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FLY-BY-WIRE VERSUS DUAL MECHANICAL CONTROLS FOR THE ADVANCED SCOUT HELICOPTER — QUANTITATIVE COMPARISON

LEVEL II

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APPLIED TECHNOLOGY LABORATORY POSITION STATEMENT

This effort is one of two parallel contractual studies to evaluate the payoffs associated with application of advanced control technology (including fly-by-wire, fiber optics, and digital control laws) to an ASH-sized helicopter. The associated study program under the same title was conducted by Boeing Vertol under the terms of contract DAAK-79-C-0008. As a parallel effort to these contracts, Sikorsky performed a similar study funded through their IR&D program. The results of the Sikorsky study may be made available to Government personnel by contacting the project engineer.

As a baseline for this study, the Medium Utility Transport (MUT) was chosen based upon similarity to ASH requirements and the use by MUT of a modern dual mechanical control system. The results of this study are useful not only for defining projected payoffs associated with the use of advanced control technology, but also for projecting a maturity rate for advanced control technology.

Mr. John W. Stephens, Jr., Aeronautical Systems Division, served as project engineer for this program.

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20. ABSTRACT (Continue on reverse side if necessary and identify by block number)

A preliminary design study comparing fly-by-wire with dual mechanical controls for the Advanced Scout Helicopter was conducted by Bell Helicopter Textron. The purpose was to determine if significant payoffs were available in the utilization of fly-by-wire design concepts. Considered were flight safety and mission reliability; vulnerability; system reliability, availability, and maintainability; control system weight; electrical and hydraulic next page

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power supply requirements; initial and life-cycle costs; and predicted handling qualities. In addition, the study investigated the payoffs from the use of innovative thinking relative to cockpit controls.

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1. INTRODUCTION

The availability of a maturing electronics technology in flight controls has established the possibility of significant payoffs in the utilization of fly-by-wire concepts for helicopter flight control systems (FCS). Whether such payoffs are to be available in an Advanced Scout Helicopter (ASH) size helicopter is the question to which this effort has been directed.

Specifically, this effort has established a preliminary design fly-by-wire/light (FBW/L) control system in sufficient detail that a quantitative comparison to a dual mechanical control system could be made.

The system resulting from the preliminary design utilizes optical transducers to measure pilot input and optical fibers to transfer command inputs to power actuators controlling the helicopter. This system is referred to as a fly-by-wire/light control system due to the fact that "wires" and "light" are used to implement helicopter control instead of the conventional push-pull tube mechanisms.

The preliminary design FBW/L system utilizes a multiarm rise/fall swashplate, independent channel concept for the main rotor and a redundant path electromechanical implementation for both the horizontal stabilizer (elevator) and tail rotor. This system was compared to a dual mechanical control system for a particular baseline helicopter in the following areas:

- Flight safety reliability
- Mission reliability
- Vulnerability
- System reliability, availability, and maintainability
- Control system weight
- Electrical and hydraulic power supply requirements
- Initial and life-cycle cost
- Predicted handling qualities

In addition to the quantitative comparison, the effort investigated the possibility of gaining additional payoffs from the use of innovative thinking relative to cockpit controls and mission displays. The FBW/L control may be viewed as part

of the total modernization of the helicopter's cockpit and instruments that is required to perform the demanding crew tasks associated with the flight controls and mission equipment of the ASH. The nap-of-the-earth flight tasks, with periodic unmasking for target detection and attack, requires automatic and crew-commanded functions. Through command augmentation, the helicopter can be made to exhibit decoupled controls, automatic turn coordination, trim compensation for power change effects, and compensation for downwash effects on the horizontal stabilizer. These features allow the pilot to concentrate his effort more on the tactical situation and less on the basic flight requirements. Special displays and controls, such as side-arm and push-button controls, may be implemented for use with the FBW/L and automatic flight control system that offers savings in weight and cockpit space.

Section 2 presents the details of the preliminary design of the FBW/L control system. Special attention was given to the determination of the size, space, and configuration constraints imposed by the baseline helicopter as described in the Reference 1 report. Both the dual mechanical and the FBW/L control systems were addressed in the predesign effort to a sufficient technical depth to insure a realistic quantitative comparison.

Section 3 presents the specific details of the quantitative comparison in the various areas. It is shown that significant benefits are predicted in all areas of comparison.

Section 4 presents a discussion of the particular redundancy management techniques proposed for the FBW/L control system. Five different states of each of the rotor control actuators were identified and quantified by a failure model. These states were then considered in various combinations to derive mission abort and flight safety critical situations. The probability of these events was derived using statistical techniques and these techniques were utilized in the reliability predictions of the quantitative comparison.

Section 5 presents a discussion of the potential for advanced cockpit controls and displays in a FBW/L implementation.

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[1]Hoffstadt, Donald J., and Swatton, Sidney, ADVANCED HELICOPTER STRUCTURAL DESIGN INVESTIGATION, VOLUME I - INVESTIGATION OF ADVANCED STRUCTURAL COMPONENT DESIGN CONCEPTS, Boeing Vertol Co., USAAMRDL TR 75-56A, Eustis Directorate, U.S. Army Air Mobility Research and Development Laboratory, Fort Eustis, Virginia, March 1976, ADA024662.

Advantages and disadvantages of between-the-legs versus side-arm sticks, and displacement-control versus force-control sticks are presented.

Section 6 presents an evaluation of the significance of a lighter control system as provided by the FBW/L control system. Basically, the weight savings may be utilized for extra fuel (more range) or extra payload.

2. PRELIMINARY DESIGN

In the following paragraphs, the definition of both a dual mechanical control system and a fly-by-wire control system will be presented. These two systems are defined to satisfy the requirements of the ASH mission and will be quantitatively compared in subsequent sections of this report.

2.1 DUAL MECHANICAL DEFINITION

In order to assess the desirability of a FBW FCS in the ASH, a baseline helicopter dual mechanical flight control system configured for the ASH mission was defined as discussed in the following paragraphs.

2.1.1 Baseline Definition

The baseline aircraft was designed during an advanced structures study of a medium-range utility transport (MUT) helicopter. This vehicle is shown in Figure 1 from USAAMRDL Report TR 75-56A (Reference 1). The MUT has a gross weight of 9544 pounds and a payload of 960 pounds.

This design features a single, main rotor system employing a hingeless rotor blade concept that is powered by twin advanced technology engines.

The pilot's compartment accommodates a crew of two with side-by-side seating; the aircraft's critical dimensions are noted on the drawing in Figure 2.

Flight control of the aircraft is accomplished by using a redundant mechanical system coupled with inputs from a redundant SCAS (Stability Control Augmentation System) to hydraulic actuators controlling the main and tail rotors (Figure 3).

The main rotor actuators impart motion to the nonrotating ring of the swashplate assembly. This motion is transferred to the rotating ring of the swashplate that provides pitch control to the rotor blades through pitch links.

The tail rotor actuators impart motion to the rotating sliding sleeve on the tail rotor shaft. The sleeve transfers pitch control to the tail rotor blades through pitch links.

The drive system consists of: two-engine right-angle-nose, main rotor, intermediate, and tail rotor transmissions; accessory gearboxes; and interconnecting sectionalized shafting (Figure 4).

Figure 1. Medium-range utility transport helicopter (MUT).

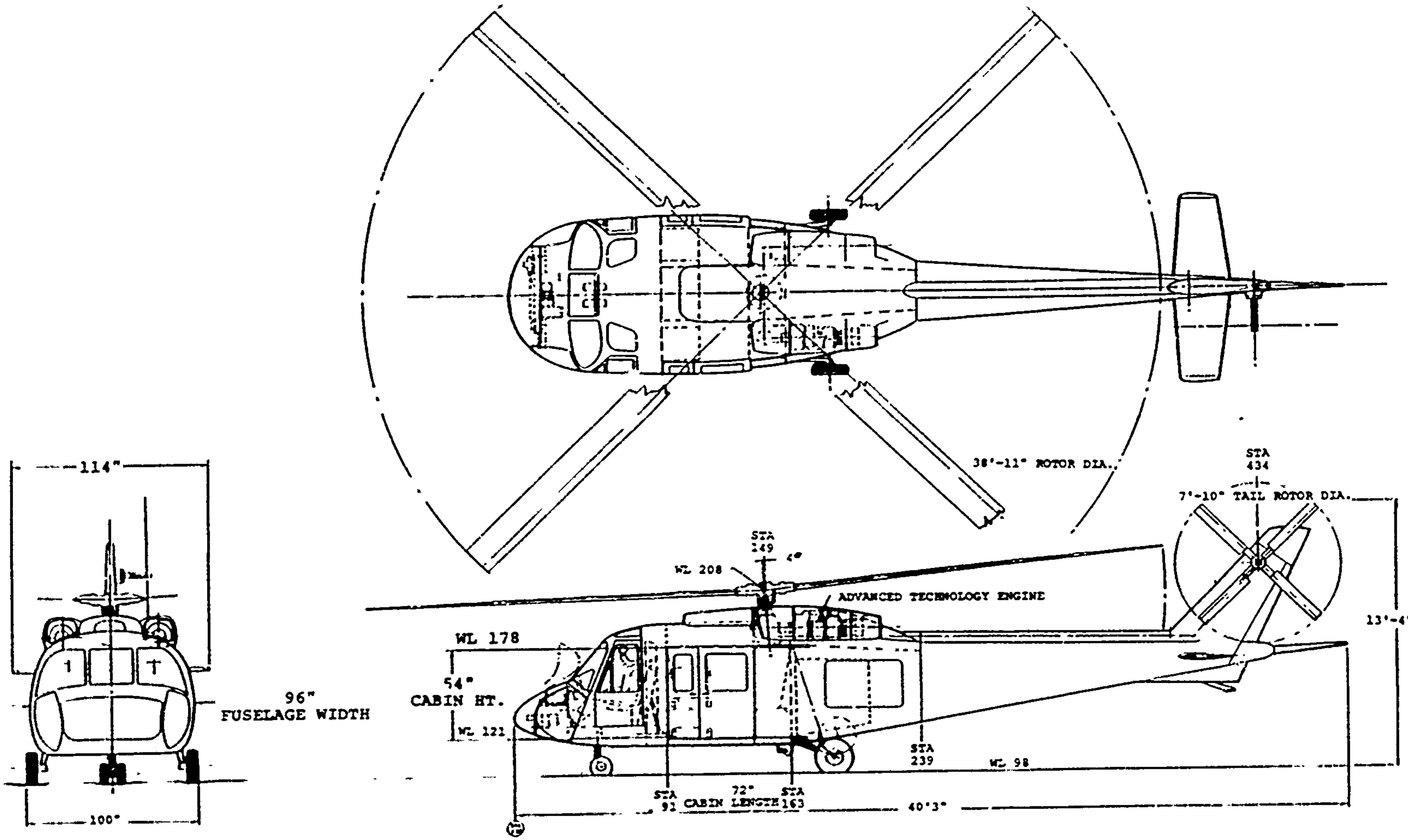


Figure 2. Baseline MUT three-view drawing.

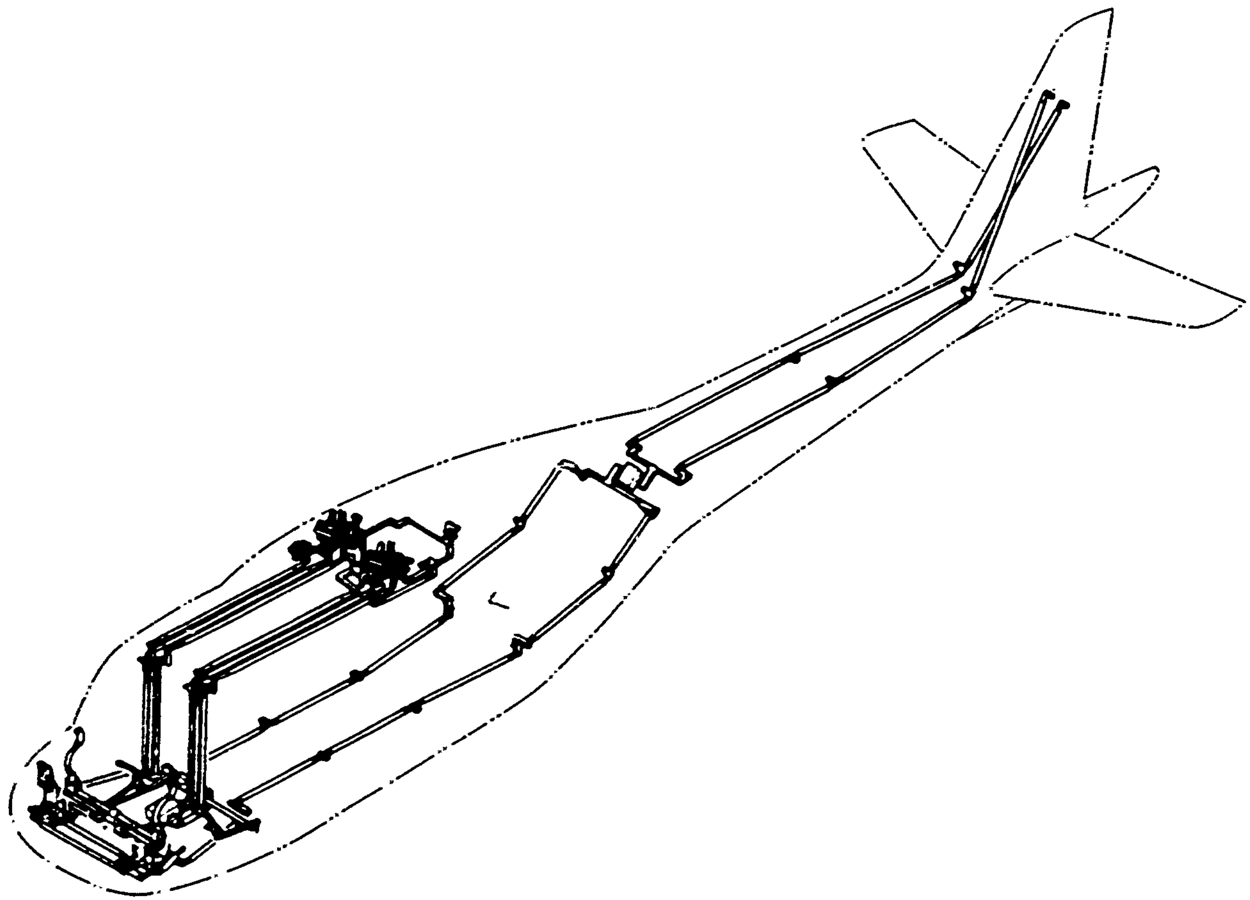


Figure 3. Dual mechanical flight control system.

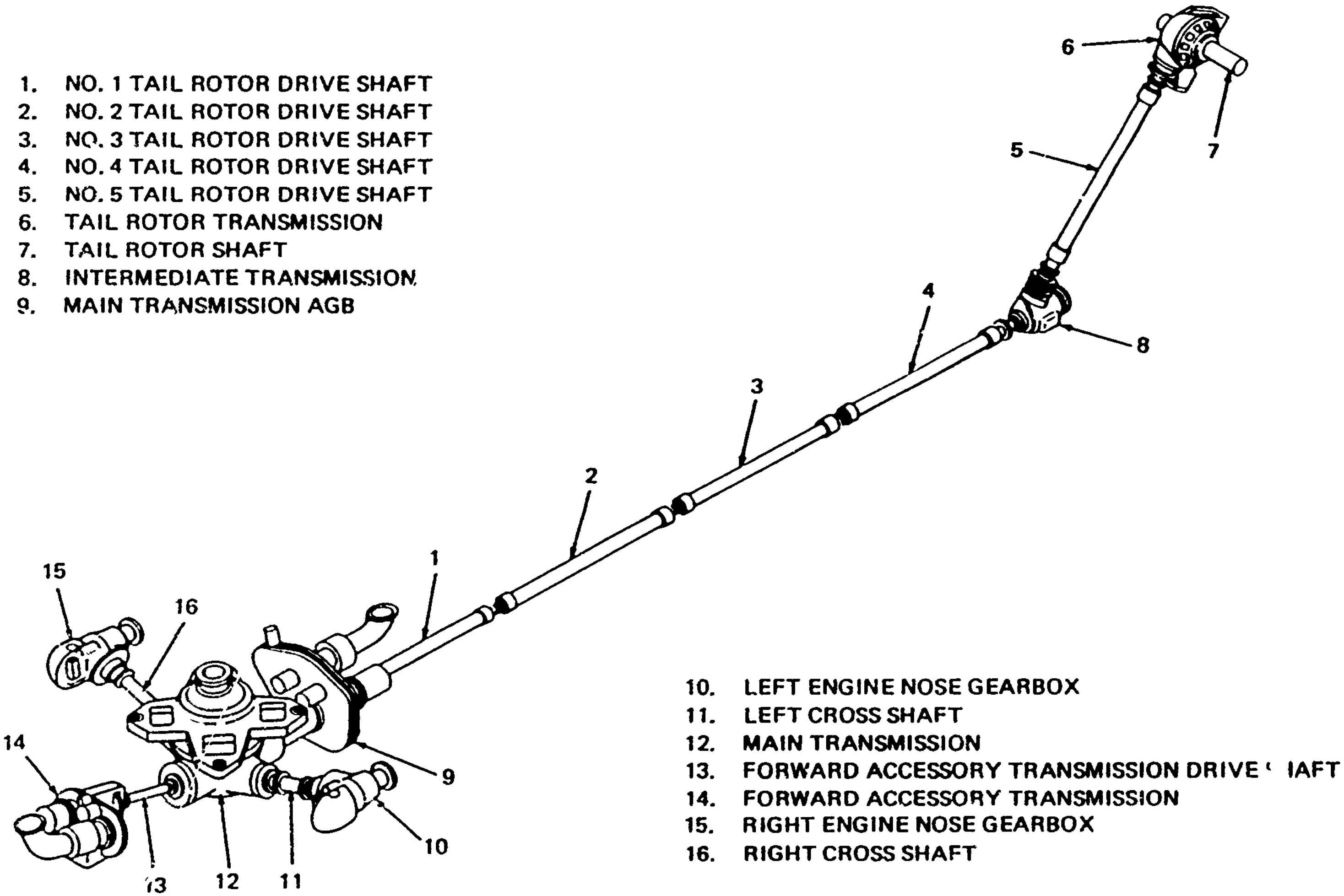


Figure 4. Drive system.

