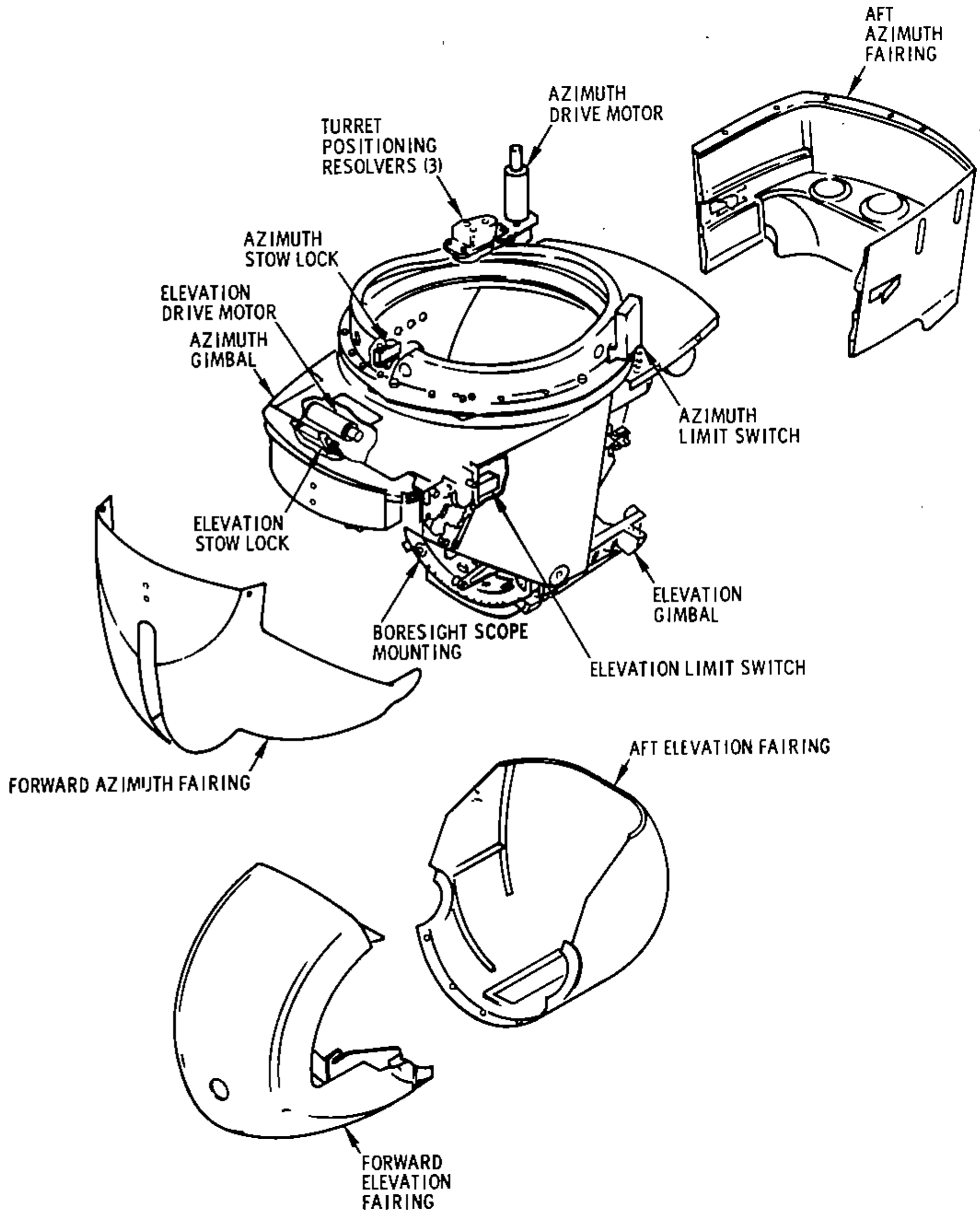


Figure 20C-2. XM-52 Turret Assembly

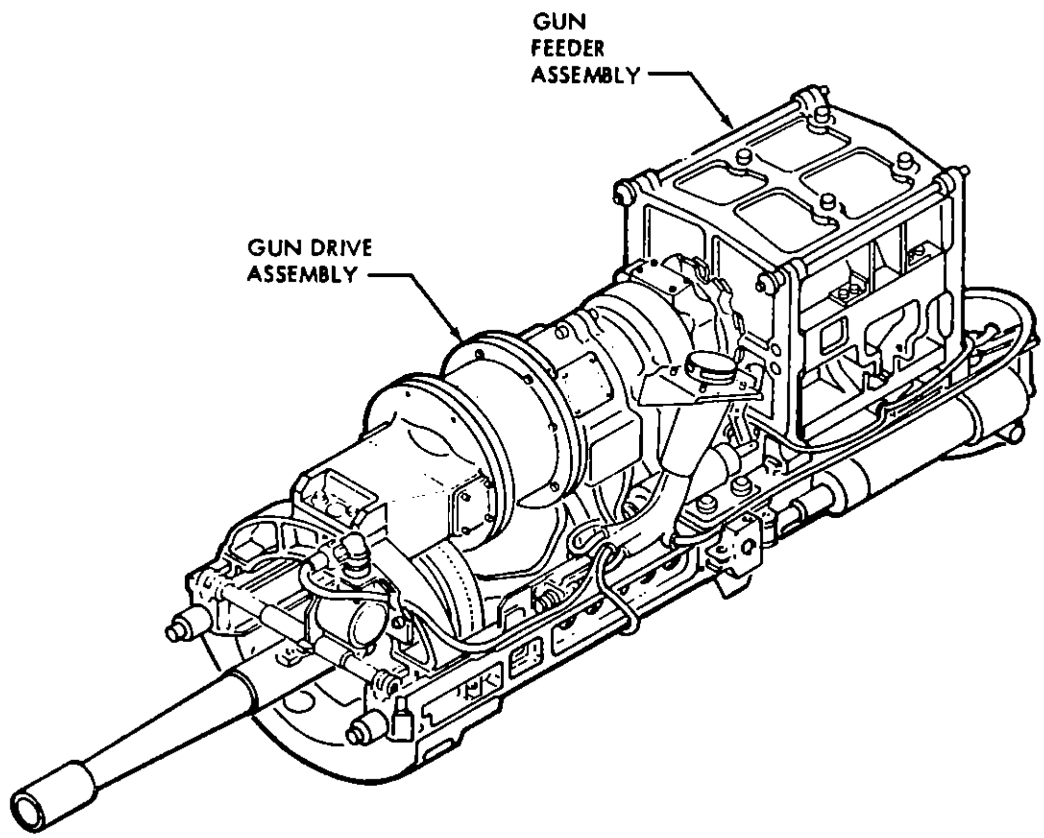
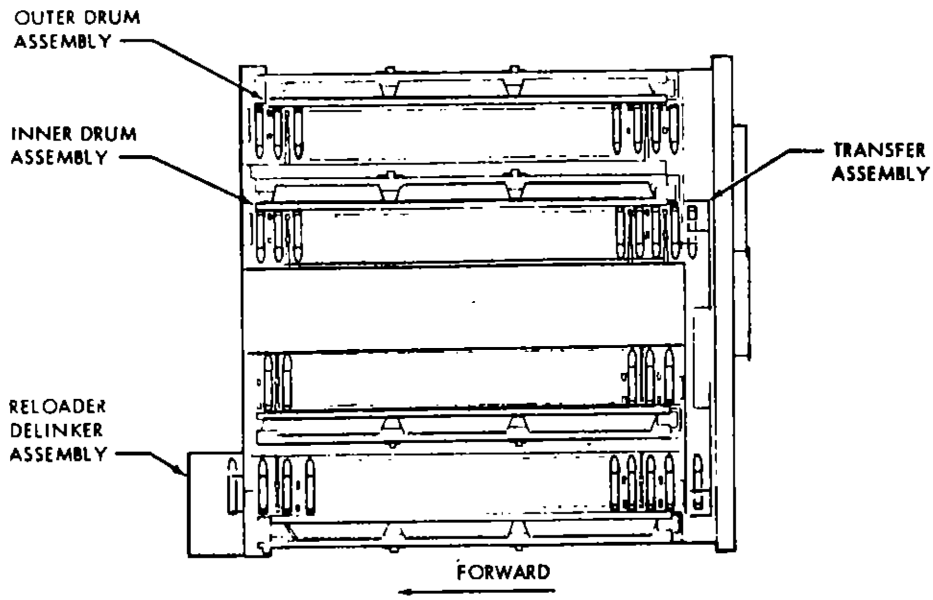


Figure 20C-3. XM-140 Gun - 30 mm



SECTION VIEW OF MAGAZINE ASSEMBLY

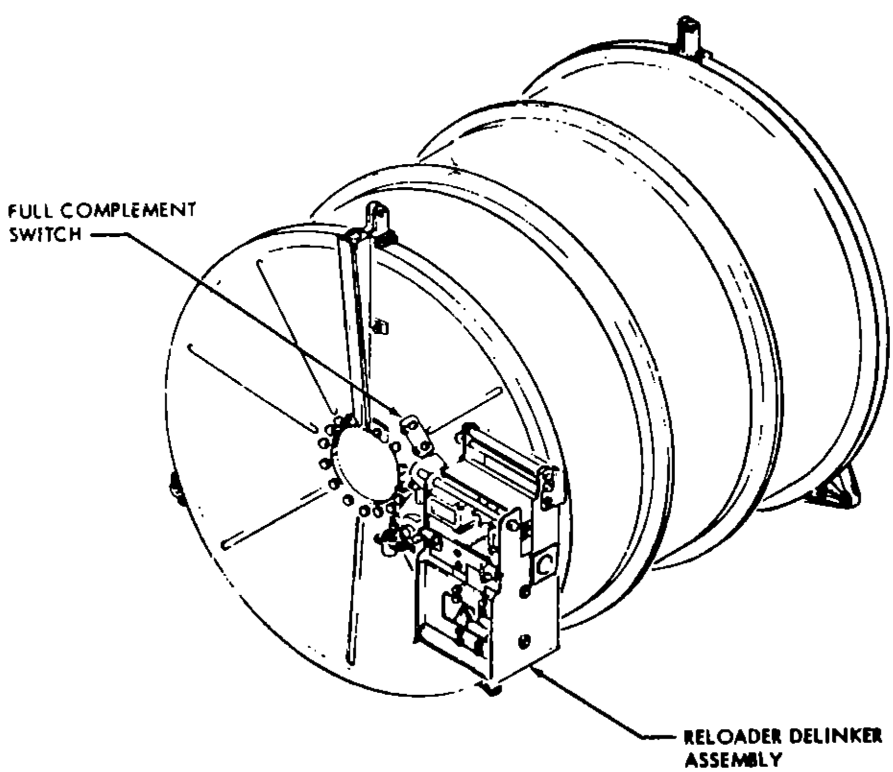


Figure 20C-4. XM-52 Ammunition Storage Magazine

E. XM-52 Control Module

Is installed in the leading edge of the left sponson, receives a weaponline command from the Fire Control System positions the belly turret in azimuth and elevation to correspond with the command. The XM-52 Control Module also contains firing logic and fire interrupt logic. (See Figure 20C-5.)

IV. SYSTEM OPERATION

The XM-52 Turret System is an electrically and hydraulically operated subsystem that provides wide angular coverage and rapid fire. The armament subsystem, as installed on the AH-56A, is used primarily as an offensive weapon. The system consists of the following major components: Turret Assembly, XM-140 Gun, Ammunition Storage Magazine, Ammunition Feed System, and an Electronic Control System.

The 30 mm turret is mounted to the bottom of the belly turret bay, which is just aft and below the pilot's compartment. The gun is fed from an ammunition storage magazine, which holds 2,010 rounds of ammunition. The magazine is mounted in the aft section of the ammunition bay, which is located behind the 30 mm turret bay. Access to the ammunition bay is through two hinged access doors in the bottom of the fuselage.

The 30 mm cartridges are fed to the gun by a feed system that includes two conveyer belts and a reservoir assembly. The cartridges are stored in the magazine in a linkless condition. During firing, the cartridges are loaded on an aft conveyer belt, transported to the reservoir assembly, transferred to the forward conveyer belt, and then transported to the gun feed assembly. The reservoir acts as a slack takeup unit for the feed system, due to the possible momentary difference in speeds between the gun and the magazine. The reservoir absorbs the extra cartridges from the magazine during fire termination, and supplies the extra cartridges to the gun during gun start-up.

After firing, the spent cases are ejected downward to clear the ship. An ejector assembly is mounted to the bottom of the breech opening to positively eject the cases downward and ensure that the spent cases will not hit any part of the aircraft or be sucked into any of the rotors.

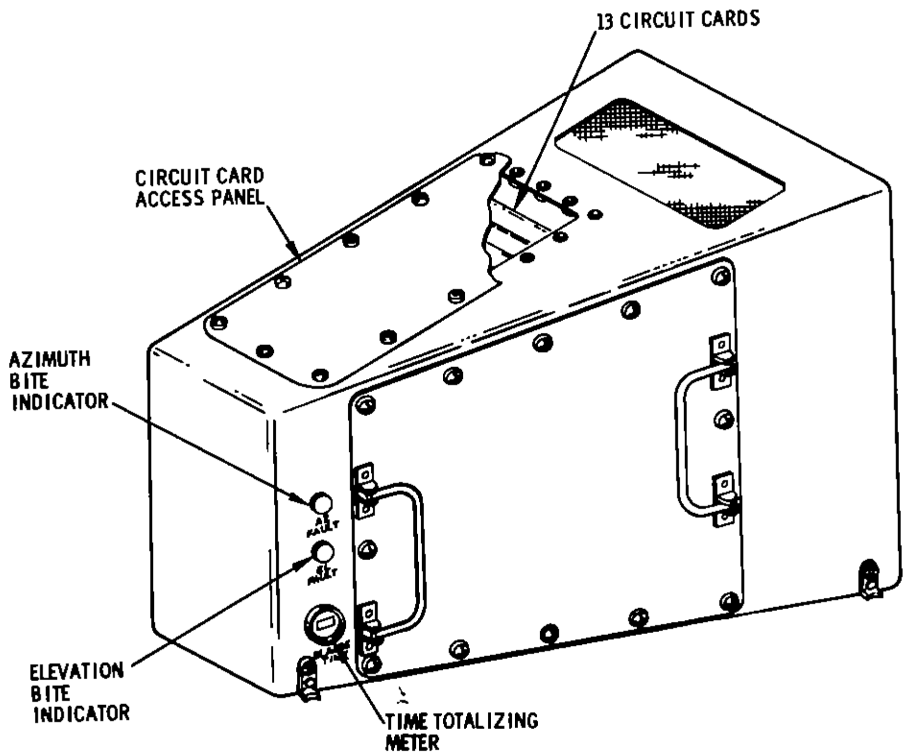


Figure 20C-5. XM-52 Turret Control Module

V. PCRS CONFIGURATION

A. The XM-52 (30 mm) gun system has been altered by reducing ammo capacity and installing a new conveyor element. The deletion of the inner bay of the ammo drum reduces the cost of the drum and improves the reliability of the feed from the drum. Eliminating the inner bay reduces the ammo capacity from 2,010 rounds to approximately 1,500 rounds. This reduction is acceptable because of infrequent requirement for more than 1,500 rounds on a single sortie. The storage drum is reoriented in a vertical arrangement which simplifies the installation of the feed chutes. The changes do not affect empty weight.

B. The control module has been moved from the left sponson to the fire control/avionics bay and combined with the XM-51 control module to save 12 pounds. The feed system was analyzed for compatibility with both brass and aluminum case ammo and found to be acceptable.

EXTERNAL STORES SYSTEM

I. GENERAL DESCRIPTION

Three types of external stores (rockets, armed stores, or fuel tanks) can be carried at the wing and/or fuselage pylon stations, and are controlled by the External Stores Control System. The TOW Missile System, which can be carried at the inboard wing pylon stations, operates independently of the External Stores Control System.

II. COMPONENTS AND LOCATION

Name of Component	Number per Aircraft	Location in Aircraft
Pylon Assemblies	6	See figure 20D-1
External Stores Control Panels	2	Right console of each cockpit
External Stores Programmer	1	Ammo debris bay

III. MAJOR COMPONENT DESCRIPTION

A. Pylon Assemblies

The six pylons on the AH-56A are not interchangeable due to the position of the pylon, wing sweepback, and wing dihedral. Aerodynamic fairings are designed to enclose the pylon, with access doors, access panels, and holes provided to allow for maintenance, servicing, and inspection.

All six pylons contain quick disconnect fuel lines and electrical connectors to allow installation of an external fuel tank. An AERO 65A1 Stores Rack, with AERO A-1 Adapters, are mounted on all pylons to provide both 14 inch and 30 inch rack hook provisions. The four wing stations contain a program director mounted in the aft end of the pylon.

A Triple Adapter Rack (TAR) can be mounted on the inboard wing pylons. (See figure 20D-2.) The TAR is designed to triple the capability of the wing pylons by allowing the installation of three rocket pods or three stores on the pylon.

B. External Stores Control Panels

The weapons control panels provided in each crew station allow selection of weapons, mode of attack, quantity and rate of weapon firing, and arming and/or fusing parameters. Both of the Control Panels interface with the External Stores Programmer.

C. External Stores Programmer

The External Stores Programmer has information which indicates the type and quantity of store installed at each pylon provided by the Pylon Control Director. The Programmer ensures that legal stores combinations are selected on the control panel, and the Programmer controls the sequencing of firing or release of the installed stores. External stores are fired or released using the rocket trigger (red R) on either the pilot's or copilot/gunner's cyclic control stick. Firing or release of external stores has priority over the internal weapons, and a fire interrupt is provided to the internal weapons systems by the Programmer when the external stores are to be fired.

The following listed external stores can be carried at the stations indicated:

1. 2.75" Folding Fin Aerial Rockets (FFAR): rockets can be carried at any of the four wing pylon stations, but cannot be carried at the fuselage stations. Impact fused, smoke warhead, or variable time fused rockets may be carried in the XM-159 (19 rocket) Pod. Triple adapter racks (TAR), which can accommodate three XM-159 Pods each, can be carried at the outboard wing pylons only. The maximum rocket loading configuration is 152 rockets; 19 at each inboard wing station +57 at each outboard wing station.

2. Armed stores: armed stores can be carried at any of the six pylon stations. Nose fused or nose-tail fused armed stores can be suspended from the Aero 65A1 rack. Each pylon is structurally capable of carrying up to 2000 pounds of external stores.

3. Fuel tanks: each of the six pylons is plumbed to carry external fuel tanks. Maximum gross weight limitations dictate a maximum of five external fuel tanks loaded at any one time.

D. TOW Pod (figure 20D-3): Each Pod houses three TOW missiles equipped with an armor piercing warhead, contains a guidance wire coiled in each of the three launcher tubes. The guidance wire transmits steering commands from the Fire Control System to the TOW missile in flight to guide the missile into the target. A modulated infrared source at the aft end of the missile transmits a tracking signal with which the helicopter's Fire Control System can track the launch and flight of the TOW missile.

Each of the four wing pylons consists of a strongback to which the Aero 65A1 rack is bolted. Sway braces at the front and the rear of the rack can be reversed when changing from 14 inch stores (XM-159 Pod, triple adapter rack, armed stores, or TOW missile launcer) to 30 inch fuel tanks. A Pylon Control Director is mounted at the aft section of each of the four wing pylons.

The Pylon Control Director electronically notifies the External Stores Control System Programmer as to the type and quantity of store installed at the pylon. Presence of a Triple Adapter Rack is indicated by a connector at the forward portion of the rack. When fuel tanks, armed stores, or the TOW missile launcher are installed, the TYPE STORE selector switch should be positioned to STORE/TANK/EMPTY. When 2.75" folding fin aerial rockets are installed, the TYPE STORE selector switch should be positioned to correspond to the type of rocket installed -- IMP for impact fused, SMK for smoke warhead, or

VT for variable time fused. A SAFETY PIN INSERT accepts a safety pin when the TYPE STORE selector is positioned to OFF. Inserting the safety pin causes the stepper (rocket sequencer) to "home", and is indicated by the STEPPER HOME light annunciator. A stray VOLT MONITOR and a Built In Test Equipment FAULT DET (detector) are provided in each of the four Pylon Control Directors.

IV. SYSTEM OPERATION

After being commanded with the selections from the External Stores Control Panel, and being triggered by the rocket fire trigger on the cyclic control stick, the External Stores Programmer controls the firing pulses to the stores and/or rockets on the pylons. The following priority sequency is controlled by the External Stores Programmer:

A. Outboard stations are depleted first, unless the outboard stations are not selected on the control panel, until the QTY selection is fulfilled.

B. If triple adapter racks are installed at the outboard stations, the center pod is fired first, the left pod second, and the right pod last until the QTY selection is fulfilled.

C. Inboard stations.

D. Fuselage stations.

Releasing the rocket trigger stops all firing of external stores, regardless of the quantity remaining to be fired. Firing of any external store will interrupt firing of the belly automatic weapon and the nose weapon for approximately 1/4 second. An empty condition at a selected pylon station will interrupt firing and present an E annunciation. The left inboard pylon station and the right inboard pylon station are automatically interlocked if the quantity of rockets selected from either station exceeds 19. The left inboard station and the right inboard station are automatically interlocked if an armed store has been selected at either of the inboard stations. Release of armed stores must be accomplished in SALVO only.

Built In Test Equipment (BITE) is provided in the External Stores Control System. A TEST INITIATE switch on the External Stores Programmer enables the ground self-test. During self-test, the RESET T annunciator on each External Control Panel should light. BITE indicators for the control panels (C1, C2), the four Pylon Control Directors (DIR), and the Programmer (PROG) are located at the front of the External Stores Programmer. Failure of any one of the four Pylon Control Directors can be verified by the FAULT DET indicator on the individual Pylon Control Director.

V. PCRS CONFIGURATION

A. 2.75-Inch FFAR Rocket System

The rocket system arrangement and capabilities are unchanged. However, the mounting pylons are simplified and the manual stores release system is eliminated.

B. Guided Missile System

The TOW control equipment (TCE) has been relocated in the main avionics bay, for a savings of 26 pounds. The space is made available by the elimination of other avionics equipments.

C. Mission Kits

Complete provisions are included for the mission kits listed. However, the manual external stores release cable system is deleted for a savings of nine pounds in empty weight. Specific kits affected are:

- Fuselage pylons - deleted
- Wing pylons - simplified to eliminate need for Aero 65A1 racks and A1 adaptors. Fuel provisions are deleted from the outboard pylons.
- Ferry kit fuselage pylon fuel tank - deleted.

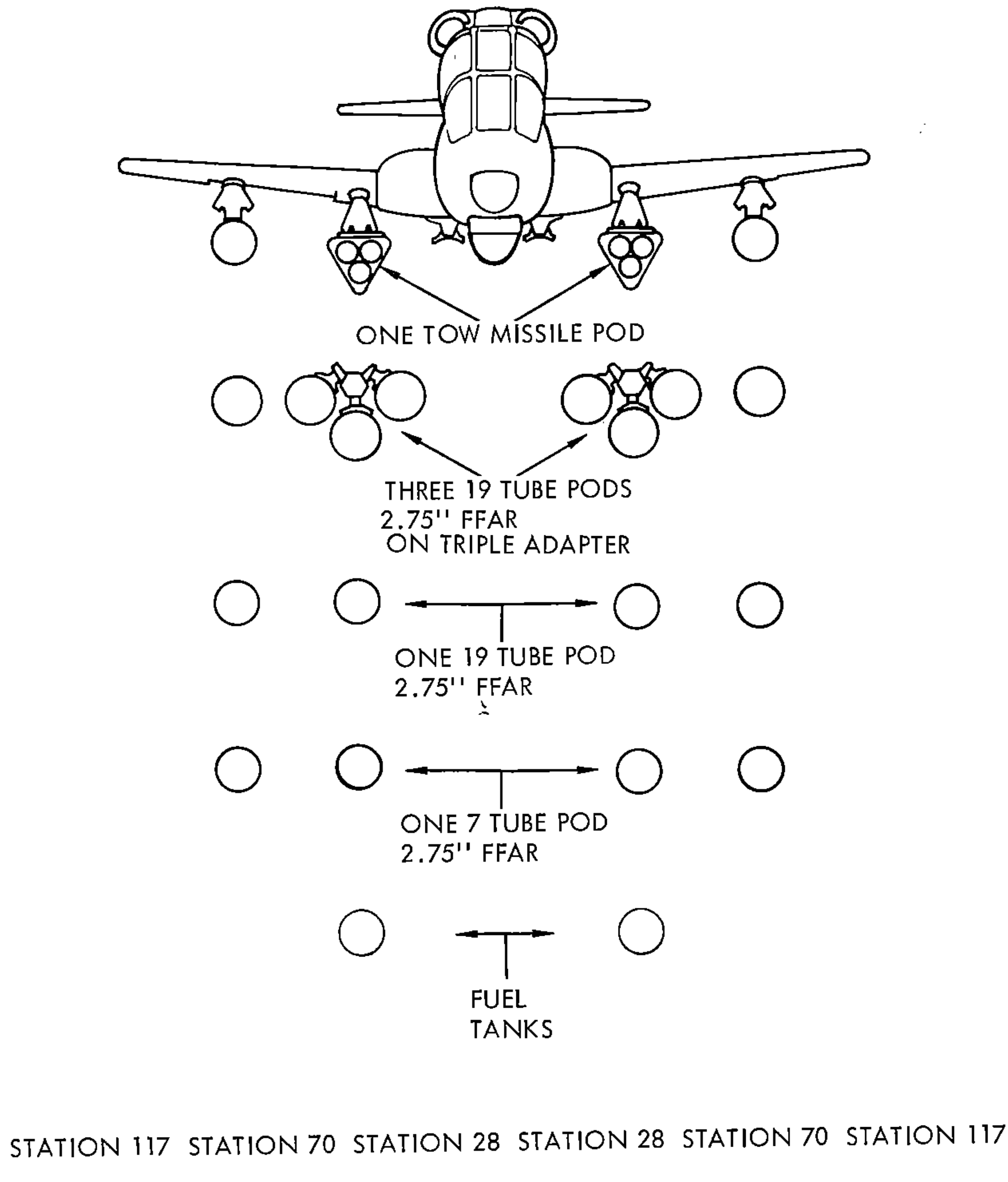


Figure 20D-1. External Stores Configuration

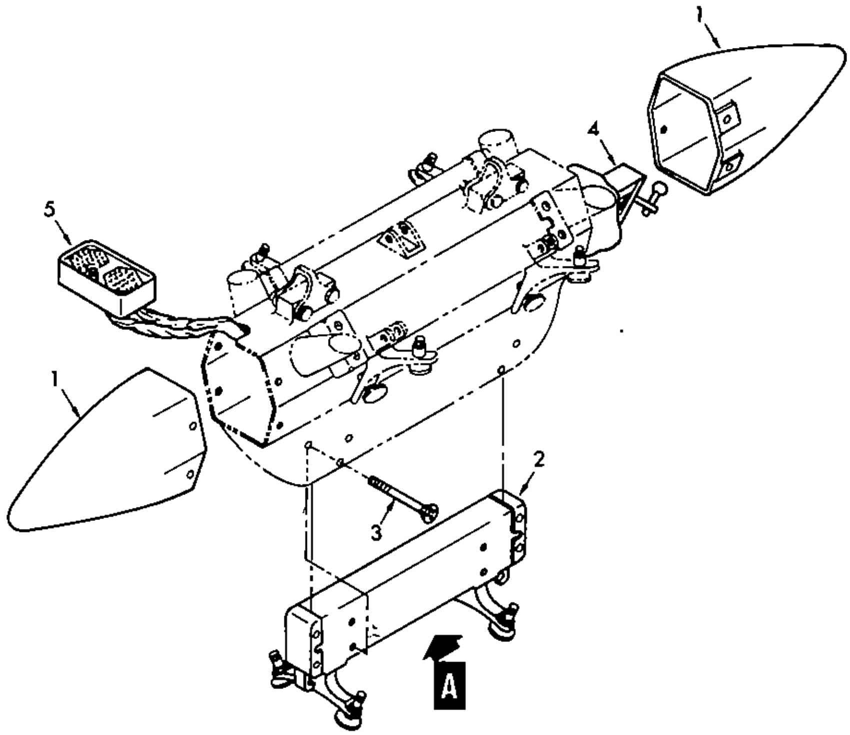


Figure 20D-2. Triple Adapter Rack

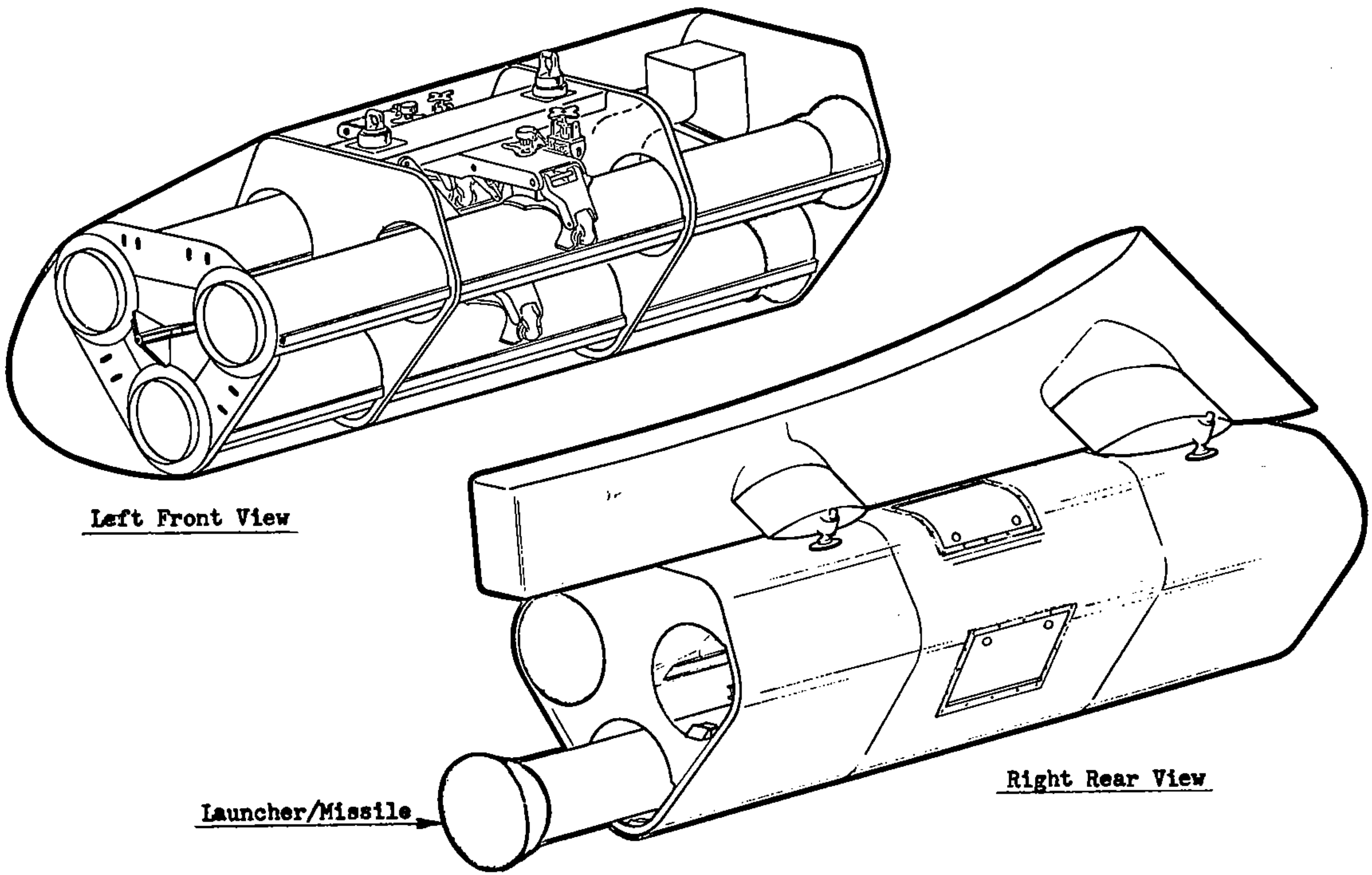


Figure 20D-3. TOW Missile Pod

FLAWS

I. GENERAL DESCRIPTION

The Fault Location and Aural Warning System (FLAWS) provides warning for 38 vehicle and 30 mission systems parameters. This frees the pilot from continuously scanning his instrument panel, thus enabling him to concentrate on the tactical problem.

Failure warning is provided by a Master Caution light in each cockpit (for flight critical failures) and an appropriate legend illuminating on the pilot's Annunciator Panel, copilot/gunner's Annunciator Panel and/or on the FLAWS Status Panel.

Five priority failures are also announced by one of two available tones into the intercommunication system.

A Voice Warning System with a 40 failure capacity is utilized as a parallel warning system.

II. COMPONENTS AND LOCATIONS

Name of Component	Number per Aircraft	Location in Aircraft
Equipment Status Panel	1	Copilot/gunner station aft bulkhead
Pilot Annunciator Panel	1	Pilot station right hand console
Copilot/gunner Annunciator Panel	1	Copilot/gunner instrument panel
Warning Lights		
- (ENG FIRE, RPM & CANOPY UNSAFE)	2 sets	One set on each instrument panel
- APU Fire Light	1	Pilot left Hand Console
- Landing Gear Unsafe Light	1	Pilot left Hand Console
Master Caution Light	2	One on each instrument panel
Warning and Caution Lights Test Button	1	Pilot right fwd. panel

III. MAJOR COMPONENT DESCRIPTIONS

A. Equipment Status Panel

An equipment status panel, accessible to the copilot/gunner by swiveling his seat clockwise, provides the flight crew with a means of quickly determining equipment status. All inputs from fault locating and advisory signal sensors are routed to the FLAWS status panel, and all outputs to annunciators and the aural warning system are generated by the panel. The panel contains 54 annunciator lights (including spares), primarily advisory in nature, for the avionics, fire control, and other aircraft systems. Power to operate the panel is from the essential DC bus.

B. Annunciator Panels

Both pilot and copilot/gunner annunciator panels provide annunciator light indications of aircraft cautionary and advisory conditions to which the crew must immediately respond. The pilot station panel contains 26 lettered caution lights (plus spares) and the copilot/gunner panel contains 10 caution lights. When energized, the light legend illuminates; when the condition is corrected, the light extinguishes. The STBY INV ON caution light is green; all others are amber. Illumination of all but seven of the caution lights is accompanied by illumination of the master caution light. The STBY INV ON, WEAPONS HOT, COWL UNSAFE, LASER HOT, ROTOR BRAKE, APU ON, and STATUS PANEL lights are considered advisory in nature and do not cause the master caution light to illuminate.

C. Warning Lights

Five warning lights on the instrument panels (four annunciator and one control knob) provide indication to the flight crew that a hazardous condition exists in the aircraft. They are engine fire, rpm low or high, canopy unsafe, APU fire, and landing gear unsafe.

D. Master Caution Light

A master caution light is located on the top center of the instrument panel in each crew station. The master caution lights in both stations illuminate for the five instrument panel warning lights and all but 7 of the 36 annunciator panel caution lights. Once illuminated, the master caution light remains on until acknowledged by pressing, serving as a visual alerting signal to the pilot and copilot/gunner to monitor their annunciator panels. The circuit, to which the master caution light is connected, will extinguish the master caution light (when pressed) without extinguishing an individual caution light on an annunciator panel. Thus, it is possible for the master caution light to indicate a subsequent malfunction. Although each of the individual caution lights on an annunciator panel will extinguish when the malfunction is corrected, the master caution light must be manually pressed to release the light circuit.

E. Warning and Cautions Light Test Button

A warning and caution lights test button is located on the right forward panel in the pilot station. The button is placarded WARN LTS, and when pressed tests the bulbs of all the warning lights, annunciator panel caution lights, and equipment status panel lights, in both crew stations. (The copilot/gunner must simultaneously press the PRESS-TO-READ button on the equipment status panel to read the panel.) Pressing the WARN LTS test button also causes the highest priority warning light voice message to be heard in the headsets, repeating as long as the button is held down. If, however, while the button is held down the master caution light is depressed, the next highest priority voice message will be heard. Continuing this procedure will test the entire tape of 40 recorded voice messages.

IV. SYSTEM OPERATION

The FLAWS Status Panel is the central junction point for the fault location Warning System. It recognizes 0 VDC BITE signals and 28 VDC switch closing signals as failures.

The monitored element characteristics, through sensors, determine which of the following conditions is used to provide failure warning:

A. A switch closing providing a 28 VDC signal

B. A switch closing providing a ground signal

C. A logic change from +5 VDC (GO) to 0 VDC (NO GO)

The "Typical Fail Signal Flow" block diagram illustrates the system operation and shows that where sensors provide a ground for failure, logic inverting relays provide the 28 VDC fail signal to the status panel.

The +5 VDC or 0 VDC BITE signal sensors are restricted to use only in the mission systems.

Once the 'Fail' signal is received by the Status Panel, it is immediately sent to the voice warning units. At the same time, it is processed through the status panel to illuminate the appropriate annunciator or status panel legend. If the 'Fail' signal is a flight critical failure, the master caution light will illuminate 0.4 second after the arrival of the 'Fail' signal. This delay precludes nuisance activation of the master caution light.

Pressing of the master caution light by either crewman will acknowledge the failure and cause the master caution light to extinguish. Also, the Voice Warning message ceases. The Annunciator or Status Panel legend, however, will remain illuminated until the failure is cleared.

The aural warning system consists of pre-recorded, taped voice messages and aural tones, heard in the crew headsets through the intercom system. (Not all caution or warning lights are accompanied by an aural signal.) Whether voice messages or aural tones are heard depends upon the position of the VO WRN/AURAL TONE switch in the pilot station. (Voice messages and aural tones cannot be heard simultaneously.) The aural tones heard when the switch is in the AURAL TONE position consist of a 400-Hz chopped tone for the landing-gear-up circuit, the canopy-open circuit, and the low- and high-rotor-rpm circuit, and a higher pitched, 800-Hz steady tone for the engine-out circuit.