The primary electrical power supply is provided by ac generators, one on each accessory gearbox (AGB). One generator is capable of supplying the entire aircraft electrical power if necessary. The generators also provide for dc power by converting through transformer/rectifiers.

A 28-volt battery (located in the nose avionics compartment) is used for engine starting and is interlocked into the electrical system for emergency use.

The flight control hydraulic system consists of two independent systems, with a utility system as an emergency backup. Each system is completely separated from the other and consists of a pump cooler unit, hydraulic component module, accumulator, and associated hydraulic lines.

A utility hydraulic system operates at 3000 psi. It supplies power for kneeling/unkneeling of the main landing gear and serves as an emergency source of hydraulic pressure for the flight control system. The system consists of items such as an accumulator, an ac electric-driven hydraulic pump, a two-stage handpump, plus filter, and relief valves.

The horizontal stabilizer is controlled electrically by means of a two-motor electromechanical actuator. Linkage between the stabilizer and fuselage structure is designed to prevent free floating of the stabilizer during mechanical, electrical, or ballistic damage failure. The control system is common to both the dual mechanical and the fly-by-wire implementations.

2.1.2 Mission Requirements Impact

Mission requirements dictate that the baseline MUT control system be augmented to include altitude, airspeed, and hover-hold functions. Control response sensitivity, effectiveness, and damping are to be optimized to allow terrain following and NOE flight, as well as cruise flight. Equipment sufficient to accomplish these functions include 4-axis autopilot servos, skewed sensor-inertial navigation system assemblies, autopilot controller assemblies, air data sensor assemblies, and multiplex remote terminal unit assemblies. These equipments will be common to both the dual mechanical mechanization and the fly-by-wire mechanization.

2.1.3 Overview of the Dual Mechanical System

Flight control system components unique to the dual mechanical mechanization includes the tail rotor, main rotor pitch, main rotor roll, and main rotor collective series servos. These series servos provide automatic flight control functions in

series with the pilot's control inputs. In addition, the dual mechanical system includes dual boost servos, dual electric power generators, dual control system hydraulic pumps, control system linkages and mixers, and a utility backup hydraulic system. Table 1 lists the type and redundancy characteristics of the mechanical baseline system. A block diagram of the system is shown in Figure 5. The controls layout for the dual mechanical system is shown in Figure 6.

TABLE 1. DUAL MECHANICAL SYSTEM REDUNDANCY CHARACTERISTIC

Function	Type	Redundancy
Cyclic and collective boost actuators	Hydraulic	Dual, fail-operate
Tail rotor boost actuator	Hydraulic	Dual, fail-operate
Cyclic and collective actuators	Electro-mechanical	Dual, fail-safe
Tail rotor series actuator	Electro-mechanical	Dual, fail-safe
Elevator servo	Electro-mechanical	Dual, fail-safe
Autopilot/trim parallel servos	Electro-mechanical	Nonredundant, fail-safe

2.2 FLY-BY-WIRE DEFINITION

The fly-by-wire system consists of equipment that controls the helicopter main rotor, horizontal stabilizer, and tail rotor. The main rotor is controlled by the STAR system that utilizes a multiarm swashplate that is positioned by electrically controlled hydraulic actuators, as shown in Figures 7 and 8. The horizontal stabilizer is controlled by means of an electro-mechanical actuator and is common to both the dual mechanical system and the fly-by-wire system implementation. The tail rotor is controlled by three electric servomotors that move the tail rotor crosshead. These systems and special features will be described in the following sections.

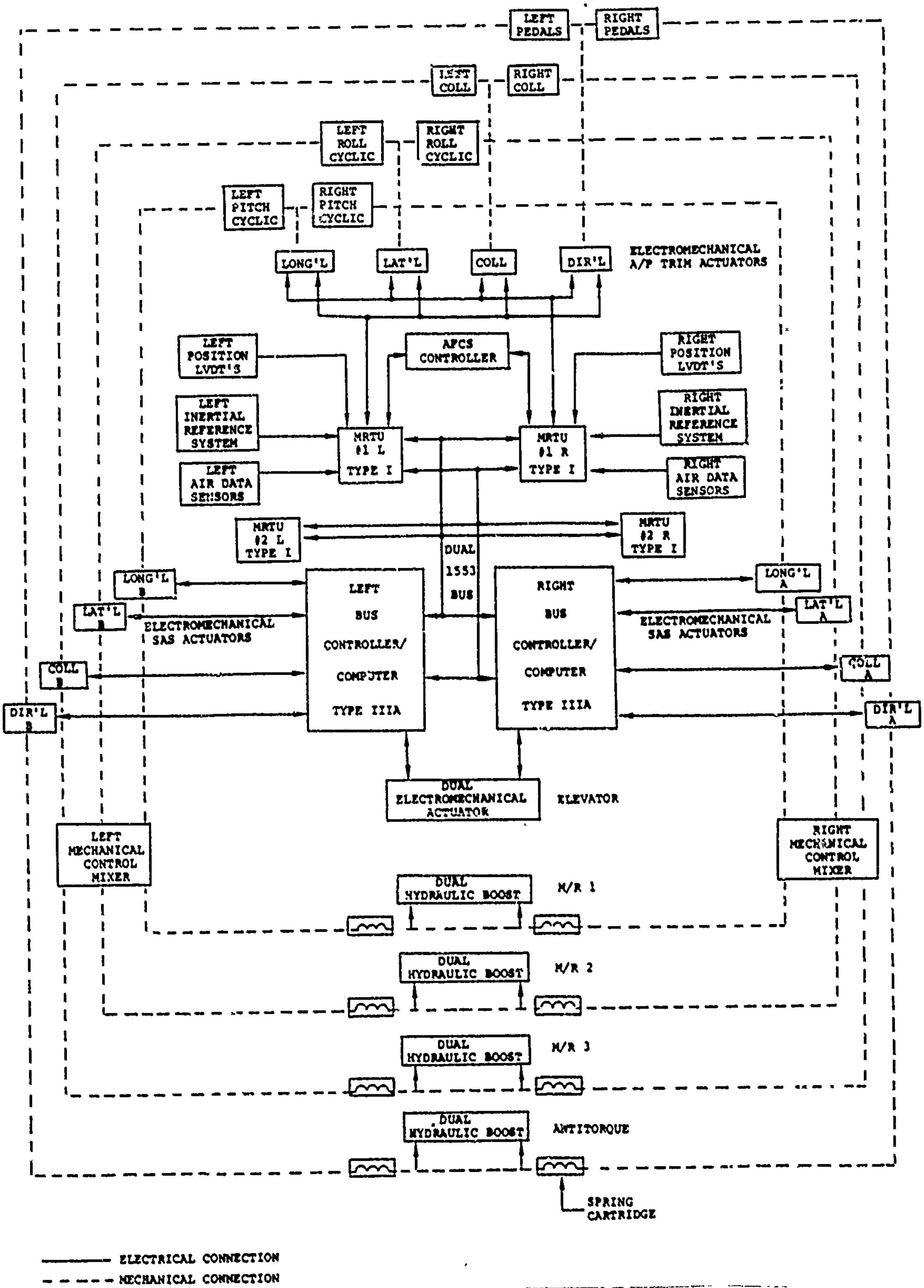


Figure 5. Block diagram of ASH mission dual mechanical control system.

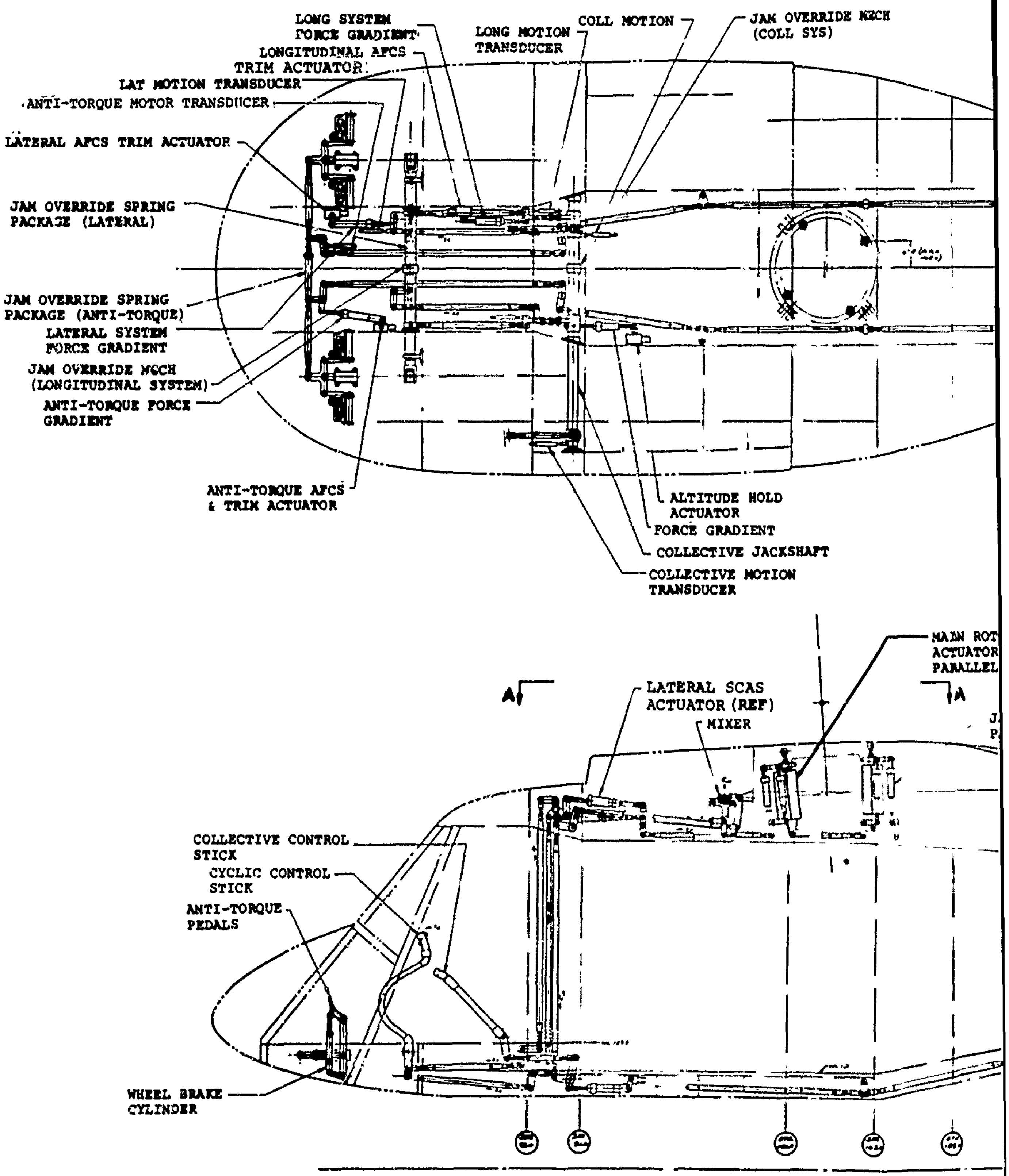
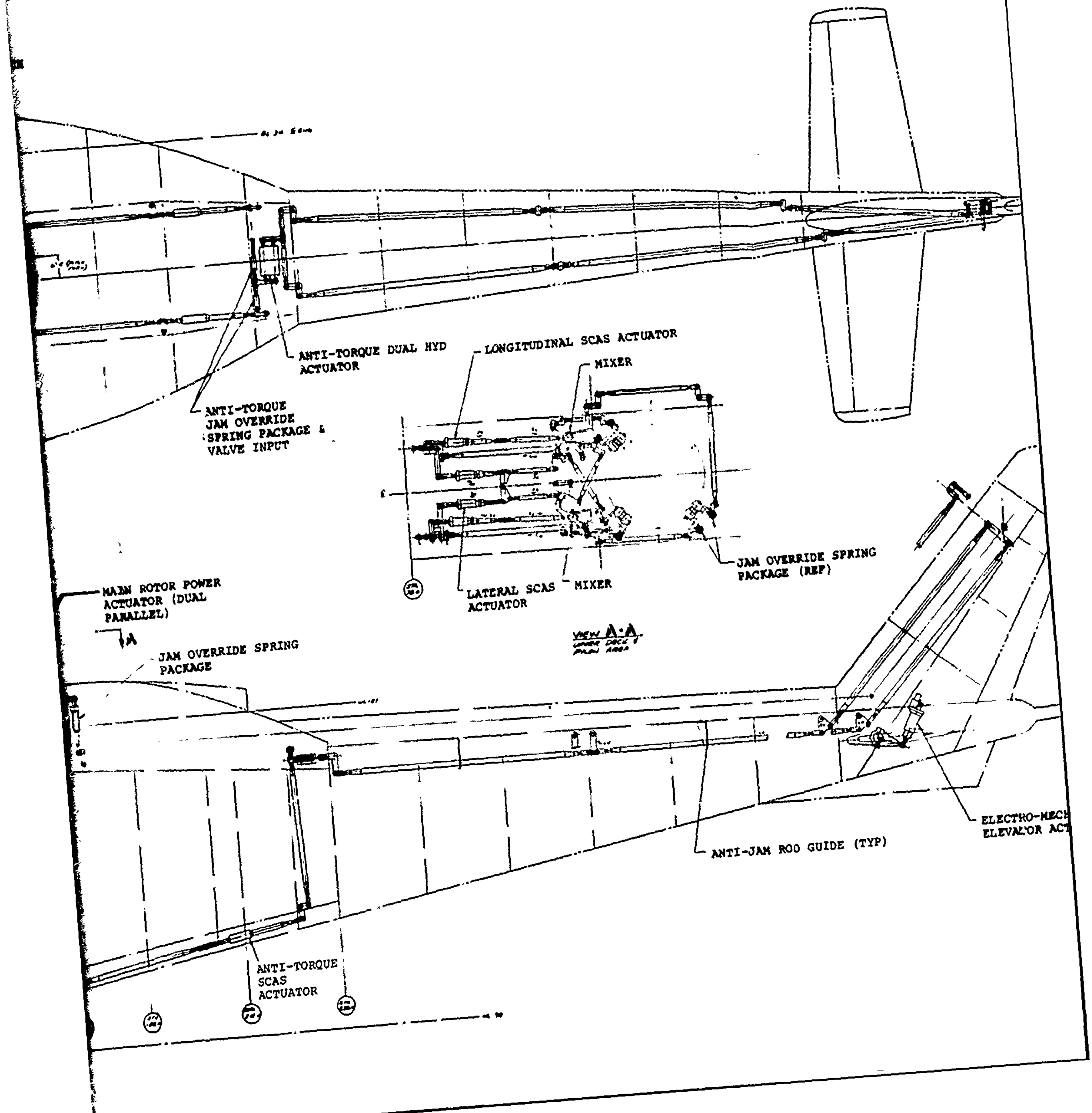


Figure 6. Layout of dual mechanical control system.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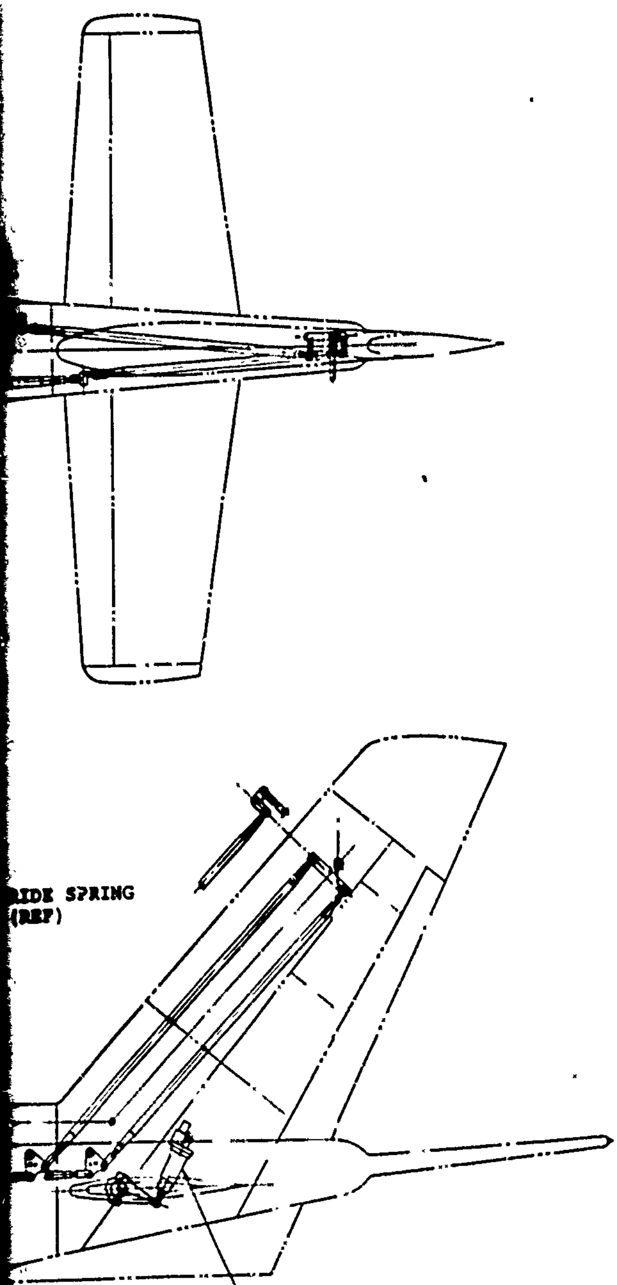
LONGITUDINAL SCAS ACTUATOR
MIXER
ANTI-TORQUE JAM OVERRIDE SPRING PACKAGE & VALVE INPUT
JAM OVERRIDE SPRING PACKAGE (REF)
LATERAL SCAS ACTUATOR
MIXER
MAIN ROTOR POWER ACTUATOR (DUAL PARALLEL)
VIEW A-A
JAM OVERRIDE SPRING PACKAGE
ELECTRO-MECH
ELEVATOR ACT
ANTI-JAM ROD GUIDE (TYP)
ANTI-TORQUE SCAS ACTUATOR



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ELECTRO-MECHANICAL
ELEVATOR ACTUATOR
IDE (TYP)

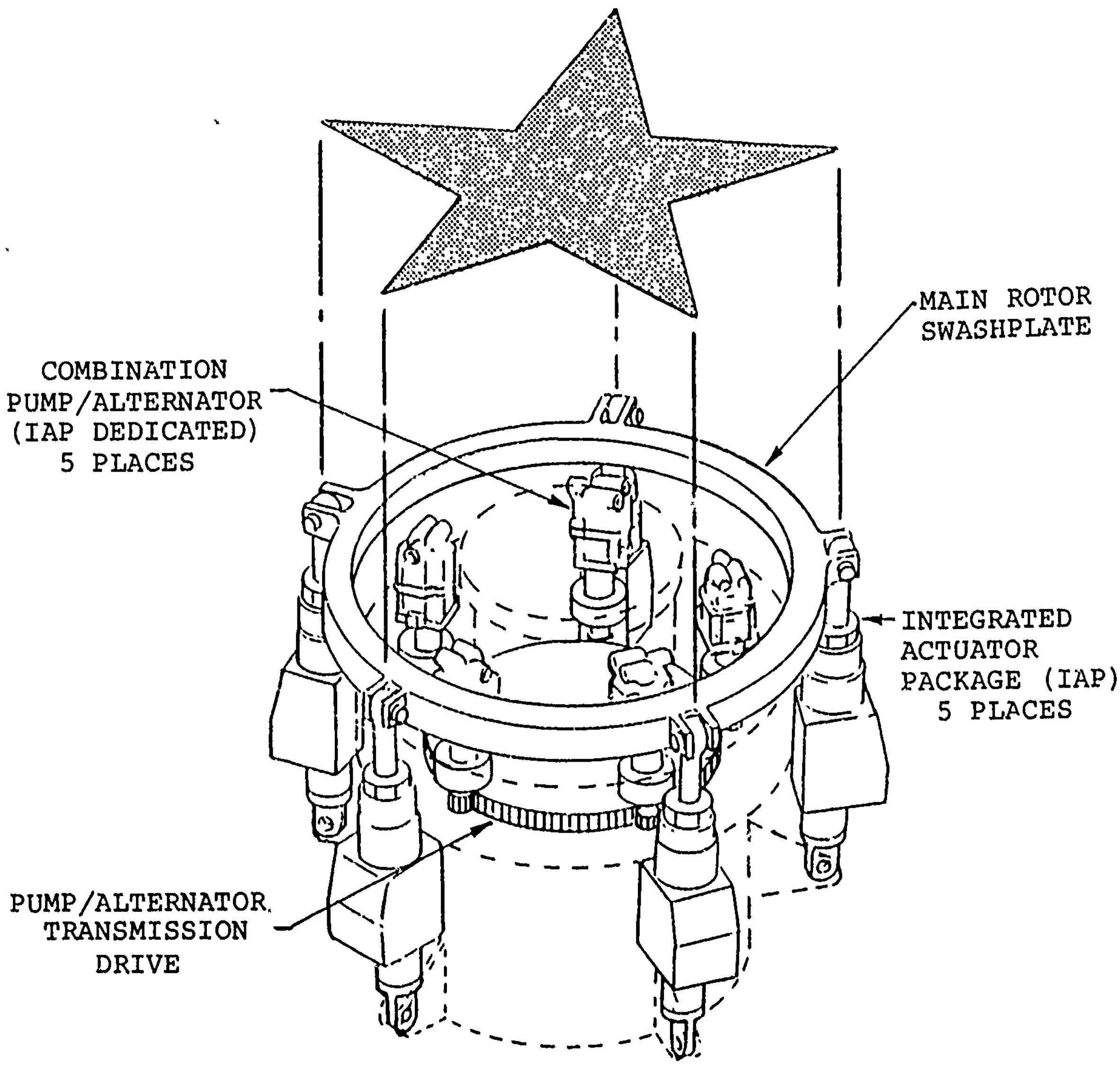

Figure 7. STAR mechanization.

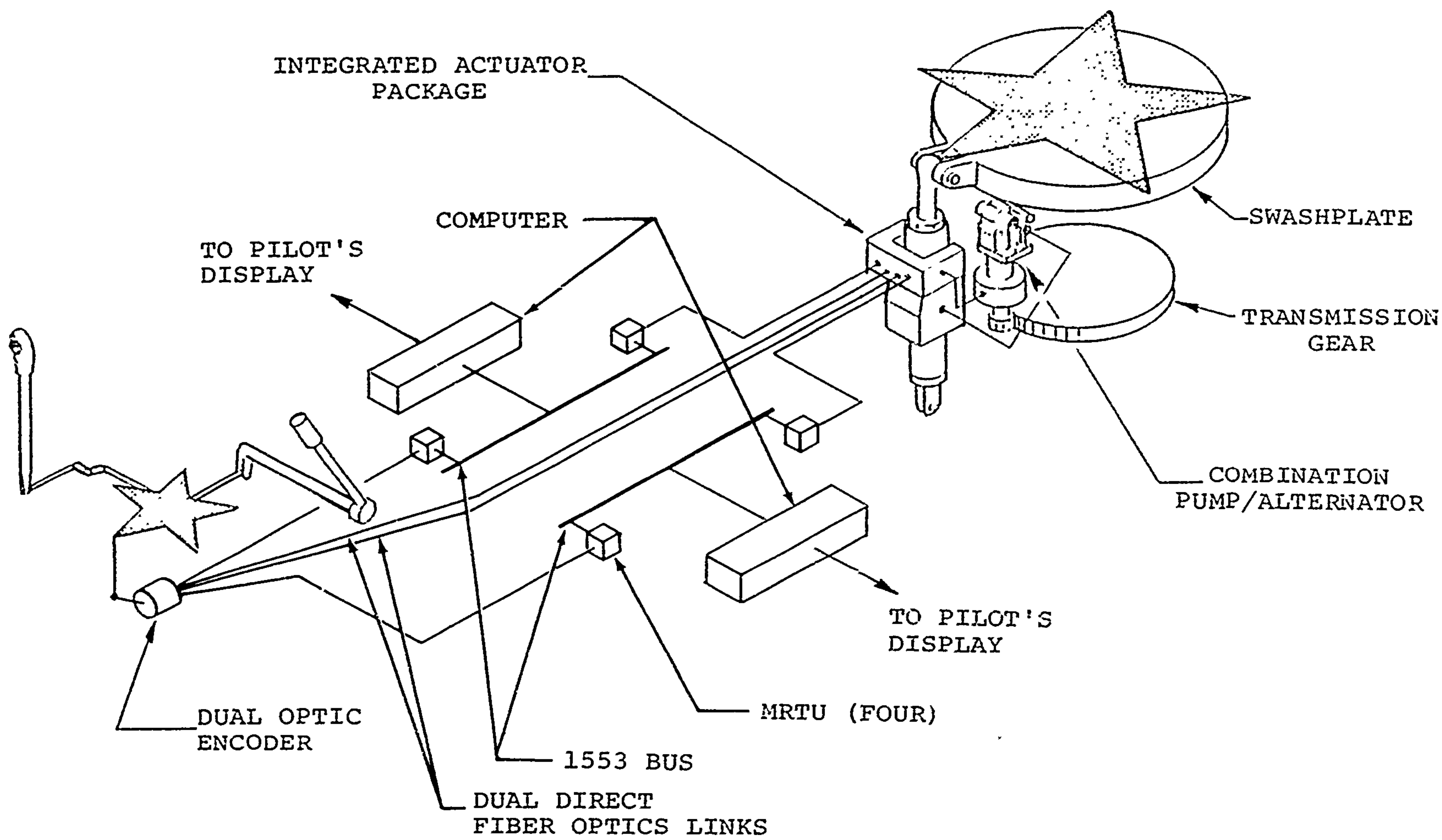

Figure 8. STAR control (typical channel).

2.2.1 Main Rotor Control Configuration

Main rotor loads are generated by the main rotor system as described below:

Four-blade rotor with rotor diameter of 38.9 feet

Solidity	0.1
Blade chord	18.33 inches
Rotor tip speed	750 fps
Helicopter velocity	150 kt
Rotor speed	368 rpm

Control of the rotor system must be insured during both level and maneuvering flight. In the predesign effort, four-, five-, and six-actuator configurations were considered for control of the main rotor swashplate.

Consideration was given to space, weight, and reliability impact of the three configurations. Each configuration was considered as utilizing single-piston, integrated actuator packages (IAPs) with single or multiple electrohydraulic servovalves (EHSVs) for control input.

The four-actuator approach lacked attractiveness because of the fundamental one-fail-operate nature of the geometrical configuration. To insure control after a second system failure, each actuator must be designed (as a minimum) to be operational after a first failure and bypassed after a second failure.

This necessitates at least two EHSVs on each IAP, triplex signal paths, with one path driving an electronic model and fail-operate hydraulic power for each of the four control channels to the main rotor swashplate arms. Such a configuration became overly complex during the predesign effort and was discarded.

The six-actuator approach presented initial attractiveness because of its apparent geometrical superiority after first and second failure situations. Each channel could be designed to operate as fail/bypass, which would switch the actuator into bypass after any channel failure.

Sizing the actuators so that four actuators would react swashplate loads after two failures resulted in smaller actuators than required using the five-actuator approach.

However, as the predesign effort progressed, it became apparent that the added hardware to implement the additional independent hydraulic and electrical supplies and the pilot

control motion sensors resulted in added weight and significant space problems around the helicopter transmission. In fact, the requirements for dual engine, tail rotor, and accessory gearbox drives prevented the six-actuator implementation from being a practical alternative.

The main rotor control system selected for quantitative comparison with the dual mechanical system is the five-actuator STAR mechanization. The transmission configuration layout is shown in Figure 9. To accommodate the installation, the MUT transmission input driveshafts are canted aft 7.2 degrees, and the forward accessory gearbox is angled 25 degrees to the right. Each of the five control channels has electrical and hydraulic power supplies, pilot control motion sensors, and actuator control and monitoring electronics. Each channel is self-sufficient and isolated from the other four channels, except for failure condition information that is shared via fiber optic signal links.

Redundant fiber optic cables are used to transmit control input information from the cockpit to the 5 IAPs, in addition to monitoring and displaying the status of one IAP relative to another. Optical encoders located at a control mixer in the cockpit transmit light signals from both pilot and autopilot control inputs to the IAPs.

Redundant computers are powered from the two main electrical buses in the aircraft and provide secondary control inputs such as SCAS and hover augmentation. A loss of one computer is not mission critical, nor is the loss of both computers critical to safety of flight.

Two 300 cfm blowers are mounted, one each, on the forward and aft accessory gearboxes for the purpose of cooling the IAPs during the hover mission. One operating blower is adequate, two are provided for redundancy, and they use the gearbox drives vacated by the dual-mechanical hydraulic pumps.

Conventional control sticks are depicted for this baseline fly-by-wire definition; however, new, smaller, and possibly combined function controllers appear very attractive for the FBW control system. These will be discussed in Section 5.

Ground check of the flight controls is accomplished with the rotor stopped. A 400-cycle ac power cart is used to operate the onboard motorpump to provide the necessary hydraulic power for ground checkout, without the requirement to power the rotor system.

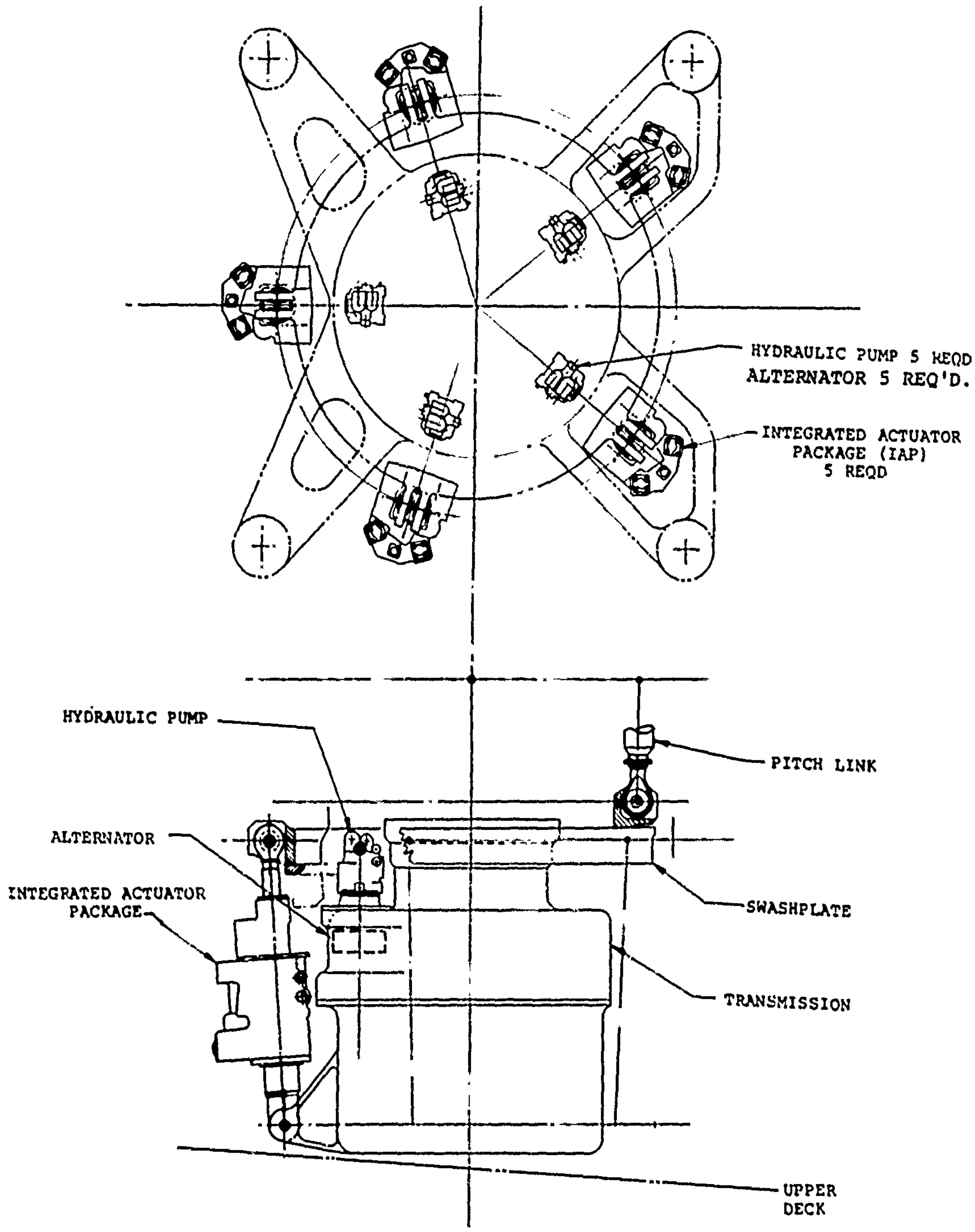

Figure 9. Five actuator STAR transmission layout.

2.2.2 Tail Rotor Control Configuration

The tail rotor control configuration is an electromechanical configuration using three electrically powered and controlled motors. These motors drive through a simple mechanical torque breakout device into a single gear that is attached to a quill that controls the sliding sleeve on the tail rotor shaft. The system is similar to the installation flight tested under contract DAAJ01-77-C-0070 (References 2 and 3).

The torque breakout devices are designed such that the torque limit is 150 ± 10 percent of the stall torque of the motor and is intended to eliminate any requirement for acceleration control of the motors. Three output gears of the three motors drive a common gear of the control quill. An Acme screw portion of the quill, which converts rotary motion to linear motion, projects into the tail rotor gearbox. Attached to the Acme screw is a control tube that passes through the tail rotor mast to connect, via bearings, to the rotating controls of the tail rotor. A layout depicting the configuration of the tail rotor transmission and control quill is shown in Figure 10.

Three passive sensors are also attached to pedals in the cockpit to provide control input to three separate electronic channels that control each of the three electric motors. In addition, commands from redundant computers provide command augmentation, which is additive to the pilot's inputs. Each channel is a simple position servosystem consisting of an error amplifier and comparison means between actual and command positions. The amplifier is a modular type having adequate amplification and power to drive the servomotors directly from the error signal. The total error signal consists of the vector sum of pedal position, actuator position, command augmentation, and motor velocity signals.

The fault-monitoring circuit is designed to compare the absolute value of each tach generator signal with the average of the three. The resultant dc signals represent tach motor

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[2] Stephens, W., and Hampton, B., FLY-BY-WIRE TAIL ROTOR CONTROLS, presented at the American Helicopter Society Specialists Meeting on Helicopter Flight Controls, Arlington, Texas, October 1978.

[3] Blount, P., RESULTS OF A GROUND AND FLIGHT TEST OF A MODEL AH-1G EQUIPPED WITH A 209-961-468-1 FLY-BY-WIRE DIRECTIONAL CONTROL INSTALLATION, Bell Helicopter Textron Report No. 299-099-930, June 1979.