With the switch in the VO WRN (voice warning) position, the tape-recorded voice messages listed in the accompanying table are heard.

V. PCRS CONFIGURATION

By utilizing increased Computer Central memory, many of the FLAWS logic functions can be eliminated. Thus, in the approved Producibility Cost Reduction Study configuration, the Status Panel is deleted. Those functions still necessary for flight safety warning would be incorporated into the pilot's Annunciator Panel. Mission type equipment failures would be recorded by the Computer Central and available for call up by air or ground crews. The Voice Warning system is deleted and the tone warning capability is retained.

FLAWS Voice Messages Table

Priority	Message	Priority	Message
1	Engine Fire	21	Hydaulic Two Pressure Low
2	Engine Out, Lower Gear	22	Generator One Failure
3	RPM Low	23	Generator Two Failure
4	Gear Not Down	24	DC One Failure
5	RPM High	25	DC Two Failure
6	Oil Cooler Bypass	26	Roll Desensitizer Out
7	Fire - APU	27	Channel 27 Spare
8	Chips - Transmission	28	YAW SAS Out
9	Chips - Engine	29	Channel 29 Spare
10	Chips - Propeller Gearbox	30	Weapons Hot
11	Transmission Oil	31	Laser Hot
12	Engine Oil Pressure Low	32	Fuel Filter
13	Boost Pump Failure	33	Pitch Desensitizer Out
14	Hot Engine Oil	34	Use Standby Attitude and Whiskey Compass
15	Canopy Unsafe		
16	Oil Cooler Fan Out	35	Propeller Gearbox Vibration[1]
17	Fuel Low	36	Engine Oil Level Low
18	Exhaust Compartment Hot	37	Transmission Oil Level Low
19	Propeller Oil	38	Belly Out of Boresight
20	Hydraulic One Pressure Low	39	Nose Out of Boresight
		40	Computer Out

[1]Not associated with a fault sensor; message heard during test only.

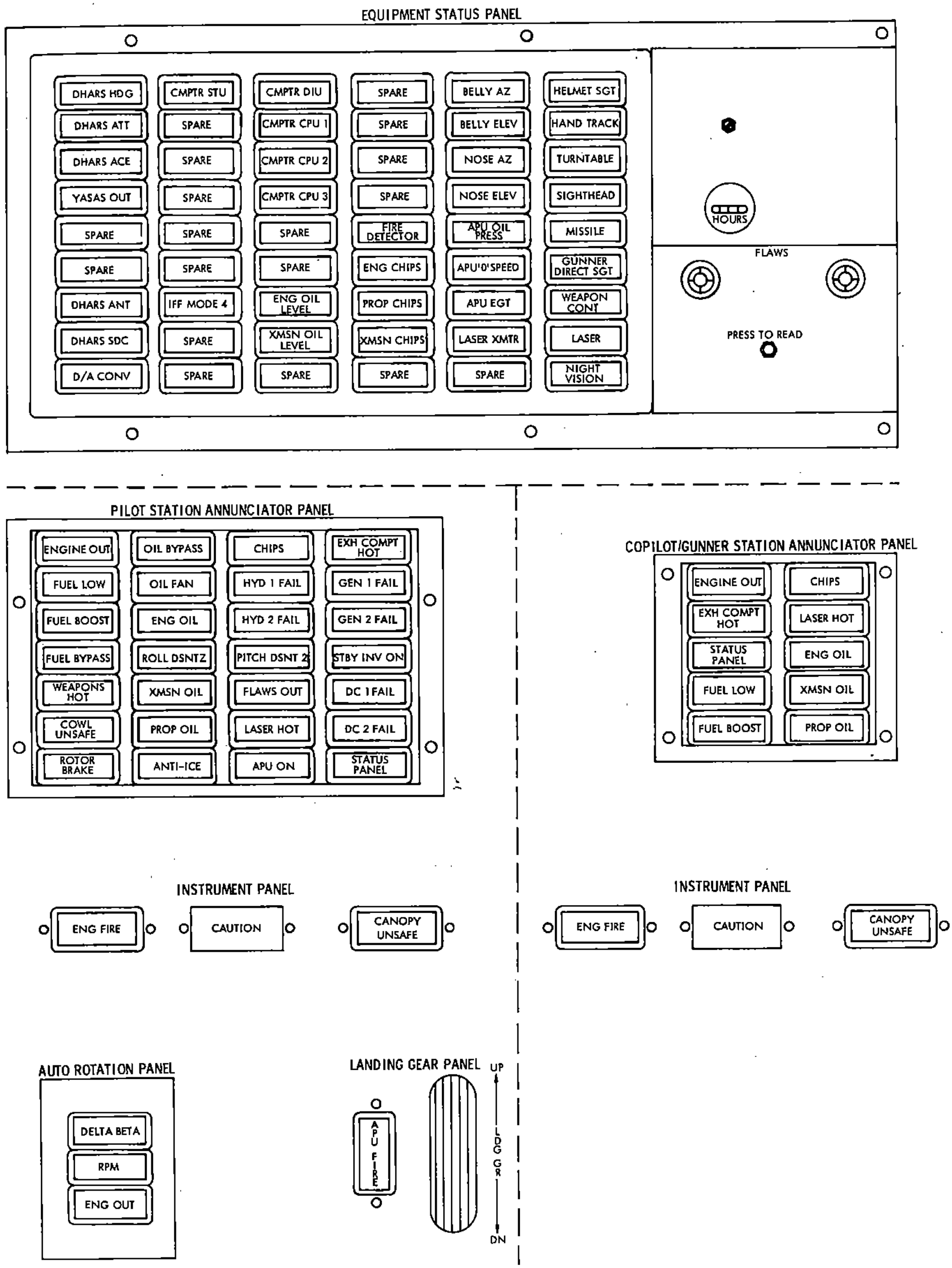
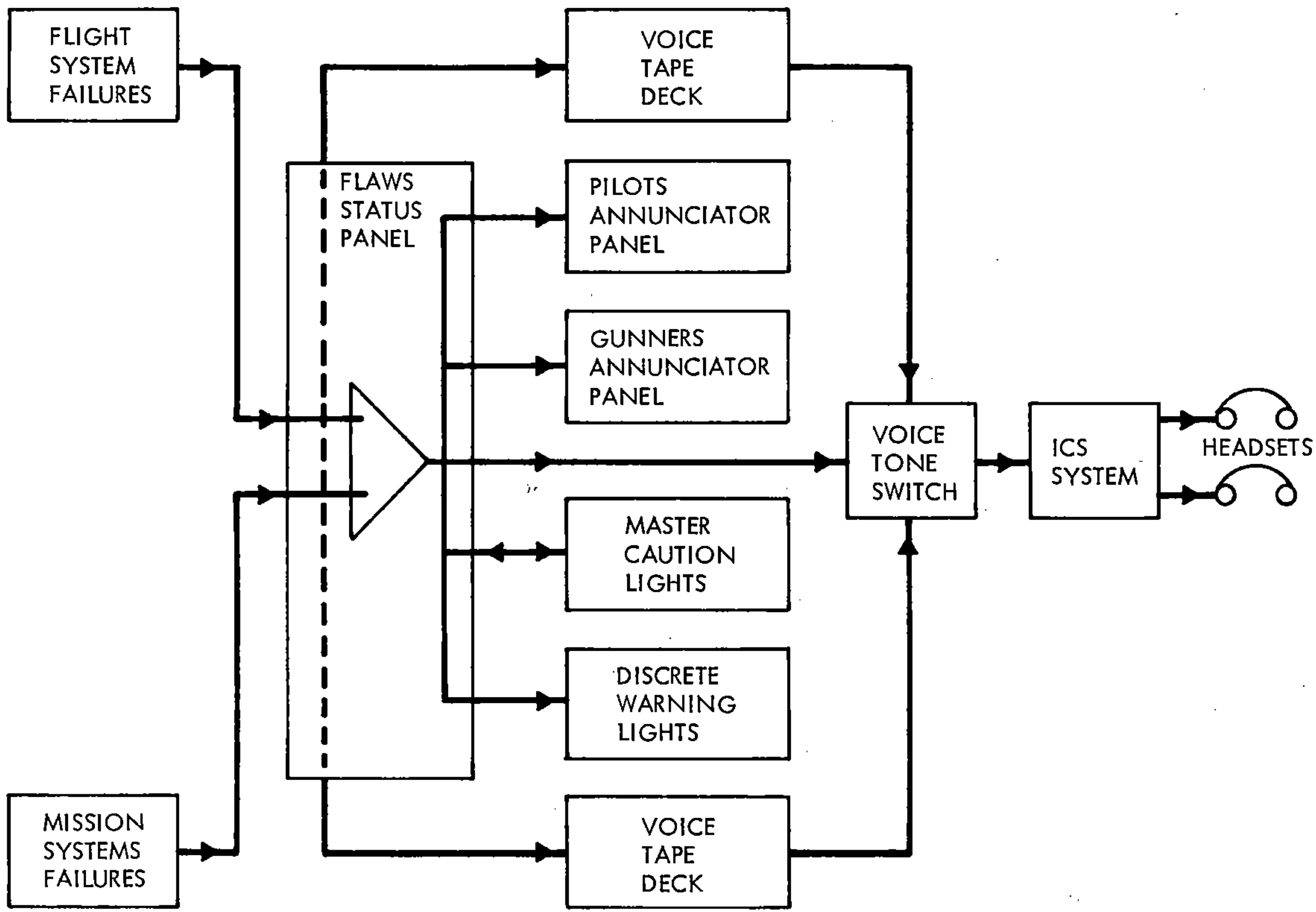


Figure 21A-1. Fault Locating Aural Warning System



FAULT LOCATION AURAL WARNING SYSTEM
VOICE WARNING SYSTEM

Figure 21A-2. Flaws

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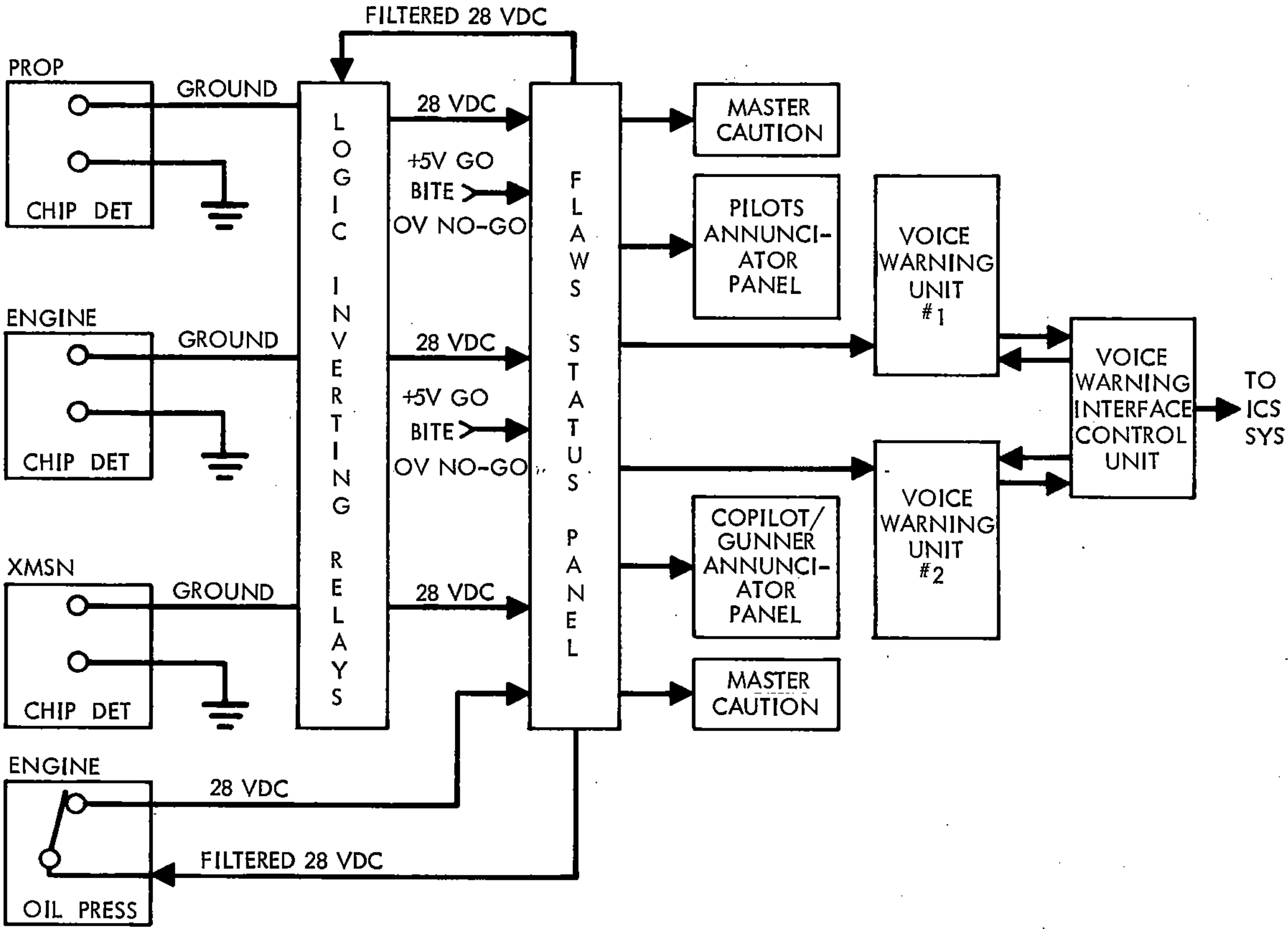

Figure 21A-3. Flaws Typical Failure Sensor and Signal Flow

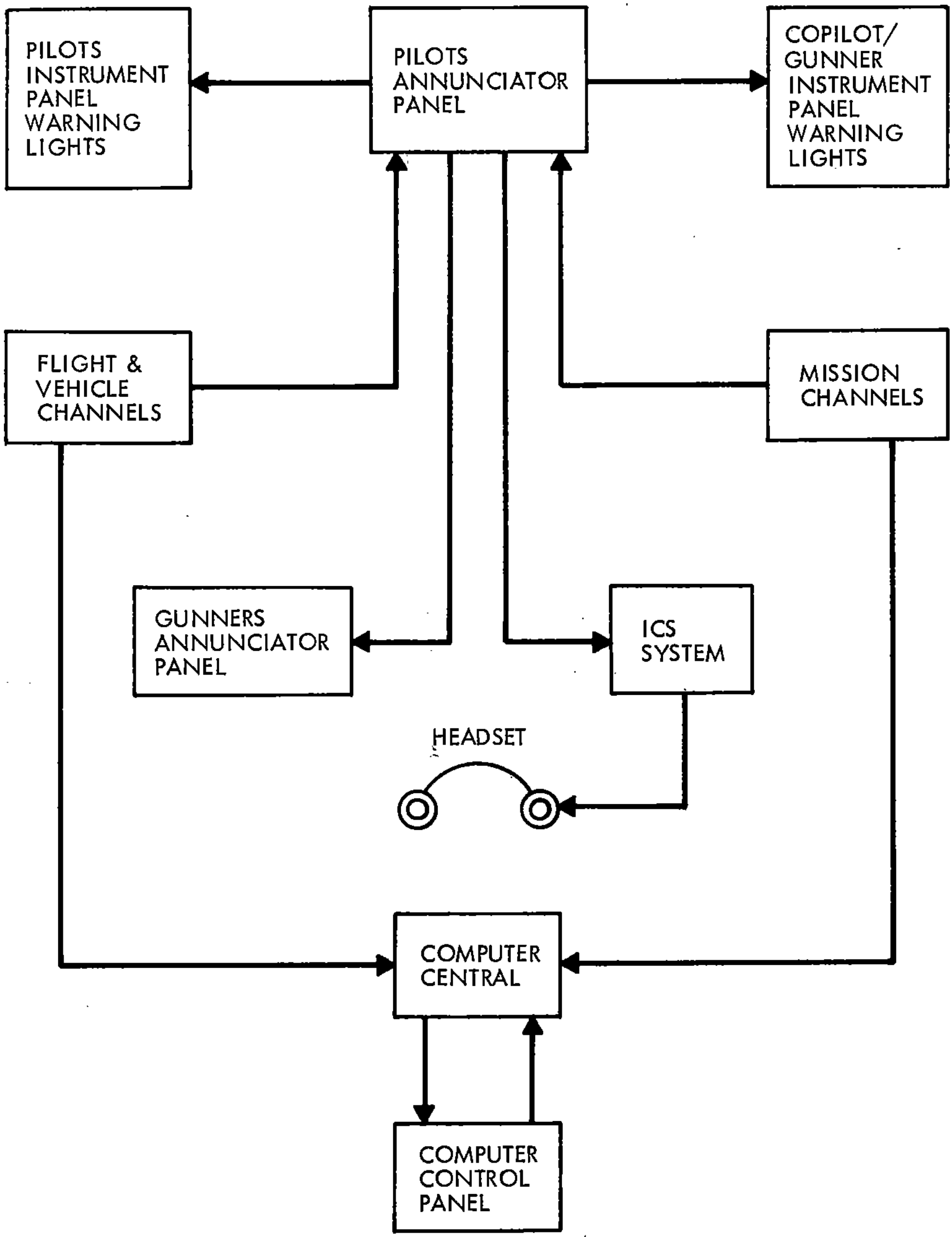

Figure 21A-4. PCRS Flaws

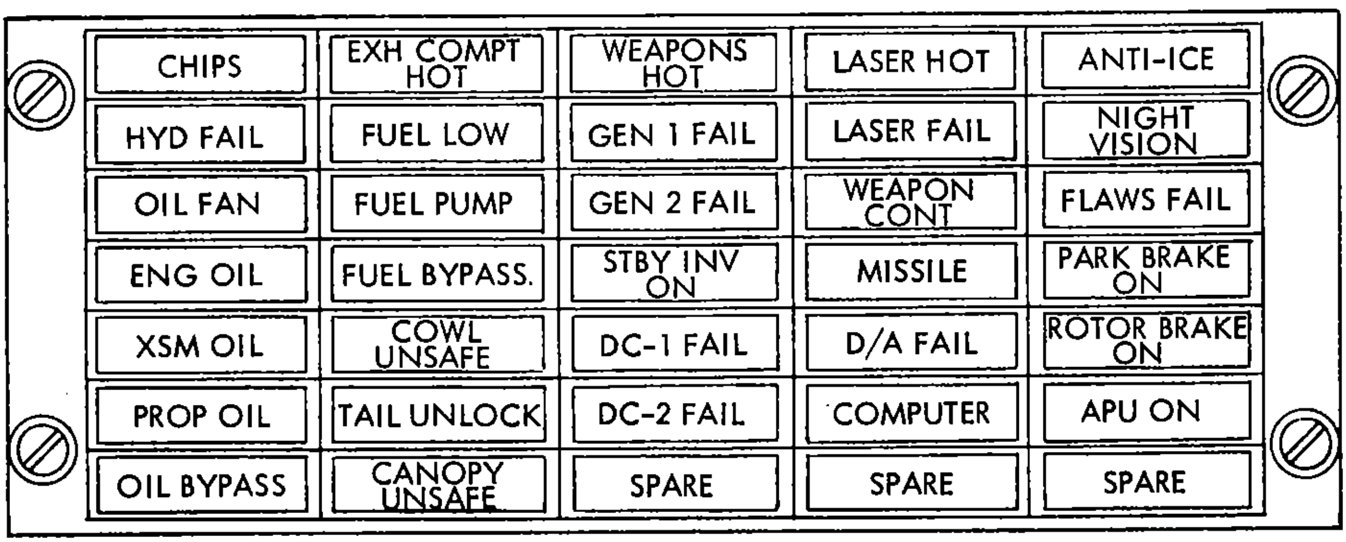

Figure 21A-5. PCRS Pilot's Annunciator Panel

FIRE DETECTION AND EXTINGUISHING

I. GENERAL DESCRIPTION

A fire and overheat detection system is provided to monitor the following three potential fire or overheat areas: the engine compressor (zone 2), engine combustion (zone 1), and the APU compartment. In the event of a fire or overheat condition, visual indication of the respective area is provided. The three monitored areas contain short discrimination circuits, and closed loop sensors provide a control signal even with an open circuit. All area monitors may be checked for operation from a central location. Fire extinguishing is provided, by an on-board pressurized bottle, to either engine or APU compartments.

II. COMPONENTS AND LOCATIONS

Name of Component	Number per Aircraft	Location in Aircraft
Fire Detection Radiation Sensor Elements	2	Engine Compartment
Fire Detection Radiation Control Unit	1	Avionics compartment
FLAWS (Fault Location and Aural Warning System)	1	Copilot/gunners station
Copilot/gunners Engine Fire Warning Light	1	Copilot/gunners station
Pilots Engine Fire Warning Light	1	Pilot's station
Fire Detector Test Switch	1	Pilot's right forward panel
Engine Vent Valve Switch	1	Aft avionics compartment
Engine Vent Valve	1	Aft avionics compartment

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Name of Component	Number per Aircraft	Location in Aircraft
Zone 1 Fire Detection Sensor	1	Engine compartment
Fire and Overheat Detection Control Unit	1	Avionics compartment
Copilot/gunners Exhaust Compartment Hot Warning Light	1	Copilot/gunners station
Pilot's Exhaust Compartment Hot Warning Light	1	Pilot's station
APU Fire Detection Sensor	1	APU compartment
Fire Extinguishing Bottle	1	Swash plate compartment
Fire Extinguishing Switch	2	One in each crew station

III. MAJOR COMPONENT DESCRIPTION

A. Fire Detection Radiation Sensor Elements

Both sensor elements, also referred to as pyrotector detectors, are identical. Each is a solid state photoconductive cell hermetically sealed in a stainless steel housing with a viewing cap of high temperature shock resistant glass. The sensor provides a signal to operate a control unit when a fire occurs.

B. Fire Detection Radiation Control Unit

The control unit is a transistorized control unit packaged to dimensions of 3-11/16" x 4-13/32" x 2-1/4". One receptacle provides electrical connections. The control unit provides signals to close a vent valve and illuminate the warning lights when either sensor detects a fire. The control unit provides operation to check all components for operation when test is selected. The control unit will remain operative if one of the sensors becomes shorted or open circuited. Failure of both sensors will not give a false indication of fire.

C. FLAWS

The FLAWS is interfaced with the fire and overheat system to provide voice warning when a fire occurs in zone 1 or zone 2, and to provide the signal to illuminate the warning lights if there is a fire in zone 2, or a fire or overheat condition in zone 1 or the APU compartment.

D. Pilot and Copilot/Gunner Engine Fire Warning Lights

Each light assembly contains two lamps. When illuminated, the red lens background permits the wording ENG FIRE to be read. Both lights are illuminated when a fire condition exists in zone 2.

E. Fire Detector Test Switch

This is a double pole, single throw push button type switch. It provides the means to initiate operation of the fire and overheat detection system without operation of interfaced emergency equipment.

F. Engine Vent Valve Switch

This is a double pole, center off, two position momentary toggle switch. It permits manual selection to check operation of the engine vent valve.

G. Engine Vent Valve

This is a 24 volt DC motor actuated butterfly valve. The motor is reversible for opening and closing the valve. Limit switches provide termination at each end of travel. The valve provides the means to shut off ventilation of zone 2 when a fire occurs.

H. Zone 1 Fire Detection Sensor

The sensor is an assembly consisting of two conductors embedded in semi-conductive ceramic and housed in an inconel tube approximately 10 feet in length. A plug type connector is attached at one end of the tube and a receptacle type at the opposite end. The tube is connected to ground and the center element forms a continuous monitoring loop. When the sensor or a portion of it is exposed to increasing temperature, the resistance of the semi-conducting ceramic decreases causing a drop in the center element-to-tube resistance. This signals the control unit to operate when a fire or overheat condition exists reducing element resistance to approximately 300 ohms. The sensor is installed on the firewall to sense a fire or overheat condition in the area of the engine combustion section.

I. Fire and Overheat Detection Control Unit
The control unit is a sealed assembly containing transistorized control circuitry to operate a warning light or lights if a fire or overheat condition exists in zone 1 or the APU compartment.

J. Pilot and Copilot/Gunner Exhaust Compartment Hot Warning Lights
Each light assembly is a component of the FLAWS and contains two lamps. It provides each crew member visual warning of a fire or overheat condition in zone 1 by illuminating the light. The lamps illuminate an amber lens background and the wording EXH COMPT HOT is visible.

K. APU Fire Detection Sensor
This sensor is an assembly like the zone 1 fire detection sensor but is approximately 8 feet in length. It is strategically routed and installed to monitor potential fire or overheat areas of the APU compartment.

L. Fire Extinguishing Bottle
This is a pressurized spherical bottle containing bromotriflouromethane as the extinguishing agent, under a minimum of 600 psi at $80^{\circ}F$. Release of the agent is accomplished by the use of an explosive squib. This is accomplished automatically with an APU fire or manually from either crew station in case of engine fire.

M. Fire Extinguisher Switch
This is a normally open cover guarded switch located in each crew station. It provides a 28 volt signal to fire the explosive squib in the line leading to the engine compartment.

IV. SYSTEM OPERATION

A fire and overheat detection system is provided to monitor three potential fire or overheat areas. The engine compressor area (zone 2) is monitored by a radiation sensor type system. In the event of a fire in this area, the pilot and copilot/gunner are visually informed of the condition, and ventilation air to the area will be shut off. Either crew member may manually operate the extinguisher for this area by actuating the fire extinguishing switch.

The engine combustion area (zone 1) and the APU compartment area employ continuous sensor loop systems to detect a fire or overheat condition of these two areas. Should a fire or overheat condition exist in zone 1, the pilot and copilot/gunner are visually informed of the condition. No extinguishing system is provided for this area. Should a fire or overheat condition exist in the APU compartment, the pilot is visually informed, the APU operation is terminated, and the extinguisher system is automatically discharged into the area.

V. PCRS CONFIGURATION

No change.

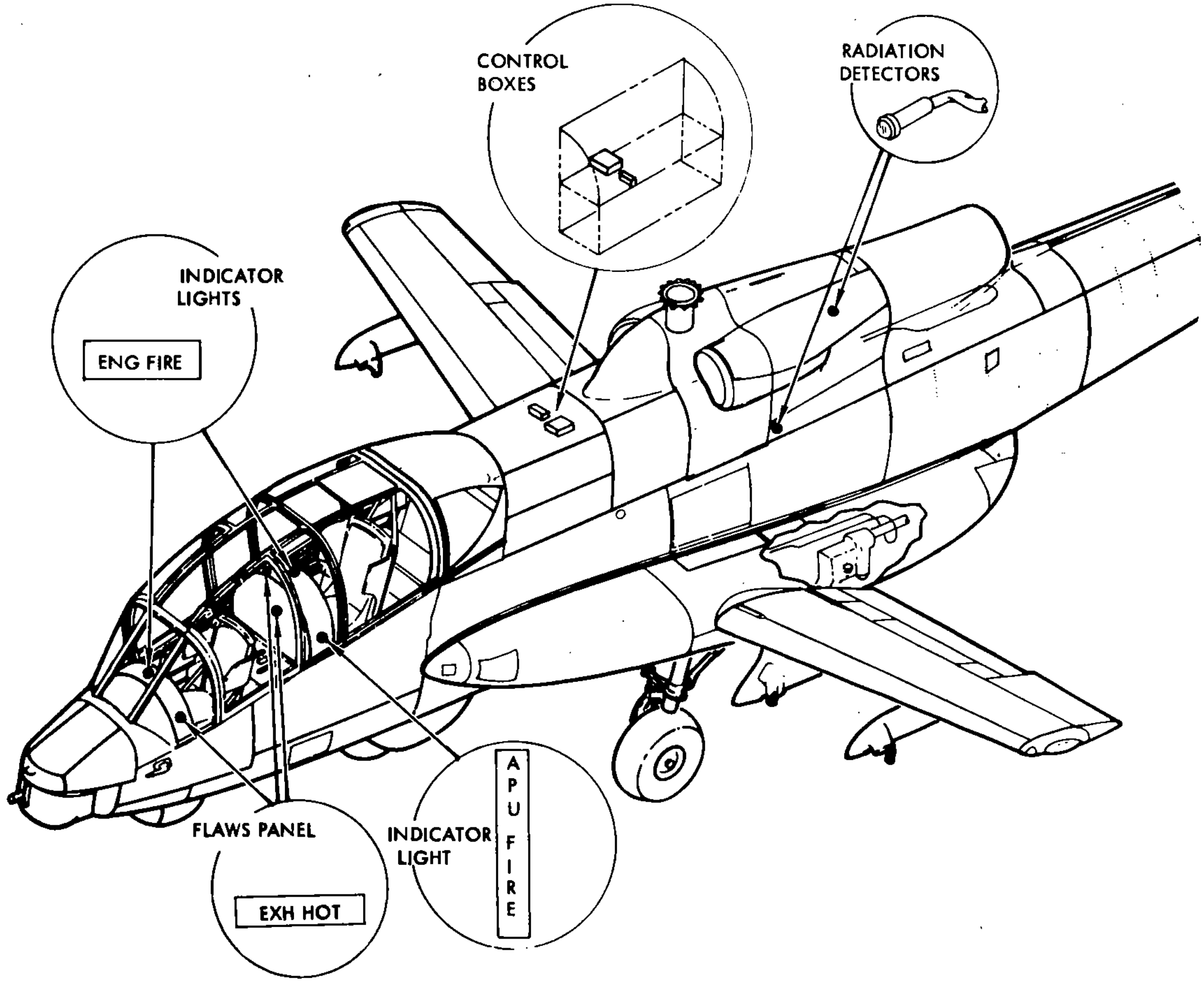


Figure 21B-1. Fire and Overheat Detection System Components

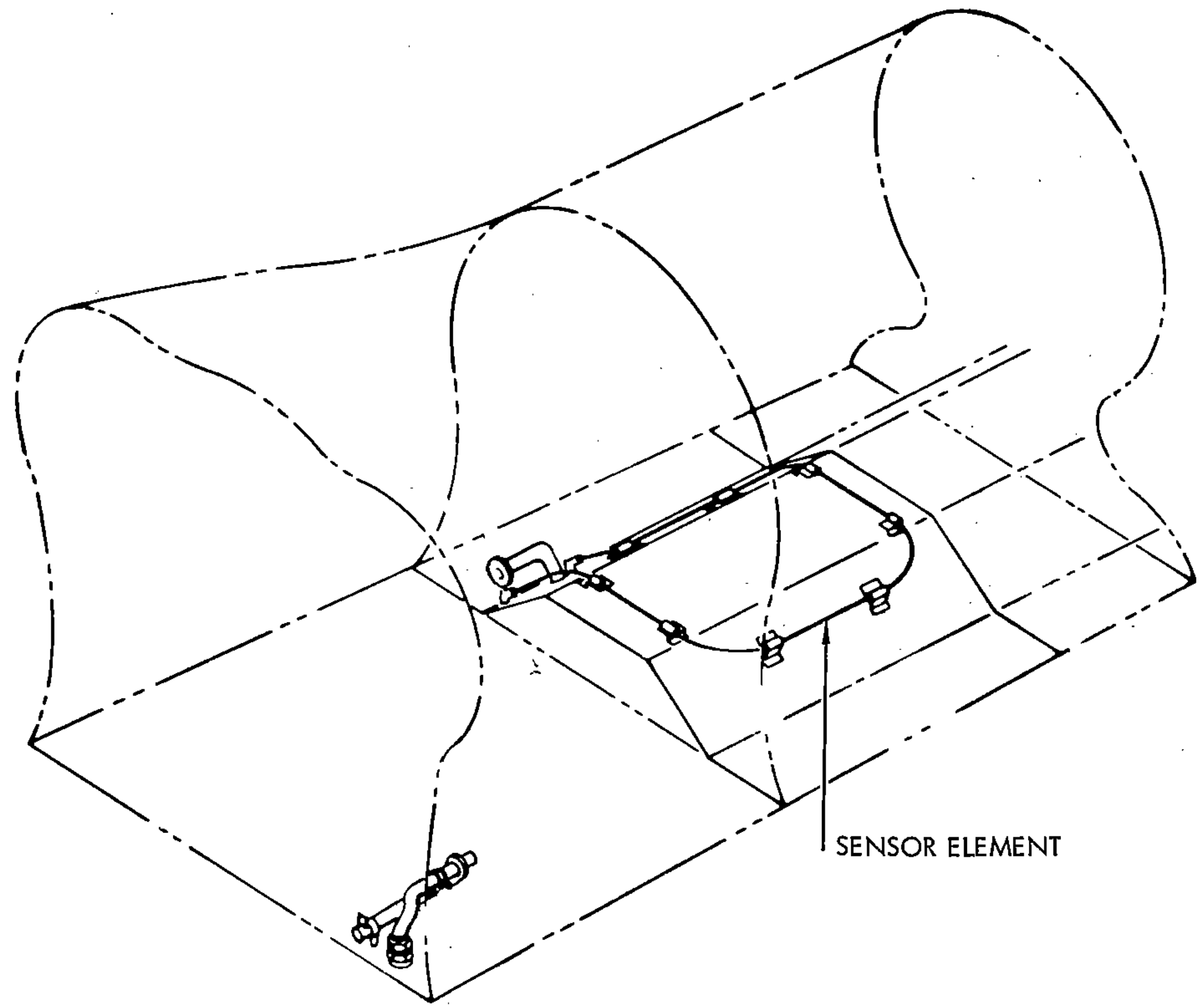


Figure 21B-2. Engine Fire Detection Element Installation

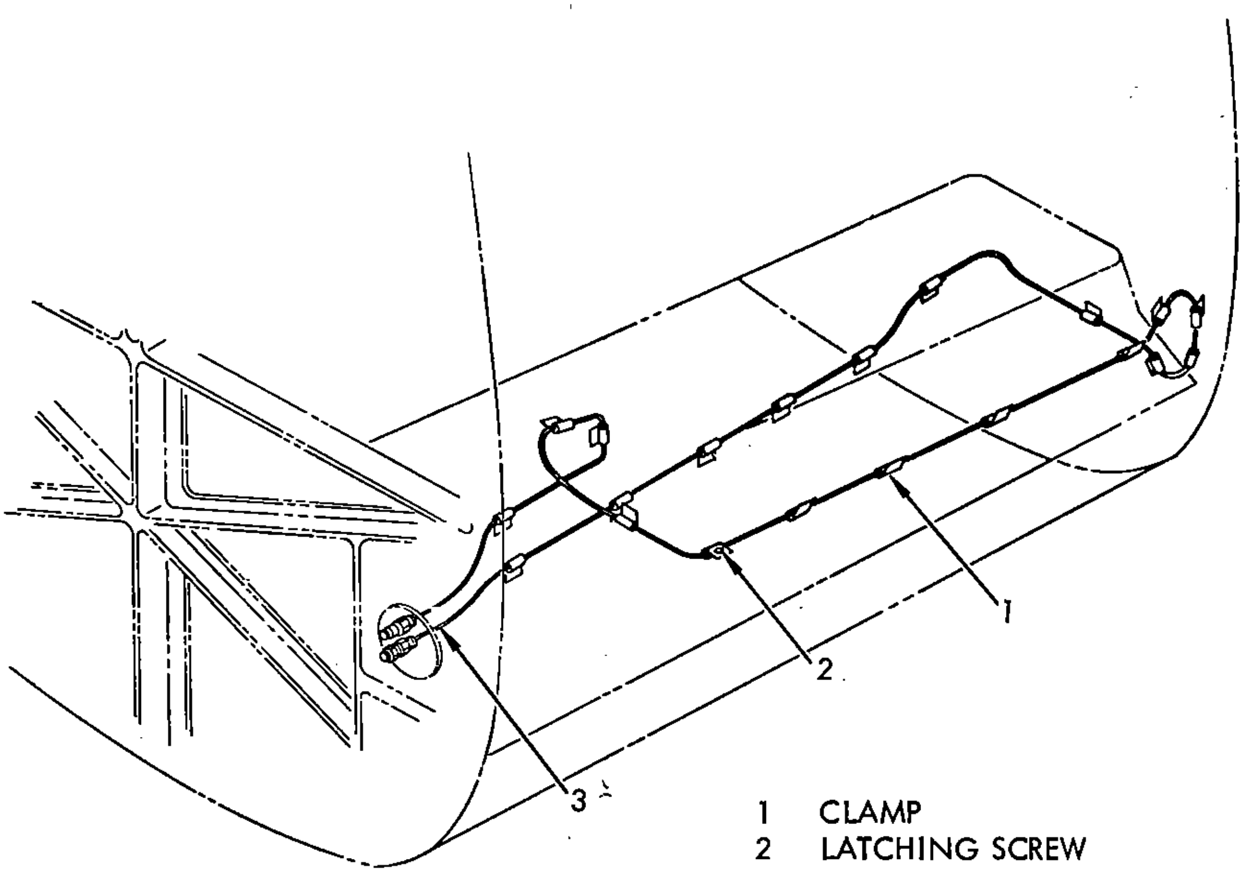
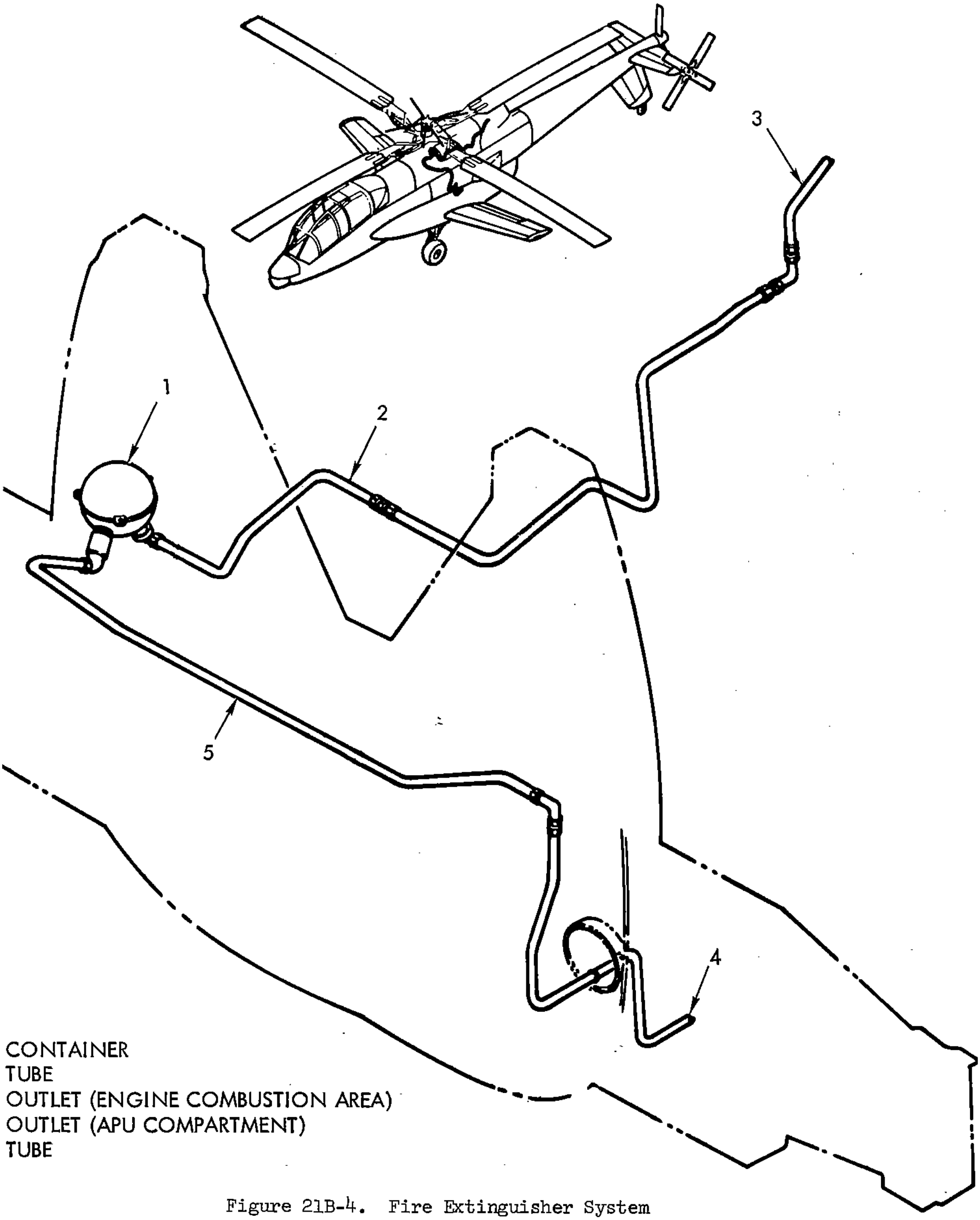