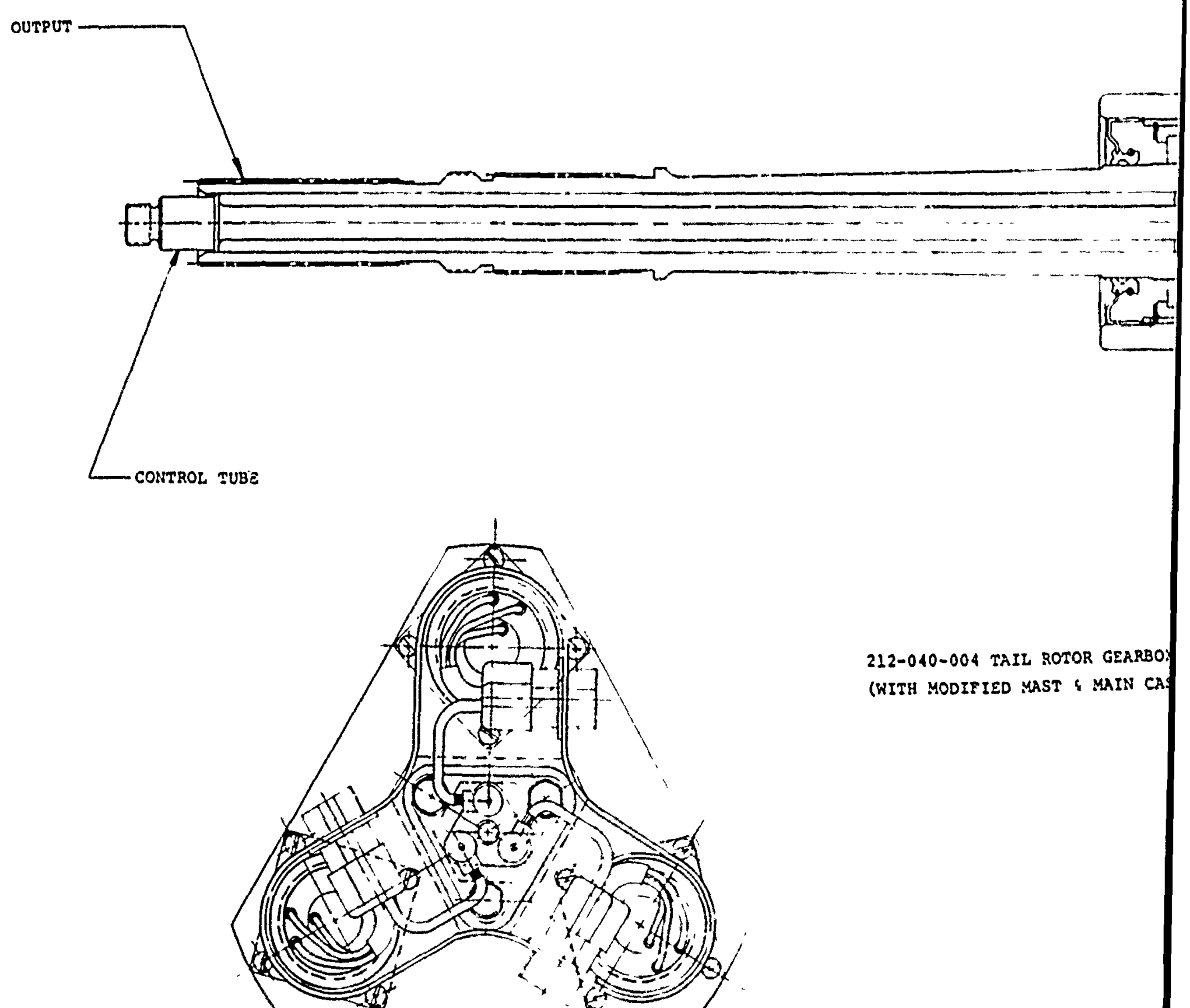
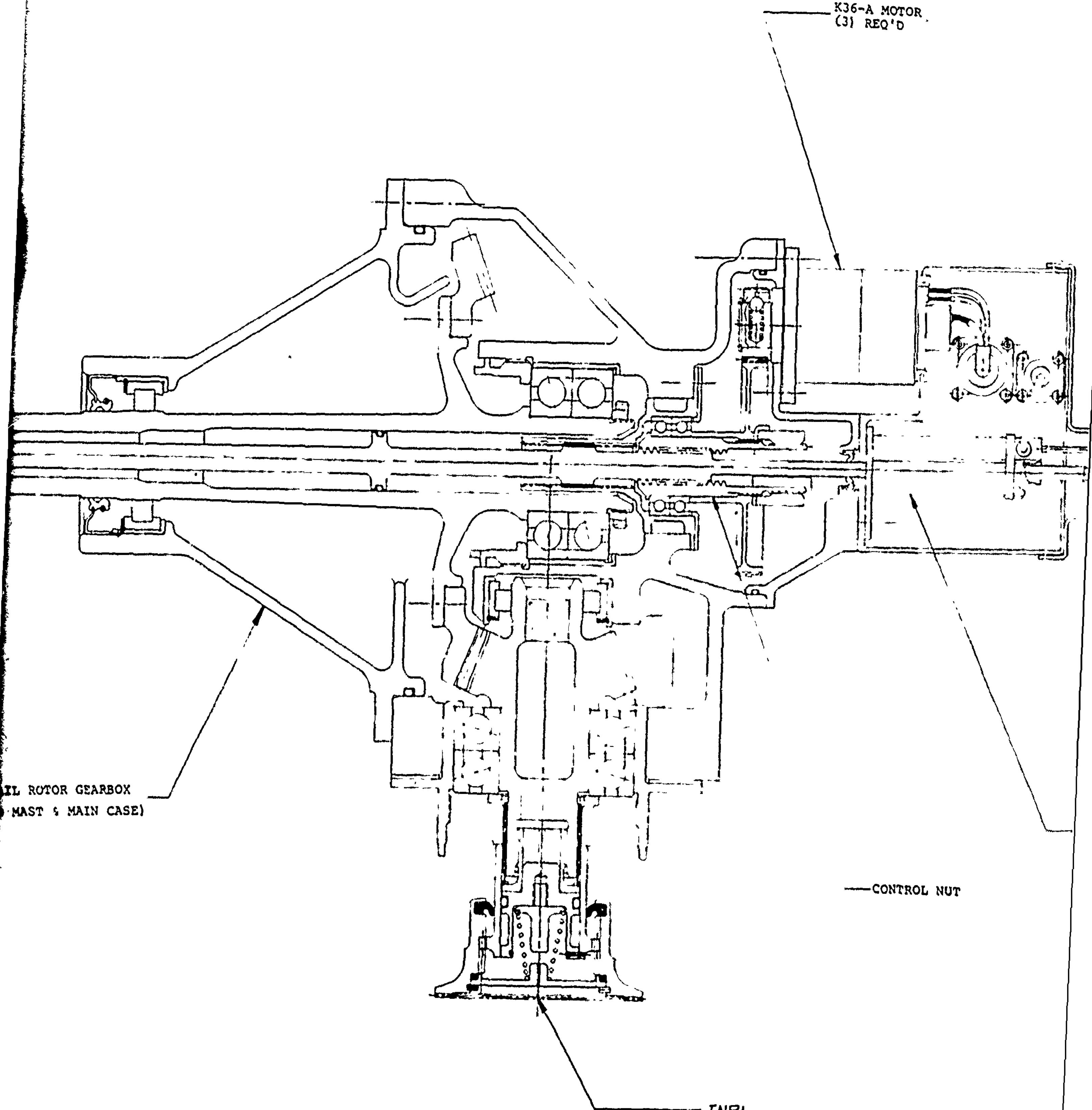


Figure 10. Fly-by-wire tail rotor transmission and control quill.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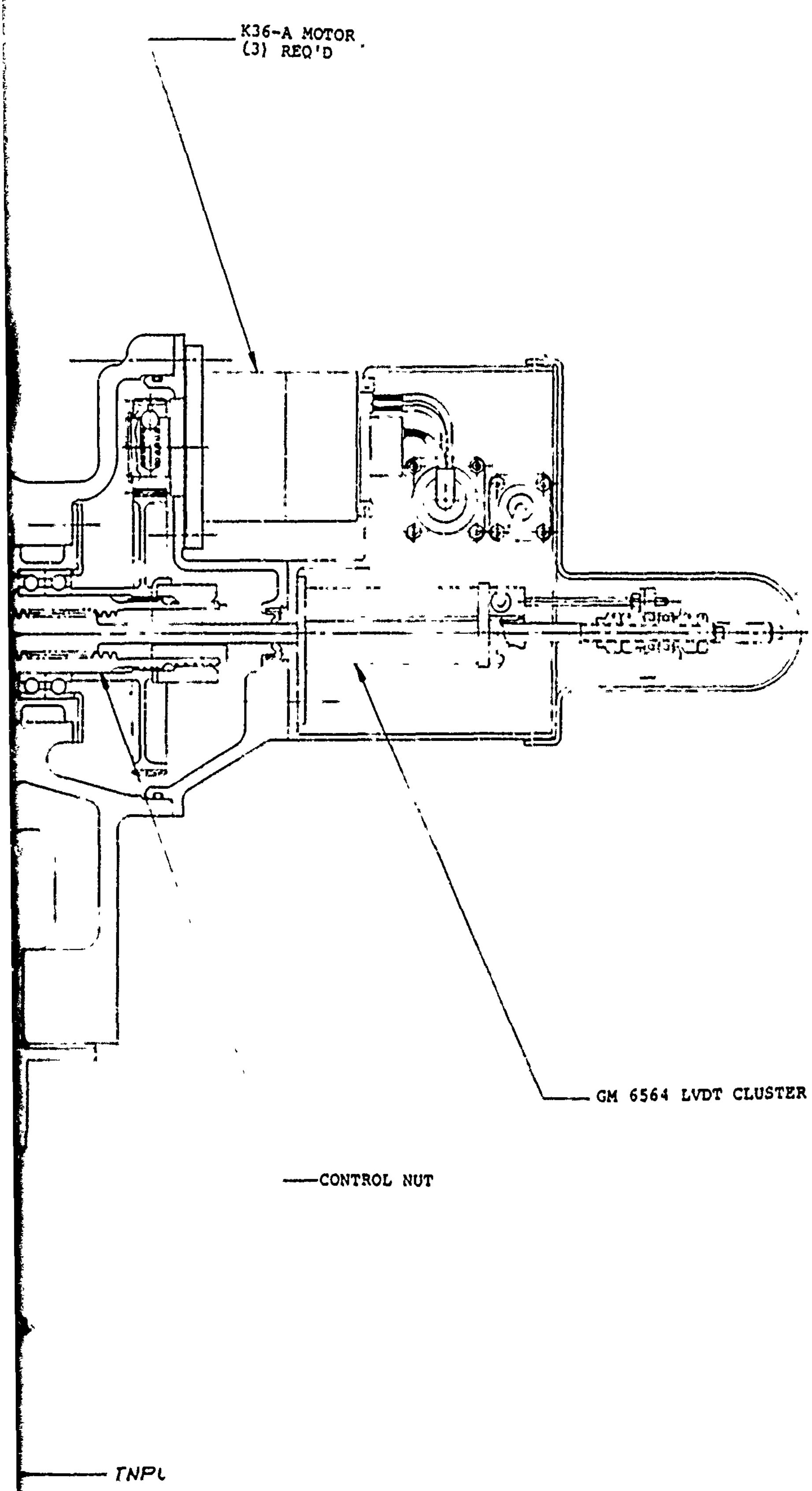

K36-A MOTOR
(3) REQ'D
GM 6564 LVDT CLUSTER
CONTROL NUT
TNPL

velocity, regardless of direction of rotation of the individual motors. Filtering provides immunity to noise by slowing the response to an acceptable level. The resultant signals are summed and divided by a resistor network to form an average signal equal to each other. Signal comparators, contained in one integrated circuit, compare each signal with the average. The three comparator outputs are then decoded by a combination of one integrated circuit and transistor logic to provide failure information. An "exclusive or" logic of an integrated circuit is combined with an "and" logic of the transistor circuits to provide a warning when any tach generator signal disagrees in absolute value with the other two. The circuit sensitivity is set to provide a warning when approximately 30 percent speed difference exists.

2.2.3 Command Augmentation System Interface

In order to perform mission-oriented flight functions, it is necessary that the pilot's primary control be augmented by a control augmentation system. This system is functionally integrated with a dual multiplex bus system that transmits augmentation signals to the main rotor, tail rotor, and horizontal stabilizer, as depicted in Figure 11.

The dual configuration of the IAP electronics allows for straightforward interface to the dual SCAS system. Fiber optic links are used in the IAP/SCAS interface to manage EMI problems. All data are transmitted in digital pulse form with the necessary encoding and decoding provided in each control sensor and IAP electronic module unit.

Stick position data required by the SCAS is obtained indirectly through the IAP, rather than directly from the stick sensors. This design choice is driven by power and light loss considerations in the passive stick sensors and by the reduction in fiber optic connectors and receivers that is obtained when compared to the direct connection configuration. In addition to the stick position data, IAP status data are also transmitted to the SCAS computers. The SCAS computers perform the IAP interconnect logic function.

SCAS commands, based on aircraft motions and stick positions are generated in each computer and returned to the IAP through the fiber optic link. The SCAS computer status and the actuator control outputs from the interconnect logic computations are also transmitted to the IAP.

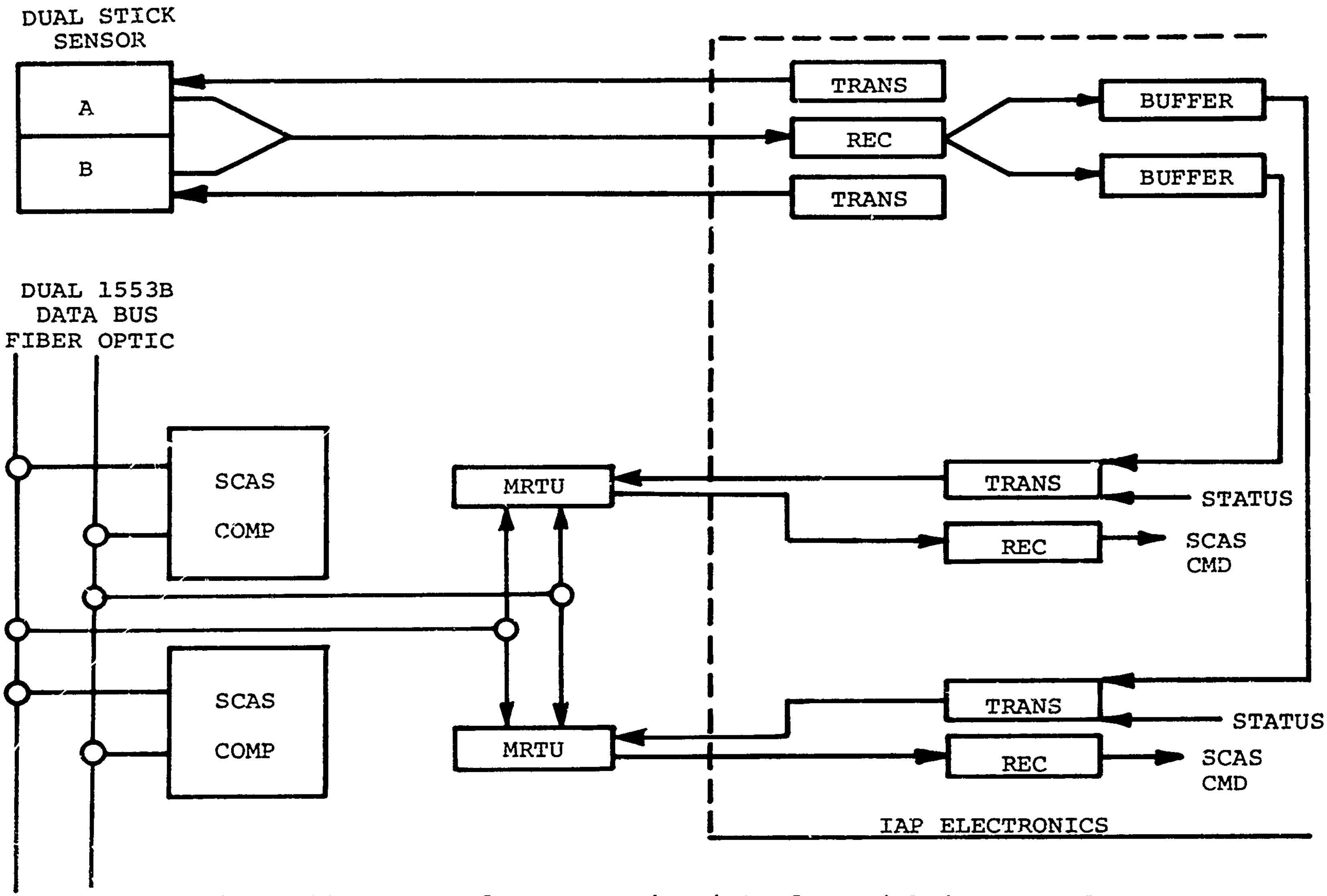


Figure 11. Command augmentation interface with integrated actuator package.

2.2.4 Overview of the Fly-By-Wire System

Flight control system components unique to the fly-by-wire/light mechanization include the triplex electromechanical tail rotor control and the STAR main rotor control that features passive sensors, optical cockpit sensors, hydraulic-mounted electronics, and dedicated hydraulic and electrical power supplies. Table 2 lists the type and redundancy characteristics of the fly-by-wire/light mechanization.