Figure 21B-3. APU Fire Detection Element Installation



1 CONTAINER
2 TUBE
3 OUTLET (ENGINE COMBUSTION AREA)
4 OUTLET (APU COMPARTMENT)
5 TUBE

Figure 21B-4. Fire Extinguisher System

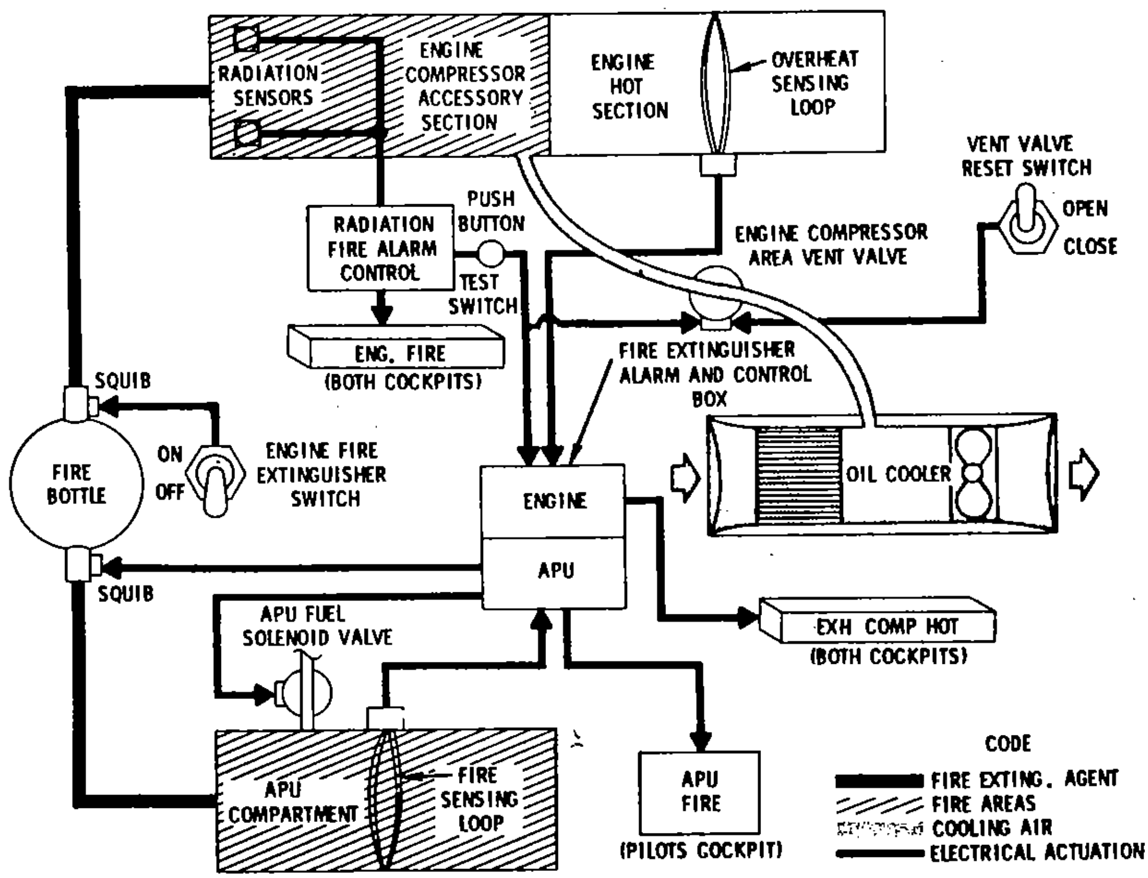


Figure 21B-5. Fire Extinguishing System

RELIABILITY AND MAINTAINABILITY PRESENTATION

I. AH-56A CHEYENNE - PROVEN RELIABILITY

The Cheyenne Reliability Program is unique as compared with previous programs, in that it provides a detailed method of assuring that developmental problems are corrected, thus preventing these problems from recurring on operational vehicles. These improvements in reliability reflect in fewer unscheduled maintenance actions and higher availability.

A. Reliability Defined.

Reliability is, in part, a mathematical science, by which systems may be designed for maximum mission accomplishments. This is highly important in space exploration systems where redundant hardware is necessary for the safe return of the crews. Redundancy is also important in inner space air vehicles for passenger safety and mission accomplishment.

The mathematical segment of reliability in calculating mission and system efficiencies as well as redundancies (dual alternators and hydraulic systems as an example) for the Cheyenne were accomplished prior to the beginning of the design, and were incorporated into that design.

These analyses indicated that the Cheyenne could successfully complete 94 of 100 long endurance (2 1/2 hours) escort missions without an abort, or in reliability terms,

Mission Reliability = 0.94

Further, 79 of these missions would be accomplished without an unscheduled maintenance action at the end of the mission, or

System Reliability = 0.79

The background statistical information required to generate these numbers was taken from data collected during several airplane reliability measurement programs or by careful analysis, using similar equipments of previous experience. The Cheyenne environment was a considered factor.

The measurement of reliability during the development program and design corrections to problems and the failures therein encountered, is the other side of the reliability coin -- the most important in weapons system development and system effectiveness.

Measurement during the development program indicates that the mature Cheyenne will achieve or even better these early mission and system reliability predictions.[1]

The following definitions will be helpful in understanding reliability:

Reliability:

> "The probability that an item will perform its intended function for a specified interval under stated conditions."

Primary Failure:

> "The item, while performing its specified function under intended operating conditions, incurred a failure which required unscheduled maintenance." In slang: "Committed suicide".

Secondary Failure:

> "A failure which is caused by the failure of an associated item". In slang: "Was murdered".

B. The Reliability Measurement Program

1. Conduct a reliability program in accordance with MIL-STD-785.

2. Establish a data collection system which provides for each component the total number of applicable test and operating hours, together with the number of failures experienced during those hours.

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1. See Bibliography

3. Analyze failures as to cause.

4. For those failures analyzed and identified as primary failures, develop an effective fix to prevent recurrence.

5. Upon mutual agreement between the Government and the Contractor, failures with effective fixes will be deducted from the chargeable failure count.

6. By computer simulation, based on time and failures collected during development tests, and using the long endurance mission criteria, solve the Cheyenne Reliability equations for both system and mission reliability.

C. Reliability Measurement Results

At the end of RDAT I, after 1,884 flight hours (GTV run time included) using the criteria agreed upon by the Government and Lockheed, shows that the Cheyenne will meet or better the 0.94 mission reliability and 0.79 system reliability.[1]

During this period (June, 1967 through December, 1971) over 18,000 individual pieces of data from all tests and all inspections were reviewed. From these, 1,682 primary failures were identified, and 1,426 documented corrective actions were developed for implementation. Of the 256 remaining residual failures, it is estimated that at least 40 percent will be corrected and these fixes incorporated into production aircraft. Those failures that then remain include:

- Parts lost, failure mode not determined.
- Vendor has not responded to analysis request.
- Vendor out of business or no longer in aerospace.
- Data not clear - background information lost.

In summation, based on reliability data collected through RDAT I, and with reliability measured in accordance with Government requirements, this measurement firmly supports a projection that the Cheyenne aerial weapons system will meet and may exceed those reliability goals established by the contract.

D. Reliability for Production

1. Continue the reliability program, in accordance with MIL-STD-785.

2. Assure that all corrective actions are implemented into the production design.

3. Review and assess designs to maintain or improve the demonstrated reliability.

4. Collect data on the initial vehicles and bench tests and correct discrepancies that will lead to reliability degradation.

5. Review Army RAMMIT or similar data and identify unreliability trends.

   a. Prepare data and recommendations as required.

6. Assist the Government in studies, failure analysis, tests, etc., as required to maintain high reliability.

E. Bibliography:

1. "Reliability Data Report, (Final)"
LCC Document No. ED296 dated 1972 February 21

2. "Reliability Handbook", W. Grant Ireson,
McGraw-Hill Book Co.

3. MIL-STD-785 "Requirements for Reliability Program"

4. "Reliability Program Plan"
LCC Document No. ED-106B dated 27 January 1967

5. MIL-STD-721 "Definitions of Effectiveness Terms......"

- <u>RELIABILITY:</u> THE PROBABILITY THAT AN ITEM WILL PERFORM ITS INTENDED FUNCTION FOR A SPECIFIED INTERVAL, UNDER STATED CONDITIONS
- <u>PRIMARY FAILURE:</u> THE ITEM WHILE PERFORMING ITS SPECIFIED FUNCTION UNDER INTENDED OPERATING CONDITIONS, INCURRED A FAILURE WHICH REQUIRED UNSCHEDULED MAINTENANCE. (SUICIDE)
- <u>SECONDARY FAILURE:</u> A FAILURE WHICH IS CAUSED BY THE FAILURE OF AN ASSOCIATED ITEM. (MURDERED)
- <u>SYSTEM RELIABILITY:</u> THE PROBABILITY THAT THE AH-56A WILL SUCCESSFULLY COMPLETE THE 2.5 HOUR LONG ENDURANCE ESCORT MISSION UNDER SPECIFIED FLIGHT CONDITIONS WITHOUT INCURRING A FAILURE IN ANY OF ITS SUBSYSTEMS REQUIRING UNSCHEDULED MAINTENANCE.
  (SYSTEM RELIABILITY = .79)
- <u>MISSION RELIABILITY:</u> THE PROBABILITY THAT THE AH-56A WILL SUCCESSFULLY COMPLETE THE 2.5 HOUR LONG ENDURANCE ESCORT MISSION UNDER SPECIFIED FLIGHT CONDITIONS WITHOUT ANY FAILURE IN ANY SUBSYSTEM/COMPONENT REQUIRED FOR THE MISSION.
  (MISSION RELIABILITY = .94)

Figure 22-1. Reliability Definitions

- CONDUCT A RELIABILITY PROGRAM IN ACCORDANCE WITH MIL-STD-785
- ESTABLISH A DATA COLLECTION SYSTEM TO SHOW TOTAL OPERATING HOURS AND FAILURES
- ANALYZE FAILURES AS TO CAUSE
- DEVELOP EFFECTIVE FIXES TO PREVENT RECURRENCE OF PRIMARY FAILURES
- UPON MUTUAL AGREEMENT WITH THE GOVERNMENT, DEDUCT FAILURES WITH EFFECTIVE FIXES FROM THE FAILURE COUNTS
- CALCULATE MISSION AND SYSTEM RELIABILITY BY COMPUTER SYNTHESIS

Figure 22-2. The Reliability Measurement Program

- DATA COLLECTED UNDER CONTROLLED CONDITIONS AND ANALYZED BY STRICT RULES
- CORRECTIVE ACTIONS DOCUMENTED FOR IMPLEMENTATION
- PROVE THAT THE CHEYENNE WILL MEET AND MAY EXCEED RELIABILITY REQUIREMENTS
- RESULTS OF HIGH RELIABILITY
    - o FEWER UNSCHEDULED MAINTENANCE ACTIONS
    - o HIGHER AVAILABILITY
    - o BETTER SYSTEM EFFECTIVENESS
    - o LOWER LIFE CYCLE COSTS

Figure 22-3. Cheyenne - Proven Reliability

* SYSTEM SIMPLIFICATION
* IMPROVED COMPONENTS

Figure 22-4. DHARS Reliability Development Activity

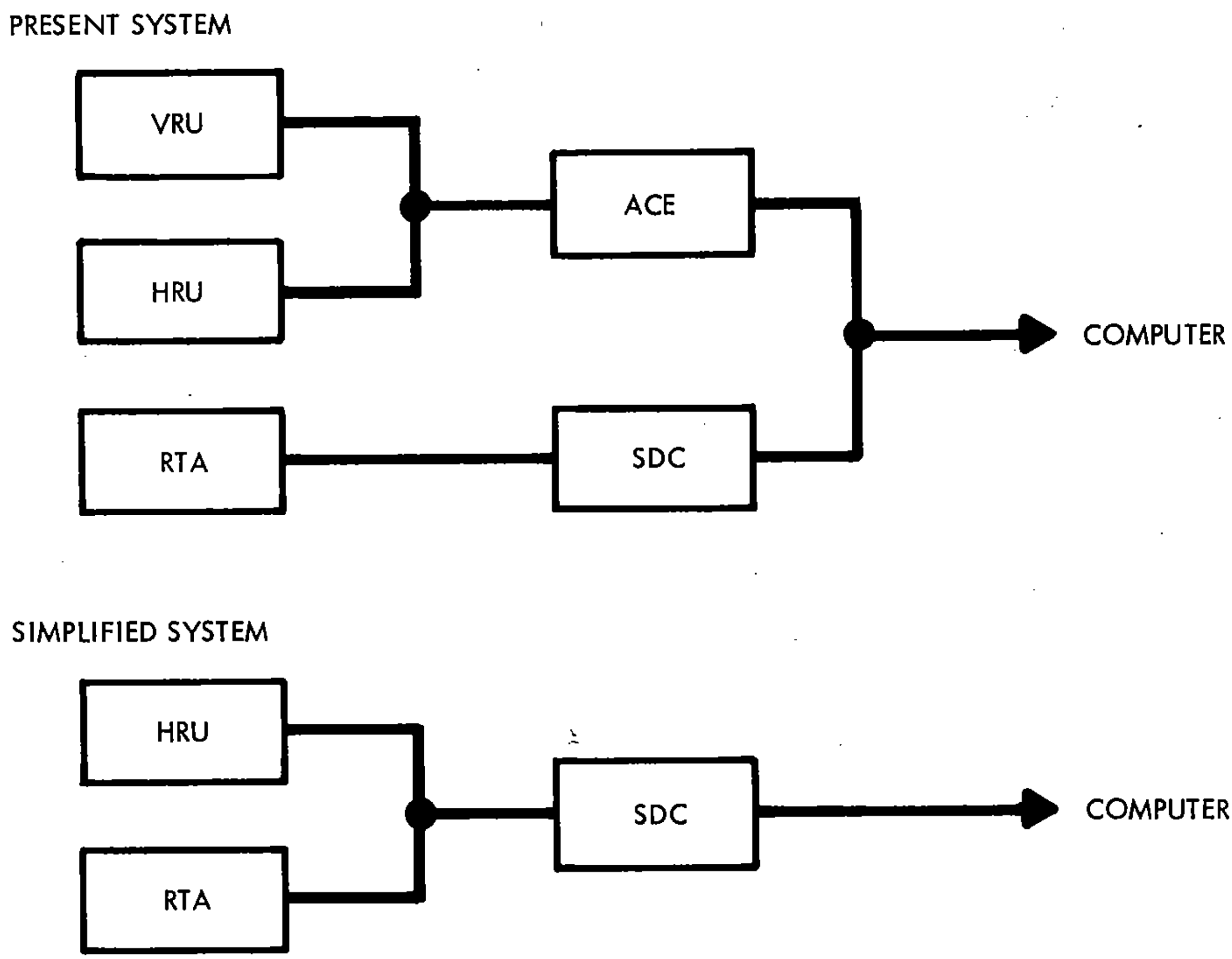


Figure 22-5. DHARS System Simplification

SUBSYSTEM		1ST GENERATION FAILURES/FLIGHT HOUR	2ND GENERATION FAILURES FLIGHT/HOUR
GUNNERS STATION		0.0870	0.0496
TOW CONTROL EQUIPMENT		0.0043	0.0017
CENTRAL PROCESSING UNIT COMPUTER CENTRAL		0.0143	0
WEAPONS	XM-51	0.1080	*
	XM-52	0.2870	*
NAVIGATION - DHARS		0.0965	*

*EQUIPMENT NOT AVAILABLE FOR TEST

Figure 22-6. Improved Reliability Through Development

RELIABILITY MODEL CODE	SUBSYSTEM	TOTAL FAILURES	TOTAL FIXES	TOTAL RESIDUAL
110000	Airframe	242	212	30
111000	Landing Gear	78	74	4
112000	Power Plant	103	95	8
113000	Transmission	201	198	3
114000	Rotors/Props	141	132	9
115000	Hydraulics	36	22	14
116000	Instruments	231	215	16
117000	Electrical Power	22	20	2
118000	Fuel	6	5	1
119000	Flight Controls	145	133	12
120000	Utilities	101	97	4
121000	Aux Power Unit	43	41	2
210000	Communications	72	65	7
211000	Navigation	74	35	39
212000	Radio Navigation	3	3	0
213000	Computer Central	92	21	71
217000	Special Instrum.	18	10	8
220000	Fault Location	22	10	12
310000	Pilots Sighting Systems	1	0	1
311000	Swiveling Gunners Stat.	48	37	11
312000	Stores Control System	3	1	2
TOTAL (GFM and CFM)		1682	1426	256

Figure 22-7. Failure Status Summary

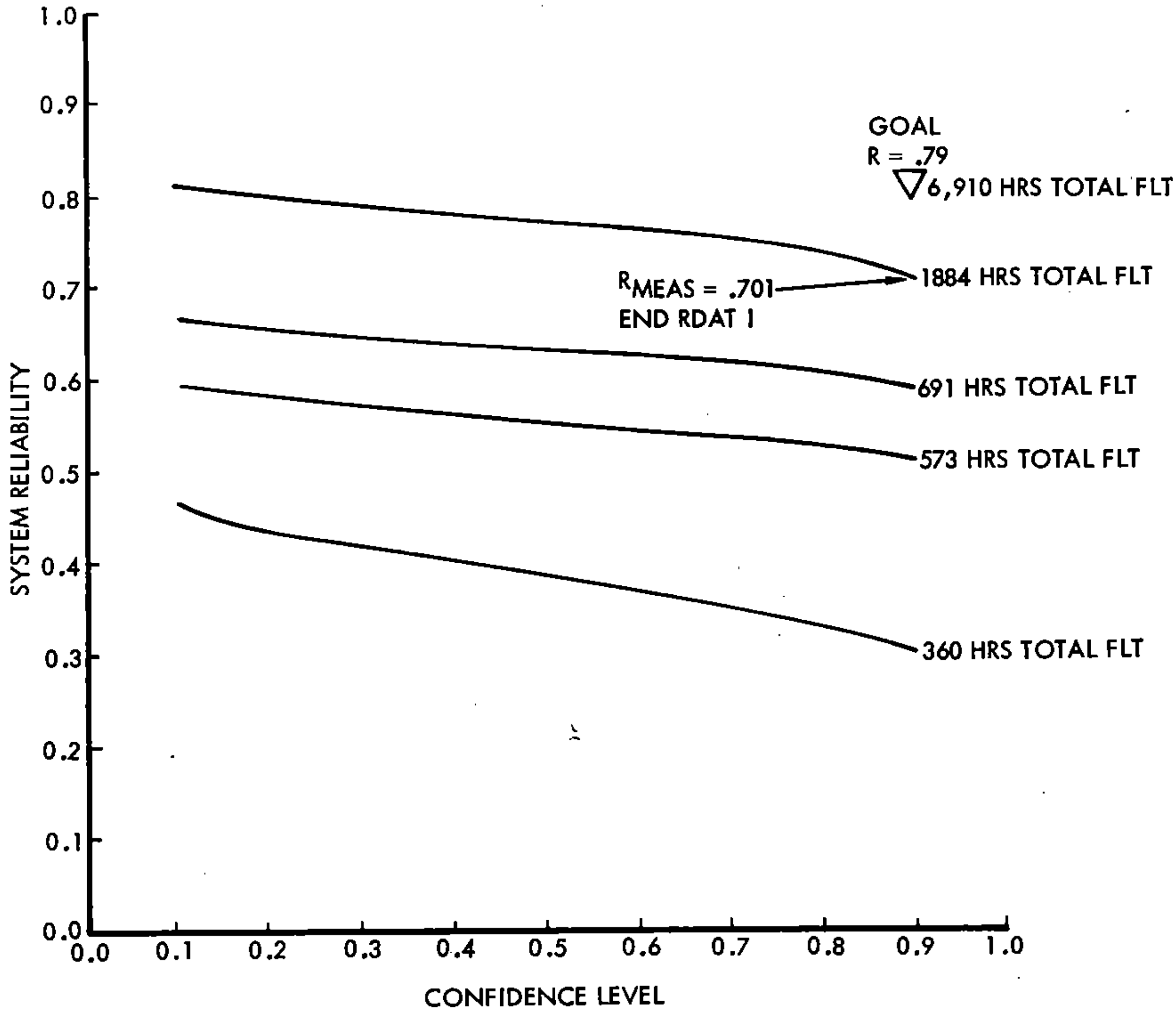


Figure 22-8. Reliability Growth by Total Flight Hours - System Reliability Values Vs. Confidence Level - GFM at Par

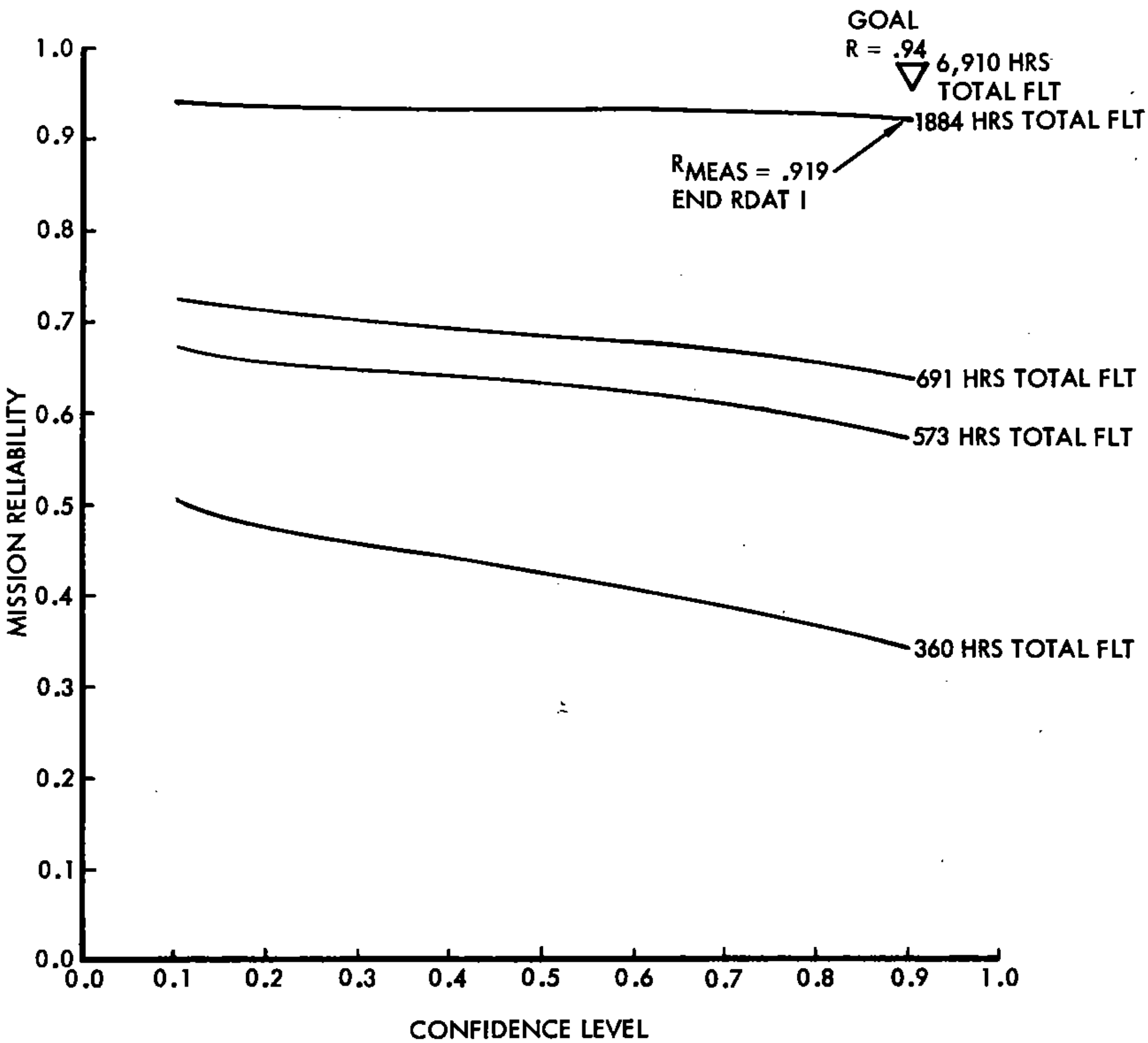


Figure 22-9. Reliability Growth by Total Flight Hours - Mission Reliability Values Vs. Confidence Level - GFM at Par

SYSTEM	TOTAL CHARGEABLE STOPPAGES	TOTAL DEDUCTED (FIXED)	TOTAL RESIDUAL (NOT FIXED)	MEASURED MEAN ROUNDS TO STOPPAGE	GOAL MEAN ROUNDS TO STOPPAGE	TOTAL ROUNDS FIRED
XM-51 40 MM FEED SYSTEM, CONTROLS, & TURRETS.	40	26	14	2,895	7,510	40,530
XM-52 30 MM FEED SYSTEM, CONTROLS & TURRETS	34	22	12	2,786	9,550	33,432
XM-53 7.62 MM FEED SYSTEM CONTROL & TURRETS	31 -	16	15	10,062	14,400	161,003

Figure 22-10. Weapon System Reliability Summary (Less Guns & Ammo)

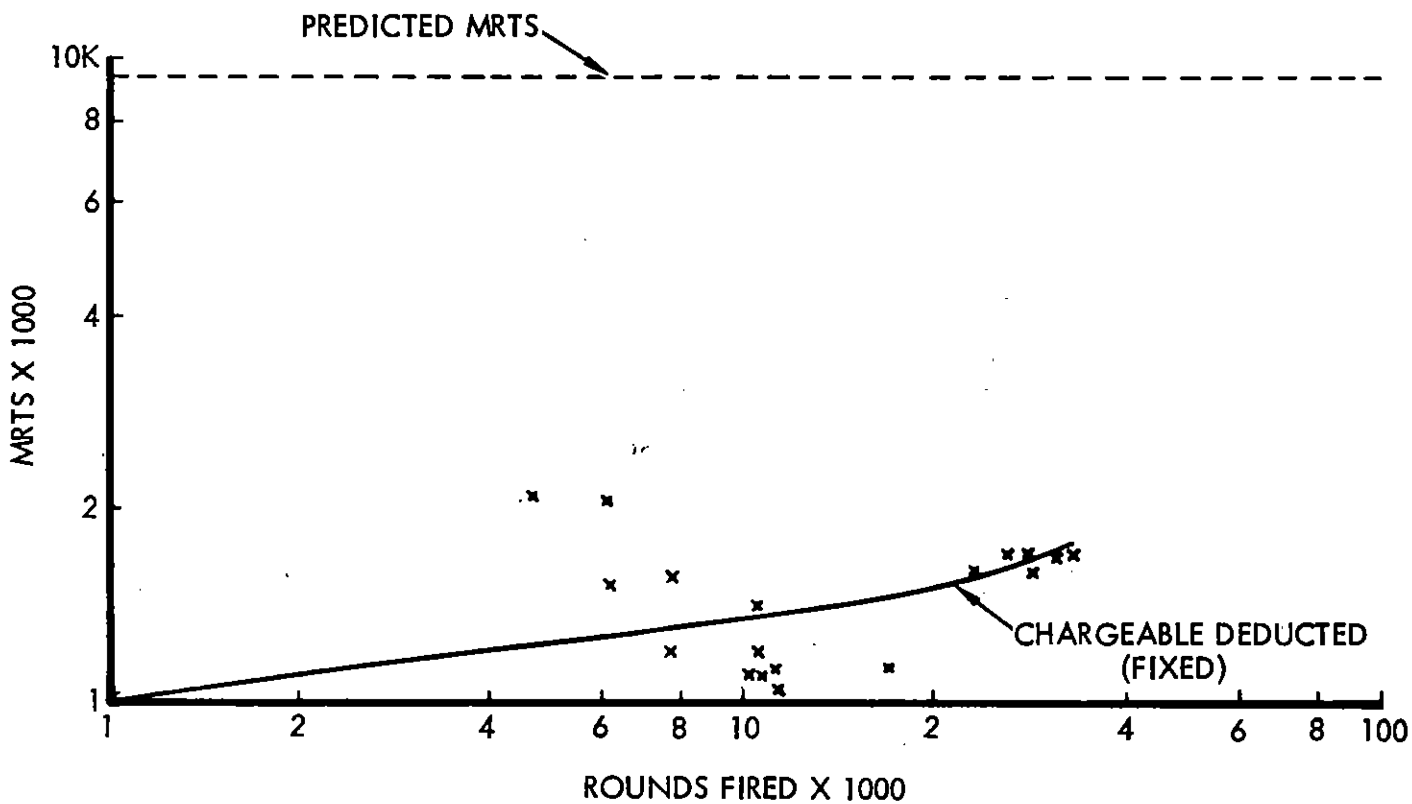


Figure 22-11. Mean Round to Stoppage Measurement (MRTS) XM 52 (30 MM)

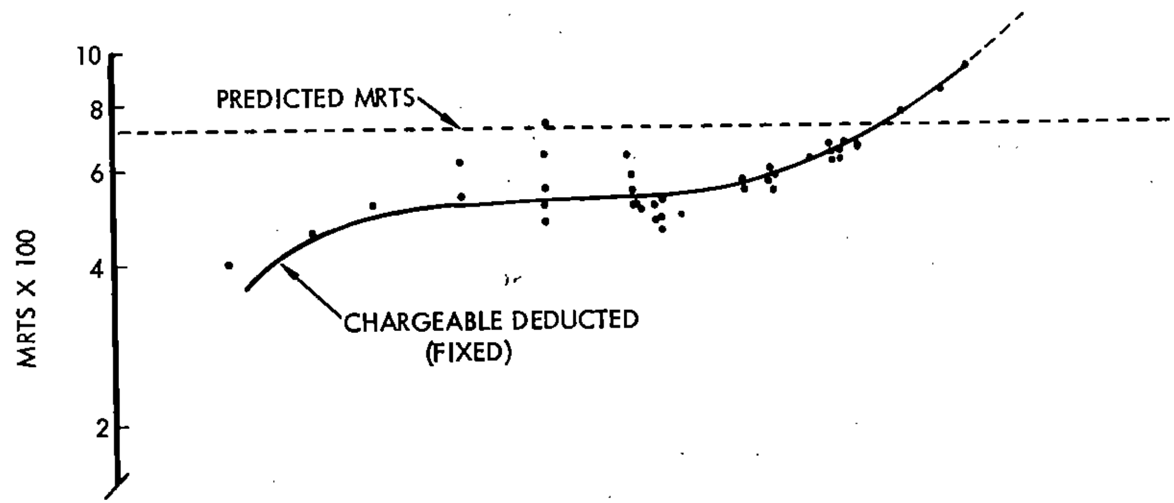


Figure 22-12. Mean Round to Stoppage Measurement (MRTS) XM 51 (40 MM)

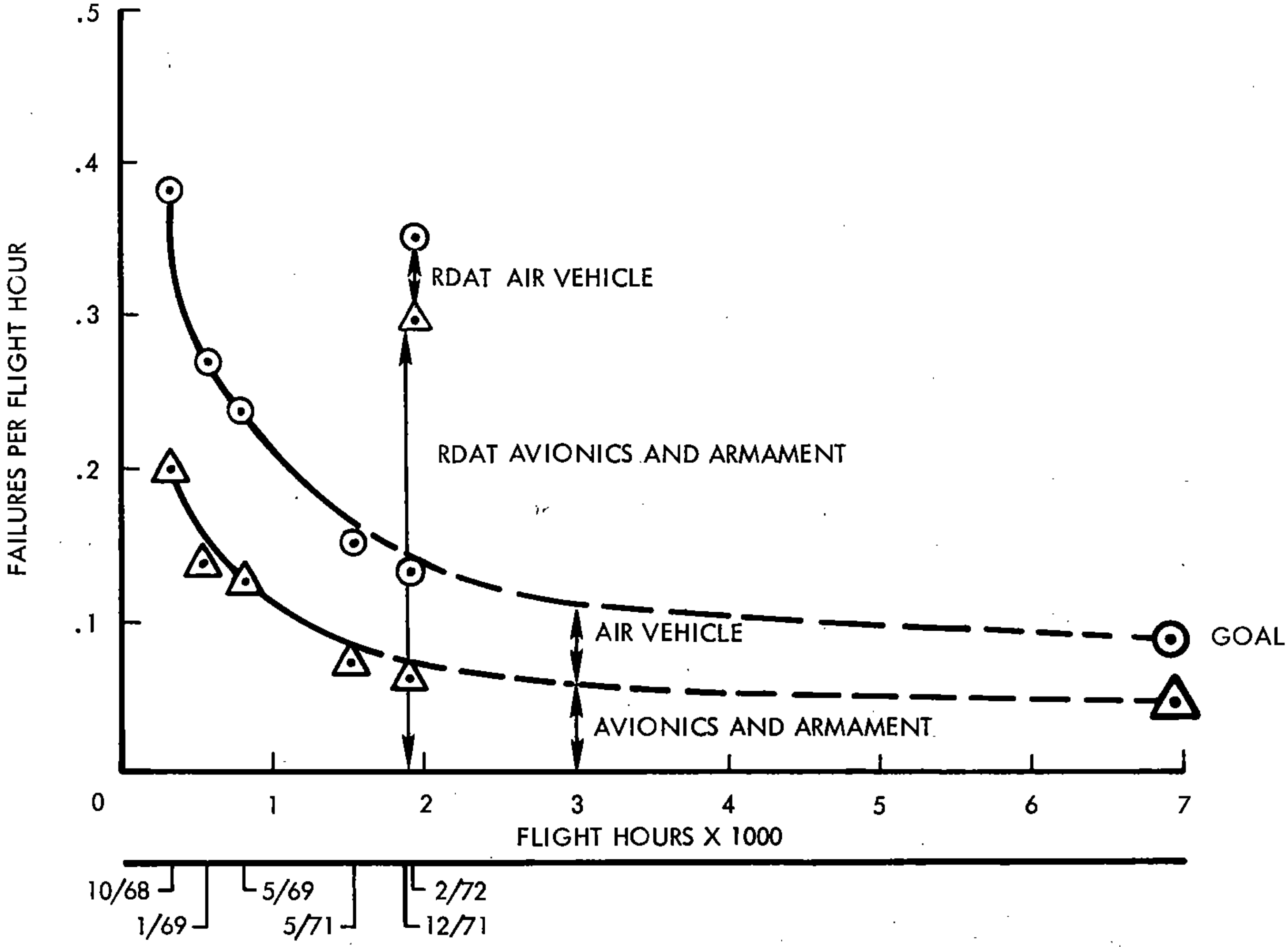


Figure 22-13. Reliability Improvements

- CONTINUE THE RELIABILITY PROGRAM PER MIL-STD-785
- ASSURE THAT ALL CORRECTIVE ACTIONS ARE IMPLEMENTED INTO THE PRODUCTION DESIGN
- REVIEW AND ASSESS DESIGNS TO MAINTAIN OR IMPROVE THE DEMONSTRATED RELIABILITY
- COLLECT DATA ON THE INITIAL VEHICLES AND BENCH TESTS, AND CORRECT DISCREPANCIES THAT DEGRADE RELIABILITY
- REVIEW ARMY RAMMIT OR SIMILAR DATA AND IDENTIFY UNRELIABILITY TRENDS
    - o PREPARE DATA AND RECOMMENDATIONS
- ASSIST THE ARMY IN STUDIES, FAILURE ANALYSES, TESTS, ETC., TO MAINTAIN HIGH RELIABILITY

Figure 22-14. Reliability for Production

II. AH-56A CHEYENNE - PROVEN MAINTAINABILITY

The AH-56A Cheyenne promises far better performance and weapons effectiveness than ever before attained by any comparable system. Perhaps even more important is the "Breakthrough in Maintainability" that has been achieved through maintenance engineering efforts so that the Cheyenne can be maintained by Army crew chiefs and technicians at organizational level, using existing Army tools and equipment, and with minimum impact on the Army's Logistics System.

During design and development of the Cheyenne, maintainability has been considered equal in importance to performance, payload and weight. Every aspect of the configuration reflects this careful consideration of maintenance requirements and available maintenance resources.

The QMR Requirement of 9.0 MMH/FH productive time for all maintenance levels and 80 percent operational availability are rather stringent maintainability requirements. These are not easy requirements for an aircraft with the Cheyenne's capabilities.

In order to satisfy them Lockheed has, since the start of preliminary design, had a top flight team of maintainability engineers working directly with designers on a day-by-day, drawing-by-drawing basis. Every installation drawing has had maintainability approval.

The maintainability engineers established design ground rules based on contractual goals and constantly reviewed all aspects of the design for conformance with these criteria. Some of the more important guidelines were:

- The Cheyenne must be completely self sufficient for operation in the Army's forward austere environment.
- Organizational maintenance shall require no tools other than those included in the standard Army Mechanics Tool Kit.
- Special support equipment requirements shall be avoided by designing-out requirements for such equipment and by deliberately designing the Cheyenne to use existing Army equipment.

- Components shall be designed so that no maintenance or servicing is required except at a 300 hour periodic inspection interval. Minimum time between overhaul for components shall be 1200 hours and minimum component life shall be 3600 hours - three times that of present day equipment.

Similar requirements for maintainability were included in specifications for all suppliers, and supplier performance is under close surveillance.

Specifications and requirements and analysis do not necessarily create a maintainable machine. But in the case of the Cheyenne the proof is in the actual hardware which has been undergoing tests at Lockheed's Van Nuys and Rye Canyon Facilities, Yuma and Hunter-Liggett.

The degree to which the maintainability engineers have been effective is evident in the following system-by-system descriptions and illustrations of the features and characteristics that drastically reduce the maintenance requirements of the Cheyenne. It is these features and many more that will enable Lockheed to meet the Army maintainability requirements.

A. Ground Handling

Ground handling of the aircraft is conventional and requires only standard support equipment already in the Army inventory.

Towing is normally from the rear wheel using a standard tow bar, but lugs on each main gear strut permit forward towing using tow cable or rope.

Jack points are built into the wing (left and right) and the fuselage nose section. With standard jacks installed at these points, the entire aircraft can be jacked for leveling or for landing gear maintenance. A built-in jack is provided at each wheel to permit wheel, tire and brake maintenance. A leveling plate and plumb bob attachment point built into the fuselage at Station 455 permit leveling in all axes.

Towing lugs can also be used as mooring points. Tie down rings can also be installed at the wing and fuselage jack points. The standard Army mooring kit can be used for tie down.

Built-in provisions for a screw jack device permit easy blade folding by a mechanic using a "Speed Handle" from the standard Army mechanics tool kit. With one blade folded forward and one blade folded aft, the Cheyenne can be hangared or parked in minimum space.

The reversible thrust propeller provides unique taxiing capability.

B. Servicing

Fuel, oil and hydraulic systems are all replenished from the right side, leaving the left side completely free for rearming and other maintenance activity. Fueling is by single point gravity flow. A fuel gage is located adjacent to the servicing point so that fuel quantity is always visible to the servicing crew.

The engine, main transmission, and propeller gearboxes all use the same type of oil. Each reservoir is equipped with an oil level sight gage, a chip detector and a separate fillerwell.

A built-in pump and filter unit permits replenishment of hydraulic oil directly from standard quart cans. This procedure eliminates the contamination associated with use of funnels, fillerwells, etc.

On the entire aircraft only two fittings require periodic greasing. These are at the two drive shaft hanger bearings and they must be greased only at the 300 hour periodic inspection interval.

C. Accessibility

Accessibility is almost synonymous with maintainability and the Cheyenne provides outstanding access to all service centers and compartments. Approximately 60 percent of the exterior surface of the fuselage is devoted to access doors and panels. All doors are hinged, have hold-open devices, and are secured by quick-operating latches or fasteners.

Separate compartments or service centers are provided for electrical equipment, avionic equipment, engine accessories, engine, swashplate, gun debris and fire control avionics, forward ammunition drum, aft ammunition drum, control servos, and communications equipment. Components common to each system are grouped within the appropriate service center to permit easier trouble-shooting and maintenance. Service centers are painted white inside and are lighted.

In addition to access doors for major compartments, seventy eight removable panels provide access to various components, cables, connectors, pulleys, turnbuckles, etc.

The sponsons on each side of the fuselage can be used as work platforms with space enough for several mechanics and their tool boxes. The sponsons provide ideal access to the upper service centers, to the engine and transmission, and for inspection of the main rotor hub, blades and gyro.

A built-in ladder is provided at the aft end of each sponson. The left ladder is manually operated from the ground, while the right ladder is operated by an electromechanical actuator which operates in conjunction with the landing gear. If the gear is up, the ladder is retracted, and if the gear is down, the ladder is extended.

A built-in platform along the right side of the forward fuselage is used for entrance to the cockpit. It also serves as a work platform for maintenance in and around the cockpit area.

D. Structure

Structure is semi-monocoque. Skin, stringers, bulkhead rings, and ribs are made of common alloys. Most repairs can be completed in the field using available materials and skills.

Major structural assemblies are designed to be replaceable. The aft fuselage section is attached by a series of bolts through a bulkhead ring and quick disconnects on cables and fluid lines allow quick removal of the aft section for repair, replacement, or if necessary, to facilitate crash retrieval.

Wings are replaceable assemblies. Wing structure is simple since there are no ailerons, flaps or other movable surfaces.

Each of the empennage surfaces is attached by four bolts so that they can be replaced easily in event of damage.

E. Landing Gear

The main landing gear is of conventional design, retracting aft into the sponson on either side. The tail wheel is faired into the contour of the vertical fin. Struts are conventional air-oil type. All joints rotate on self lubricated bearings that require no grease fittings. However, a grease fitting is installed at each wheel bearing so the bearing can be completely flushed with grease after operation in extremely dusty areas.

Main struts, drag struts, and most other main gear components are designed to be interchangeable left to right in order to minimize logistics requirements. The 29 x 11-10 tires permit operation from unprepared soft ground having a CBR of 2 1/2.

F. Power Plant

The T-64 engine is completely exposed for maintenance when the cowl is rolled aft. The cowl is secured in place with two interconnected latches. One mechanic, working from either side, can unlatch and slide back the cowl.

Uniball mounts, fluid and electrical quick disconnects, a slip-fit firewall and other design features permit engine replacement in 45 minutes. A portable hoist installed in suitably located structural hard points can be used for handling the engine, as well as main rotor blades, hub, and transmission.

A specially designed bank of "Donaldson" filters will remove 85 percent of all dust and dirt particles over 40 microns in size from engine inlet air. This bank of filters will drastically reduce guide vane and compressor blade erosion and other engine damage caused by sand and dirt encountered when operating from austere forward areas.

G. Auxiliary Power Unit

A built-in auxiliary power unit drives the complete accessory section of the main transmission upon which are mounted the aircraft's two 20 KVA alternators, one of the two hydraulic power packages, and the transmission oil pump. Thus, when the APU is operating, electrical and hydraulic power is available for trouble shooting and checkout of the electrical, avionics, hydraulic and flight controls systems.

Through the hydraulic system the APU provides power for engine starting, and since the APU drives the transmission oil pump, the transmission is pre-oiled prior to every start up.

The APU is installed adjacent to the left wheel well and is serviced and inspected through the wheel well opening. An access opening provided in the sponson skin above the APU permits use of a hoist cable for lowering or raising the APU for maintenance or replacement.

H. Main Transmission

Many features have been designed into the main transmission to make both scheduled and unscheduled maintenance easier. All input and output shaft seals are split so that replacement can be made on the flight line with no disassembly. Provision is made for visually checking gear patterns without disassembly. Parker seals are used so the need for gaskets and "O" ring seals is obviated. Oil passages are designed for easy cleaning. A chip detector is installed as a warning device. The accessory gearbox and forward part of the transmission are accessible through the accessory compartment. With skin panels removed, the complete transmission and its four attachment bolts are fully exposed.

I. Rotors

The Lockheed rigid rotor is inherently simple. There are no flapping hinges, no lead-lag hinges, no dampers and no grease fittings. Each of the four blades is attached to a movable hub section by two bolts. One of these bolts is a special expansion type that can be removed easily for blade folding. The movable hub is attached to an arm of the fixed hub by two fully exposed, self-lubricated, replaceable hinge-type feathering bearings, and a tension-torsion pack that is also replaceable.

Blades are prebalanced and tracked. A vernier adjustment device in the pitch link permits accurate indexing so that blades can be interchanged providing the replacement blades have the same usage history.

Cyclic and collective control rods are located inside the 12-inch diameter main mast where they operate in an oil mist environment; where they are not vulnerable to battle damage; and where they cannot be used as hand-holds by mechanics.

The tail rotor is a conventional 4-blade articulated system. Blades are interchangeable, and, as in the main rotor, there are no grease fittings. Feathering bearings are oil lubricated with a reservoir and sight gage for each blade.

All operating components are designed for at least 1200 hours between overhaul, and the blades, hub, gyro, etc., are all designed for at least 3600 hours of fatigue life.

J. Propeller

The Hamilton Standard Thrust Propeller has been substantially simplified and employs a modular design approach which dramatically reduces maintenance times. The blades, control, and actuator can be replaced separately. Replacement can be made on the flight line by mechanics with ordinary skills using only the tools found in the standard Army Mechanics tool box.

Propeller blades incorporate a quick disconnect feature that permits a blade to be replaced in five minutes. Only a screwdriver is required.

The blade has a steel core with a fiberglass shell that can be easily repaired without removing the blade from the aircraft and without rebalancing.

The propeller control can be replaced as a module in 15 minutes. The actuator module is a sealed, self-contained package containing an oil supply, hydraulic pumps and distributor valve. It can be completely replaced in 10 minutes without loss of oil, with no special tools, and no subsequent adjustments are required.

The complete propeller and gearbox are mounted to the aft section by four bolts. It can be replaced in 1 man-hour.

Other maintenance features unique to the Cheyenne propeller include an oil level sight gage visible from the ground, a gearbox port for viewing gear mesh, externally accessible oil pumps, and a split, individually replaceable gearbox shaft seal.

The Cheyenne propeller has approximately one-third the parts of a conventional propeller, is considerably lighter, and more reliable, and requires about 10 percent as much maintenance.

K. Hydraulic System

Two completely separate hydraulic systems are used for redundancy and safety. The heart of each system is a hydraulic power package that includes pump, pressure regulator, reservoir, quantity gage and filter in a single package. The power packages are clamp mounted to the accessory drive pads for easy replacement.

A specially designed filler device, previously described, prevents contaminants from entering the system and thus eliminates the cause of most hydraulic system maintenance problems.

A ground hydraulic power unit is not necessary since the built-in APU drives one of the aircraft hydraulic power packages to develop pressure for system checkout.

L. Flight Controls

Only conventional helicopter controls plus a propeller pitch control (BETA) are necessary. No rudder, elevator or ailerons are required. The primary element for control of the Lockheed rigid rotor system is the control gyro. No Stability Augmentation System (SAS) is needed, so maintenance problems associated with this system are nil.

Collective and cyclic servos are identical and are mounted as a package below the swashplate, accessible through a large door in the fuselage lower skin. Servos can be replaced individually or the three servos can be replaced as a complete package. Hoisting provisions are built in to facilitate replacement of the complete package.

The yaw servo for tail rotor control is mounted on the forward face of the horizontal stabilizer beam. With the leading edge of the stabilizer removed, the servo is completely exposed.

M. Avionics

The Cheyenne has a number of communications systems and has remarkable navigation capabilities. Historically, maintenance of avionic equipment of this nature has consumed a high percentage of total maintenance time. Furthermore, identification of the faulty unit (trouble shooting) has been the avionic maintenance task requiring the most skill and the most man-hours.

Achievement of a low overall maintenance man-hour per flight hour ratio for the Cheyenne necessitated a unique approach that would drastically reduce maintenance and trouble shooting requirements for the avionic systems. This improvement has been achieved by designing the avionic equipment to meet the following criteria:

- To the extent feasible, equipment is micro-modularized so that it is smaller, lighter, and far more reliable.

- It is possible to immediately isolate any failure to a Line Replaceable Unit (LRU).

A status panel in the cockpit indicates whether or not a system is operating properly. If a system is not operating properly, it is then necessary to identify the faulty Line Replaceable Unit (LRU). For some components a failure is obvious or visible. For most components, however, BITE (Built-In Test Equipment) within the component provides an automatic go-no go indication of the components ability to function properly. If the component fails to operate properly, a white dot or pattern appears on the BITE indicator. The failure indication remains even after power is removed from the aircraft so the mechanic need not start the engine or apply electrical power, but can immediately identify the failed unit.

Fire control and armament systems have had the same degree of attention given these systems to make sure that they can be properly maintained even under the most difficult operating conditions.

N. MEADS

An integral part of the Cheyenne Maintainability Program prior to production cancellation in May 1969 was a comprehensive maintenance analysis required by the Army's Maintenance Engineering Analysis Data System (MEADS). The analysis, the first conducted under Army contract, identified organizational and direct support maintenance requirements and the resources required to complete the task such as personnel, spares and repair parts, ground support equipment, data, and facilities. We were in process of identifying general support and depot requirements when the program was terminated. The MEADS data generated was mechanized into a computer tab run and data was inputted periodically and reported to the Army. The computer tab runs provided an excellent tool for prediction and tracking of such maintainability parameters as maintenance man-hours per flight hour (MMH/FH), availability and scheduled/unscheduled man-hours. The last computer tab run accomplished prior to termination of MEADS effort was dated May 1969. This tabulation listed a MMH/FH figure for the total vehicle of 9.4 at all levels of maintenance. At the time there were design improvements on the drawing board that reduced this figure significantly below the current QMR requirement. Some of these improvements have been incorporated into the development aircraft, such as Phase III Rotor System, reverse rotation tail rotor and sprag clutch, and others are proposed for production incorporation such as the P/CRS transmission cockpit improvements.

Although the MEADS effort was discontinued, Lockheed has continued to feed information into the computer data band by recording all component failures that have occurred during the flight test program. In addition, stop watch monitors have recorded organizational and direct support maintenance task times during Army flight testing and selected contractor testing. The results of these efforts are being reported to the Army in accordance with reliability and maintainability contractual reporting requirements. Besides providing reliability and maintainability engineers with excellent factual data which can be compared to previous analyses, a history of components/areas requiring improvement or redesign is established for future reference. The P/CRS study has utilized these data to a large extent in modifying or redesigning components/subsystems to further improve overall system reliability and maintainability.

- NO LEAD-LAG HINGES, DAMPERS, OR FLAP RESTRAINING DEVICES
- NO LUBRICATION REQUIRED
- PRETRACKING & MASTER BALANCING PERMIT ROTOR BLADE INTERCHANGE-ABILITY AT ORGANIZATIONAL LEVEL
- 3600 HOURS FINITE LIFE
- EYE LEVEL ACCESSIBILITY FOR HUB AND BLADE INSPECTION
- MINIMUM INSPECTION REQUIREMENTS THROUGH USE OF CORROSION-RESISTANT MATERIALS IN MAIN BLADE & HUB STRUCTURE - TITANIUM HUB, STAINLESS STEEL STRUCTURAL AND MULTIPLE LOAD PATH MEMBERS
- LOWER COMPONENT STRESSES AT HIGH SPEEDS BECAUSE OF RIGID ROTOR COMPOUNDING - RESULTS IN LONGER LIFE

Figure 22-15. Lockheed Rigid Rotor System

ARMY UH-1A, B, C &
EARLY D
TEETERING
MAIN ROTOR HEAD

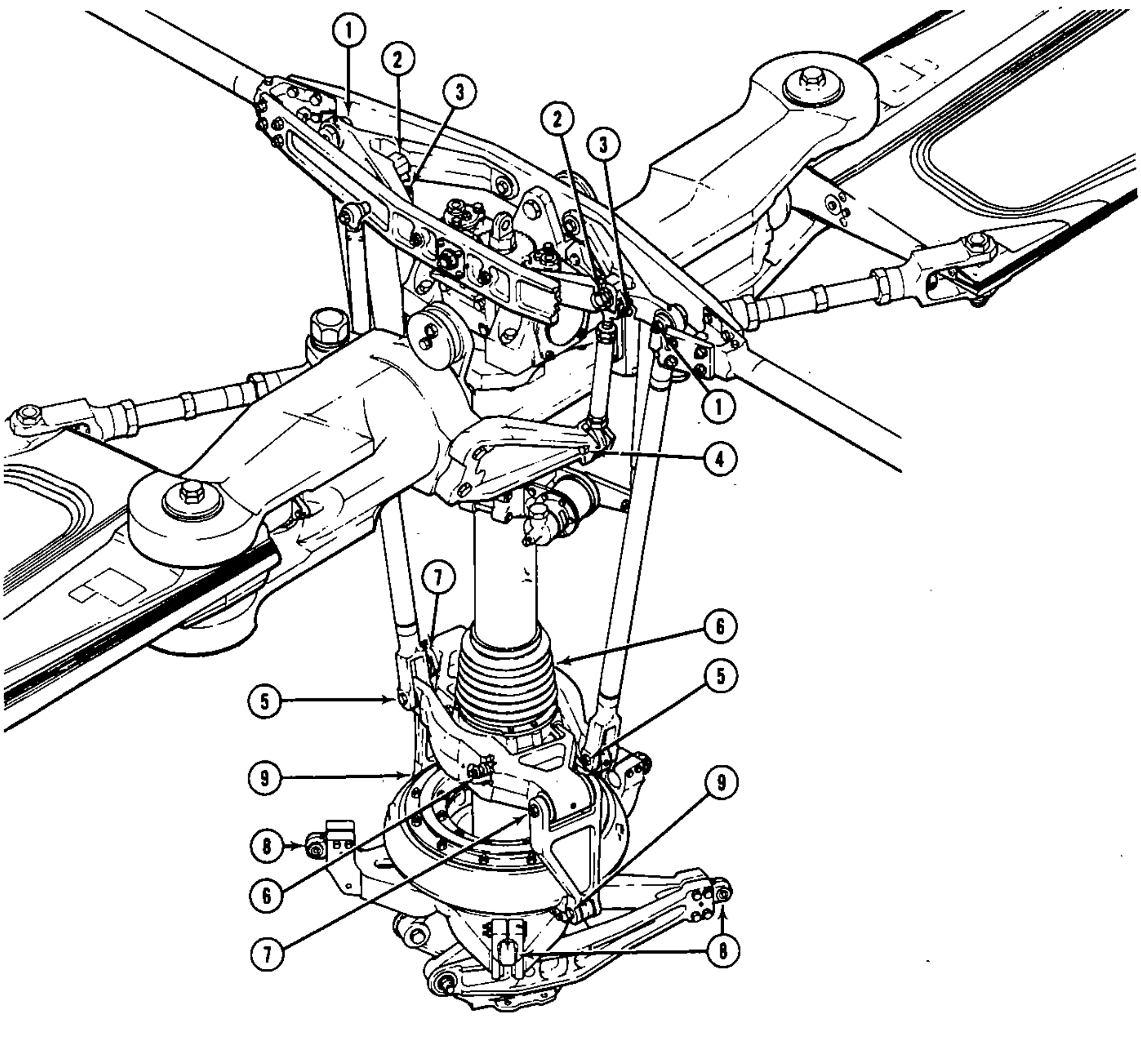
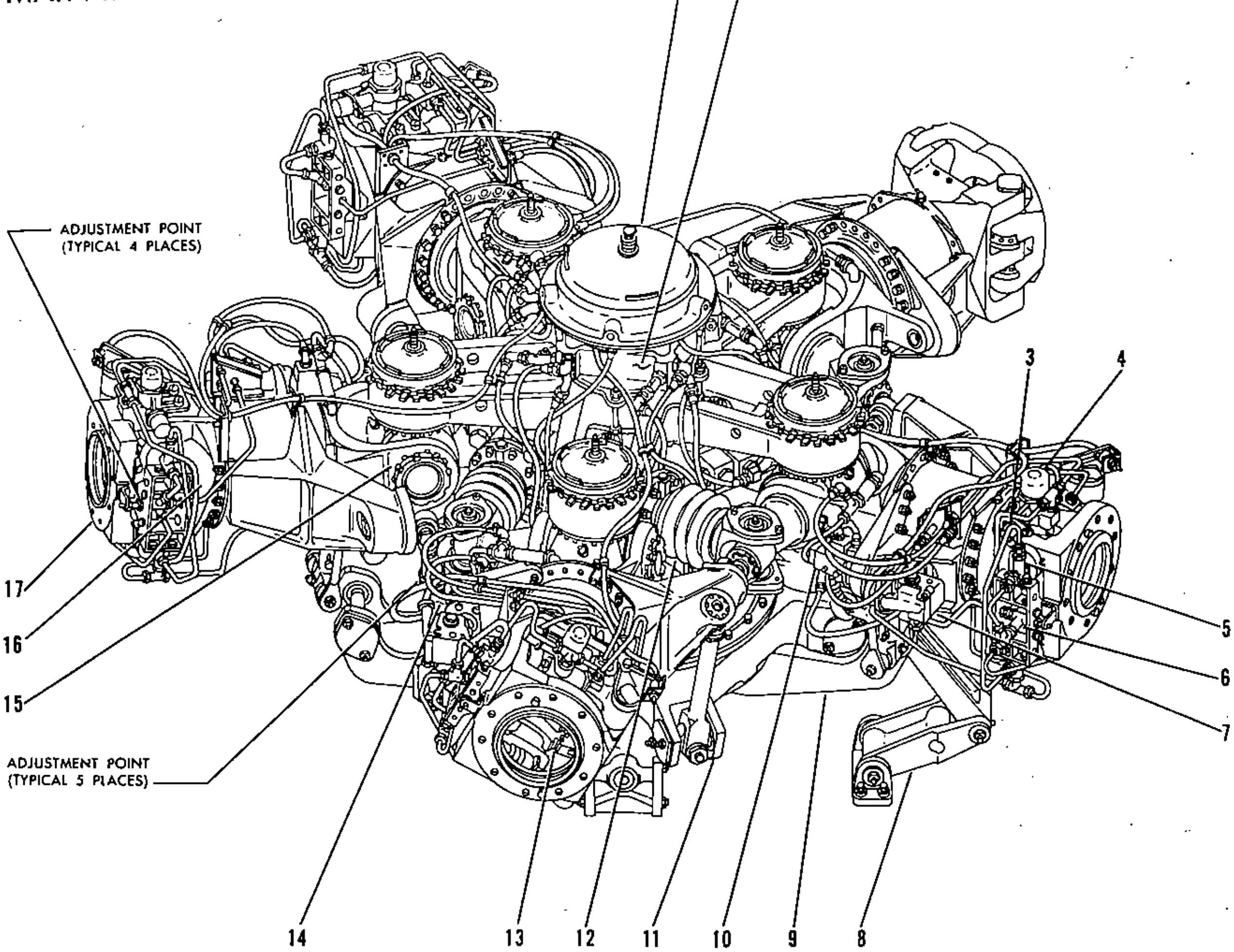


Figure 22-16.

LOCKHEED
CALIFORNIA COMPANY

NAVY SH-3A FULLY ARTICULATED MAIN ROTOR HEAD



1. Fluid Tank
2. Manifold
3. Blade Lockpin Limit Switch
4. Blade Lock Cylinder
5. Blade Lockpin Sequence Valve
6. Blade Spread Sequence Valve
7. Control Lock Cylinder
8. Stationary Scissors
9. Stationary Swashplate
10. Control Lock Sequence Valve
11. Rotating Swashplate
12. Damper-Positioner
13. Sector Gears
14. Adjustable Pitch Control Rod
15. Spindle
16. Sleeve
17. Hinge

Figure 22-17.

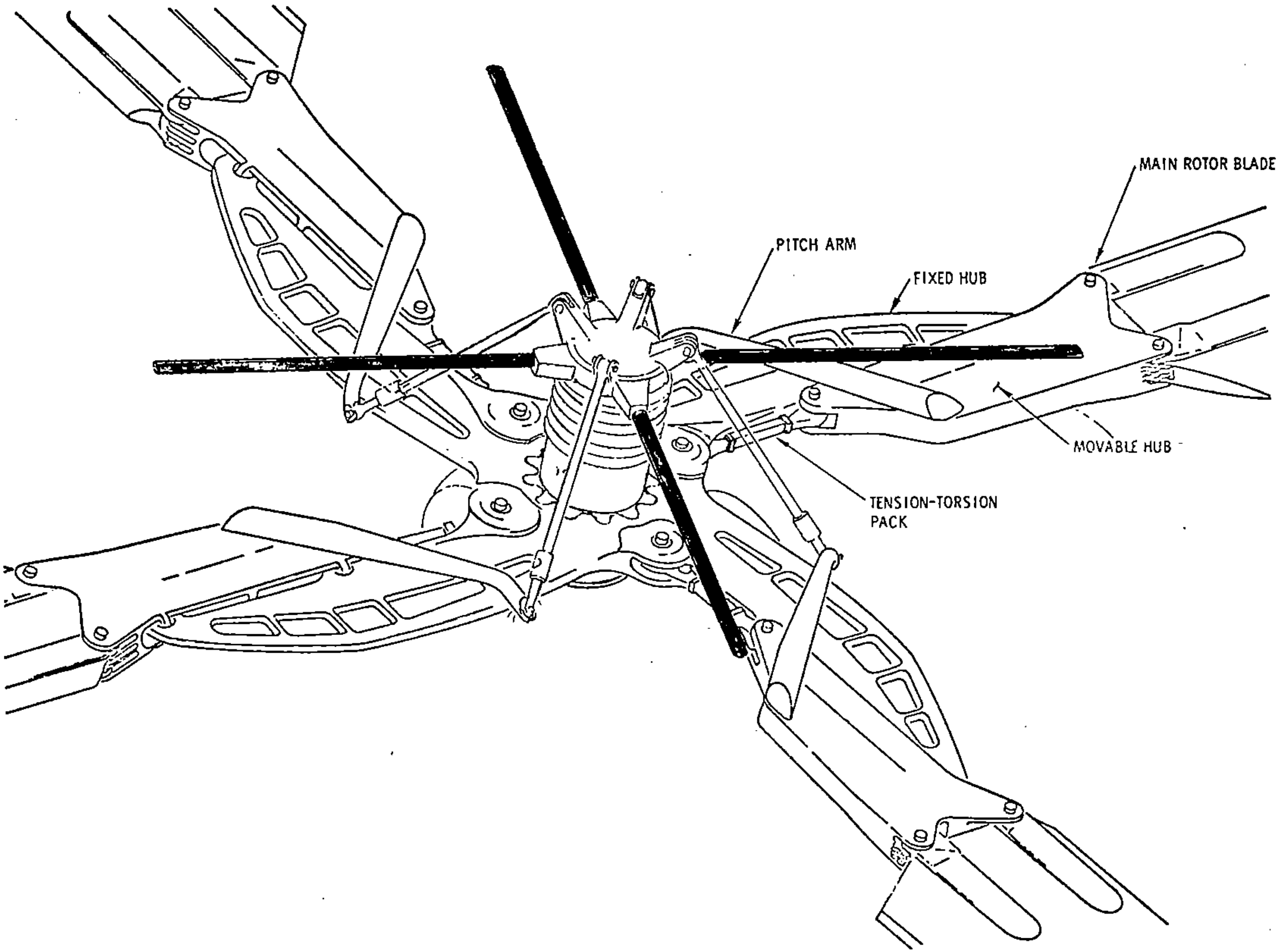


Figure 22-18. AH-56A Rigid Rotor Head

ARMY UH-1A, B, C &
EARLY D
TEETERING ROTOR
FLIGHT CONTROL SYSTEM

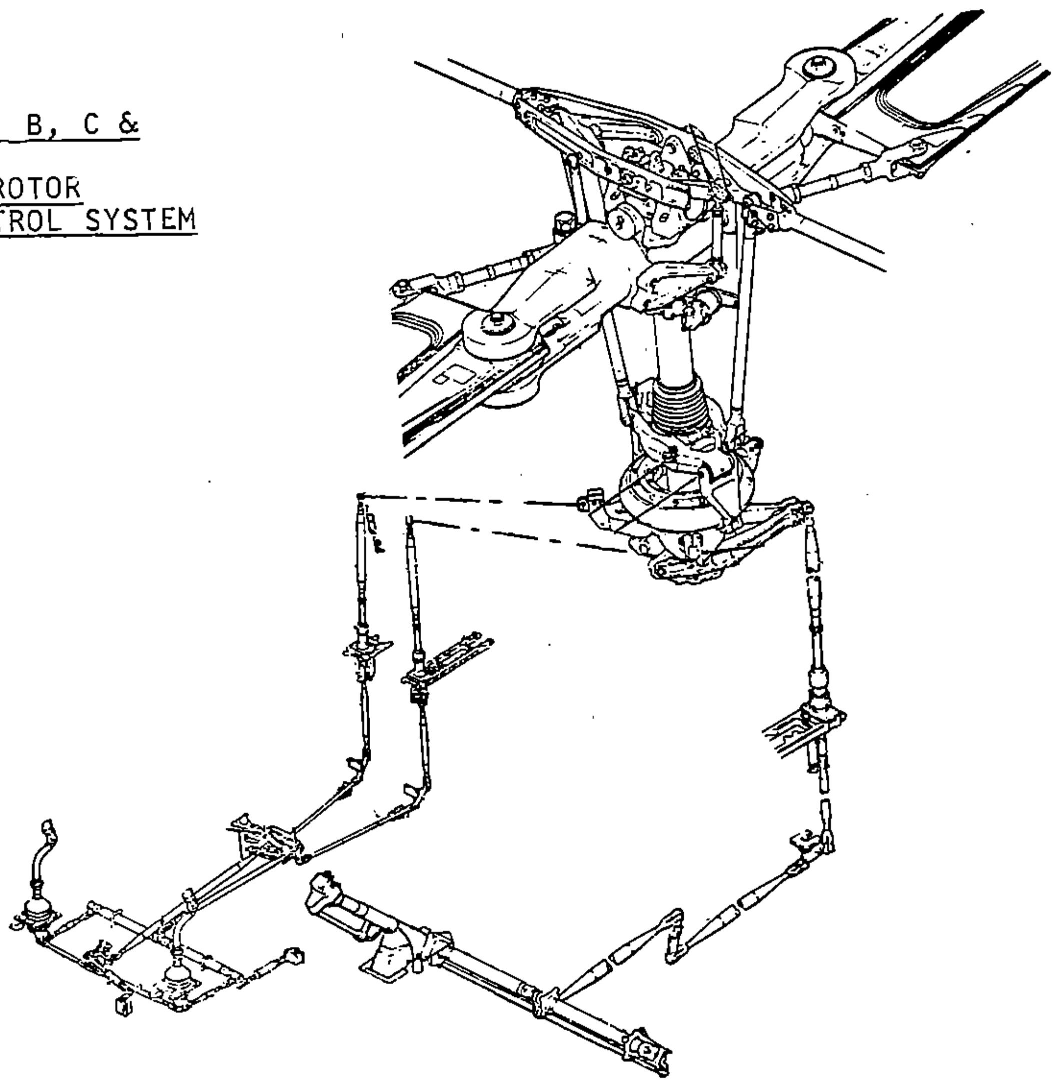


Figure 22-19.

ARMY AH-56A
RIGID ROTOR
FLIGHT CONTROL SYSTEM

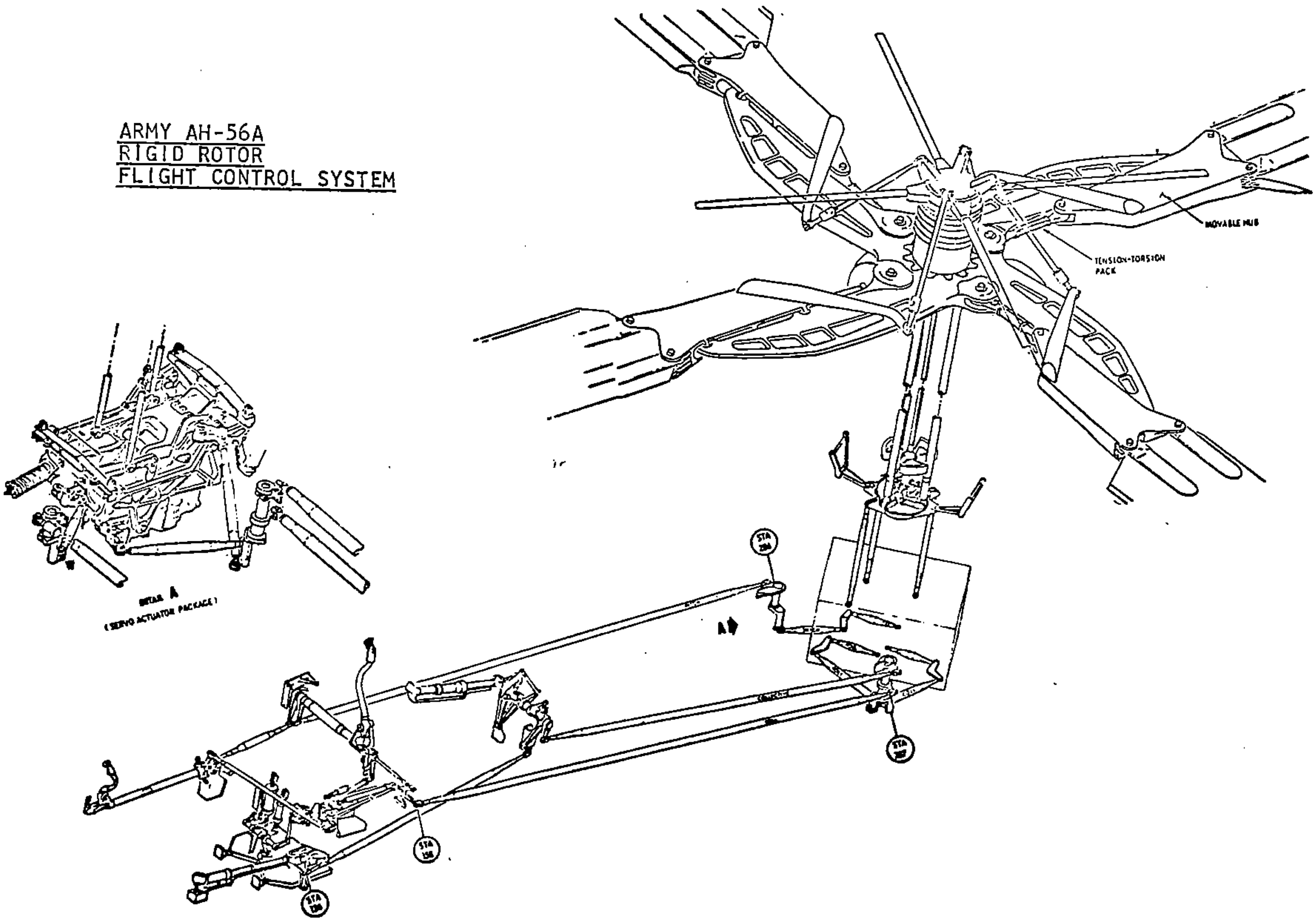


Figure 22-30.