TABLE 2. FLY-BY-WIRE SYSTEM REDUNDANCY CHARACTERISTICS

Function	Type	Redundancy
Cyclic & collective	Electrohydraulic	Five channel (Fail-Operate)^2
Tail rotor	Electromechanical	Triplex Fail-Operate
Elevator servo	Electromechanical	Dual Fail-Safe
Autopilot/trim Parallel servos	Electromechanical	Nonredundant Fail-Safe

Table 3 lists the size, weight, and reliability characteristics of various electronic subassemblies. A block diagram of the system is shown in Figure 12. The controls layout is depicted in Figure 13.

2.2.5 Hydraulic System Features

The main rotor control system fly-by-wire/light implementation utilizes five integrated actuator packages, five dedicated hydraulic pumps, and one auxiliary hydraulic pump that provides fill-and-bleed capability and ground check capability without the requirement for the engine to be running.

2.2.5.1 Actuator Load Requirements. The design requirements for the main rotor power actuators depend upon the size of the vehicle, the type of rotor system employed, and the particular criteria selected for post-system-failure operation. For purposes of this preliminary design, the criteria listed in Table 4 are considered.

TABLE 3. DESCRIPTION OF FLY-BY-WIRE SUBASSEMBLIES

Sub-assembly	Packaging description	WEIGHT (lb)	SIZE (in.)	POWER (Watts)	RELIABILITY (MTBF) (hr)	Qty
Electronic module	Horseshoe shaped box with optic and electrical connectors	2.0	4 X 5 X 2 3/8	8.5	30,300	5
Trim Actuator (Plessey)	Drive motor and mag brake package with rotary arm and electrical connector	3.3	6.6 X 3.6 X 4.75	AC 13.6 (max) DC 15 (max)	56,600	4
AFCS Controller (Holcis)	Mode selector panel with electrical connector	2.3	3 1/4 X 3 1/4 X 6	9 (max)	14,350	1
MRTU Type I	Computer package	10	5 X 7 X 7.42	25	3,590	2
MRTU Type IIIA	Computer package	15	5 X 7 X 10.25	40	2,598	2
Bus Coupler	Transformer cube with one electrical connector	.12	1.57 X 1.2 X 1.8	Negligible	2,000,000	12
Auto Elevator Control Panel	Mode select and annunciator panel with electrical connector	2.0	5 3/4 X 1 1/2 X 2	3 (max)	62,500	1
FBW Tail Rotor Control Panel	Mode select and annunciator panel with electrical connector	2.0	5 3/4 X 3 X 2	5 (max)	62,500	1
Passive Optical Position Sensor (Dual)	Cube with optic connector and protruding rotary shaft	2.5	4 X 1 1/2 X 2 3/8	None	31,250	5
Passive Optical Position Sensor (Single)	Cube with optic connector and protruding rotary shaft	1.5	3 X 1 1/2 X 2 3/8	None	100,000	3

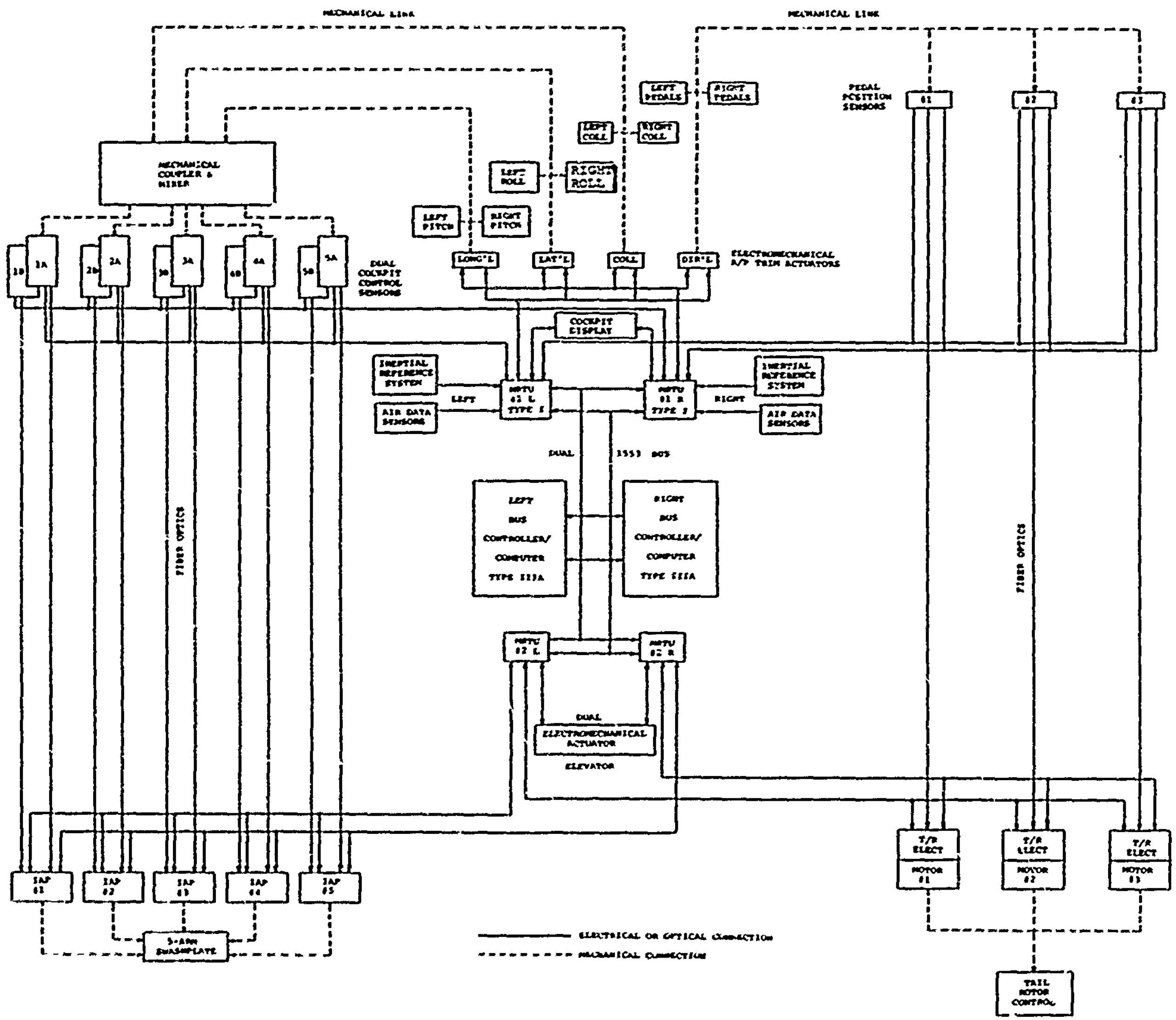


Figure 12. Block diagram of ASH mission FBW/L control system.

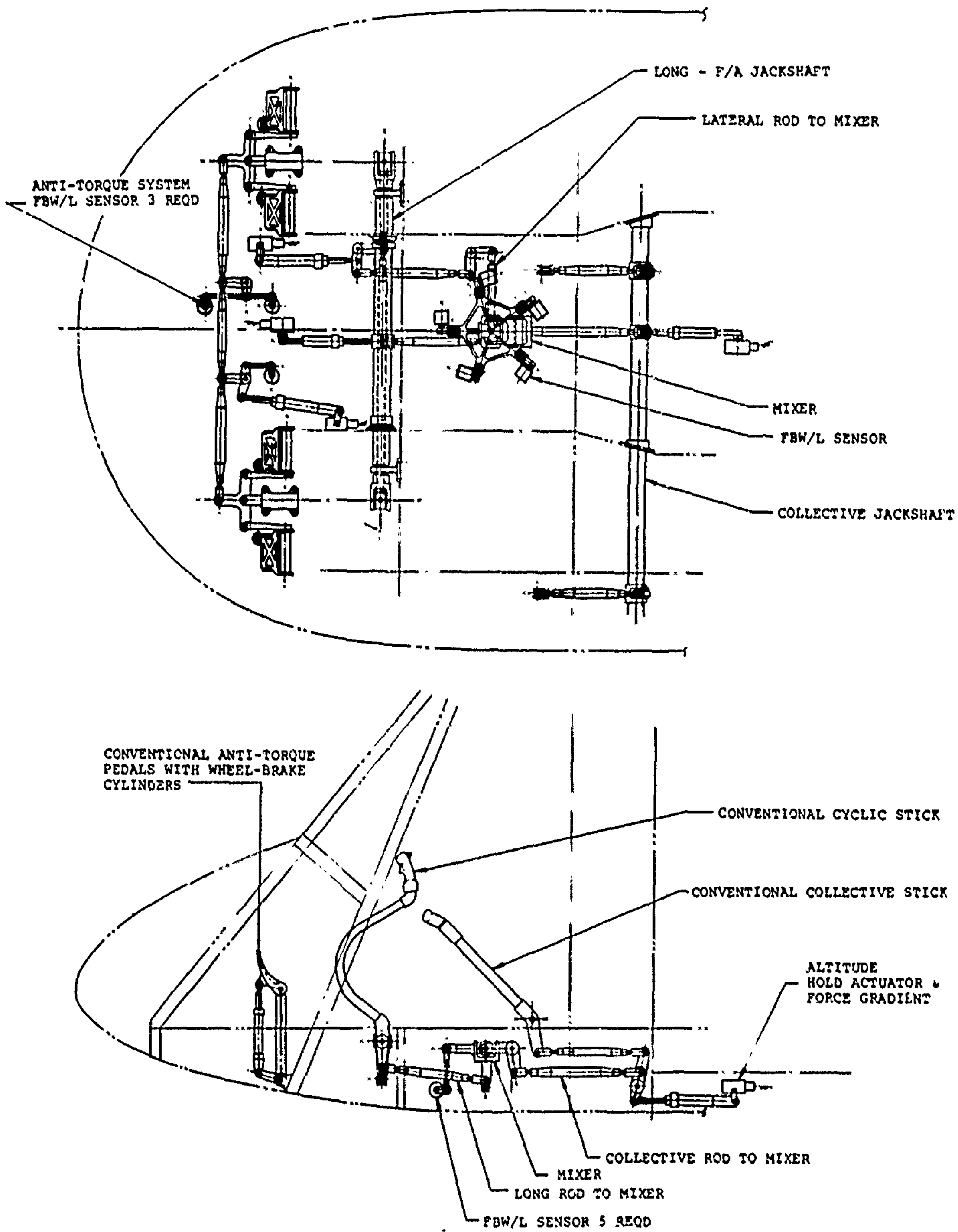


Figure 13. Mechanical layout for FBW/L control system.

TABLE 4. OPERATIONAL CRITERIA FOR STAR MAIN ROTOR CONTROL

Case	Condition	Criteria
I	All IAPs Normal	Maneuver Loads, V_H
II	One IAP bypassed	Maneuver Loads, V_H
III	Two IAPs bypassed	Cruise, V_H
IV	One IAP failed and force fighting (multiple failure of electronic monitors)	Maneuver Loads, 0.6 V_H
V	Two IAPs bypassed	Maneuver Loads, 0.6 V_H

Exact properties of the four-bladed rotor system for the MUT vehicle were not available. Thus, the specific harmonic allocation of the swashplate loads were not known. Since the magnitude of oscillatory loads are much larger for a two-bladed rotor system, it appeared conservative to reference BHT two-bladed data when establishing the maximum load requirements for the MUT IAPs. Under the condition that two side-by-side IAPs are disabled and bypassed, the remaining IAPs are required to react steady loads of 2620 pounds and oscillatory loads of ±1610 pounds. The IAP designed to this load requirement is depicted in Figure 14.

The force of 4230 pounds was established for the MUT IAPs using similar analyses as were used to design the AH-1 Cobra IAPs. In the following paragraphs, a discussion of results from previous Model AH-1 predesign activities is presented for use as substantiating design data.

IAP reactions were calculated from fixed system swashplate shears and moments. Final histories of these shears and moments were generated from measured pitch link loads combined (after resolution into the fixed system) with actuator forces resulting from the conditions of Table 4. Tables 5 through 9 present a tabulation of the calculated actuator reactions shown as R1-R5 as a function of reference blade azimuth. The IAP design load was determined to be 5100 pounds.

2.2.5.2 Integrated Actuator Package. To maintain isolation between the various hydraulic systems, the concept of multiple integrated actuator packages and dedicated pumps is used. This concept provides true control redundancy from the pilot's control input to the output of the power actuator.

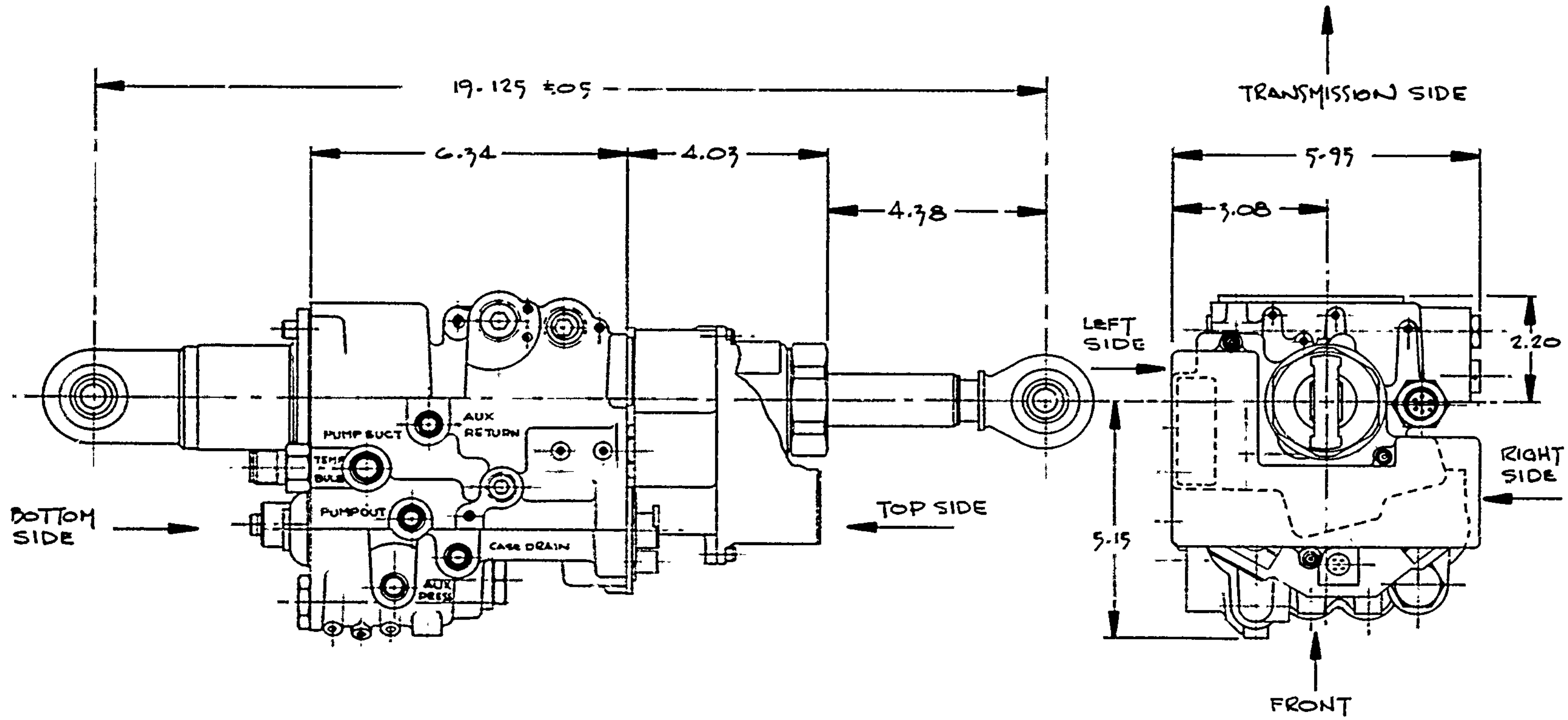


Figure 14. Envelope drawing of FBW/L integrated actuator package.