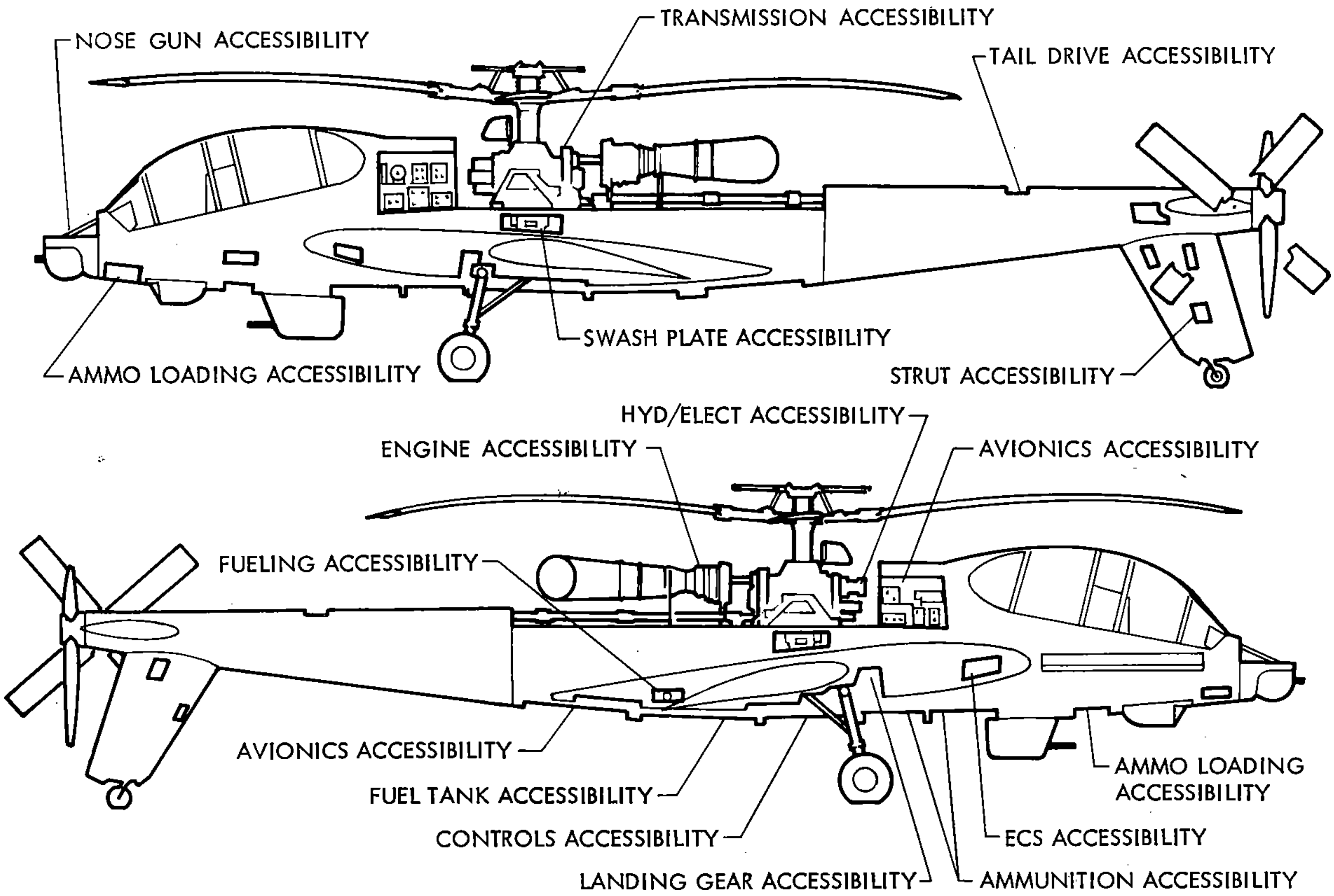


Figure 22-21. Accessibility Means Low Maintenance

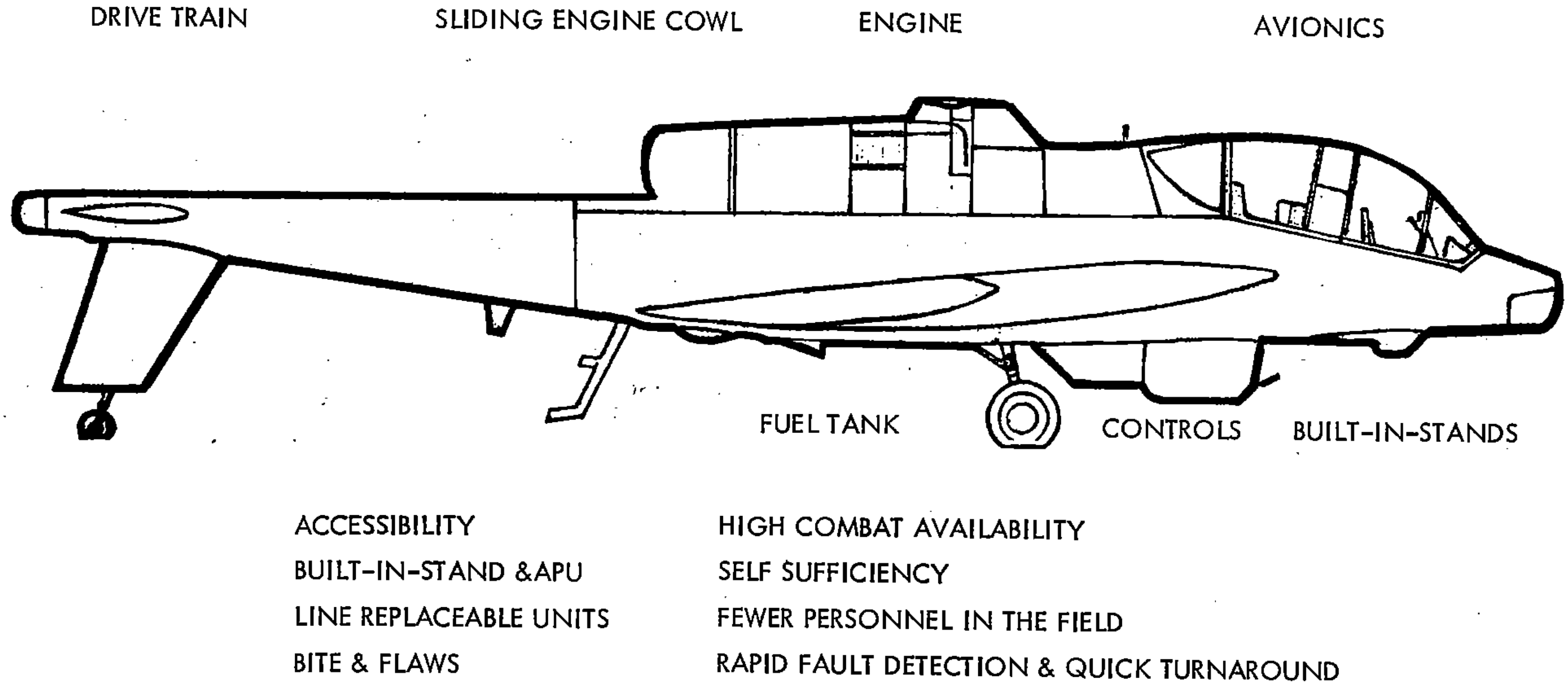


Figure 22-22. Development Maintenance Features

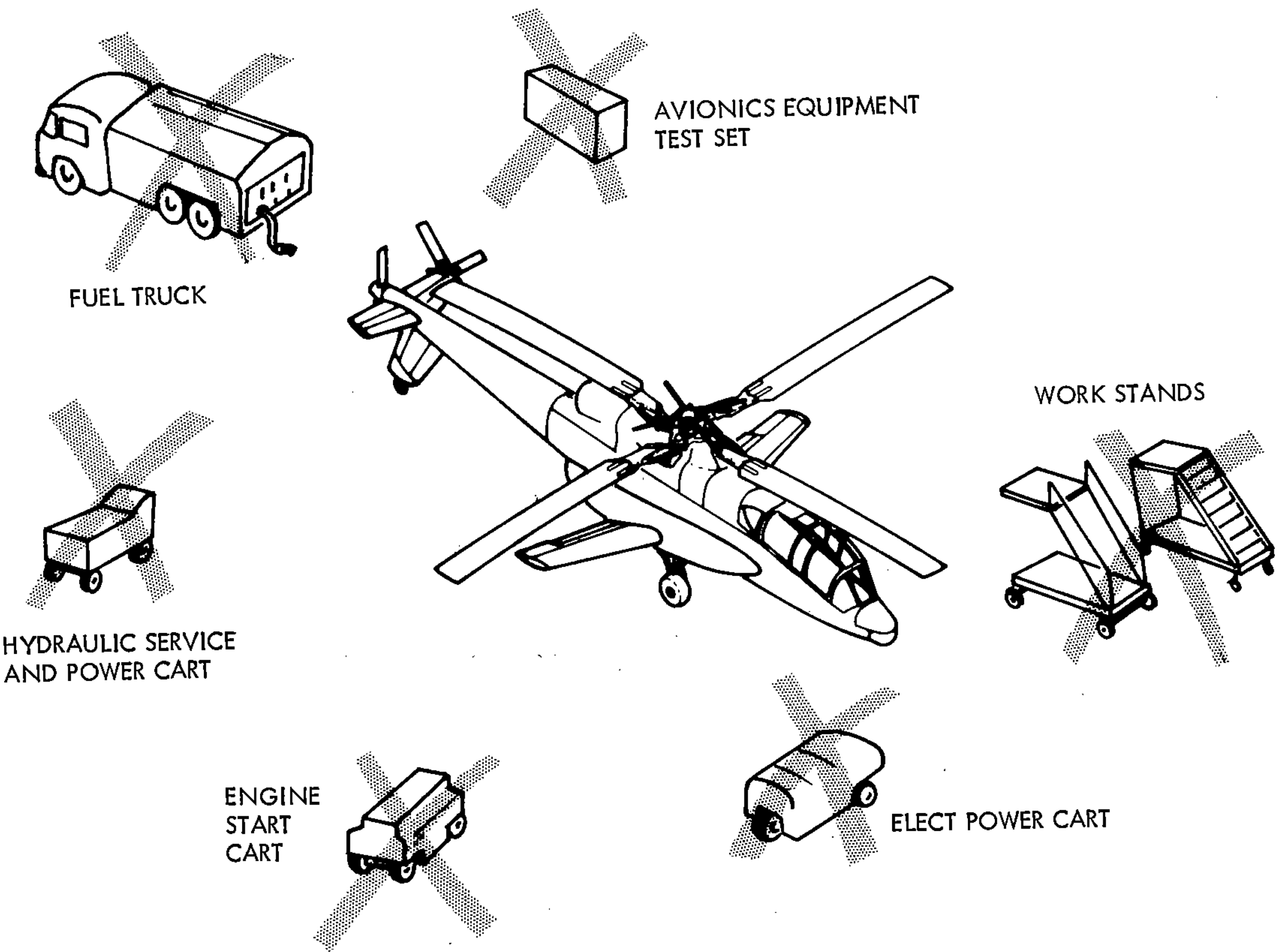


Figure 22-23. Self - Sufficiency

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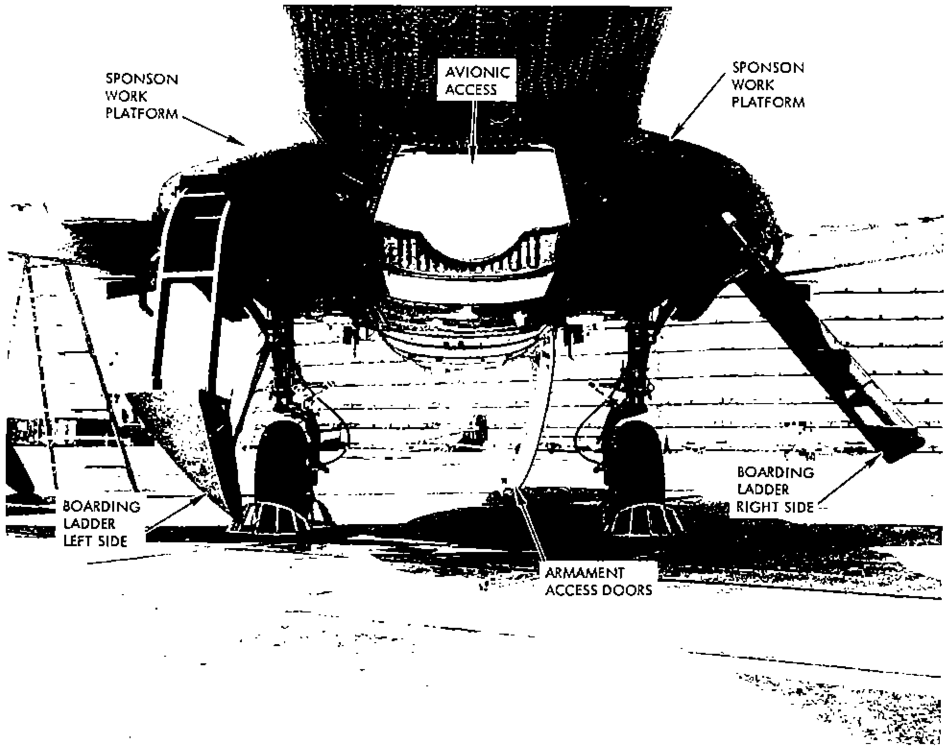

Figure 22-24. AH-56A Design for Maintainability

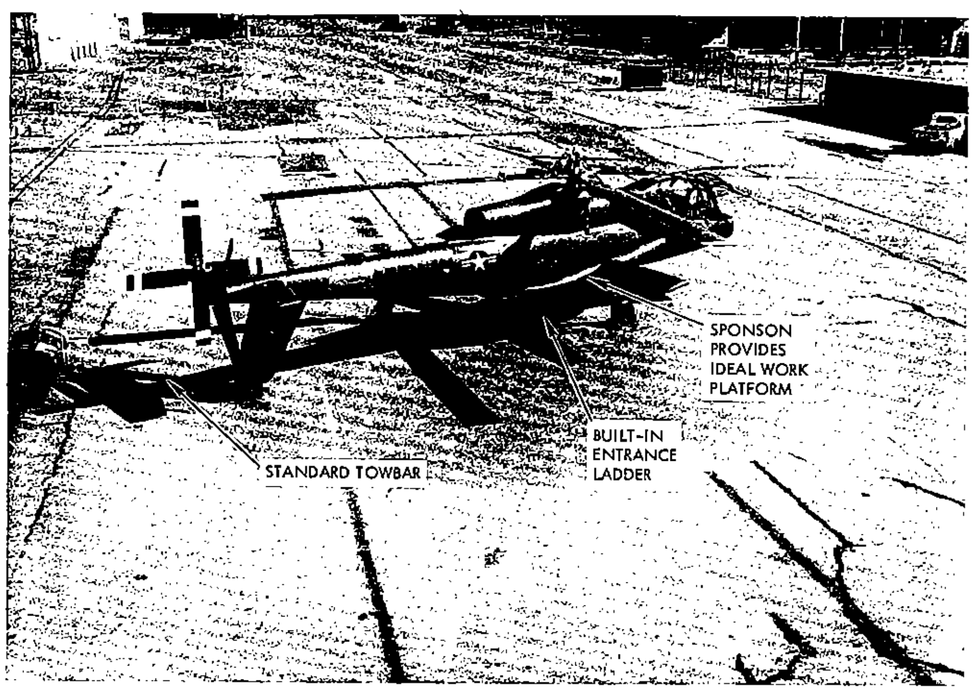

Figure 22-25. Ground Handling is Conventional. Sponsons Serve as Built-On Work Platforms

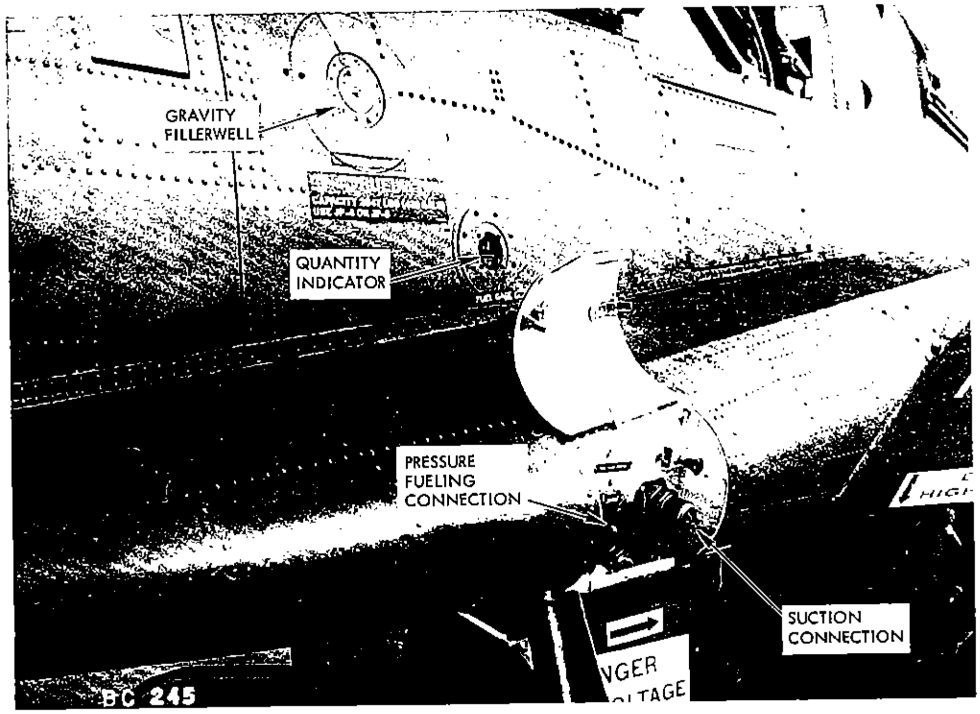

Figure 22-26. Refueling Provisions Include Gravity Fillerwell, Pressure Connection and Suction Connection for Self-Refueling From 55 Gallon Drums

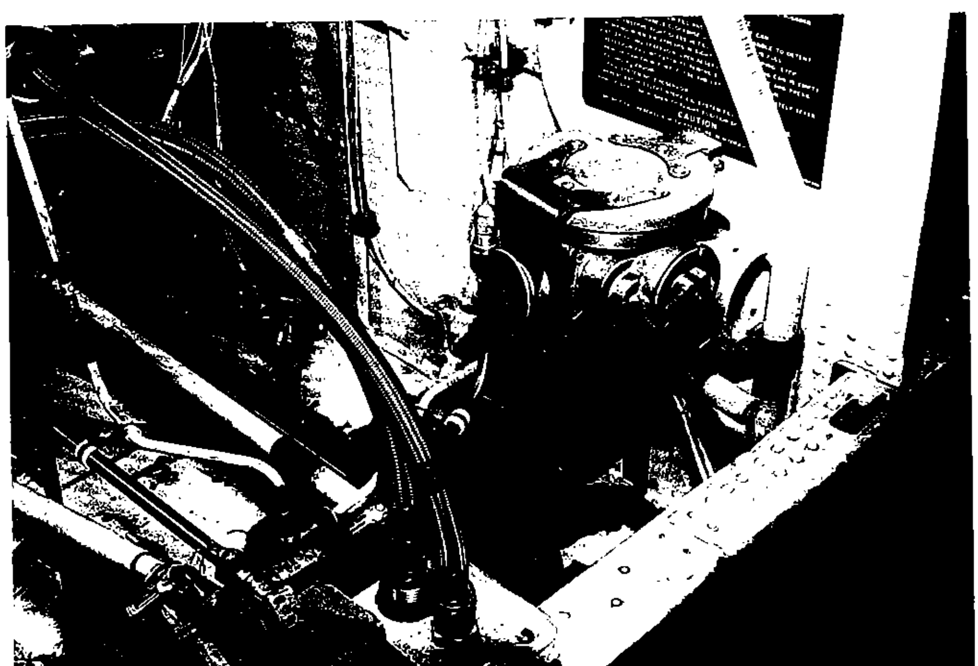

Figure 22-27. Built-In Device for Replenishing Hydraulic Oil Directly From a Standard Quart Can Eliminates Contamination

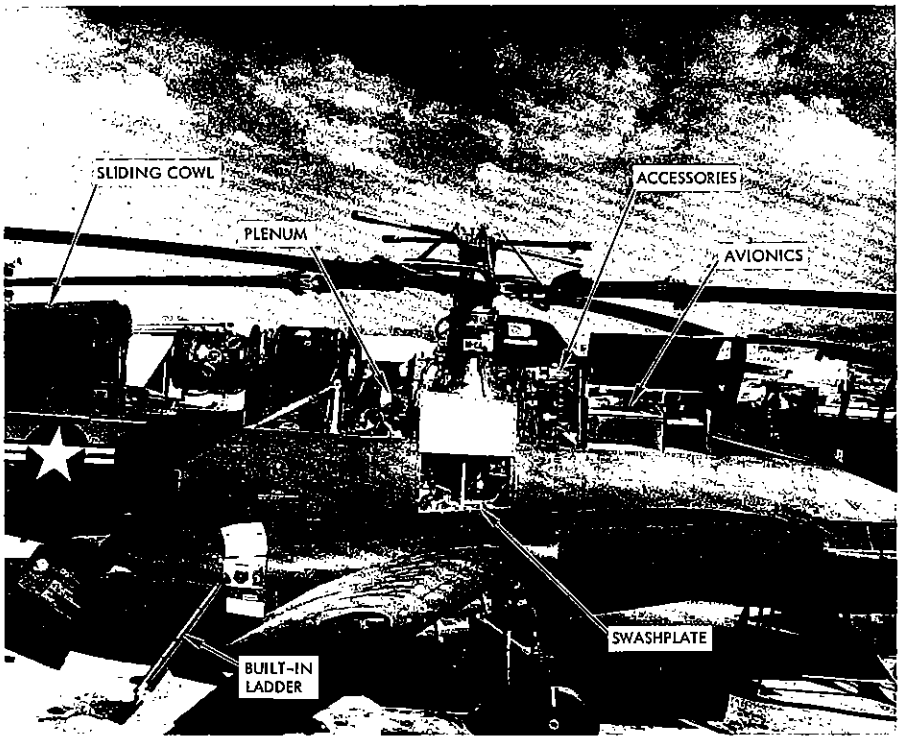

Figure 22-28. Sponsons, Built-In Ladders and Work Platforms, Hinged Doors and Panels Provide Access for Service and Maintenance

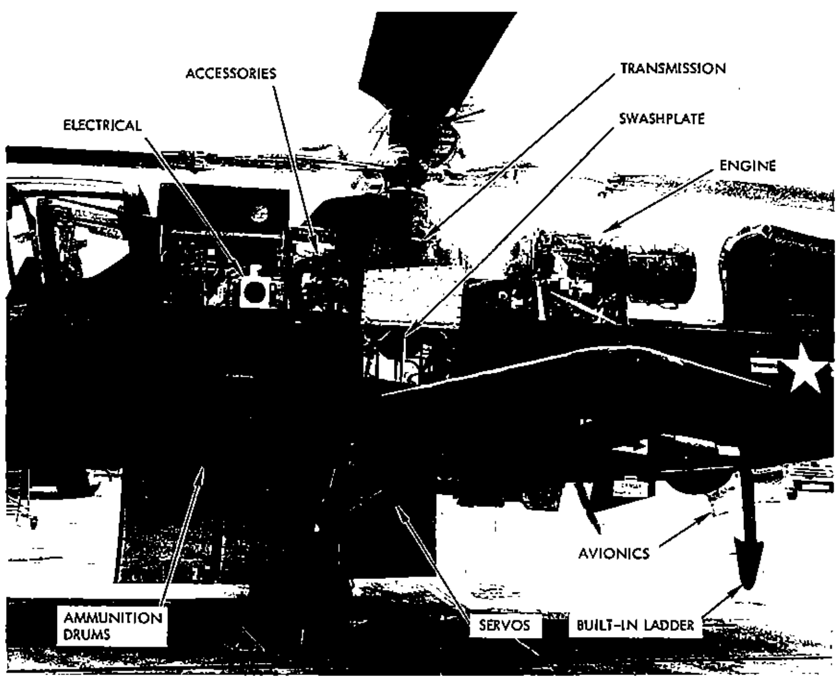

Figure 22-29. Electrical Power, Accessories, Transmission, Engine, Swashplate, Ammo Drums, and Control Servo Compartments are Accessible From Left Side

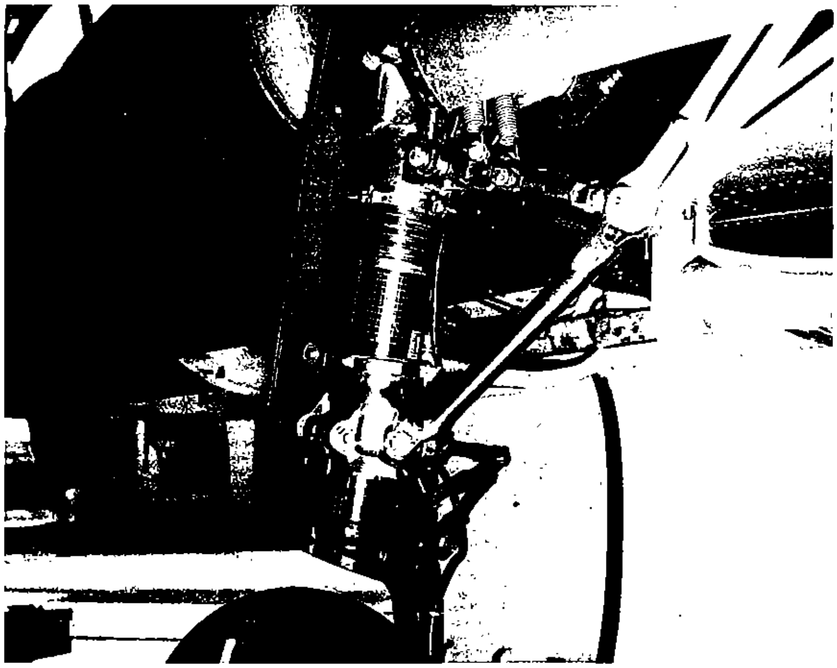

Figure 22-30. Simple Landing Gear Mechanism Requires no Lubrication

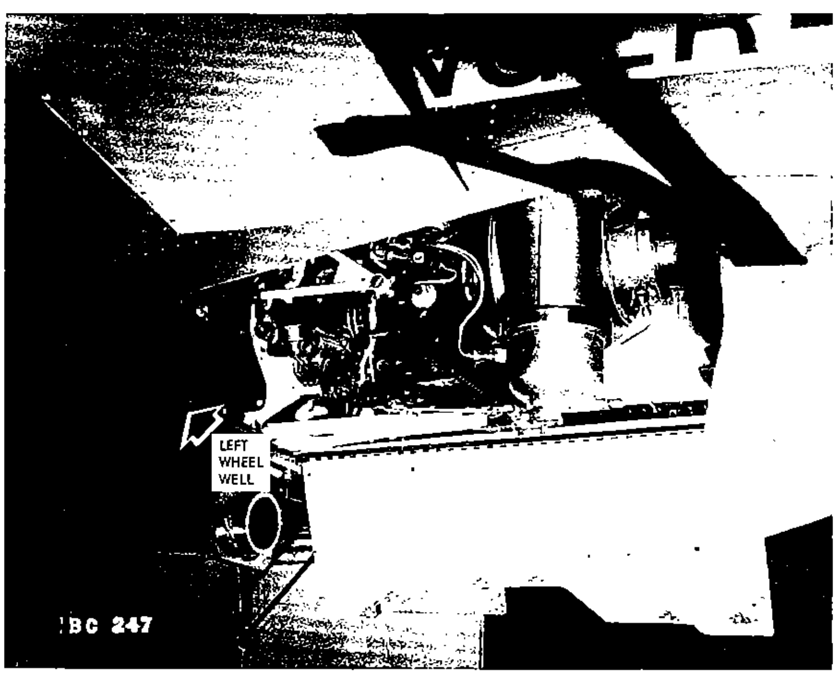

Figure 22-31. Built-In Auxiliary Power Unit Provides Power For Starting and For Hydraulic and Electrical Ground Checkout

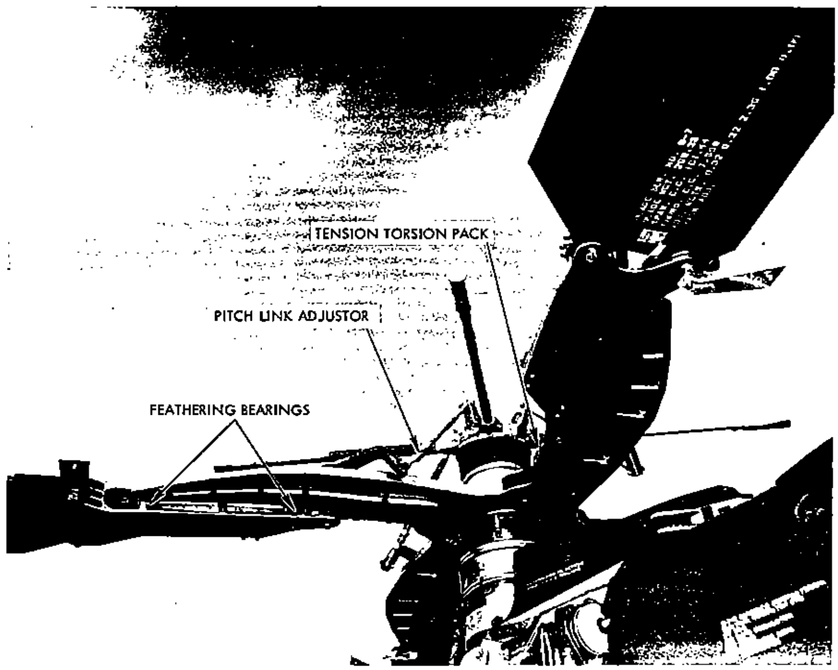

Figure 22-32. The Lockheed Rigid Rotor is Inherently Simple. There Are no Flapping Hinges, no Lead Lag Hinges, no Snubbers, no Lubrication Fittings.

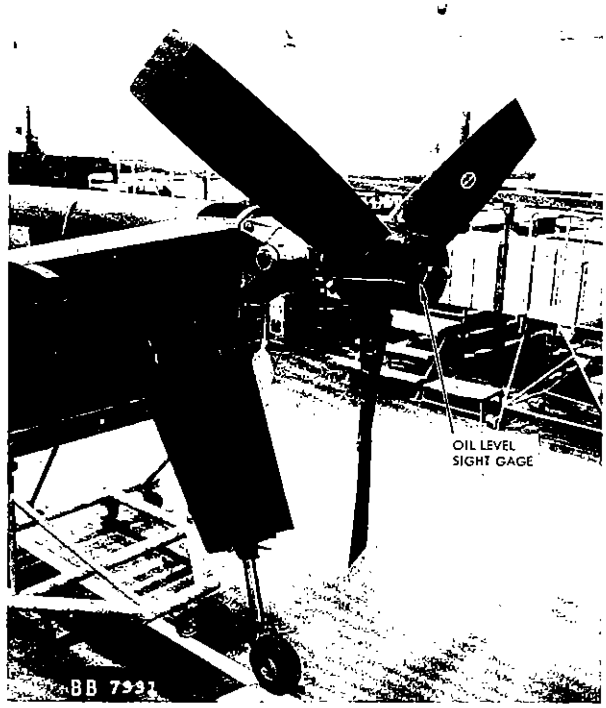

Figure 22-33. Fiberglass Propeller Blades Can be Repaired or Replaced in the Field. No Special Tools Required.

SUBSYSTEM		1ST GENERATION MMH/FLT HOUR	2ND GENERATION MMH/FLT HOUR
GUNNERS STATION		0.2633	0.1500
TOW CONTROL EQUIPMENT		0.0049	0.0019
CENTRAL PROCESSING		0.0008	0.0003
WEAPONS	XM51	0.0821	*
	XM52	0.1874	*
NAVIGATION DHARS		0.1943	*
ROTOR TRACK & BALANCE		0.1240	0.0276

*EQUIPMENT NOT AVAILABLE FOR TEST

Figure 22-34. Development Accomplishment AH-56A Demonstrated Maintainability

PERIOD	FLIGHT HOURS	TRACKING FLIGHT HOURS	TRACKING HOURS PER FLIGHT HOUR
PHASE II BLADES	570	71	0.1240
PHASE III BLADES	322	23	0.0714
PHASE III BLADES - NEW TRACKING METHOD	145	4	0.0276

TRACKING METHODS:

OLD: STROBE - MIRRORS

NEW: CHICAGO AERIAL INDUSTRIES ELECTRONIC TRACKER

Figure 22-35. Blade Tracking Development

	MAINTENANCE ACTIONS			ADJUSTED FOR ESTABLISHED FIXES		
	SCHEDULED	UNSCHEDULED	TOTAL	SCHEDULED	UNSCHEDULED	TOTAL
ORGANIZATION	6.9	2.5	9.4	4.0	1.5	5.5
DIRECT	4.0	11.6	15.6	2.5	7.5	10.0
TOTAL	10.9	14.1	25.0	6.5	9.0	15.5

Figure 22-36. Maintenance During RDAT Maintenance Manhours Per Flight Hour

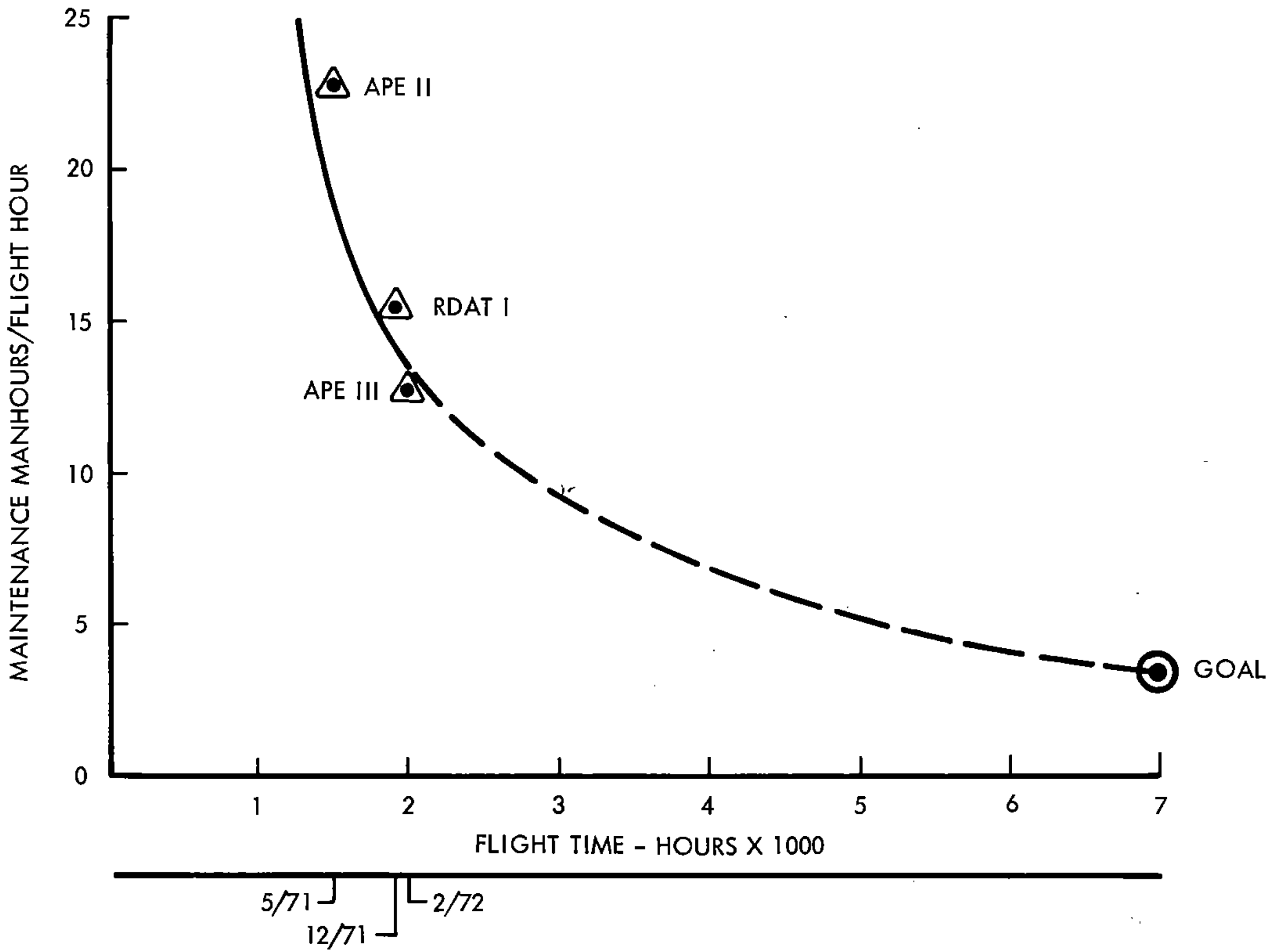

Figure 22-37. Maintenance Improvement - Organization & Direct (As Measured)

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VULNERABILITY ANALYSIS

INTRODUCTION

Vehicle vulnerability is an important factor in the final analysis of vehicle survivability, mission success, and obviously crew safety. The measure and assessment of vehicle vulnerability requires an integrated assessment of the data bases listed below.

- System operations and functions
- Performance characteristics
- Design data - down to component level
- Susceptability of components to damage
- Characteristics of enemy ordnance
- Destructive mechanisms of ordnance
- Designer's estimate of allowable damage
- Historical evidence of component/system response to damage
- Firing tests
- Protective devices
- Judgment factors

For the past few days, you have been indoctrinated on the details of the AH-56A Helicopter, including its functions, operations and performance capabilities. At this time, I wish to briefly introduce you to that mysterious Art of Vulnerability Analysis.

Before proceeding on into these details, I would like to offer the following word of caution. Different analysts will arrive at differing values or assessments -- not because of some omission, although this may occur inadvertently -- but rather due to the amount and degree of detailed information available or not available to the analyst. When adequate information is available, a good firm and reasonably accurate assessment can be made. When the condition of inadequate information prevails, the analyst is restricted by his related experience with presumably similar vehicles -- even though design philosophies are not similar. Therefore, to appreciate the significance of the assessments, it is important to know the basis of the evaluation -- including the level of detail design considerations, judgement criteria, and other assumptions which constitute the composite results.

Vulnerable Area - The term vulnerable area is a term with a unique definition. First of all, vulnerable area does not mean that area which is vulnerable, but rather it is an expression which attempts to give a weighted value to that area which, when affected by some ordnance, will produce a particular type of damage. Mathematically, vulnerable area is defined as:

(1) A_V(vulnerable area) = A_P (presented area) x $P_{K/H}$ (probability kill given a hit)

and the total vulnerable area over i components, partitions etc. is:

(2) $$A_{V_{TOTAL}} = \sum^{i} (A_P P_{K/H})_i$$

where,

(3) $A_P \geq 0$ and $0 \leq P_{K/H} \leq 1$.

(4) $\overline{P}_{K/H}$ (average kill probability given a hit) = $A_{P_{TOTAL}} / A_{V_{TOTAL}}$

where,

(5) $A_{P_{TOTAL}} = \sum^{i} (A_P)_i$, (the totality of partitions).

As an example, consider a target which has been completely divided into n unique partitions. Then, using equation (2) we have:

(6) $$A_{V_{TOT}} = \sum_{i=1}^{n} (A_P P_{K/H})_i$$

(7) $$A_{V_{TOT}} = (A_P P_{K/H})_1 + (A_P P_{K/H})_2 + (A_P P_{K/H})_3 + \ldots . (A_P P_{K/H})_n$$

When there are some $(P_{K/H})_i = 0$, those terms in equation (7) can be rearranged and grouped such that the first m terms are zero and the remaining n-m terms are non zero as below.

$$(8)\quad A_{V_{TOT}} = \sum_{a=1}^{m} (A_P P_{K/H})_a + \sum_{b=m+1}^{n} (A_P P_{K/H})_b$$

and since

$$(9)\quad \sum_{a=1}^{m} (A_P P_{K/H})_a = 0; \text{ for each } (P_{K/H})_a = 0, \text{ then}$$

$$(10)\quad A_{V_{TOT}} = \sum_{b=m+1}^{n} (A_P P_{K/H})_b, \text{ and with a change of subscript}$$

$$(11)\quad A_{V_{TOT}} = \sum_{j=1}^{n} (A_P P_{K/H})_j, \text{ then since those}$$

j^{th} partitions have a non zero $P_{K/H}$, one can say that the partition represented by $(A_P)_j$ is susceptable to some form of kill category, and that the non j partitions are not susceptable. Therefore, it is appropriate for the analyst to uniquely select a partitioning scheme convenient to his method of analysis.

To obtain an average probability of kill given a hit we obtain the equivalence of equation (4) from (11) and (5).

$$(12)\quad \overline{P_{K/H}} = \frac{\sum_{j=1}^{n} (A_P P_{K/H})_j}{\sum_{i=1}^{n} (A_P)_i} = \frac{A_{V_{TOT}}}{A_{P_{TOT}}}$$

Partitioning - The target vehicle can be conveniently partitioned according to subsystems, functions, operations, compartments, or combinations thereof. The key parameter to the selection of the partitions is heavily dependent upon the nature of the ordnance, its functioning characteristics, the general nature of the response of the selected partition to the ordnance, and the type of kill being assessed. Whatever scheme is selected for partitioning, care must be exercised in the accounting or bookkeeping process so as to assure completeness and to avoid redundant assessment. Examples are given to exemplify the above.

Kill Categories - For helicopter evaluations three generalized categories are:

> Attrition - That damage which causes the aircraft to crash, this would include uncontrolled flight which leads to a crash.
>
> Forced Landing - That damage which forces the crew to land, this includes that damage which had the pilot continued to fly, the aircraft would have been destroyed.
>
> Mission Abort - That damage which will not allow the aircraft to complete its defined mission.

Vehicle Kill Criteria and Thresholds - Table I - presents in summary form some of the more prominent criteria and regions of damage which are assumed to lead to kills.

The principal subsystems used for summing the vulnerable areas are briefly described below.

Structure - Those elements required to maintain the physical integrity of the vehicle when exposed to flight loads.

Crew - Pilot and Copilot/gunner.

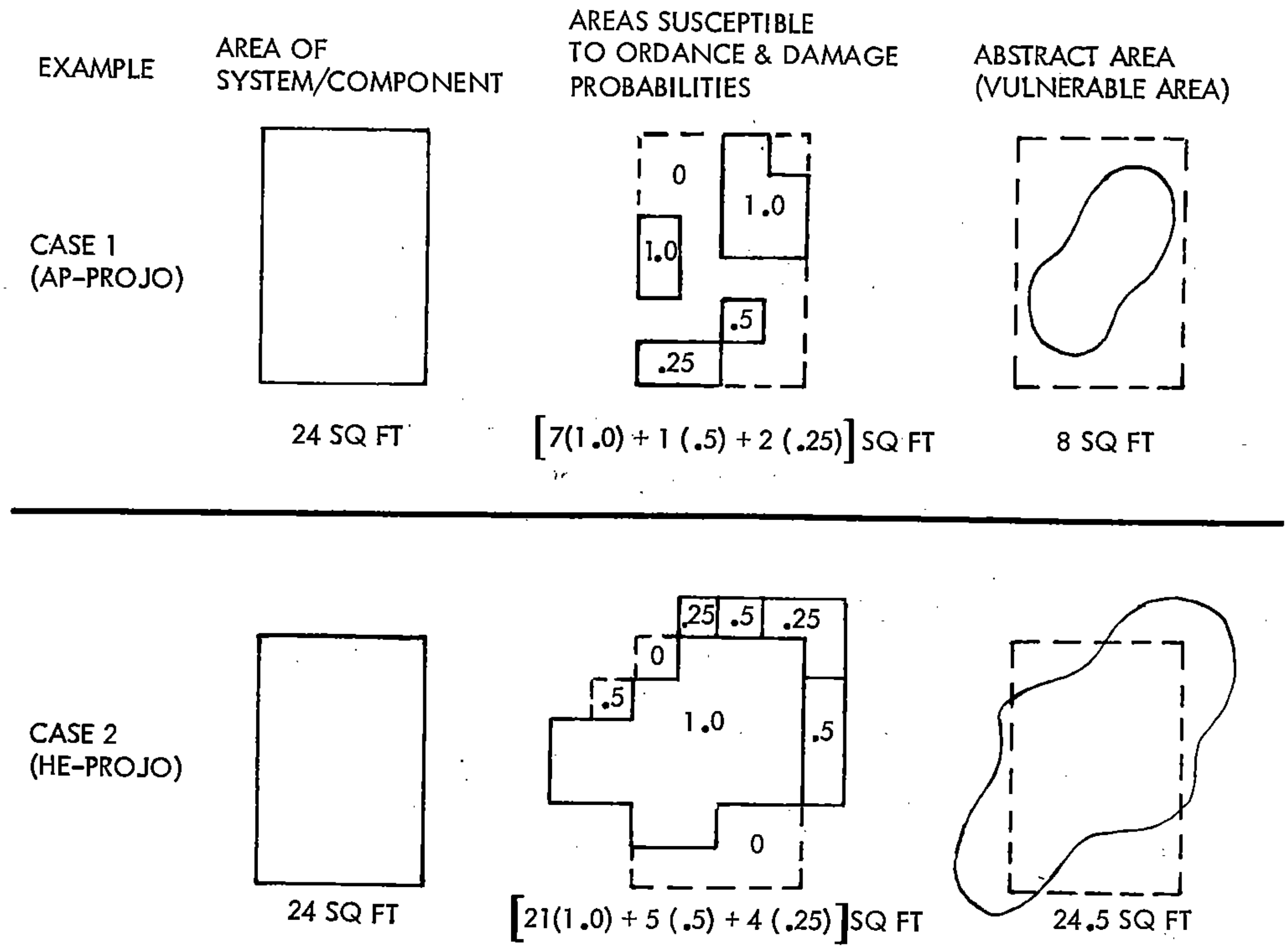


Figure 23-1. Measuring Vulnerable Area

Table I - Generalized Damage Criteria

Region	Attrition	Forced Landing	Mission Abort
Crew Compartment	Pilot and Copilot	---	Either Pilot or Copilot
Main Rotor Blade	Trailing Edge; Blade Root; Hinges; Internal Detonations	Balance Bar; External Detonation; Spars	Tip Weight; External Detonations
M.R. Linkage, Mast, Hub	Hinges; Linkage; Detonations; Internal of Movable Hub or Mast Control Rods	Control Rods, Gyro Arms Non Rotating Control Rod	---
Anti Torque Rotor	Hub; Flanges; Doublers	Tip Weight; Servo Package; Yaw Actuator (Hover Only)	Leading Edge; Trailing Edge; Spar, Direct. Conts.
Pusher Prop	Loss of Blade	Hub; Gear Box, Severe Damage Outboard, 50% Blade	Damage Prop, Reduction in Performance
Servo Assy & Transmission	Transmission Main Bearing Swash Plate, Control Rods	Surface or Internal Detonations on Transmission	---
Flight Controls	Loss of Cyclic	Loss of Collective	---
Drive Shaft, Bearings, Gear Boxes	Hanger Bearings	Gear Box and Servo for Anti Torque	(Prop Gear Box-Reduced Performance)
Engine Compartment	Fire	Plumbing; Engine Accessories; Compressor	---
Ammo Cans & Weapons	Internal Explosion of Ammo Can	Fire/Gasing of Rounds	Damaged Rounds/Feed System; Weapon
Fuel Cells	Fire; Explosion; Hydraulic Ram	Loss of Fuel; Damaged Fuel Pump	Fuel Line Leaks
Structure	Hydraulic Ram in Fuel Cells	Internal Detonations in Aft Fuselage	Internal Detonations Aft Fuselage
Avionics & Electrical	Fire	Fire	Damaged Equipment

Engine - The turbo shaft engine which drives the rotor systems and power consuming accessories - Fuel systems after the engine-driven high pressure pump are considered as part of the engine -- mechanical power elements are considered to be part of the engine up to the output spline which drives the torque meter -- the air passages from compressor inlet to turbine outlet are part of the engine; outside these boundaries they are part of the structure system. Bleed air is part of the engine to the outflow pods, beyond which points it becomes part of the pneumatic system.

Secondary Power - These systems include the hydraulic system from the pump to the actuated device; the electrical system from the generator to the avionics or electrical component using power; and the pneumatic system from the engine bleed air pods to the system exhaust.

Rotor Systems - Consists of main rotor, tail rotor, and propeller systems -- in each case, rotor system consists of hub, rotor blades, and pitch links; in the case of the main rotor the rotor system category also includes the gyro. Components which functionally precede these elements are considered as part of the mechanical power system.

Avionics and Electrical - All electrically operated equipment from the point of power supply onward. Optical systems are also carried in this category for convenience.

Ammunition and Weapons - This category includes the guns, ammunition feed mechanisms, internally carried ammunition, and externally carried stores.

Fuel Systems - Includes fuel storage and transfer system components, such as stored fuel, fuel lines, pumps, valves, and other plumbing fixtures. This system extends functionally to the high pressure fuel pump mounted on the engine.

Mechanical Power - This system begins at the engine output spline, and includes the torquemeter connection to the transmission, the transmission itself, the main rotor mast to the hub, the tail rotor shaft, the tail rotor gear box, and several lesser mechanical connections.

Control System - Both engine and flight controls are included in this category. A flight control element is assumed to begin at the point of operator actuation and end at the control surface or controlled device.

Summary of Vulnerable Areas[1]

	ASPECT				
Subsystems	Front	Side	Rear	Bottom	Average
Structure	X	X	X	X	X
Crew	X	X	X	X	X
Engine	X	X	X	X	X
Rotor Systems	X	X	X	X	X
Secondary Power	X	X	X	X	X
Avionics	X	X	X	X	X
Ammunition	X	X	X	X	X
Fuel	X	X	X	X	X
Mechanical Power	X	X	X	X	X
Controls	X	X	X	X	X
TOTALS	XX	XX	XX	XX	XX

[1] Vulnerable Areas are computed and tabulated for each combination of threat and kill category

Kill Category	Threat	
Attrition	7.62 mm API	23 mm API
Forced Landing	12.7 mm API	23 mm HEI
Mission Abort	14.5 mm API	

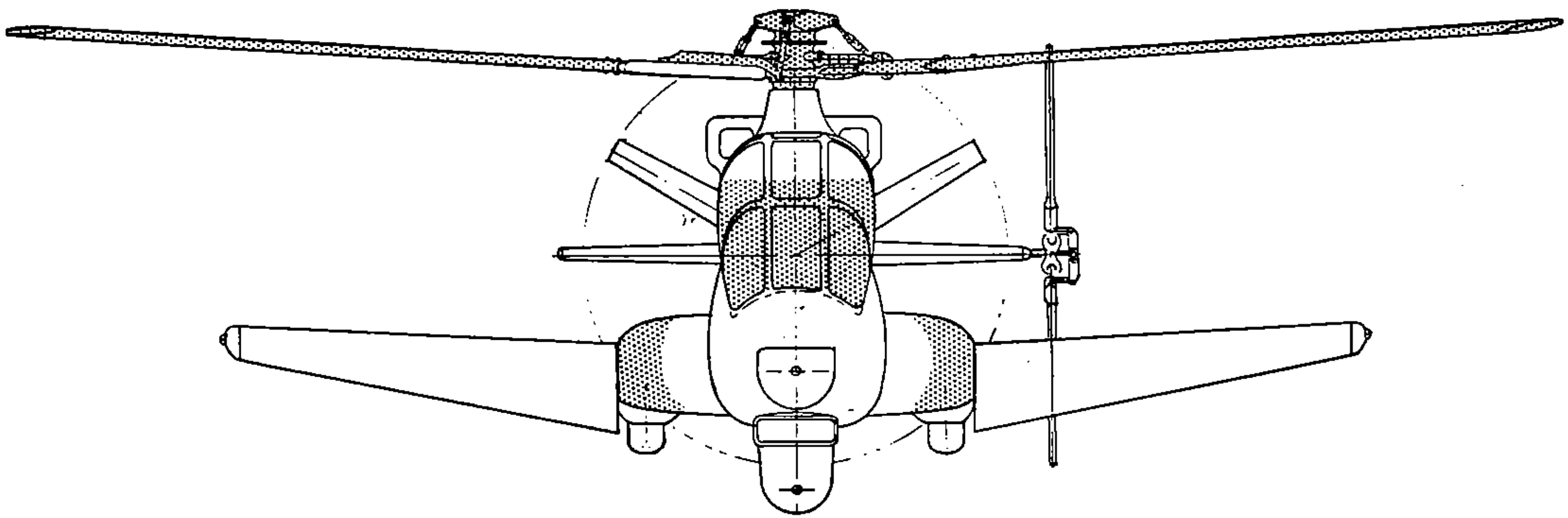
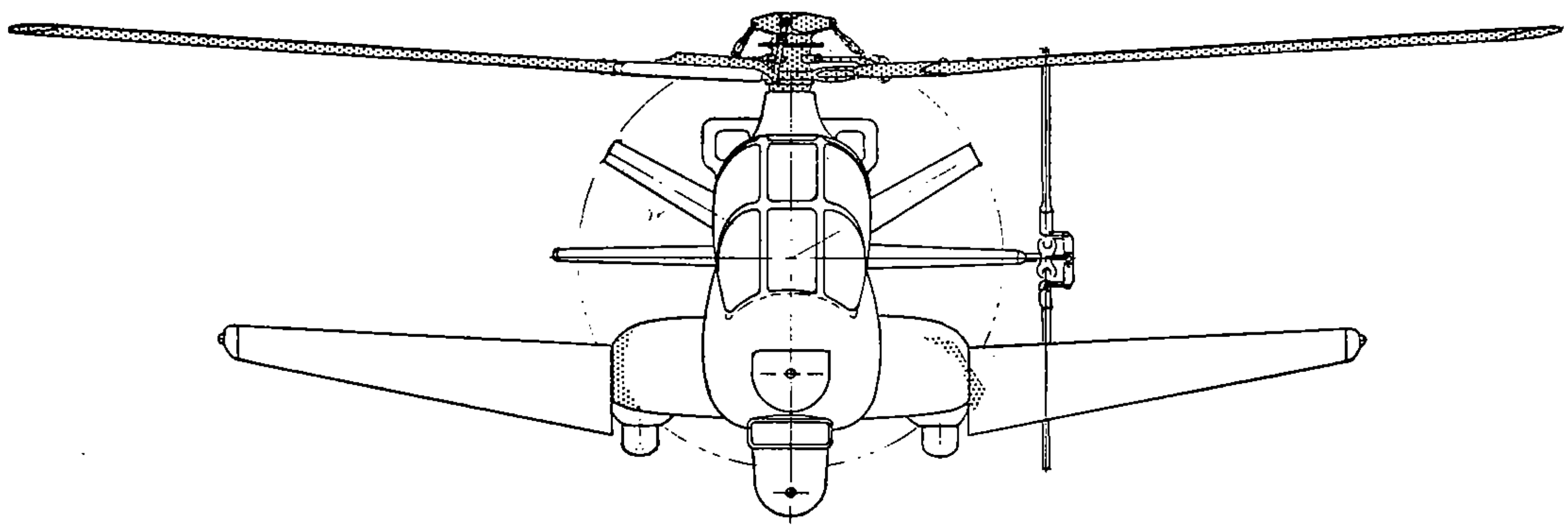


Figure 23-2. Attrition - 23 MM HEI Characteristics Vulnerable Areas - Front View (Present Configuration)



QUICK SOLUTION

CREW	- TRANSPARENT ARMOR 12.7
FUEL SPONSON	- PURGE MATS

Figure 23-3. Attrition - 23 MM HEI Characteristics Vulnerable Areas - Front View (Modified Configuration)

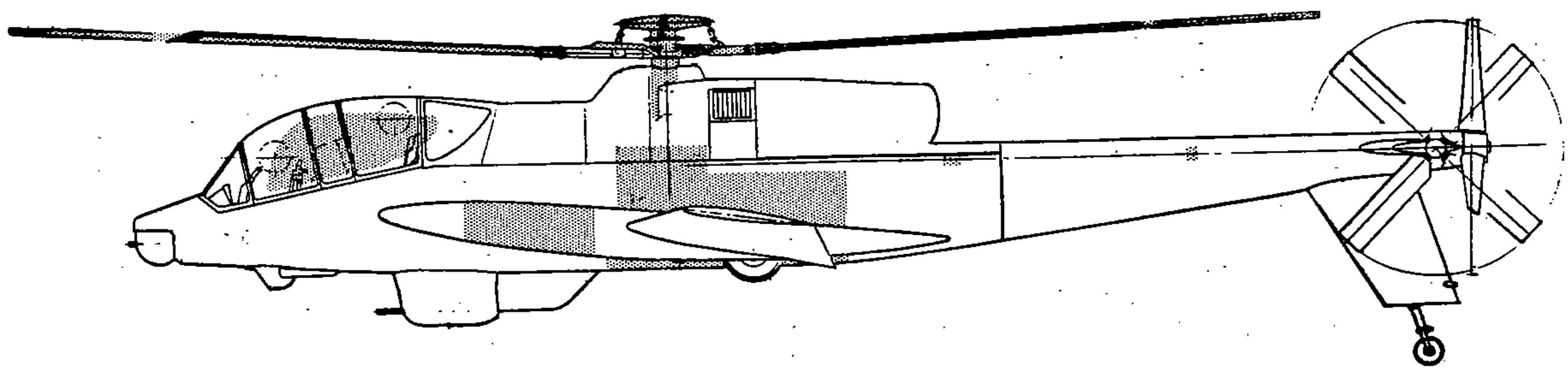
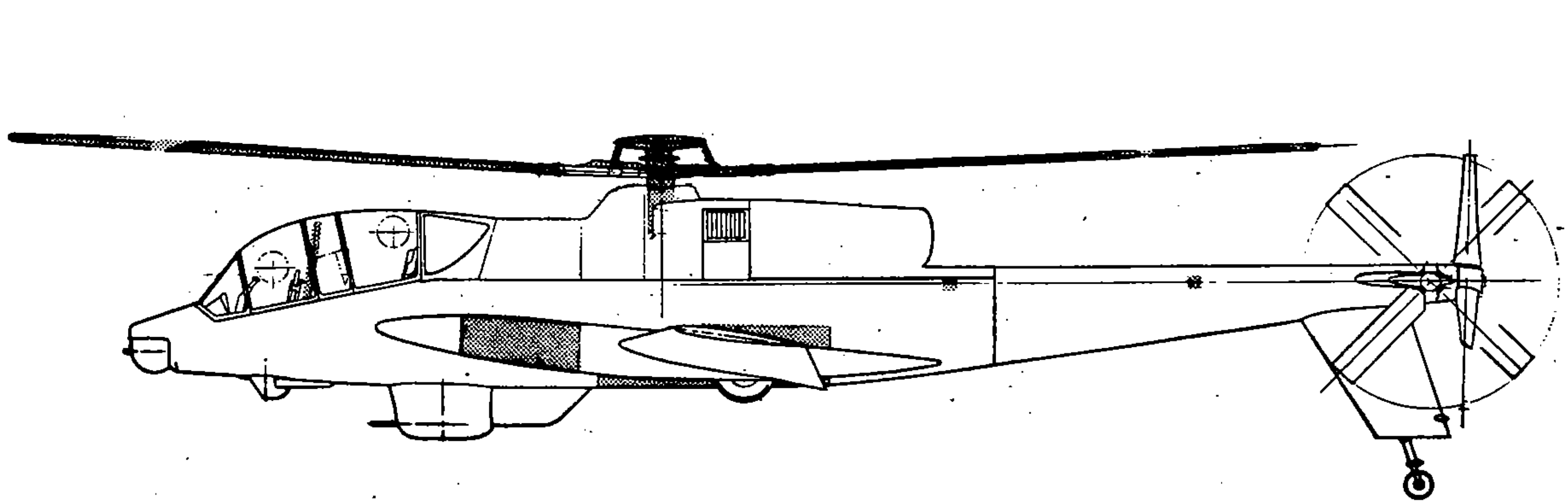


Figure 23-4. Attrition - 23 MM HEI Characteristic Vulnerable Areas - Side View (Present Configuration)



QUICK SOLUTION

CREW	- TRANPARENT ARMOR - 12.7 MM SEPARATE CREW
SERVO & TRANS	- ARMOR PLATE - 12.7 MM ISOLATE CYCLIC ASS'Y & SWASHPLATE
FUEL MAIN	- INERT GAS/COLLAPSABLE LINER
(FUEL CELLS)	- (PURGE MATS ON ENDS OF CELLS)
	- APPLICABLE TO API

Figure 23-5. Attrition - 23 MM HEI Characteristic Vulnerable Area - Side View (Modified Configuration)

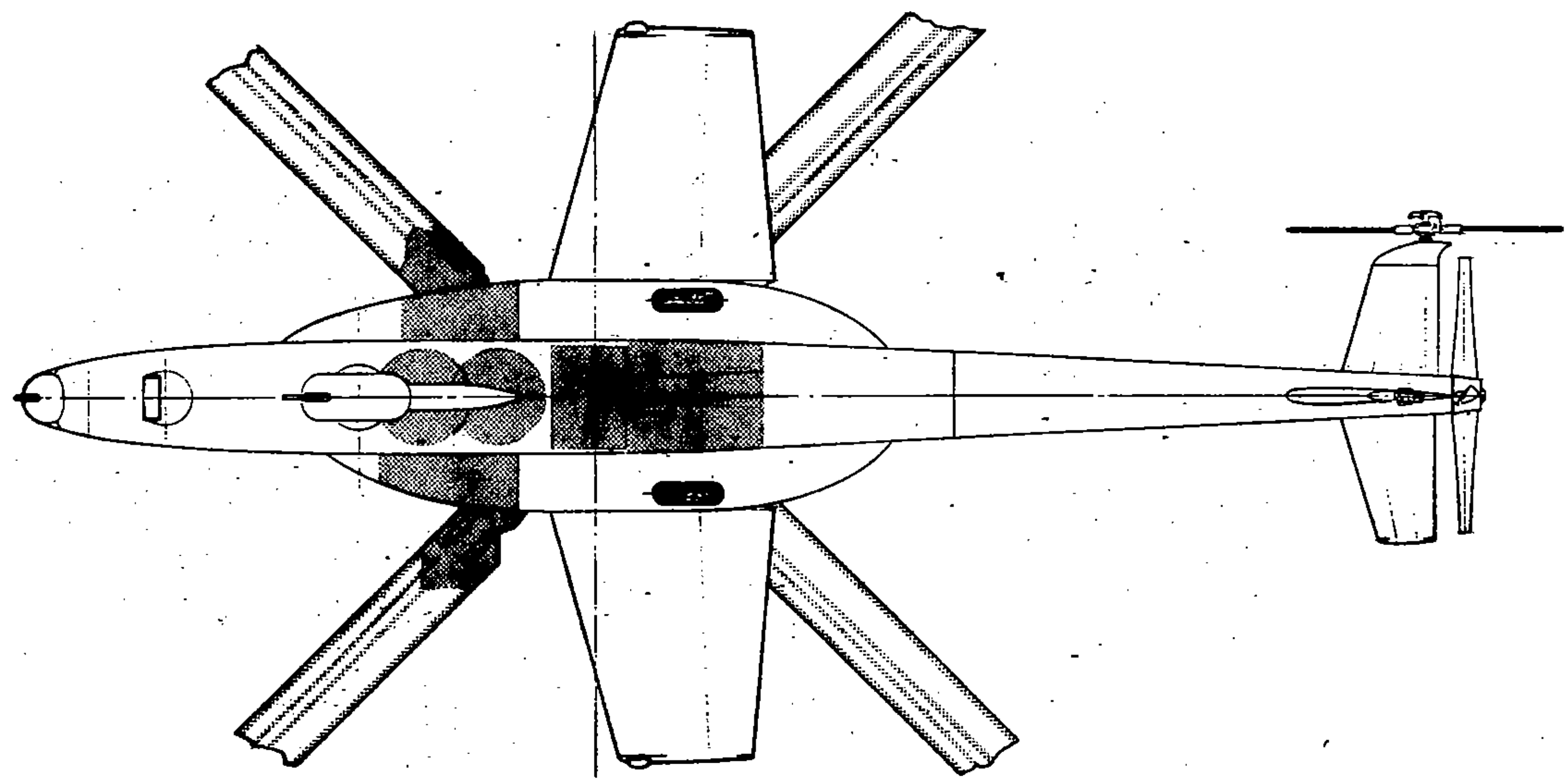


Figure 23-6. Attrition - 23 MM HEI Characteristic Vulnerable Areas - Bottom View (Present Configuration)

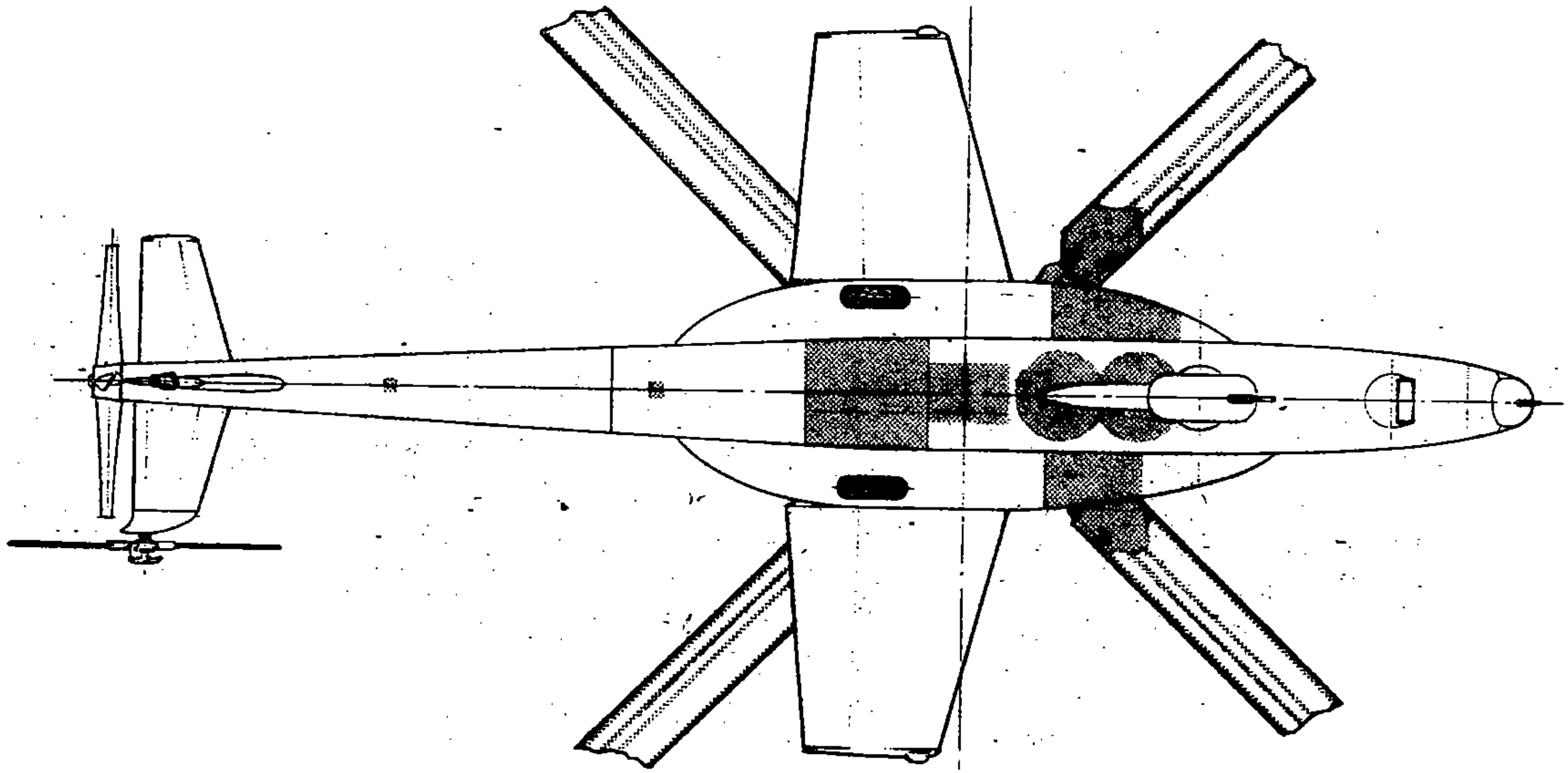


Figure 23-7. Attrition - 23 MM HEI Characteristic Vulnerable Areas - Bottom View (Modified Configuration)

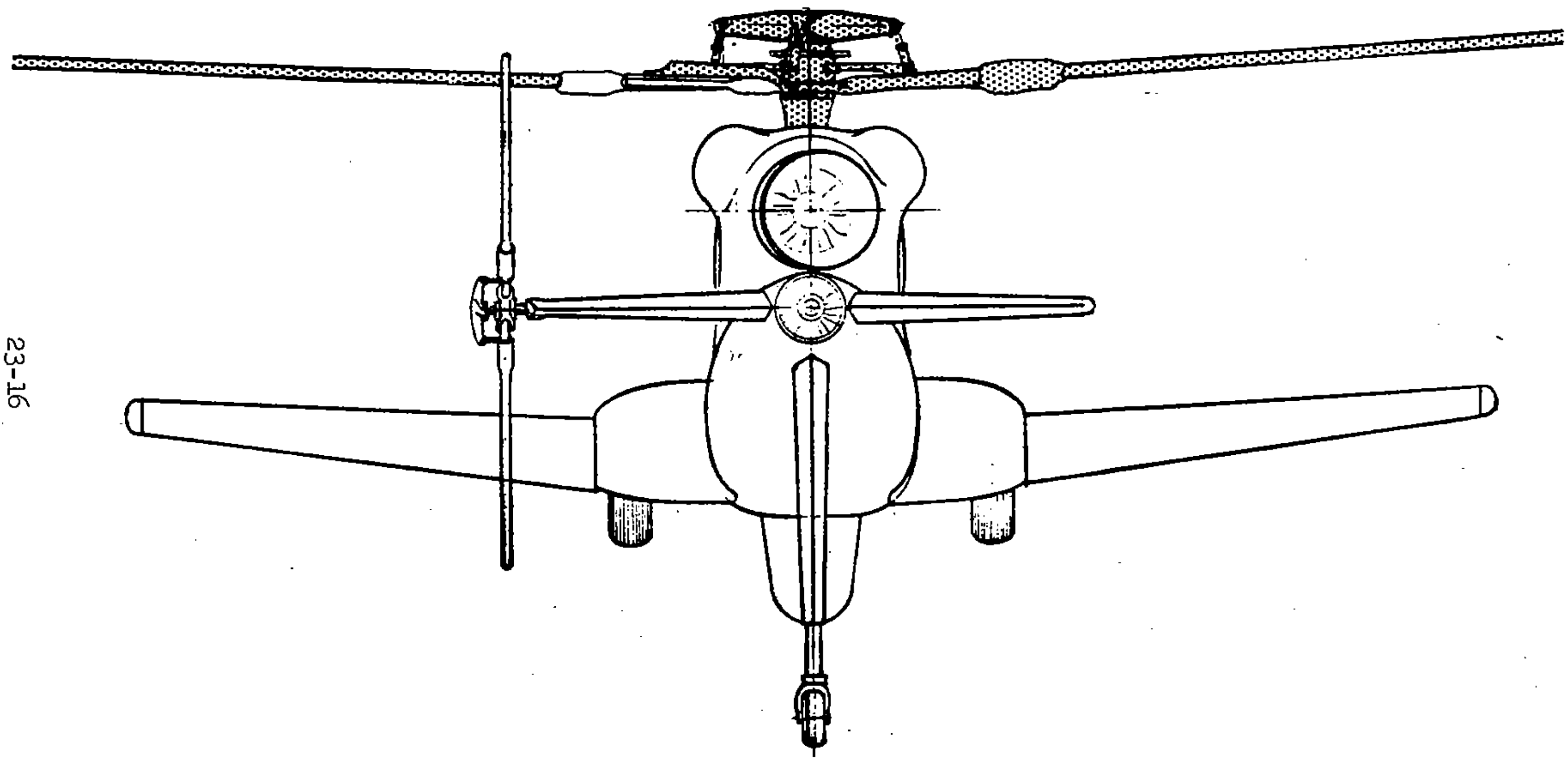


Figure 23-8. Attrition - 23 MM HEI Characteristic Vulnerable Areas - Rear View (Present & Modified Configuration)

AH-56A QUALITATIVE MATERIAL REQUIREMENT

A comparison of the Department of the Army-Approved Qualitative Material Requirement (QMR) for and Advanced Aerial Fire Support System (AAFSS) (dated 17 December 1965 including revisions through 30 March 1970) with the configuration of the AH-56A is made herein. The AH-56A configuration used for this comparison is the "Approved Producibility/Cost Reduction Study (PCRS)" configuration defined in Volume II of the Producibility/Cost Reduction Study (PCRS) Phase I Final Report (LR 24739) as revised 29 October 1971. This is an Army-approved document.

The QMR contains many requirements which are applicable to an Integrated Weapons System. Some of the requirements are specifically oriented to the aerial vehicle performance but have been included not merely as aircraft performance but of the Total Integrated System Performance. There are approximately 200 requirements considered in this report which will include

- The 188 requirements that have been met or exceeded
- The original requirements which have been deleted as a result of the PCRS
- Requirements not fully satisfied but essentially achieved (70-99%)
- Requirements not yet achieved
- Deleted Requirements

Specific information on the comparison of the requirements will not be included in this manual because of security classification.