TABLE 5. MANEUVER LOADS @ V_H
FIVE ACTUATORS OPERATING

AZIMUTH	IAP				
OMEGAT	R1	R2	R3	R4	R5
0.00	1517	1725	1646	1388	1308
5.62	1472	1966	1843	1273	1044
11.25	1336	2207	2094	1154	686
16.87	1104	2420	2392	1060	263
22.50	777	2576	2721	1011	-190
28.12	361	2648	3053	1017	-647
33.75	-129	2616	3361	1077	-1080
39.37	-666	2469	3612	1186	-1458
45.00	-1206	2209	3779	1334	-1747
50.62	-1695	1856	3835	1507	-1911
56.25	-2072	1444	3764	1682	-1925
61.87	-2287	1017	3561	1830	-1784
67.50	-2316	621	3240	1922	-1512
73.12	-2168	296	2831	1934	-1156
78.75	-1885	64	2379	1861	-774
84.37	-1528	-74	1932	1717	-421
90.00	-1156	-129	1532	1532	-129
95.62	-816	-126	1207	1340	91
101.25	-527	-84	965	1171	249
106.87	-287	-19	799	1036	365
112.50	-82	65	694	935	456
118.12	104	169	636	860	531
123.75	281	294	618	805	596
129.37	447	435	636	774	657
135.00	601	581	693	781	724
140.62	740	722	783	839	813
146.25	870	848	899	952	935
151.87	999	959	1025	1105	1089
157.80	1132	1065	1149	1267	1257
163.12	1268	1182	1263	1399	1401
168.75	1392	1327	1374	1468	1479
174.37	1483	1508	1495	1461	1454
180.00	1517	1725	1646	1388	1308

TABLE 6. MANEUVER LOADS @ V_H

FOUR ACTUATORS OPERATING/ONE BYPASSED

AZIMUTH	IAP			
OMEGAT	R1	R2	R3	R4
0.00	2430	1089	1298	2769
5.62	2074	1080	1573	2874
11.25	1618	981	1851	3029
16.87	1119	778	2093	3252
22.50	625	466	2263	3542
28.12	172	49	2333	3881
33.75	-213	-460	2282	4237
39.37	-505	-1029	2101	4577
45.00	-675	-1616	1796	4864
50.62	-699	-2158	1390	5060
56.25	-571	-2589	923	5130
61.87	-309	-2850	450	5047
67.50	37	-2907	27	4800
73.12	404	-2764	-302	4400
78.75	727	-2459	-512	3888
84.37	965	-2057	-604	3323
90.00	1107	-1628	-603	2772
95.62	1173	-1229	-540	2292
101.25	1195	-888	-446	1912
106.87	1203	-607	-339	1637
112.50	1212	-370	-223	1451
118.12	1227	-161	-96	1332
123.75	1247	33	46	1269
129.37	1283	209	196	1262
135.00	1356	360	341	1324
140.62	1492	482	464	1462
146.25	1705	577	555	1669
151.87	1983	659	610	1919
157.50	2282	742	675	2173
163.12	2533	837	752	2394
168.75	2667	940	875	2561
174.37	2636	1033	1058	2677
180.00	2430	1089	1298	2769

TABLE 7. CRUISE FLIGHT LOADS @ V_H

THREE ACTUATORS OPERATING/TWO ADJACENT BYPASSED

AZIMUTH	IAP		
OMEGAT	R1	R2	R3
0.00	1259	285	1671
5.62	1065	437	1659
11.25	872	527	1646
16.87	693	546	1622
22.50	522	516	1568
28.12	335	476	1466
33.75	108	465	1313
39.37	-170	497	1133
45.00	-480	555	974
50.62	-775	590	898
56.25	-991	540	958
61.87	-1069	356	1181
67.50	-973	18	1558
73.12	-701	-457	2047
78.75	-285	-1023	2584
84.37	224	-1618	3100
90.00	765	-2175	3533
95.62	1280	-2638	3835
101.25	1724	-2958	3977
106.87	2062	-3105	3946
112.50	2274	-3065	3752
118.12	2356	-2846	3423
123.75	2321	-2482	3008
129.37	2201	-2031	2571
135.00	2040	-1564	2176
140.62	1884	-1144	1875
146.25	1765	-812	1689
151.87	1695	-570	1611
157.50	1656	-395	1607
163.12	1619	-243	1637
168.75	1548	-81	1665
174.37	1427	101	1676
180.00	1259	285	1671

TABLE 8. MANEUVER LOADS @ 0.6V_H

FOUR ACTUATORS OPERATING/ONE FAILED HARDOVER

AZIMUTH	IAP			
OMEGAT	R1	R2	R3	R4
0.00	-2706	2142	2255	-2524
5.62	-2893	2138	2403	-2466
11.25	-3143	2084	2553	-2382
16.87	-3412	1975	2684	-2262
22.50	-3680	1806	2776	-2105
28.12	-3926	1580	2814	-1923
33.75	-4135	1305	2787	-1730
39.37	-4293	997	2690	-1545
45.00	-4384	680	2525	-1389
50.62	-4397	387	2306	-1281
56.25	-4327	154	2054	-1242
61.87	-4185	13	1798	-1287
67.50	-3997	-17	1569	-1422
73.12	-3799	61	1391	-1640
78.75	-3625	227	1278	-1919
84.37	-3497	446	1229	-2225
90.00	-3419	677	1230	-2522
95.62	-3383	893	1264	-2780
101.25	-3370	1076	1314	-2983
106.87	-3365	1226	1371	-3129
112.50	-3359	1352	1432	-3229
118.12	-3352	1464	1499	-3294
123.75	-3341	1568	1575	-3329
129.37	-3323	1664	1657	-3334
135.00	-3285	1748	1737	-3302
140.62	-3212	1815	1805	-3228
146.25	-3098	1868	1856	-3117
151.87	-2949	1912	1892	-2982
157.50	-2789	1957	1922	-2845
163.12	-2653	2007	1962	-2726
168.75	-2581	2062	2028	-2636
174.37	-2596	2112	2126	-2573
180.00	-2706	2142	2255	-2524

TABLE 9. MANEUVER LOADS @ $0.6V_H$

THREE ACTUATORS OPERATING/TWO ADJACENT BYPASSED

AZIMUTH	IAP		
OMEGAT	R1	R2	R3
0.00	2764	-1458	2859
5.62	2449	-1002	2725
11.25	1999	-455	2564
16.87	1454	111	2415
22.50	853	635	2305
28.12	231	1071	2241
33.75	-380	1384	2225
39.37	-942	1543	2251
45.00	-1404	1529	2315
50.62	-1715	1333	2409
56.25	-1833	970	2517
61.87	-1743	486	2613
67.50	-1469	-45	2665
73.12	-1070	-542	2645
78.75	-623	-937	2541
84.37	-199	-1194	2363
90.00	156	-1319	2145
95.62	428	-1345	1924
101.25	630	-1317	1734
106.87	787	-1270	1593
112.50	921	-1219	1499
118.12	1044	-1164	1443
123.75	1165	-1106	1418
129.37	1290	-1055	1432
135.00	1435	-1038	1502
140.62	1619	-1087	1648
146.25	1856	-1220	1874
151.87	2142	-1427	2157
157.50	2446	-1656	2455
163.12	2715	-1834	2709
168.75	2888	-1887	2871
174.37	2912	-1765	2919
180.00	2764	-1458	2859

Each IAP contains the hydraulic and electrical components, as shown in Figure 15. If one of the five IAPs jammed, the swashplate would be restricted in its response and normal travel. The geometry of the STAR control, however, provides a possibility of breaking the jam by using the combined force output of the other IAPs. The success of breaking the jam is dependent upon the material properties and design characteristics of the IAP.

Figure 16 depicts the cross section of an IAP which exposes the piston head, piston rod, and cylinder barrel. The slotted piston head would be fabricated from high tensile strength (260-290 ksi) steel. The piston and barrel would be made frangible so as to break upon impact with high caliber projectiles. The barrel may utilize fibrous material in an epoxy or metal matrix.

When fully developed, these concepts will do much to minimize the threat of ballistic jam (see Section 4.3.4).

2.2.5.3 Hydraulic Supply. The hydraulic power for each IAP is derived from a dedicated pump driven from the main rotor transmission. A cross section of the installation is shown in Figure 17. The pump is a 3000 psi variable displacement type that supplies 2.0 gpm at 4000 rpm.

In addition to the dedicated hydraulic pump, there is an auxiliary pump that is used for ground checkout and fill and bleed. The total hydraulic installation for the pumps/IAPs is shown in Figure 18.

2.2.5.4 Fill-and-Bleed Procedure. Hydraulic systems require servicing provisions for filling, bleeding, and ground checkout. This is normally accomplished by providing a set of self-sealing quick disconnects that allows a hydraulic ground cart to be connected to the system. Since each IAP, along with its dedicated hydraulic pump, is a separate system, it would not be practical to provide five separate sets of quick disconnects.

The baseline MUT vehicle contained an auxiliary hydraulic system powered by a 400-cycle electric-motor driven pump. This system was used for the landing gear kneeling function and also served as a backup system for the flight controls. This system has been retained for the landing gear kneeling function. Provisions incorporated in the IAPs allow use of this auxiliary system for filling, bleeding, and ground checkout.

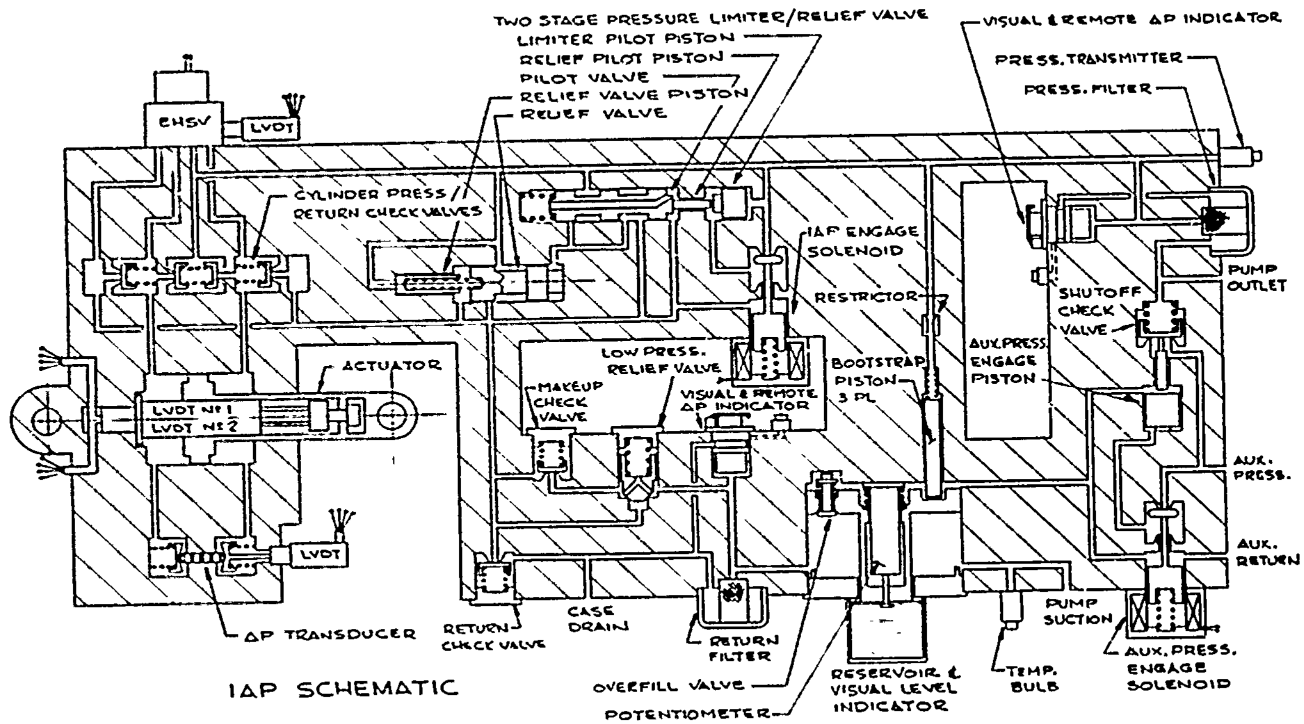
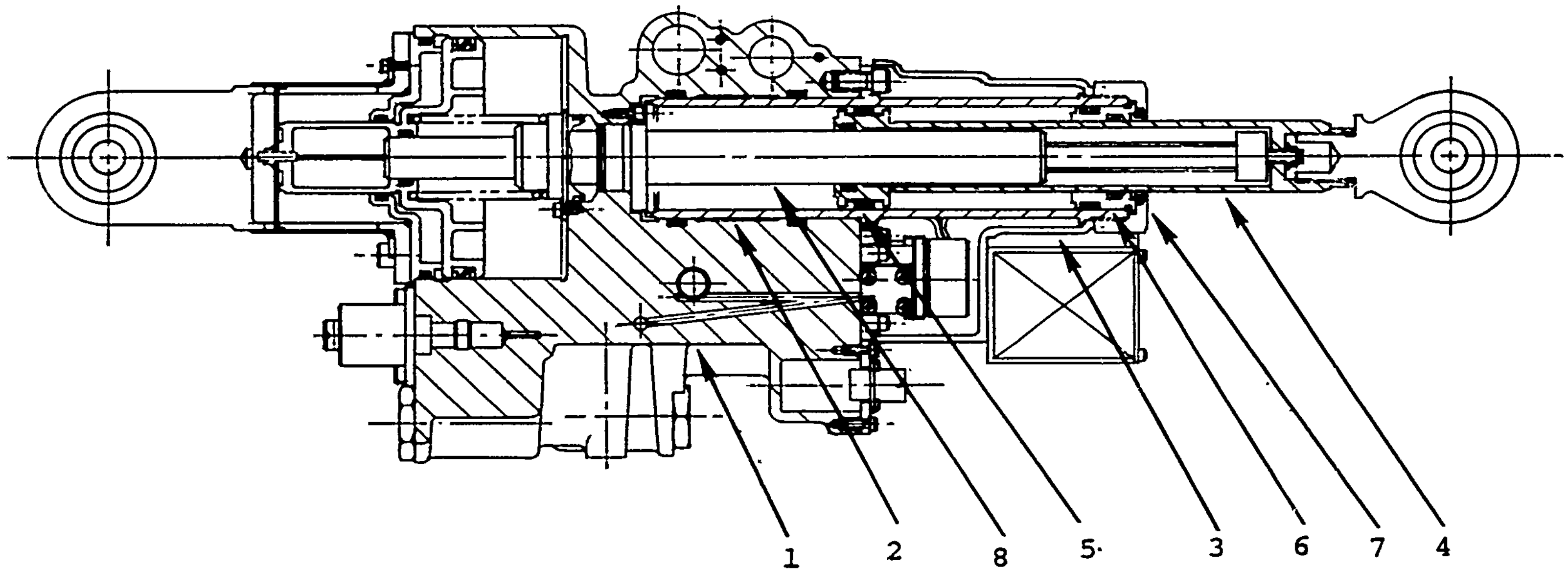


Figure 15. Integrated actuator package schematic.



No.	Description	Material
1	Housing	7075-T73 Al Forging
2	Barrel	H11 Steel 260-290ksi HT
3	Cover	356-T6 Al
4	Piston Rod	H11 Steel 260-290ksi HT
5	Piston Ring	Al Bronze
6	Piston Rod Gland	Al Bronze
7	Gland Retainer	2024-T6 Al
8	Dual LVDT	15-5 PH Cres 190ksi HT

Figure 16. IAP antijam provisions.