AH-56A ABBREVIATIONS

A

A/C	AIRCRAFT
ACE	ADAPTER COMPASS ELECTRONICS
ADC	AIR DATA CONVERTER
ADF	AUTOMATIC DIRECTION FINDER
ADI	ATTITUDE DIRECTOR INDICATOR
ADL	ARMAMENT DATUM LINE
ALT	ALTITUDE OR ALTERNATE
AM	AMPLITUDE MODULATION
AMCS	ADVANCED MECHANICAL CONTROL SYSTEM
ANT	ANTENNA
APU	AUXILIARY POWER UNIT
ARC	AUTOMATIC RESPONSIBILITY CONTROL
ATC	ATTACK
ATT	ATTITUDE
AUTO	AUTOMATIC
AZ	AZIMUTH

B

BAT	BATTERY
BDHI	BEARING DISTANCE HEADING INDICATOR
BIT	BUILT IN TEST
BITE	BUILT IN TEST EQUIPMENT
BRT	BRIGHT
BYP	BYPASS

C

CCC	COMPUTER CENTRAL COMPLEX
CCP	COMPUTER CONTROL PANEL
CCR	CLOSED CIRCUIT REFUELING
CHG	CHARGED
CLR	CLEAR
COMM	COMMUNICATION
COMP	COMPASS, COMPUTER, COMPARTMENT OR COMPENSATOR
COMPT	COMPUTER
CONT	CONTROL
CP	CHECKPOINT
CPG	COPILOT GUNNER
CPU	CENTRAL PROCESSOR UNIT
CR	CRYSOGENIC REFRIGERATOR
CRT	CATHODE RAY TUBE
CTR	CENTER
CW	CONTINUOUS WAVE

D

D/A	DIGITAL TO ANALOG
DAT	DELIVERED AIR TEMPERATURE
DDA	DIGITAL DIFFERENTIAL ANALYZER

D

DEC	DECREASE
DECR	DECREASE
DET	DETECTOR
DEV	DEVIATION
DFCS	DIRECT FEEDBACK CONTROL SYSTEM
D/F	DIRECTION FINDING
DHARS	DOPPLER-HEADING ATTITUDE REFERENCE SYSTEM
DIR	DIRECT
DIU	DIGITAL INTERFACE UNIT
DOP	DOPPLER
DRO	DESTRUCTIVE READOUT
DSNTZ	DESENSITIZER

E

E	EMPTY
ECS	ENVIRONMENTAL CONTROL SYSTEM
ECSC	ENVIRONMENTAL CONTROL SYSTEM COCKPIT
ECU	ENVIRONMENTAL CONTROL UNIT
ED	ERROR DETECTOR
EL	ELEVATION
EMER	EMERGENCY
ENG	ENGINE
EPS	ELECTRONIC POWER SUPPLY
ERR	ERROR
EXH	EXHAUST
EXT	EXTERNAL OR EXTINGUISHER

F

FAT	FREE AIR TEMPERATURE
FC	FIRE CONTROL
FCV	FLOW CONTROL AND SHUTOFF VALVE
F/CP	FRIENDLY CHECK POINT
FF	FRIEND TO FOE
FI	FIRE INTERRUPT
FLAWS	FAULT LOCATING AURAL WARNING SYSTEM
FLIR	FORWARD LOOKING INFRARED
FM	FREQUENCY MODULATION
FOV	FIELD OF VIEW

G

G	GUNNER
GEN	GENERATOR
GFE	GOVERNMENT FURNISHED EQUIPMENT
GFM	GOVERNMENT FURNISHED MATERIAL
GP	GENERAL PURPOSE
GPCM	GENERAL PURPOSE CONTROL MODULE

G

GR	GEAR
GRD	GROUND
GS	GROUND SPEED
GS/MB	GLIDE SLOPE/MARKER BEACON
GT	GROUND TRACK
GYRO	GYRO COMPASS

H

HARS	HEADING ATTITUDE REFERENCE SYSTEM
HDG	HEADING
HEL	HELMET
HF/SSB	HIGH FREQUENCY/SINGLE SIDE BAND
HOV	HOVER
HR	RADAR ALTITUDE
HRU	HEADING REFERENCE UNIT
HSSA	HELMET SIGHT SENSOR ASSEMBLY
HYD	HYDRAULIC

I

ICS	INTERCOMMUNICATION SYSTEM OR IMPROVED CONTROL SYSTEM
IDENT	IDENTIFICATION
IFF	IDENTIFICATION - FRIEND OR FOE
ILS	INSTRUMENT LANDING SYSTEM
IMP	IMPACT
INBD	INBOARD
INCR	INCREASE
IND	INDICATOR
INST	INSTRUMENT
INT	INTERCOMMUNICATION
I/O	INPUT AND OUTPUT SECTION
IR	INFRARED
IRT	INFRARED TRACKER

J

J-BOX	JUNCTION BOX
JETT	JETTISON

K

L

L	LARGE OR LEFT
LDG	LANDING
LMC	LINEAR MOTION COMPENSATION
LOC	LOCALIZER
LOS	LINE OF SIGHT
LRU	LINE REPLACEABLE UNIT

L

LSA	LIGHT SOURCE ASSEMBLY
LTS	LIGHTS

M

M	MEDIUM
MAG	MAGNETIC
MAN	MANUAL
MANU	MANEUVER
MB	MARKER BEACON
MEGA-HERTZ	MILLION CYCLES PER SECOND
MCA	MISSILE COMMAND AMPLIFIER
MFOV	MEDIUM FIELD OF VIEW
MEM	MEMORY
MIC (MIKE)	MICROPHONE
MON	MONITOR
MR	MILLIRADIAN
MS	MOISTURE SEPARATOR OR MODE SELECT
MSL	MISSILE
MP	MAP PLOTTER

N

N	**ROTATIONAL SPEED**
Nf	ROTATIONAL SPEED - FREE TURBINE
N_g	ROTATIONAL SPEED - GAS GENERATOR
Nr	ROTATIONAL SPEED - MAIN ROTOR
NARC	NARCISSUS CONTROL
NAV	NAVIGATION
NFOV	NARROW FIELD OF VIEW
NOR or NORM	NORMAL
NVS	NIGHT VISION SYSTEM

O

OPT	OPTICAL
ORIDE	OVERRIDE
OUTBD	OUTBOARD

P

P or PLT	PILOT
PC	PRINTED CIRCUIT
PCRS	PRODUCIBILITY COST REDUCTION STUDY
PDS	PILOTS DIRECT SIGHT

P

PHS	PILOTS HELMET SIGHT
PINE	PASSIVE INFRARED NIGHT VISION EQUIPMENT
PNL	PANEL
POMM	PRELIMINARY OPERATIONAL/MAINTENANCE MANUAL
POS	POSITION
PROP	PROPELLER
PWR	POWER

Q

QTY	QUANTITY

R

R	RIGHT OR RESET
RAC	REMOTE ARMAMENT CONTROL
RAD	RADIATION
RAI	RADAR ALTITUDE INDICATOR
R/B	RANGE/BEARING
R/C	RATE OF CLIMB
RCVR	RECEIVER
RECP	RECEPTACLE
RDY	READY
RKT	ROCKET
RDD	RECEIVER, DIGITAL DATA
RMR	RECEIVER MODULE, REMOTE
RPC	RECEPTACLE
RSG	RATE SWITCH GYRO
RTA	RECEIVER TRANSMITTER ANTENNA

S

S	SMALL
SAI	STANDBY ATTITUDE INDICATOR
SCIB	SIGNAL CONDITIONING INTERFACE BOX
SCS	STORES CONTROL SYSTEM
SDC	SIGNAL DATA CONVERTER
SEA	SENSOR ELECTRONICS ASSEMBLY
SEL	SELECT
SGS	SWIVELING GUNNER'S STATION
SGT	SIGHT
S/H	SIGHT HEAD
SLAE	STANDARD LIGHT WEIGHT AVIONICS EQUIPMENT
SMK	SMOKE
SPA	SIGNAL PROCESSING AMPLIFIER
SSB	SINGLE SIDE BAND
STAB	STABILIZED
STBY	STANDBY
ST	STORE

S

STOR	STORED
STU	SIGNAL TRANSFER UNIT
SYS	SYSTEM
SVG (SBVG)	STANDBY VERTICAL GYRO

T

T	TEST
TAS	TRUE AIR SPEED
TCE	TOW CONTROL EQUIPMENT
TCK ERR	TRACK ERROR
TCV	TEMPERATURE CONTROL VALVE
TDD	TRANSMITTER, DIGITAL DATA
TMR	TRANSMITTER MODULE, REMOTE
TOW	TUBE LAUNCHED, OPTICALLY TRACKED, WIRE GUIDED MISSILE SYSTEM
TRANS	TRANSFER
T/R	TRANSMITTER RECEIVER OR TRANSFORMER RECTIFIER
TRG	TURN RATE GYRO
TRK	TRACK

U

UHF	ULTRA HIGH FREQUENCY
UTM	UNIVERSAL TRANSVERSE MERCATOR

V

VAR	VARIATION
VEH	VEHICLE
VHF	VERY HIGH FREQUENCY
VOL	VOLUME
VOR	VHF OMNI RANGE
VRU	VERTICAL REFERENCE UNIT
VT	VARIABLE TIME (PROXIMITY)
Vd	CROSS HEADING PORTION OF AIRCRAFT VELOCITY VECTOR
Vh	ON-HEADING PORTION OF AIRCRAFT VELOCITY VECTOR
Vv	VERTICAL HEADING PORTION OF AIRCRAFT VELOCITY VECTOR

W

WCU	WEAPONS CONTROL UNIT
WE	WATER SEPARATOR
WPN	WEAPON
WSHLD	WINDSHIELD
WFOV	WIDE FIELD OF VIEW

X

XM-51	TURRET SYSTEM WITH XM-129 (40mm) GUN
XM-52	TURRET SYSTEM WITH XM-140 (30mm) GUN
XM-53	TURRET SYSTEM WITH XM-134 (7.62mm) GUN
XMSN	TRANSMISSION
XMTR	TRANSMITTER

Y

YASAS	YAW STABILITY AUGMENTATION SYSTEM

Z

FLIGHT TEST AND FLIGHT TEST DATA

The AH-56A Compound Helicopter has been tested in various configurations during development over the past few years. The results of these flight test performance evaluations and the flight test data reductions have provided a formidable amount of scientific data. These data are usable on existing configurations and available for future producible configurations which have designated mission requirements in performance. Some of the specific performance evaluations have been selected to provide an overview of the characteristics of the Cheyenne as an attack helicopter and as a stable mission ready weapons system.

These curves and tables include:

1. Compound Helicopter Basic Curves
2. Thrust and Power Propeller Characteristics (2 of 2)
3. Collective Envelope
4. Longitudinal Maneuver Envelope
5. Hover and Beta Curves
6. V-n Diagram
7. Maneuvering Stability - 120 KCAS
8. Maneuvering Stability - 203 KCAS
9. Sideward Flight
10. Autorotation R/D vs. A/S
11. Low Speed Auto Entry
12. High Speed Auto Entry
13. Autorotative Landing
14. H-V Diagram
15. Level Flight Performance
16. Climb Performance

17. Level Maximum Acceleration

18. Level Maximum Deceleration

19. Structural Measurement Table

A PROPELLER THRUST AND POWER RELATIONSHIPS

B WING – ROTOR LOAD SHARING

TRADEOFFS

1 LOWER VEHICLE POWER REQUIRED	}	HIGH COLLECTIVE ADVANTAGES
2 MORE NOSE DOWN ATTITUDE		
3 INCREASED MANEUVERING CAPABILITY	}	LOW COLLECTIVE ADVANTAGES
4 LOWER VIBRATION LEVEL		
5 MORE NOSE-UP ATTITUDE		

Figure 25-1. Compound Helicopter Basic Concepts

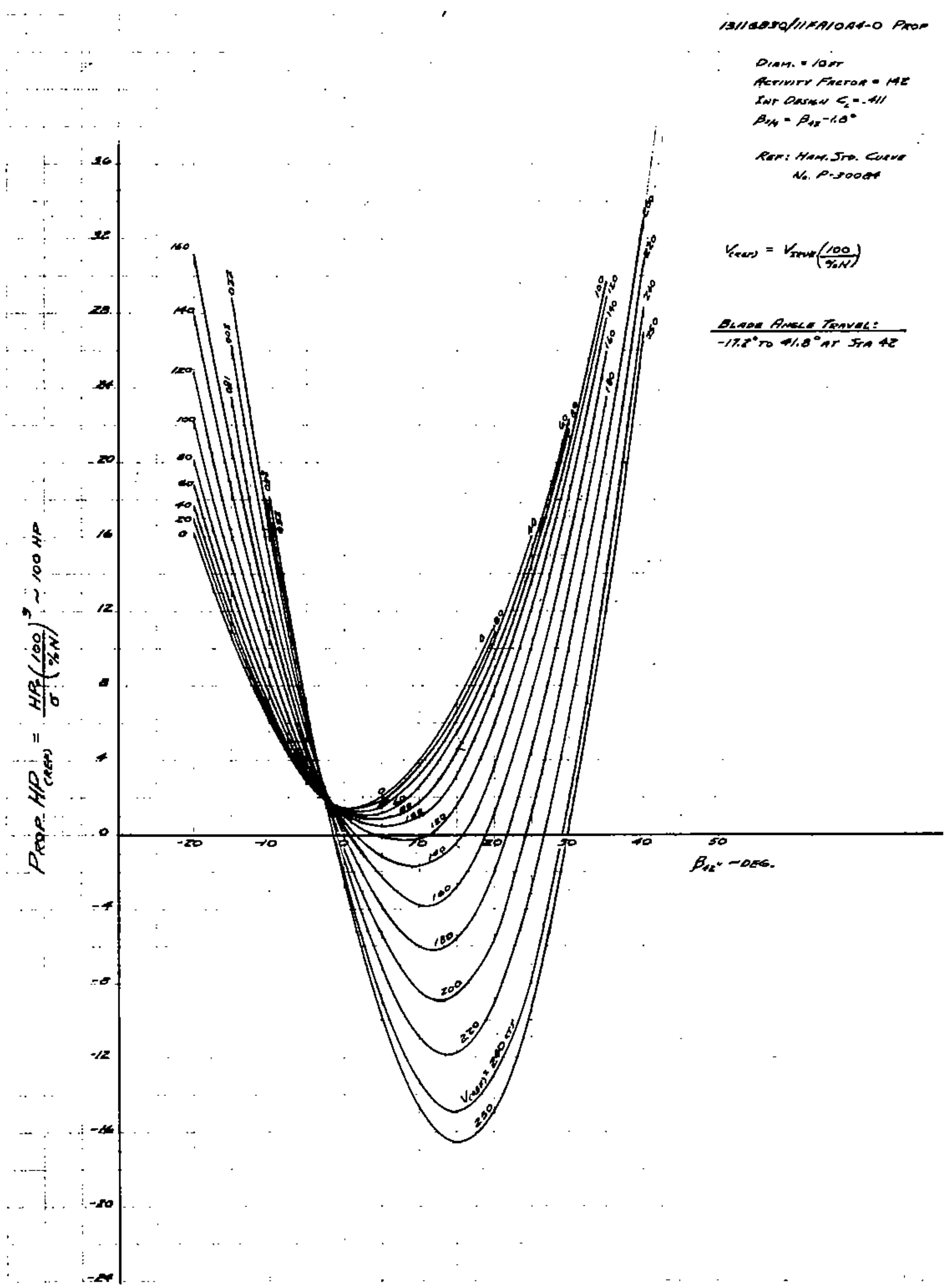


Figure 25-2. Variation of Prop Horsepower with Blade Angle and Airspeed (1 of 2)

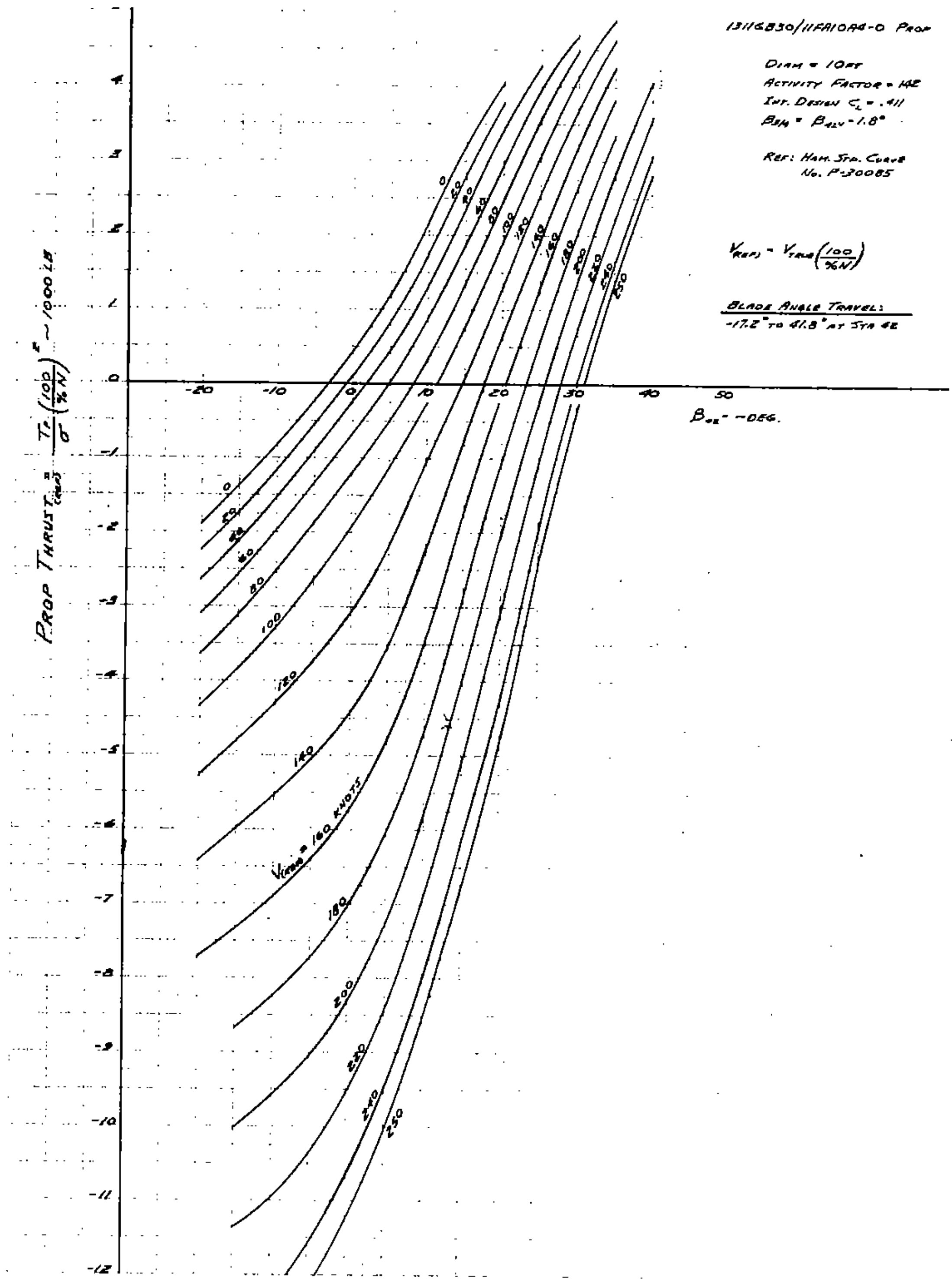


Figure 25-2. Variation of Prop Thrust Horsepower with Blade Angle and Airspeed (2 of 2)

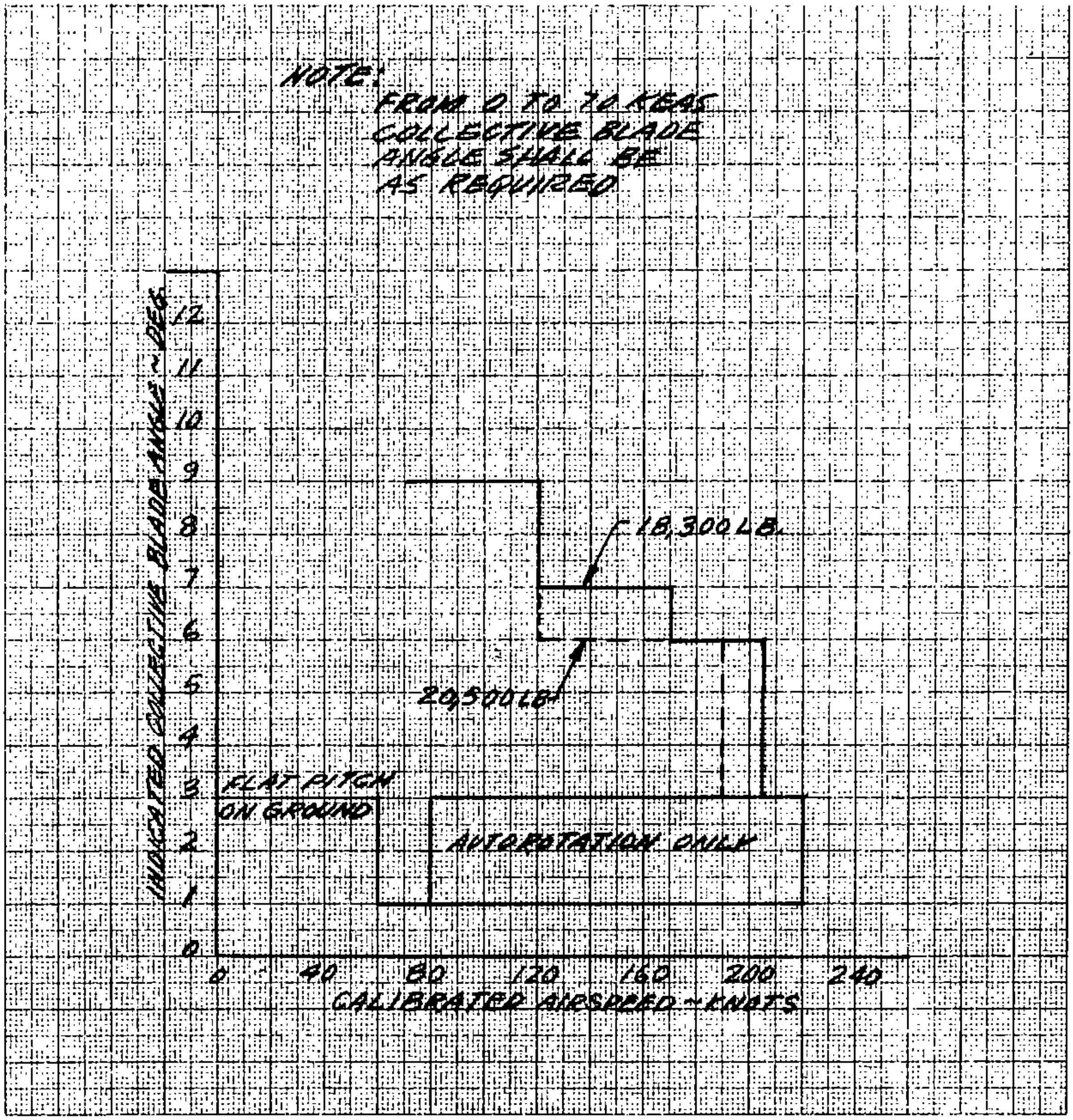


Figure 25-3. AH-56A Collective Blade Angle ~ Airspeed Envelope
Effective 30 Aug. 71

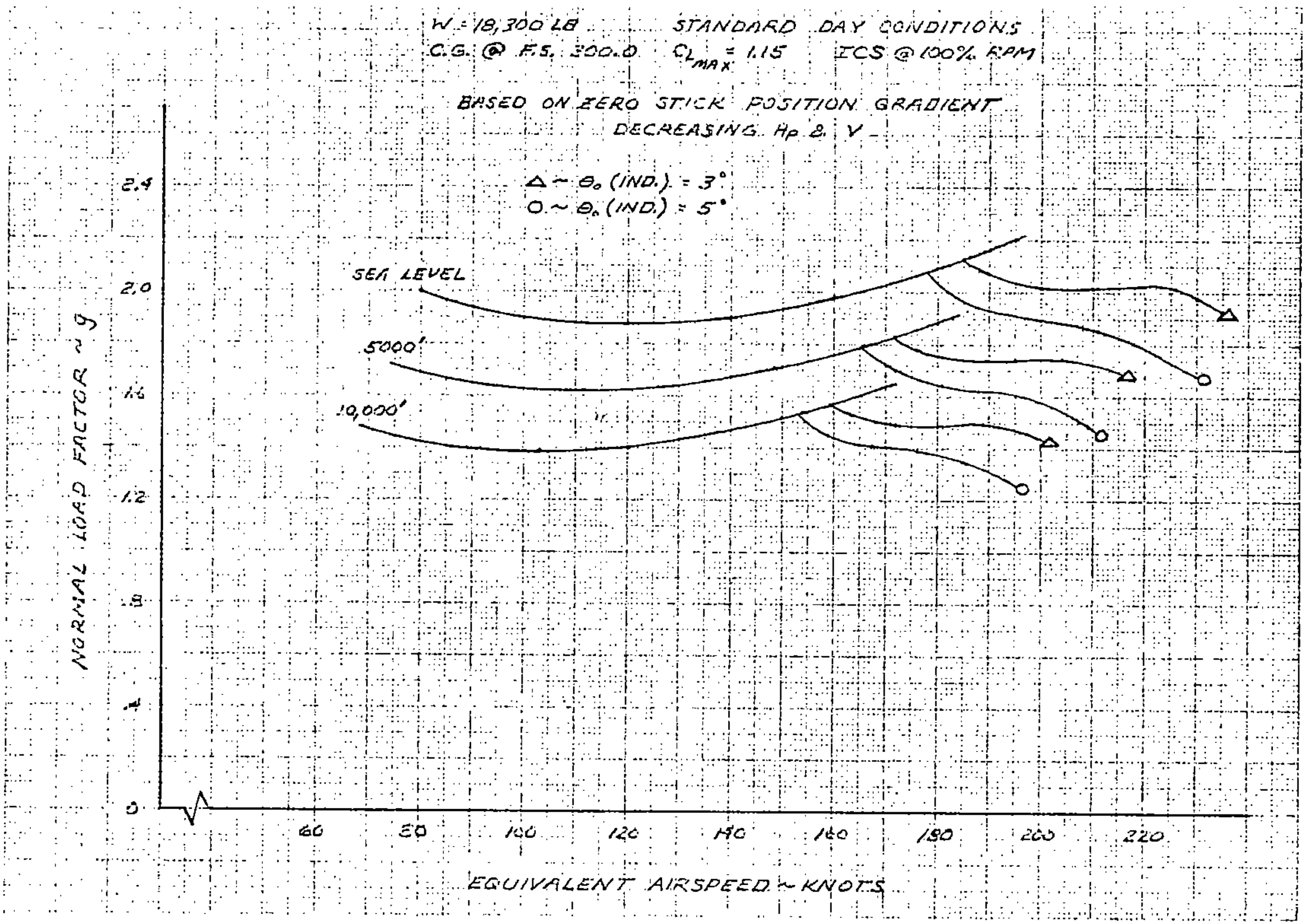


Figure 25-4. AH-56A Longitudinal Maneuvering Boundary

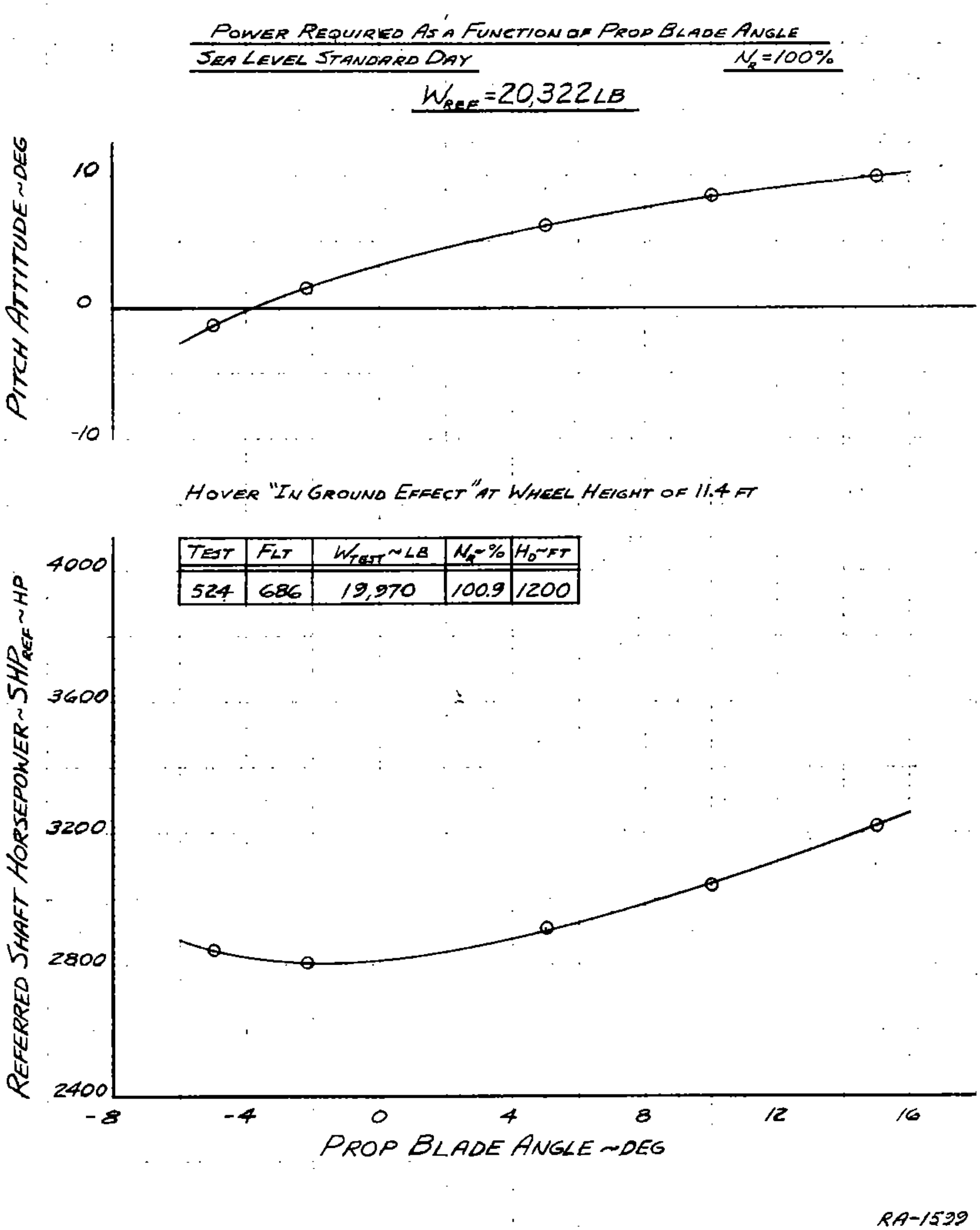


Figure 25-5. Hover Performance

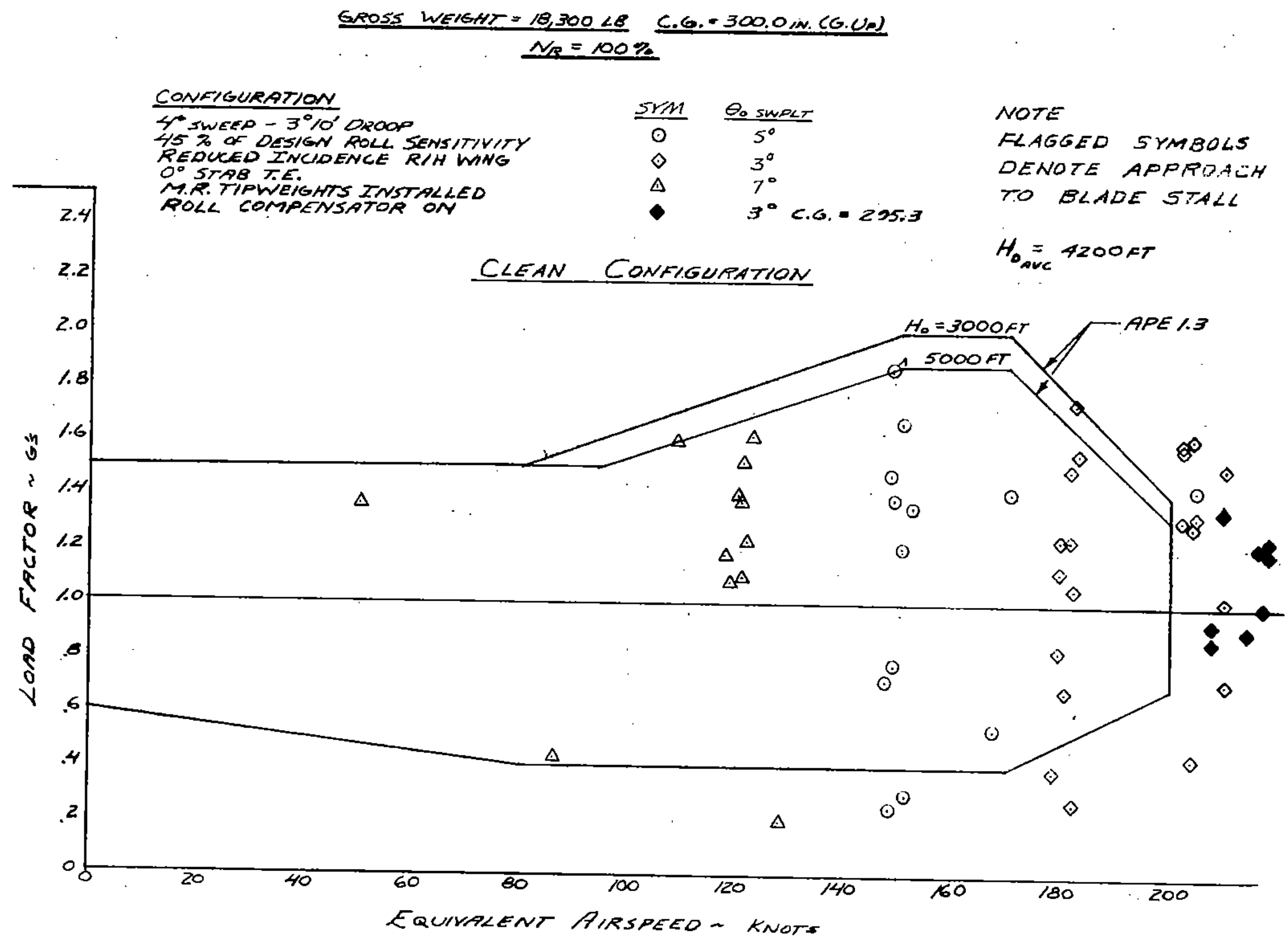


Figure 25-6. Load Factor - Airspeed Envelope

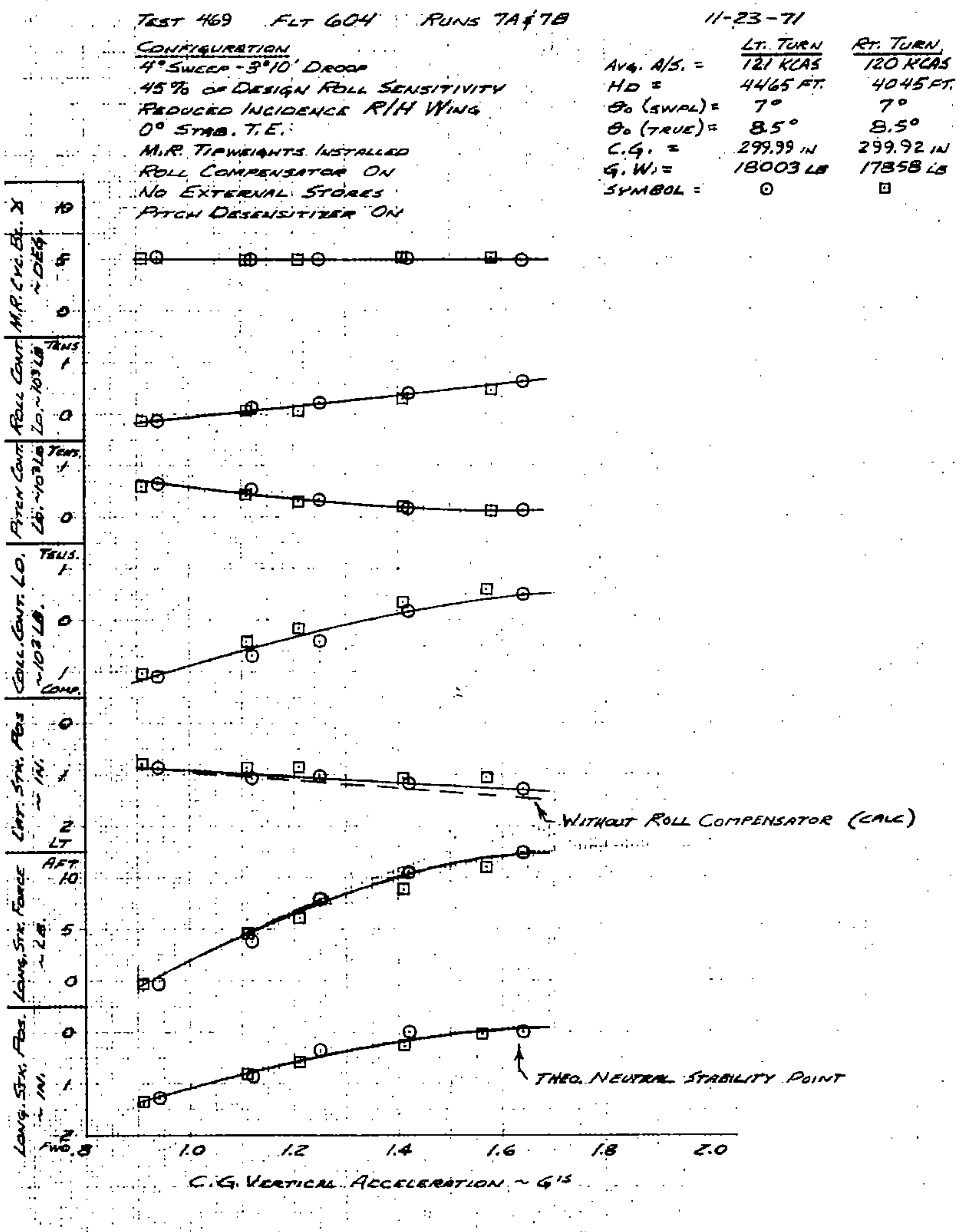


Figure 25-7. Maneuvering Stability - Left and Right Wind-Up Turn (Test 469)

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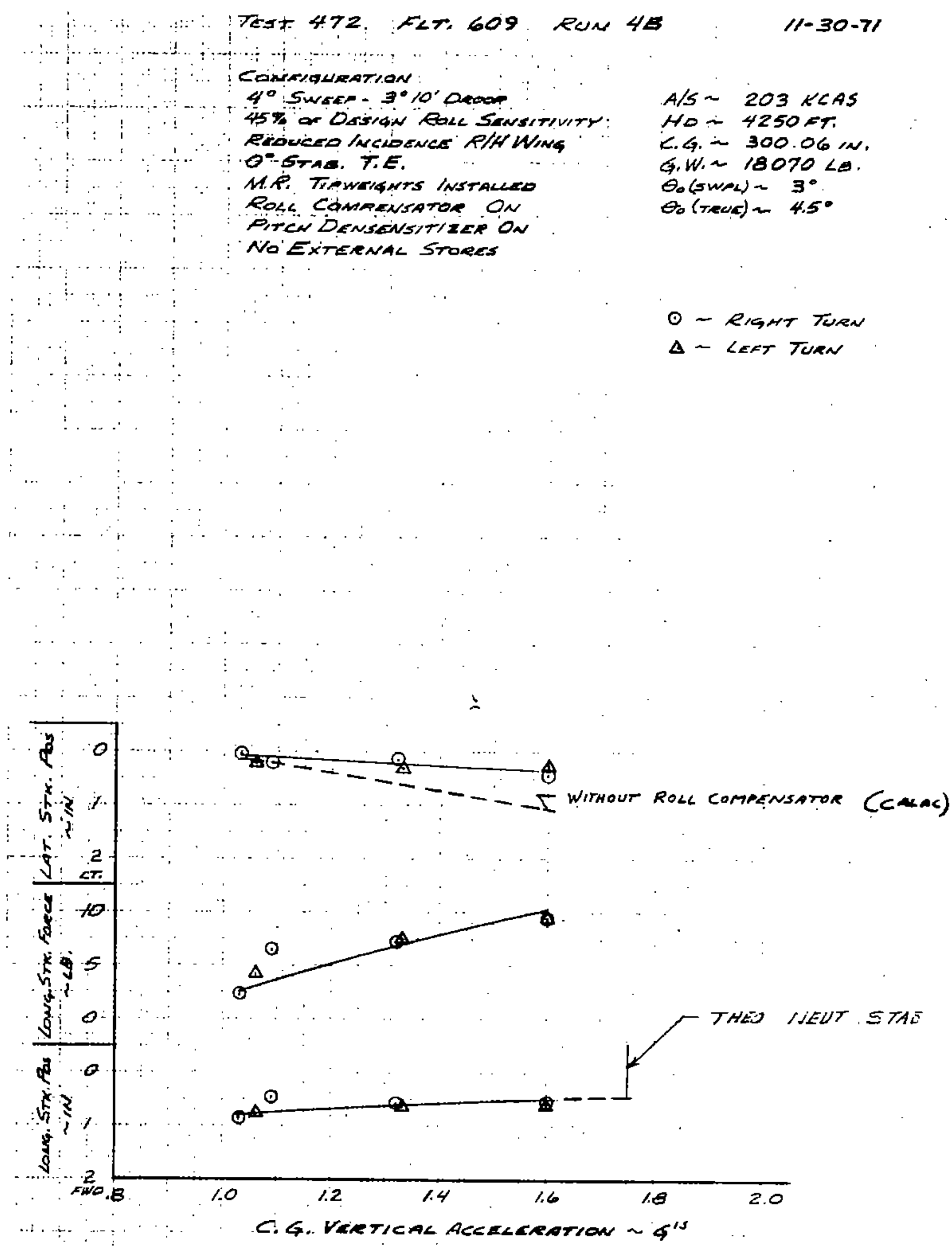

Figure 25-8. Maneuvering Stability - Left and Right Wind-Up Turn (Test 472)

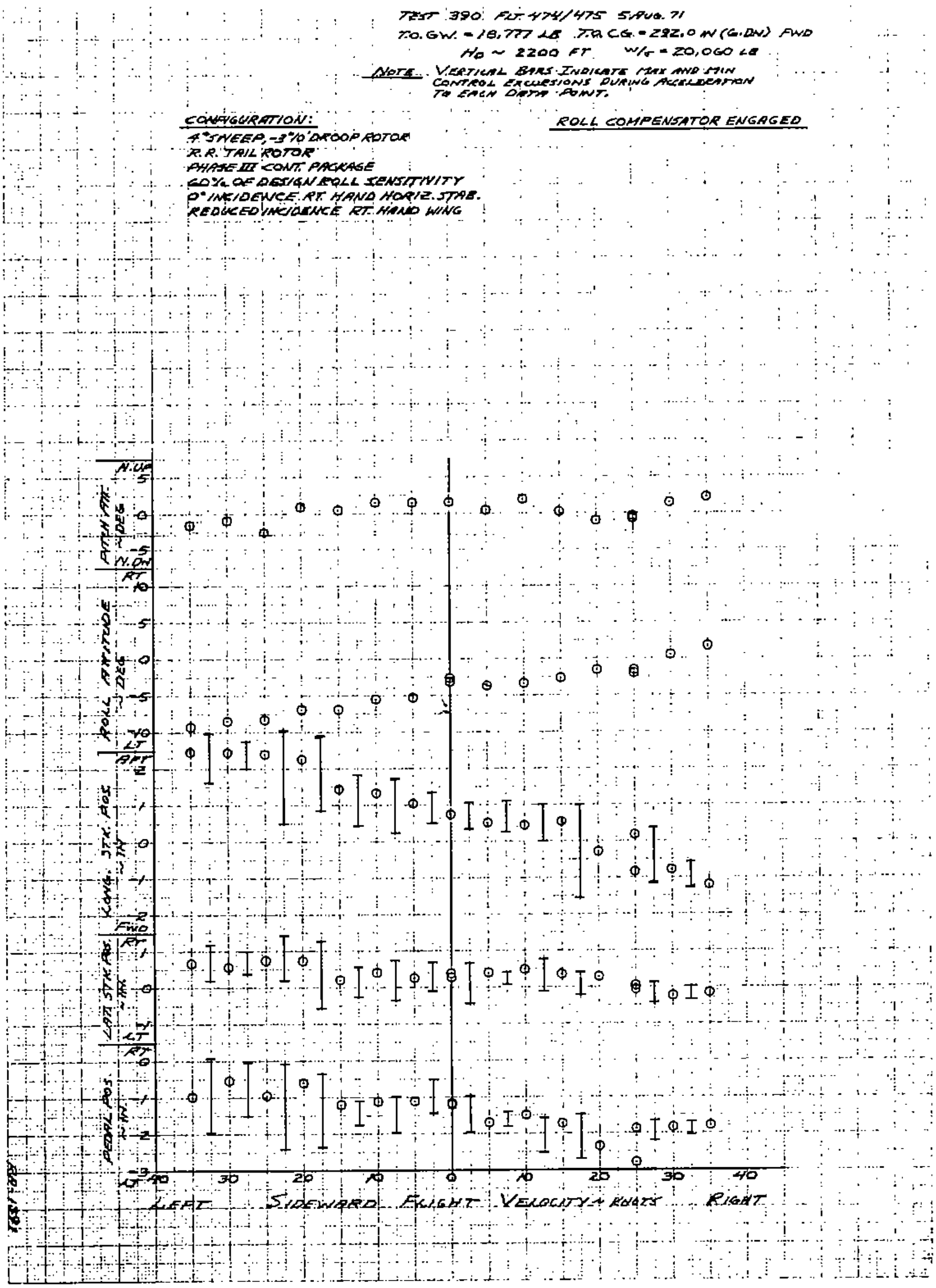

Figure 25-9. Sideward Flight

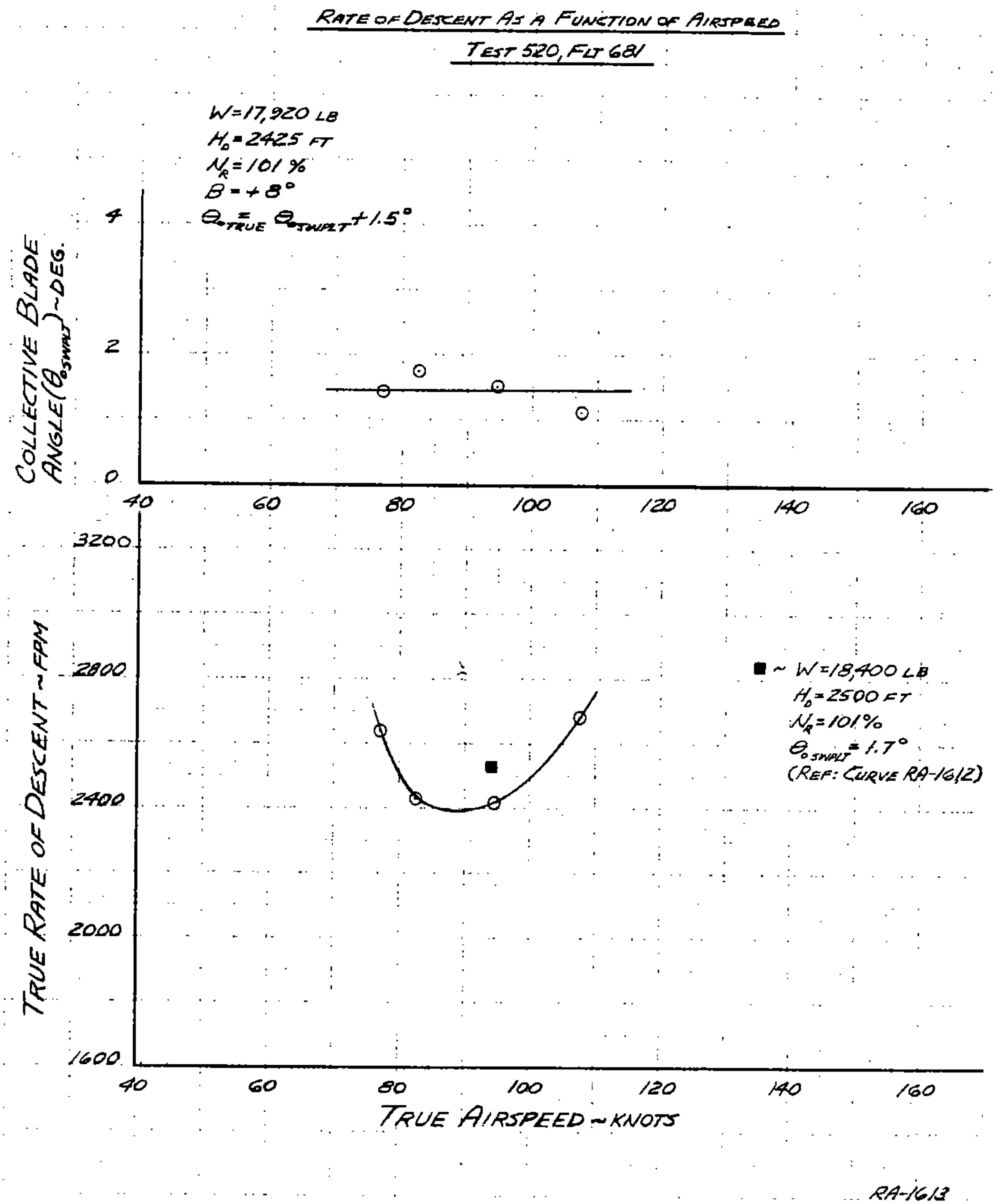

Figure 25-10. Autorotation Performance

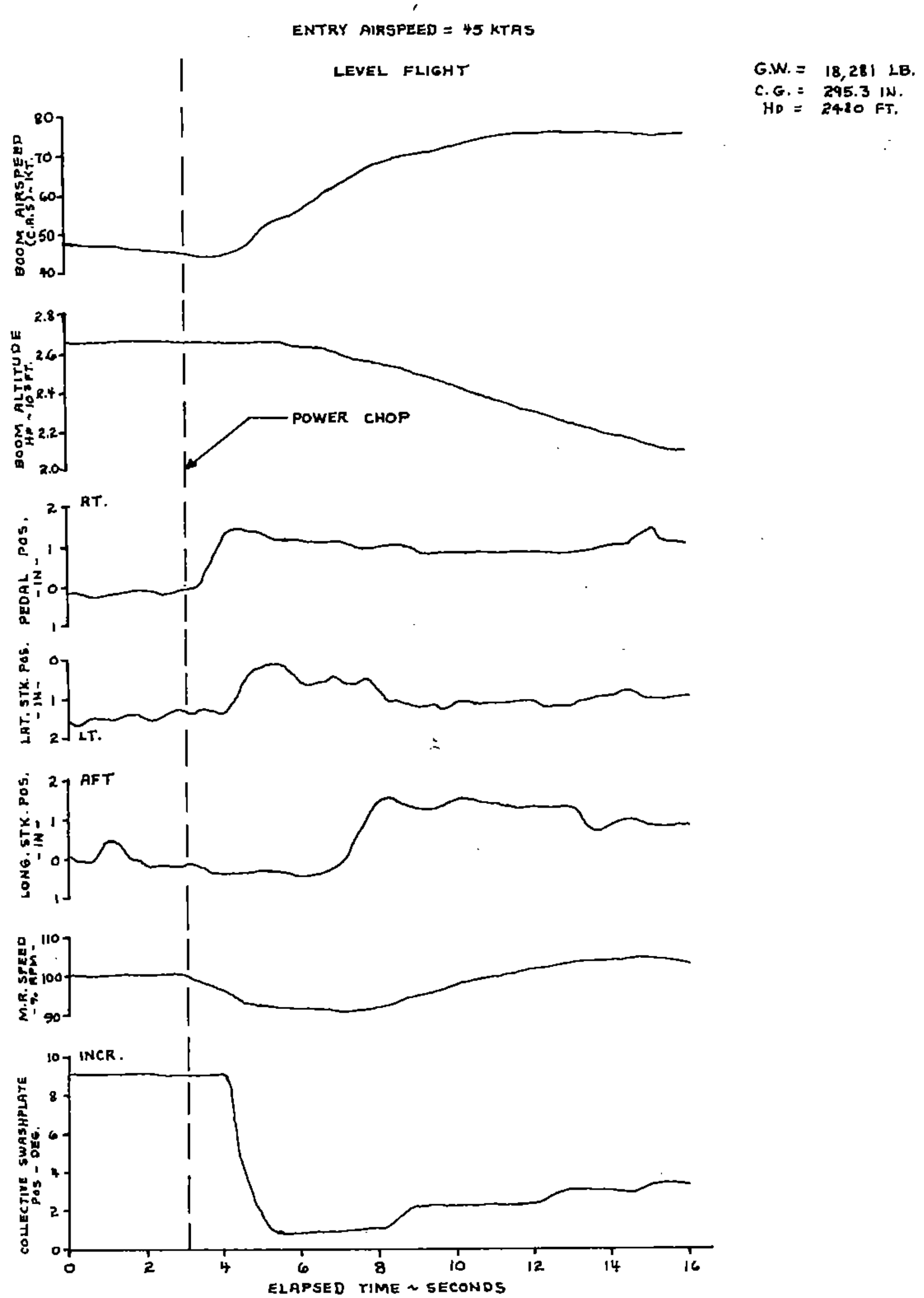

Figure 25-11. Low Speed Autorotation Entry

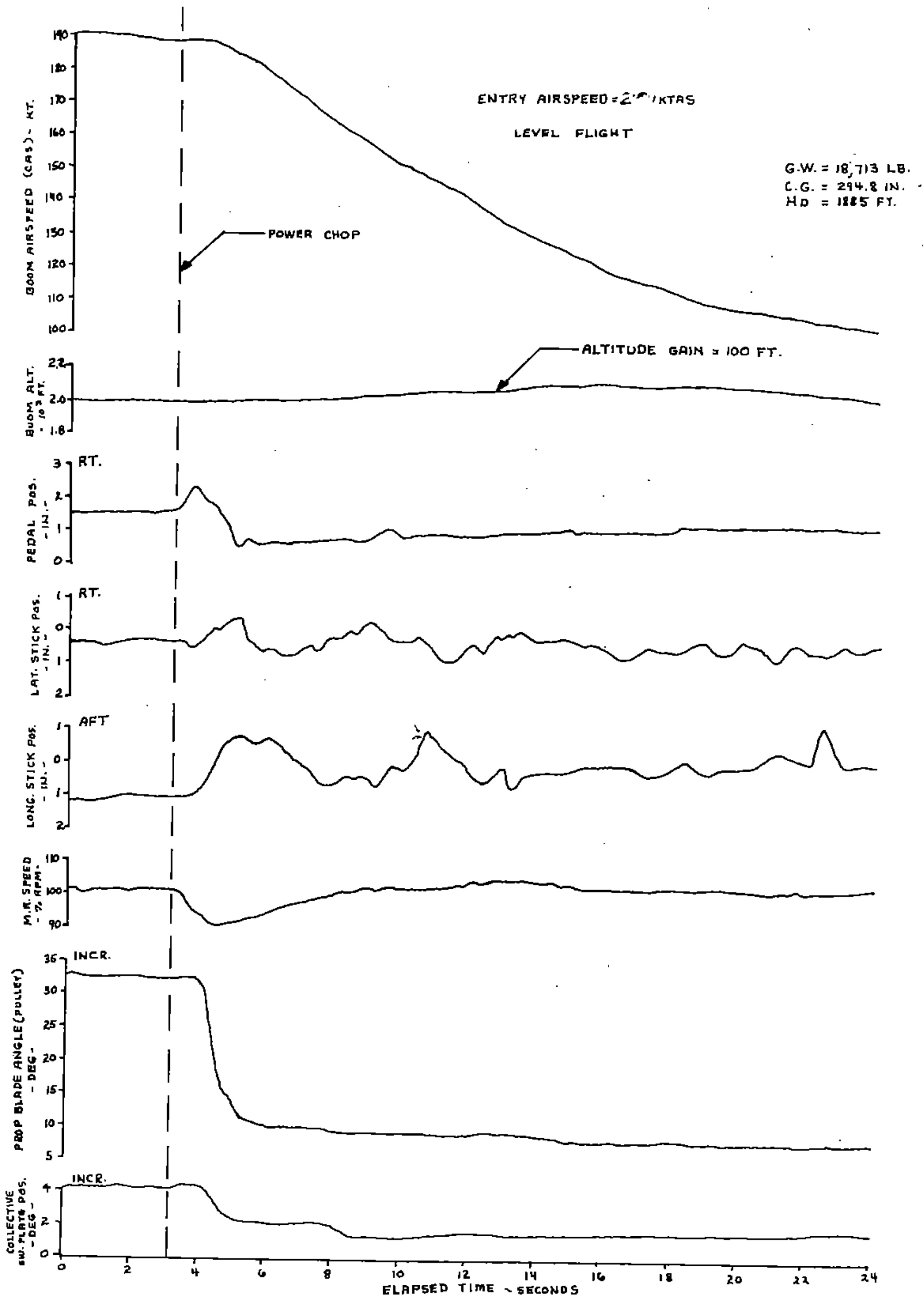

Figure 25-12. High Speed Autorotation Entry

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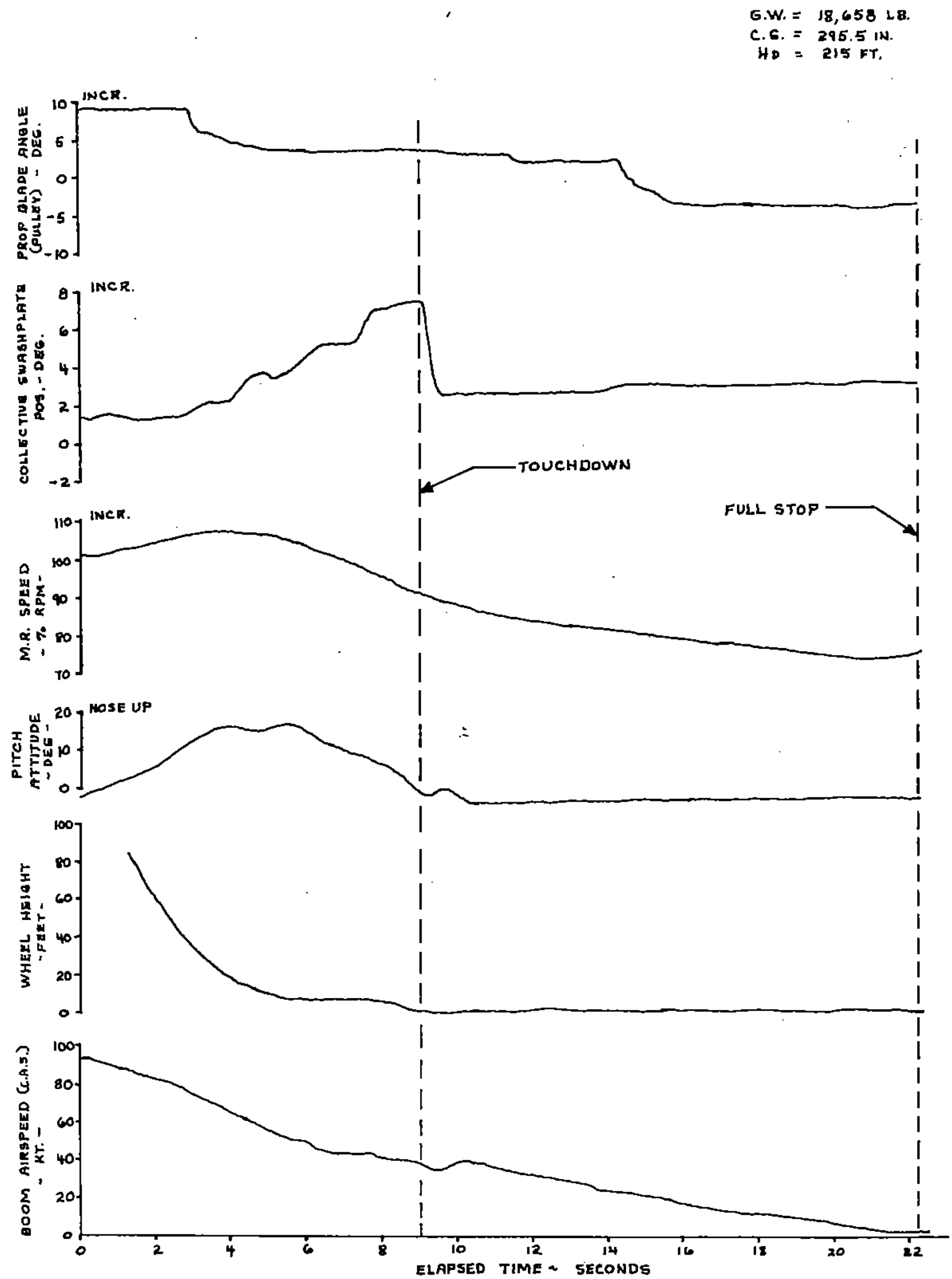

Figure 25-13. Autorotation Landing/Full Stop

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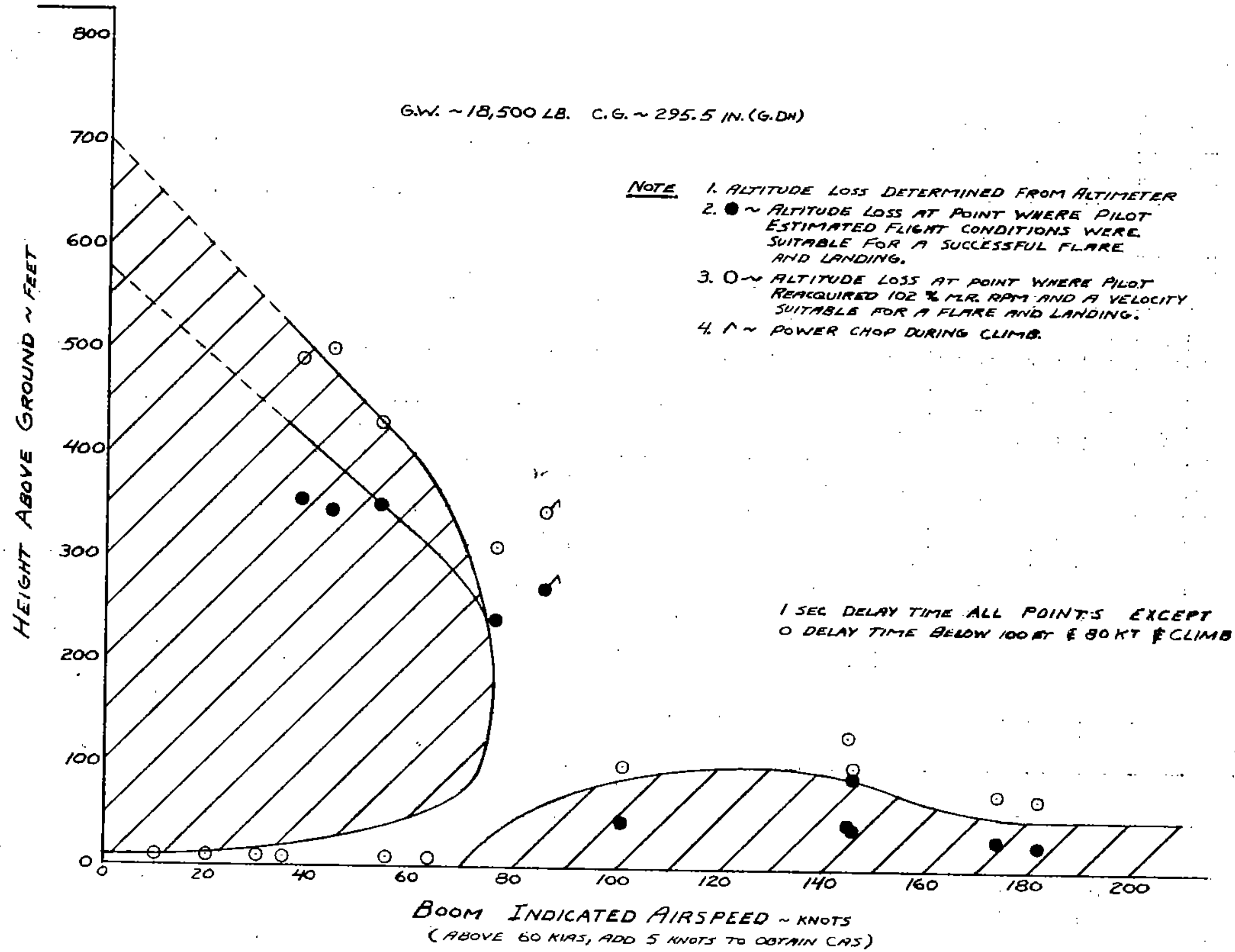


Figure 25-14. Height - Velocity Diagram

RA-1611

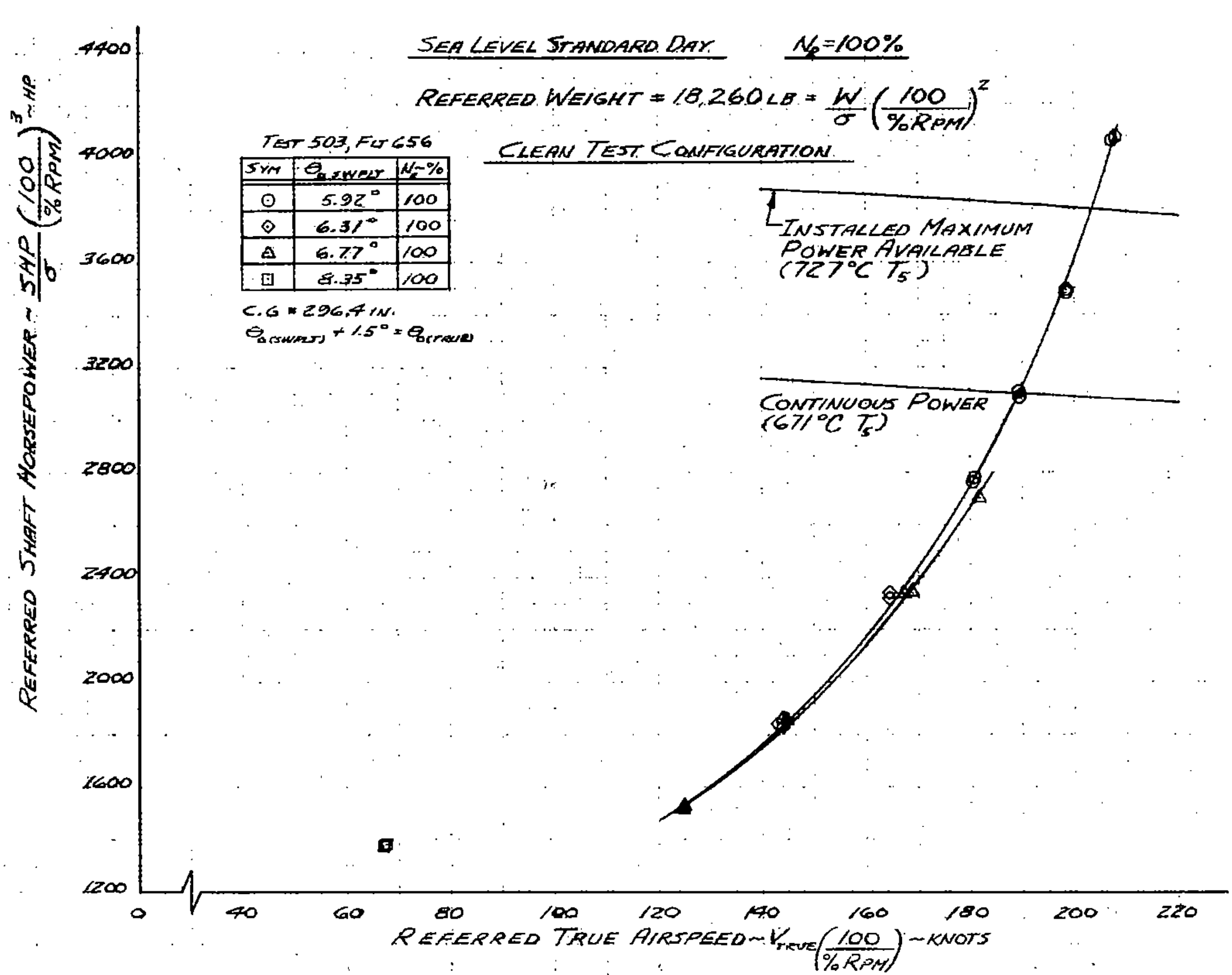


Figure 25-15. Level Flight Performance

RATE OF CLIMB AS A FUNCTION OF TRUE AIRSPEED

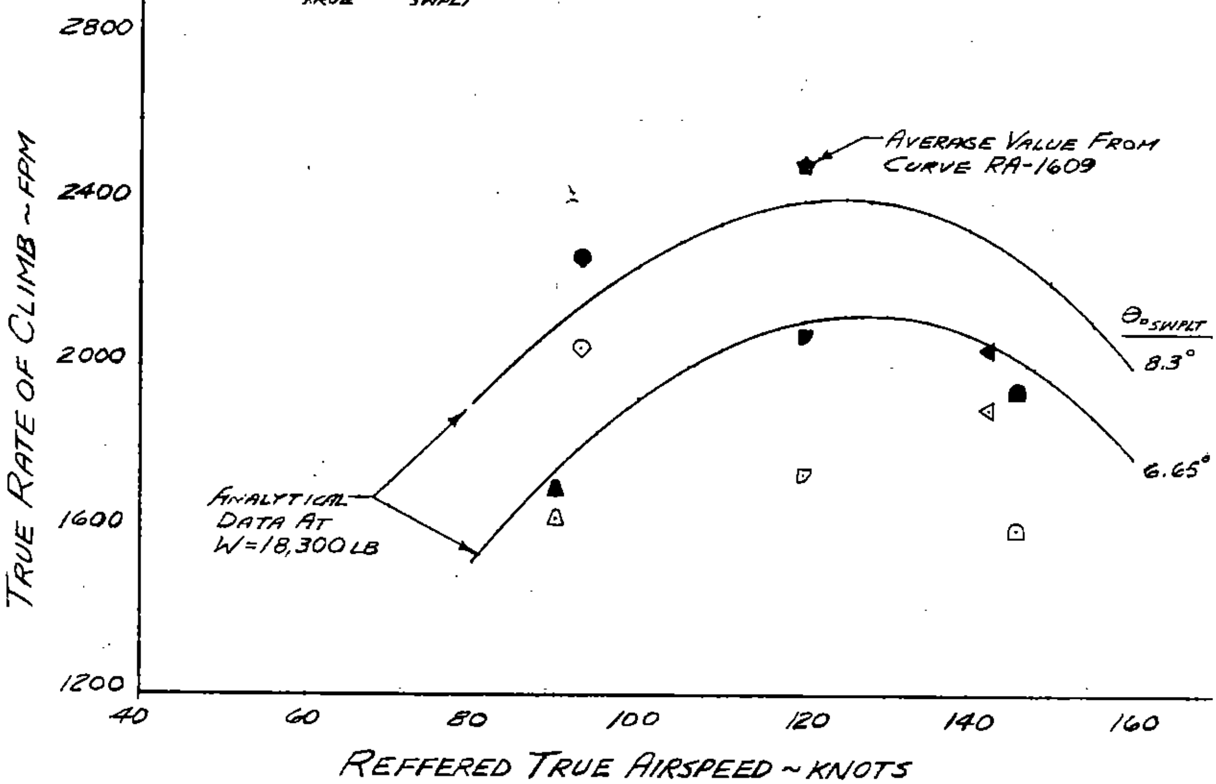
SEA LEVEL STANDARD DAY N_R=100%

W = 18,300 LB AT MAXIMUM INSTALLED POWER

SYM	TEST	FLT	W_{REF} ~ LB	SHP_{REF} ~ HP	$\theta_{o\,SWPLT}$	H_D ~ FT
△	507	662	18,600	3785	6.7°	1700
▽	507	662	19,850	3830	6.7°	2825
◁	507	662	19,080	3860	6.6°	1750
⌂	507	662	20,050	3935	6.6°	3425
◇	507	662	19,050	3805	8.3°	2100

NOTES:

1. RPM WAS 100% FOR ALL TESTS CONDITIONS.
2. SOLID SYMBOLS INDICATE RATE OF CLIMB DATA ADJUSTED TO 18,300 LB AND MAXIMUM INSTALLED POWER AVAILABLE BY APPLYING INCREMENTAL ANALYTICAL CORRECTIONS.
3. $\theta_{o\,TRUE} = \theta_{o\,SWPLT} + 1.5°$.



RA-1610

Figure 25-16. Climb Performance

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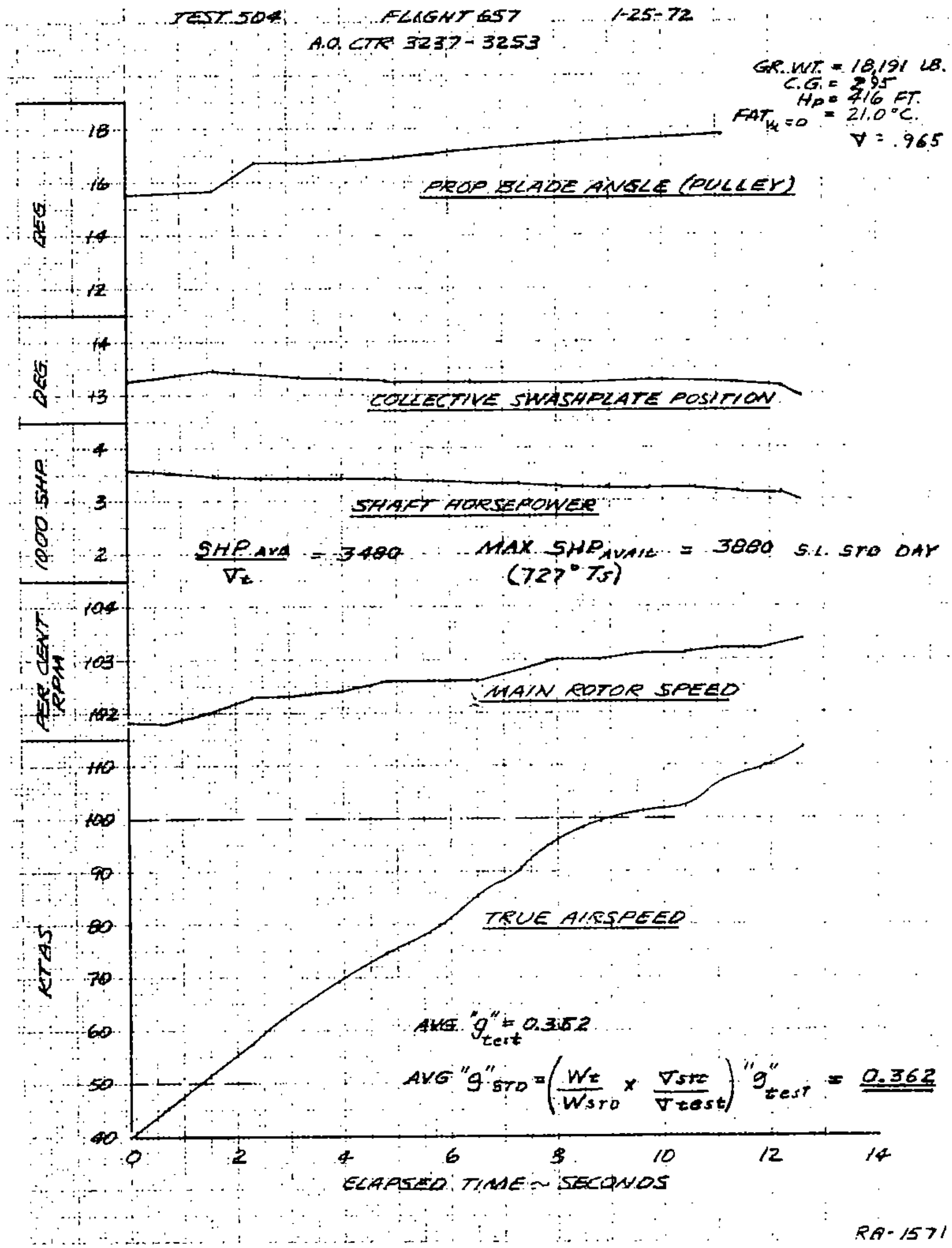

Figure 25-17. Level Maximum Acceleration

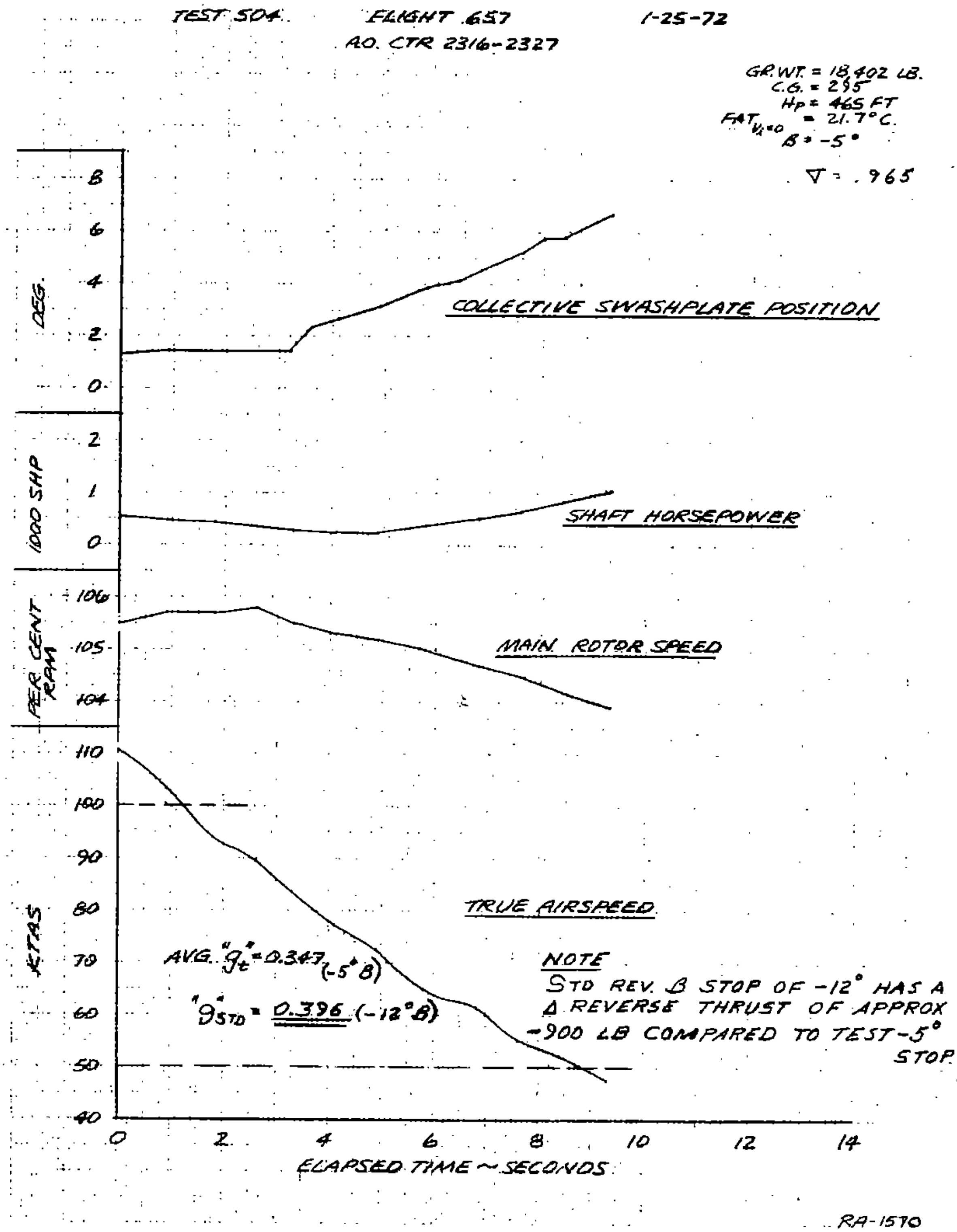

Figure 25-18. Level Maximum Deceleration

MAIN ROTOR

TAIL ROTOR

PROPELLER

MAIN ROTOR CONTROL SYSTEM

TAIL ROTOR CONTROL SYSTEM

GYRO

DRIVE TRAIN

WING

HORIZONTAL STABILIZER

Figure 25-19. AH-56A Structural Measurements