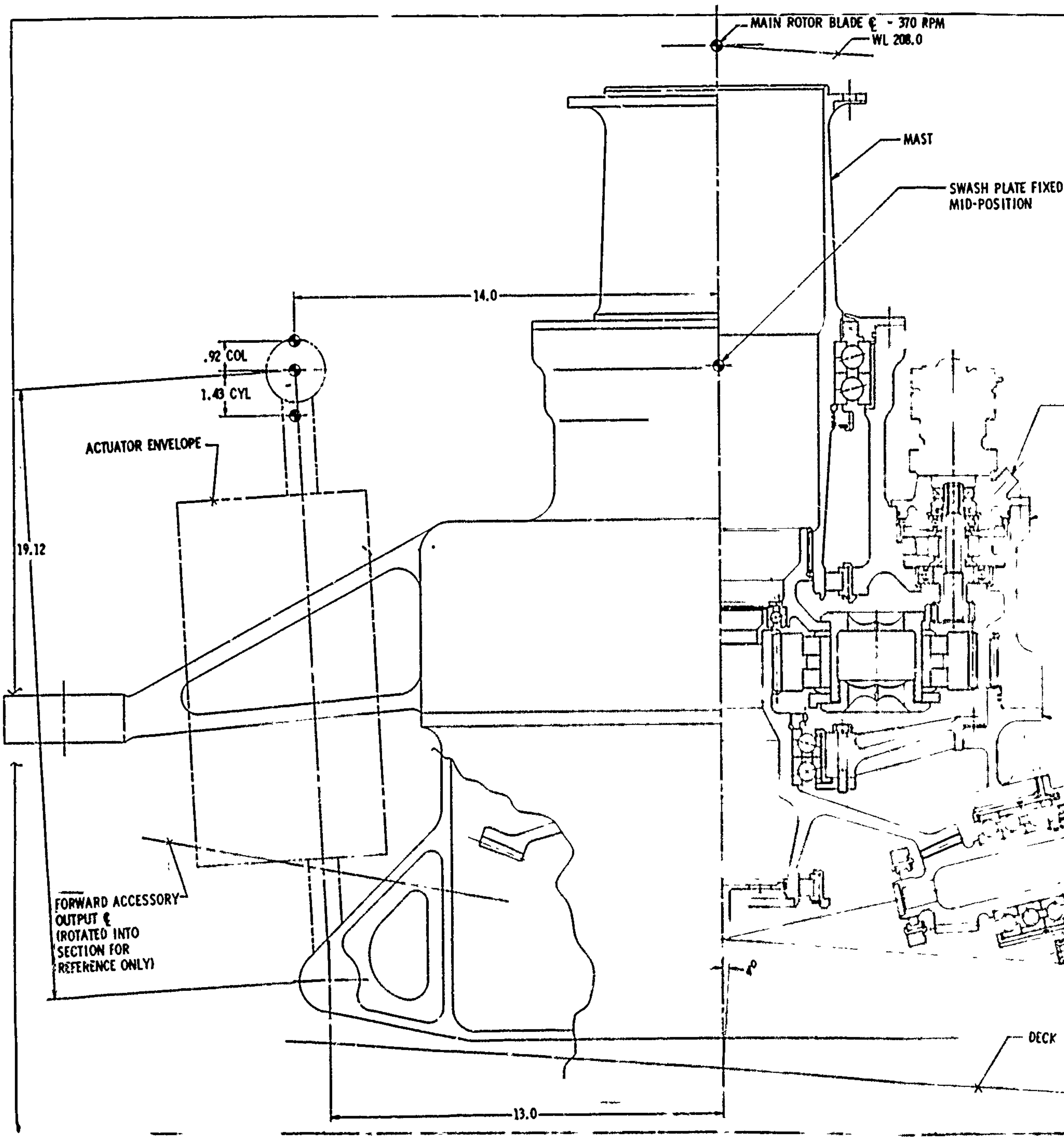
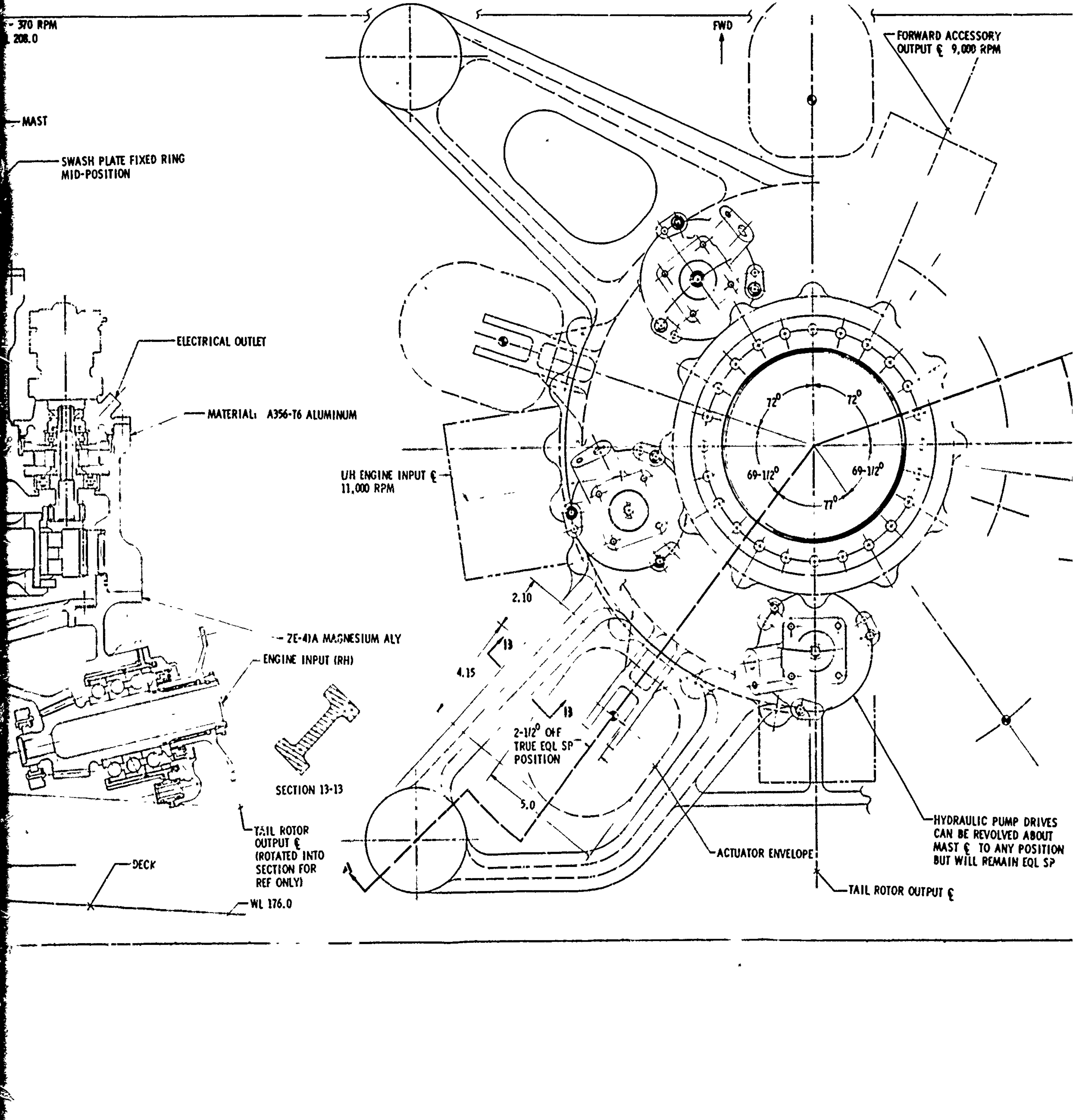
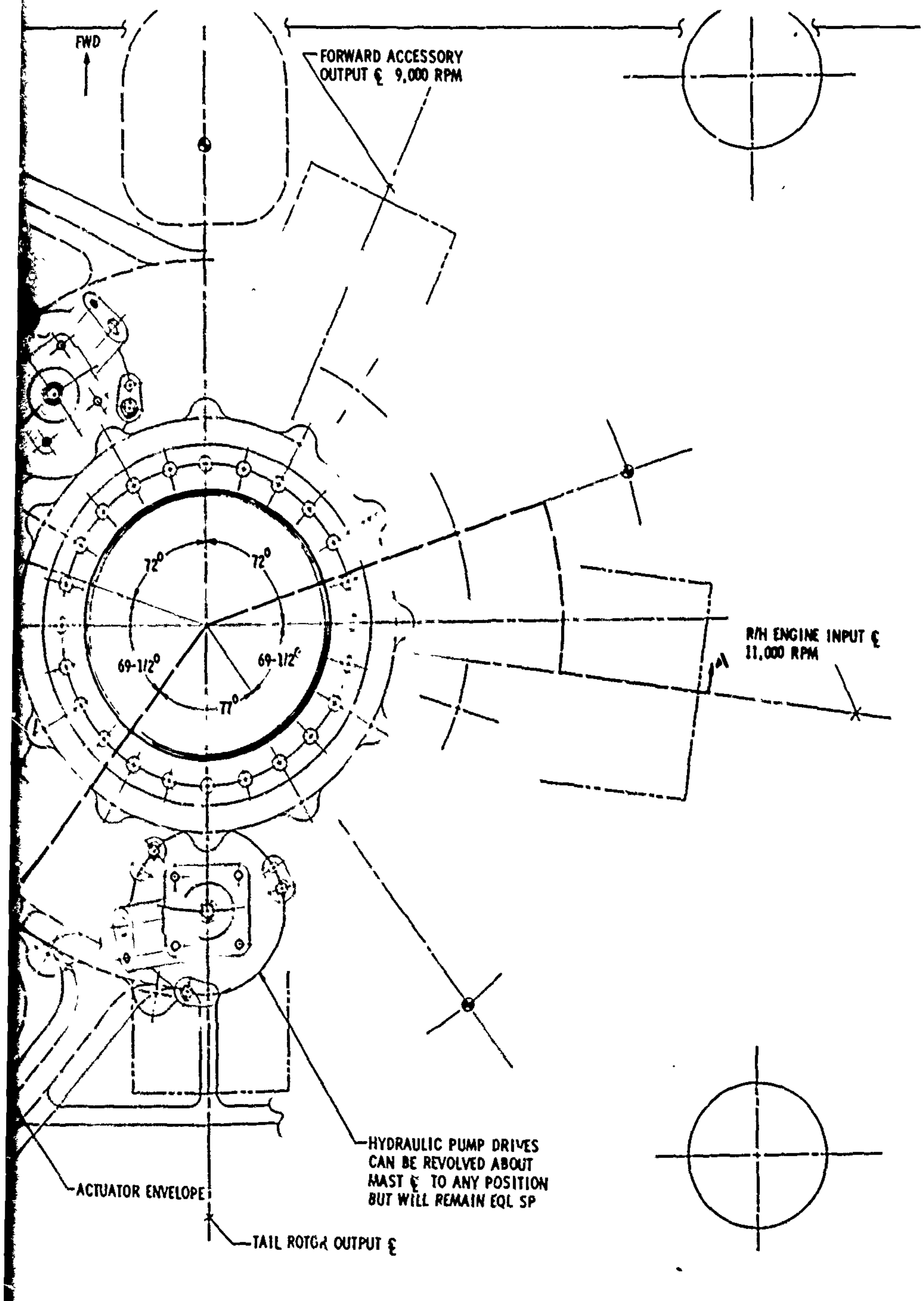


Figure 17. Installation drawing for STAR hydraulic pump and electrical alternator.



370 RPM
208.0
MAST
SWASH PLATE FIXED RING MID-POSITION
ELECTRICAL OUTLET
MATERIAL: A356-T6 ALUMINUM
ZE-41A MAGNESIUM ALY
ENGINE INPUT (RH)
SECTION 13-13
TAIL ROTOR OUTPUT ℄ (ROTATED INTO SECTION FOR REF ONLY)
DECK
WL 176.0
FWD
FORWARD ACCESSORY OUTPUT ℄ 9,000 RPM
LH ENGINE INPUT ℄ 11,000 RPM
72°
72°
69-1/2°
69-1/2°
77°
2.10
4.15
13
13
2-1/2° OFF TRUE EQL SP POSITION
5.0
ACTUATOR ENVELOPE
TAIL ROTOR OUTPUT ℄
HYDRAULIC PUMP DRIVES CAN BE REVOLVED ABOUT MAST ℄ TO ANY POSITION BUT WILL REMAIN EQL SP



FWD
FORWARD ACCESSORY
OUTPUT ℄ 9,000 RPM
72°
72°
69-1/2°
69-1/2°
77°
R/H ENGINE INPUT ℄
11,000 RPM
HYDRAULIC PUMP DRIVES
CAN BE REVOLVED ABOUT
MAST ℄ TO ANY POSITION
BUT WILL REMAIN EQL SP
ACTUATOR ENVELOPE
TAIL ROTOR OUTPUT ℄

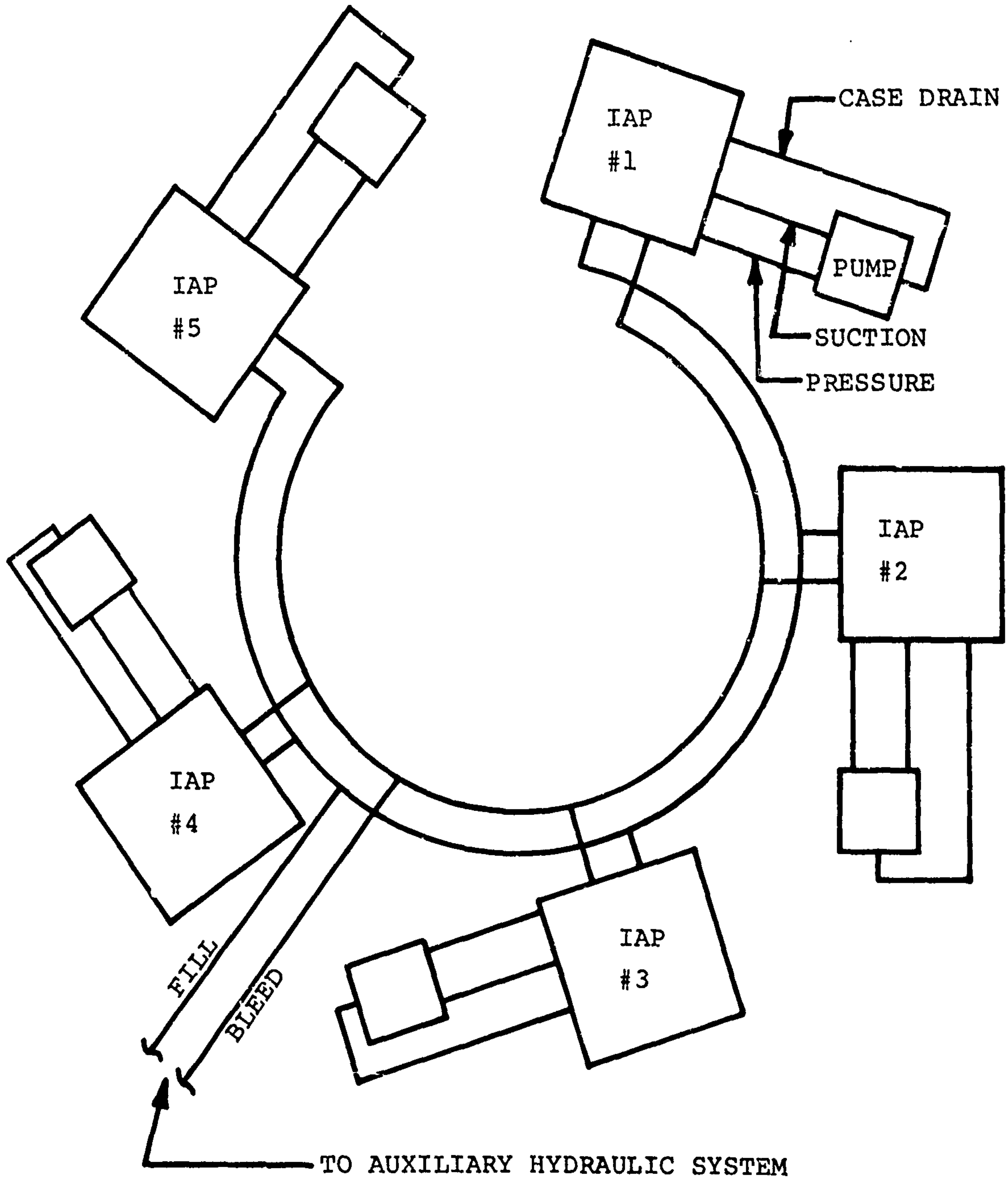


Figure 18. Sketch of dedicated and auxiliary hydraulic lines.

The provisions added to the IAP include a solenoid valve that allows the auxiliary hydraulic system to pressurize the IAP and a fluid tight reservoir overflow connection that allows a fluid return from the IAP to the auxiliary reservoir. A sketch of the layout is shown in Figure 19. When the transmission-driven pumps are operating, a bias spring prevents the solenoid valve from opening and auxiliary pump pressure is not available to the IAP. Anytime the IAP reservoir drops below 100 percent full, the mechanically operated reservoir overflow valve closes and fluid cannot leave the IAP. This means that the fill-and-bleed lines are isolated and not vulnerable during flight.

The following procedure would be used for fill and/or ground checkout:

- Turn motor/pump on.
- Engage IAP solenoids.
- Operate collective and cyclic sticks so that each IAP is exercised through full stroke (10 times minimum) to bleed system. (Note: If pressure does not build up, some malfunction exists in one or more IAPs. Malfunction may be isolated by switching off the IAPs. Do not switch off more than 2 IAPs at one time.)
- Operate system as required for checkout.
- Turn motor/pump off.
- Check that all IAP reservoirs are at least 90 percent full.

The following bleed procedure with rotor turning would be required on initial installation, or anytime a pump or IAP has been changed:

- Complete ground check before turning rotor.
- Start rotor.
- Observe that all pumps develop pressure and all IAP reservoirs stay above 20 percent. If not, stop rotor.
- Operate collective and cyclic sticks to the maximum extent possible with rotor turning so that each IAP is exercised several times.
- Stop rotor.
- Repeat ground check procedure.

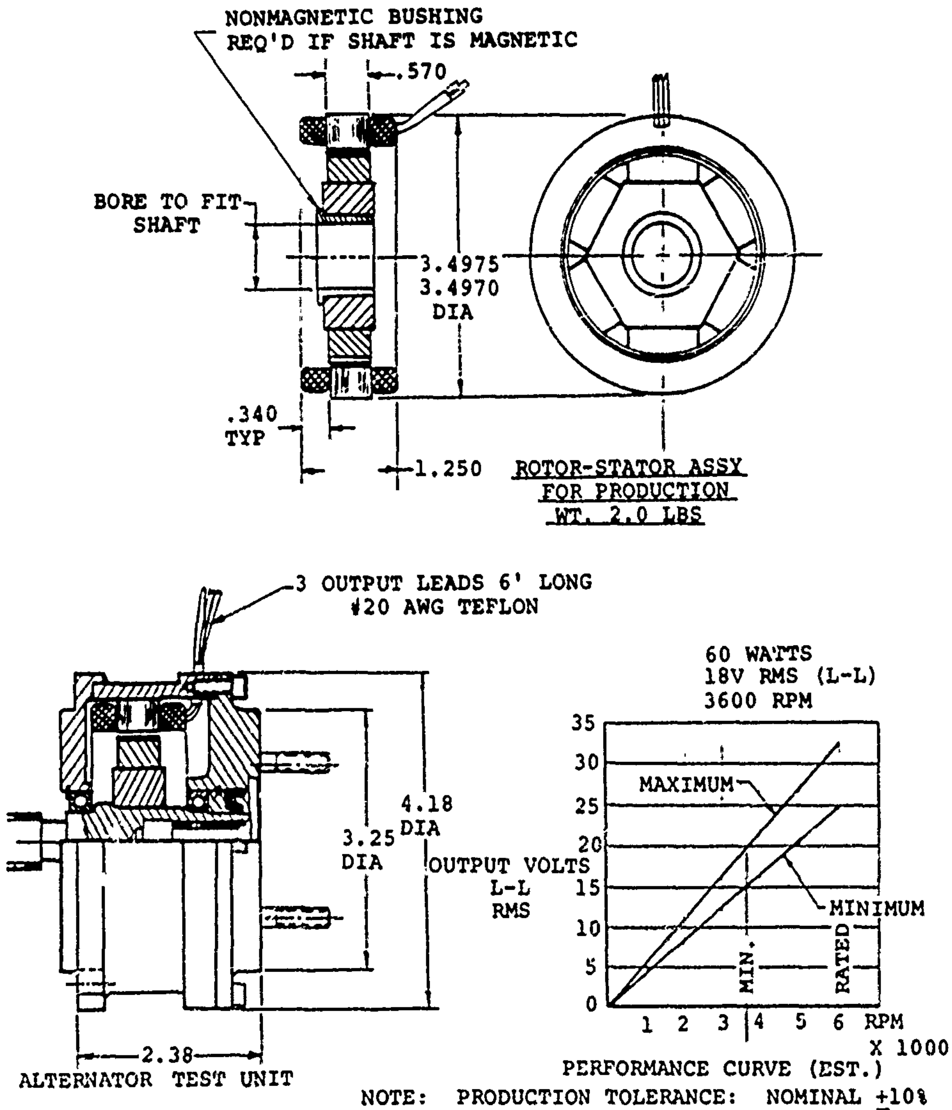


Figure 19. Prototype dedicated alternator.

2.2.6 Electrical and Electronic System

The electrical and electronic requirements for the fly-by-wire/light system mechanization are satisfied by utilizing common helicopter electrical supply hardware, additional dedicated alternators, and various electronic black boxes and modules.

2.2.6.1 Electrical Supply Schematic. The electromechanical actuators used in the horizontal stabilizer and tail rotor mechanization require redundant electrical power sources. Two power sources used are the AC buses of the MUT helicopter. Power from these buses is converted to dc by two transformer/rectifiers. A third source of dc power is the ship's battery. Dedicated alternators, shown in the installation drawing Figure 17, provide dedicated power to the IAP electronics modules. Characteristics of the alternator are shown in Figure 19.

Prior to reaching engine idle speed, power for the module is provided from the ship's battery. Automatic switch-over occurs when alternator voltage exceeds battery voltage. The electrical supply layout is depicted in Figure 20.

2.2.6.2 Electronic Module Characteristics. Early in the pre-design effort, two candidate configurations were considered for the STAR channel implementation. Conceptually, these are shown in Figure 21. The fundamental difference between the two concepts consists of the location and physical characteristics of the electronic module (EM). In one case, the module is mounted directly to the IAP and, in effect, becomes a part of the IAP assembly. In the other case, the module is located remote to the IAP and must be electrically connected to the various IAP valves and sensors through multiple electrical connections. The advantage of the IAP-mounted module is the significant inherent immunity to EMI, EMP, and lighting noise due to the use of fiber optic signal input and the elimination of exposed metallic signal interconnect wires between the electronic module and the IAP.

Design analyses indicate that IAP-mounted electronics are indeed feasible and are therefore proposed for the fly-by-wire/light mechanization. Thermal analysis for a U-shaped module is shown in Table 10. The shape was selected because it provides a reasonable volume and a large amount of surface area for heat dissipation while staying inside the allowable IAP envelope for all mounting positions. Since the mounting surface on the actuator will be at a high temperature, thermal insulators will be used between the actuator and the EM package.

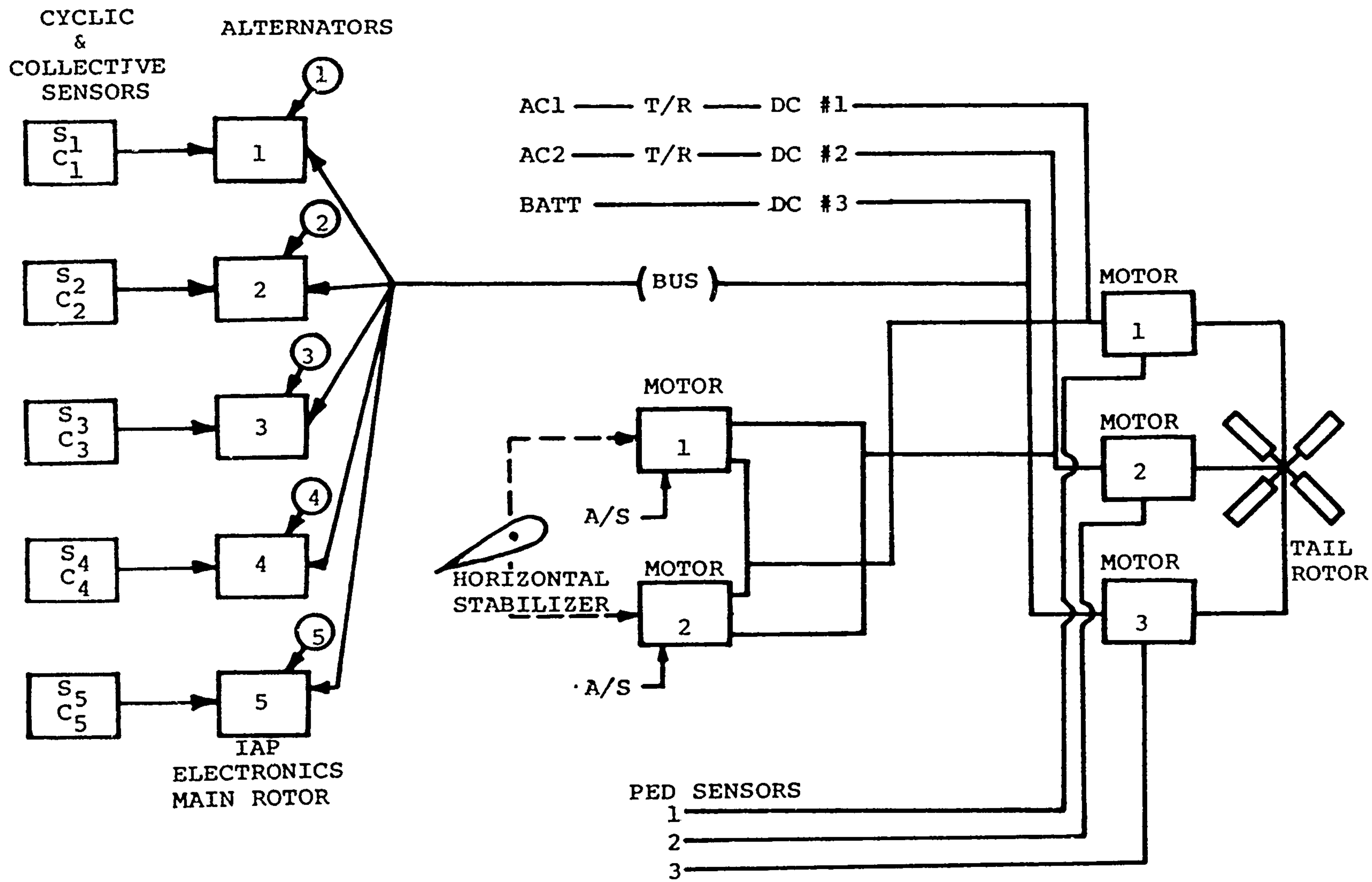
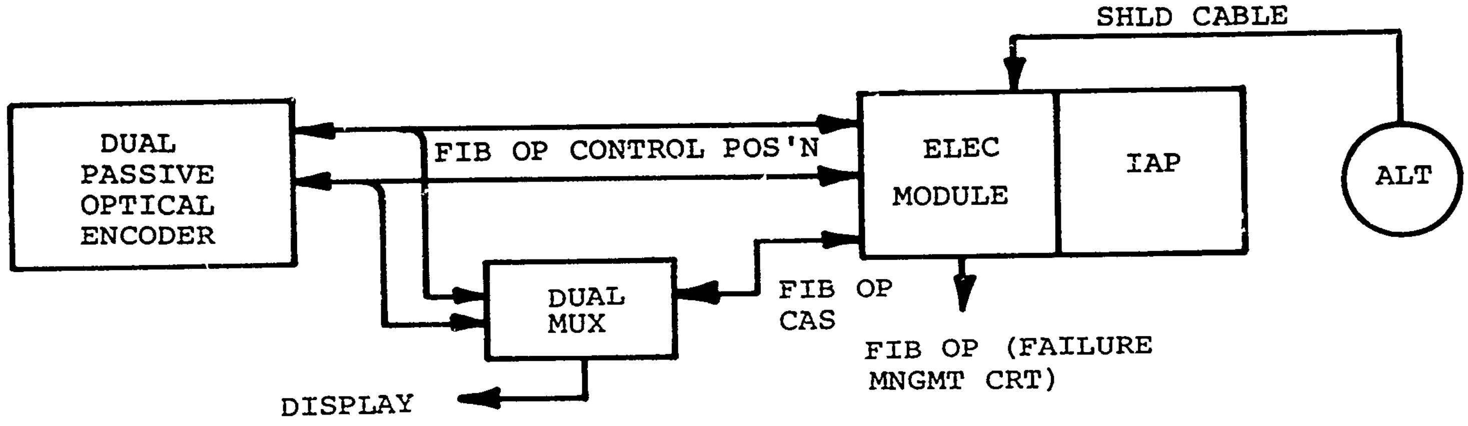
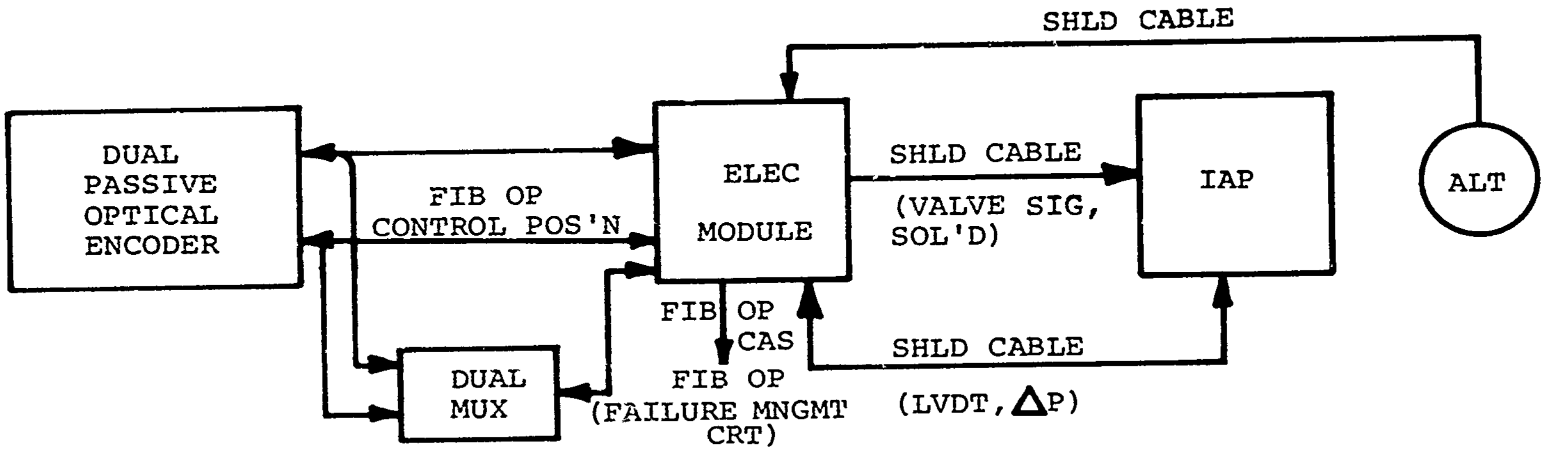


Figure 20. ASH FBW/L control system electrical supply.



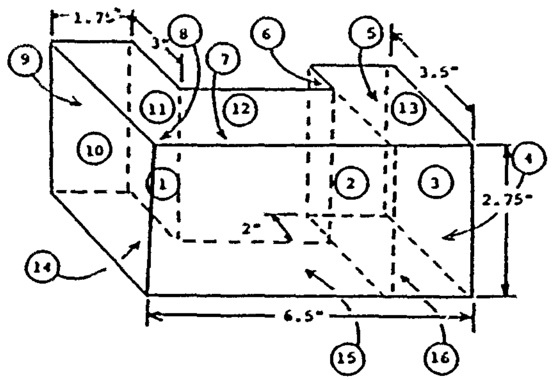
CANDIDATE 1 OPT



CANDIDATE 2 OPT

Figure 21. STAR FBW/L candidate configurations.

TABLE 10. ELECTRONICS MODULE THERMAL ANALYSIS



Location	Surface	Temp (°C)	Notes
Front	1	77.6	Convection
Front	2	78.5	
Front	3	77.4	
Side	4	77.8	
Rear	5	79.6	
Side	6	84.5	260°F Radiation from actuator - 3/16" spacing
Rear	7	85.9	
Side	8	84.6	
Rear	9	79.7	Convection
Side	10	78.2	
Top	11	78.5	
Top	12	79.7	
Top	13	77.5	
Bottom	14	83.7	Mounted to actuator with four 1/4" steel bolts 0.25" separations, also, radiation from actuator
Bottom	15	84.6	
Bottom	16	82.9	

Conditions:

8.5 Watts dissipated

30 CFM @ 152°F cooling air

260°F mounting surface

Averaged Temperatures:

$T_{radiation} = 84.4°C$

$T_{convection} = 78.5°C$

$T_{total} = 80.6°C$

$T_{junction} = 80.6 + 15 = 95.6°C$

A plug-in-type electrical connector will be located at the bottom of the electronic module. This connector will be used to take three-phase ac power and 28 VDC power into the module and to provide the interconnect between the electronic module and the actuator for transfer of LVDT excitations, LVDT position information, EHSV command lines, and pressure transducer information. The chassis connector design was selected to eliminate external wires, thus reducing EMI susceptibility.

The physical design characteristics of the electronic module package design are driven by the high temperature environment that will be encountered and by the relatively small volume available. Hybrid electronic packages are used to provide the density needed. The packages are mounted on multilayer printed wiring boards with extensive on-board heat sinking. The circuit boards slide into metallic rails on the side of the chassis to provide good thermal conduction to the chassis. A motherboard design with slide-in connectors will provide all the required board-to-board interconnections. Access to the circuit cards is provided through a removable top cover on the electronic module package. The power converter will be mounted in one of the "legs" of the U-shaped box in conjunction with the power connector interface with the actuator.

The internal circuitry of the electronic module includes a rectifier and voltage regulator for the dedicated alternator, logic circuitry to switch from battery to alternator power when alternator has sufficient rpm, optical encoder/decoding circuitry, electronic servoloop circuitry for controlling the IAP electrohydraulic servovalve, monitor and IAP bypass control circuitry, and circuitry to provide display information.

The block diagram of Figure 22 depicts the relationship of the various circuits housed in the electronic module. Detailed schematics are shown in Figures 23 through 28. To adequately house the circuitry in a small package, a number of hybrid circuits are considered.

Figure 12 depicts the block diagram of the entire ASH mission FBW/L control system. A typical electronic module interfaces with the other system components and assemblies, as shown in Figure 29. Control inputs are derived from two sources. One source is the pilot's control motion input. The other source is the control augmentation input. Control input is provided from dualized sensors over direct link fiber optic cables. Control augmentation input is provided from dual computers via Multiplex Remote Terminal Units (MRTU). The dual computers are self-monitoring and communicate with each other via MRTUS over dualized MIL-STD-1553B data buses.

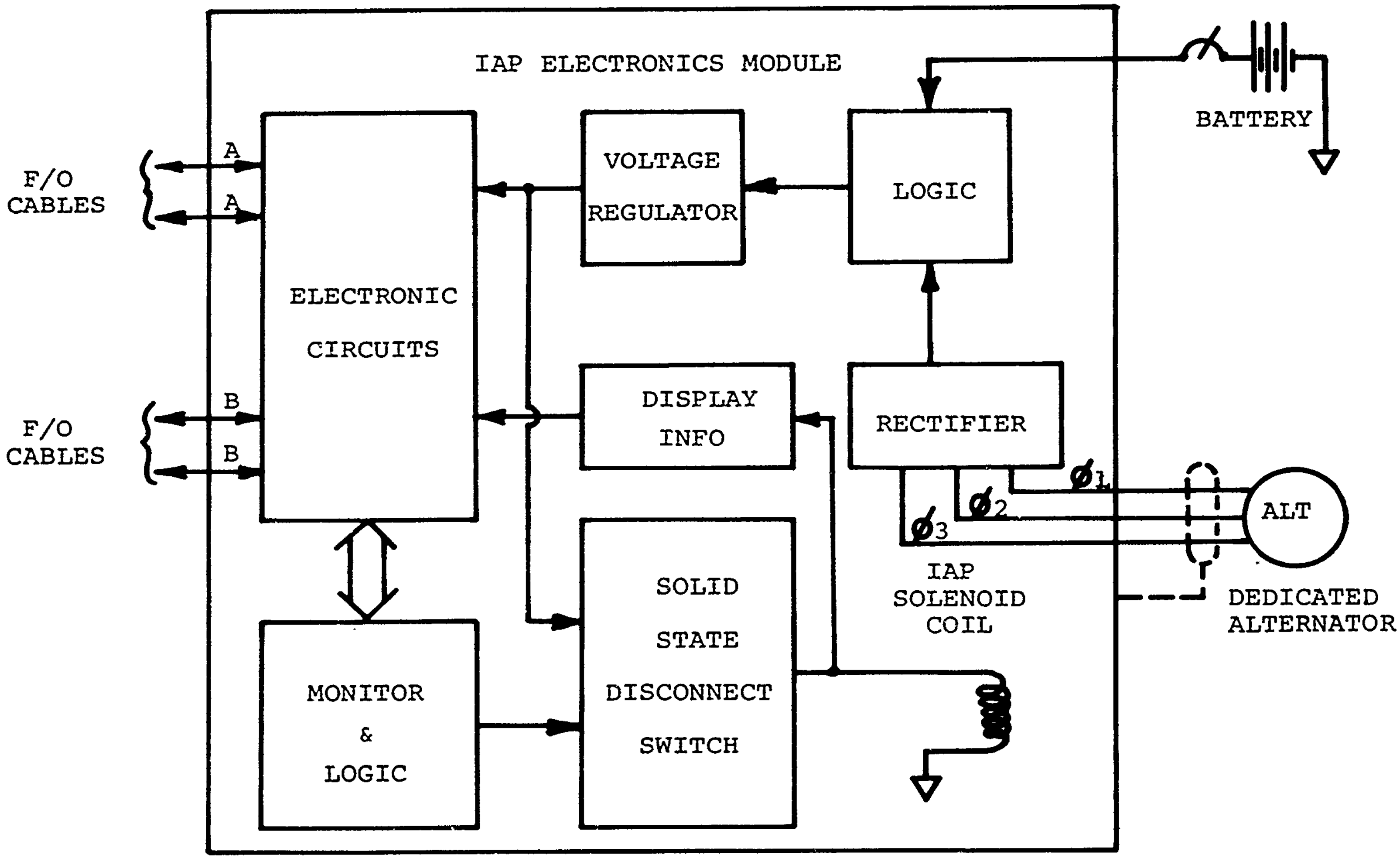


Figure 22. Architectural structure of the IAP electronics module.

ALL CAPACITANCES IN FARADS
ALL RESISTANCES IN OHMS
ALL VOLTAGES IN VOLTS

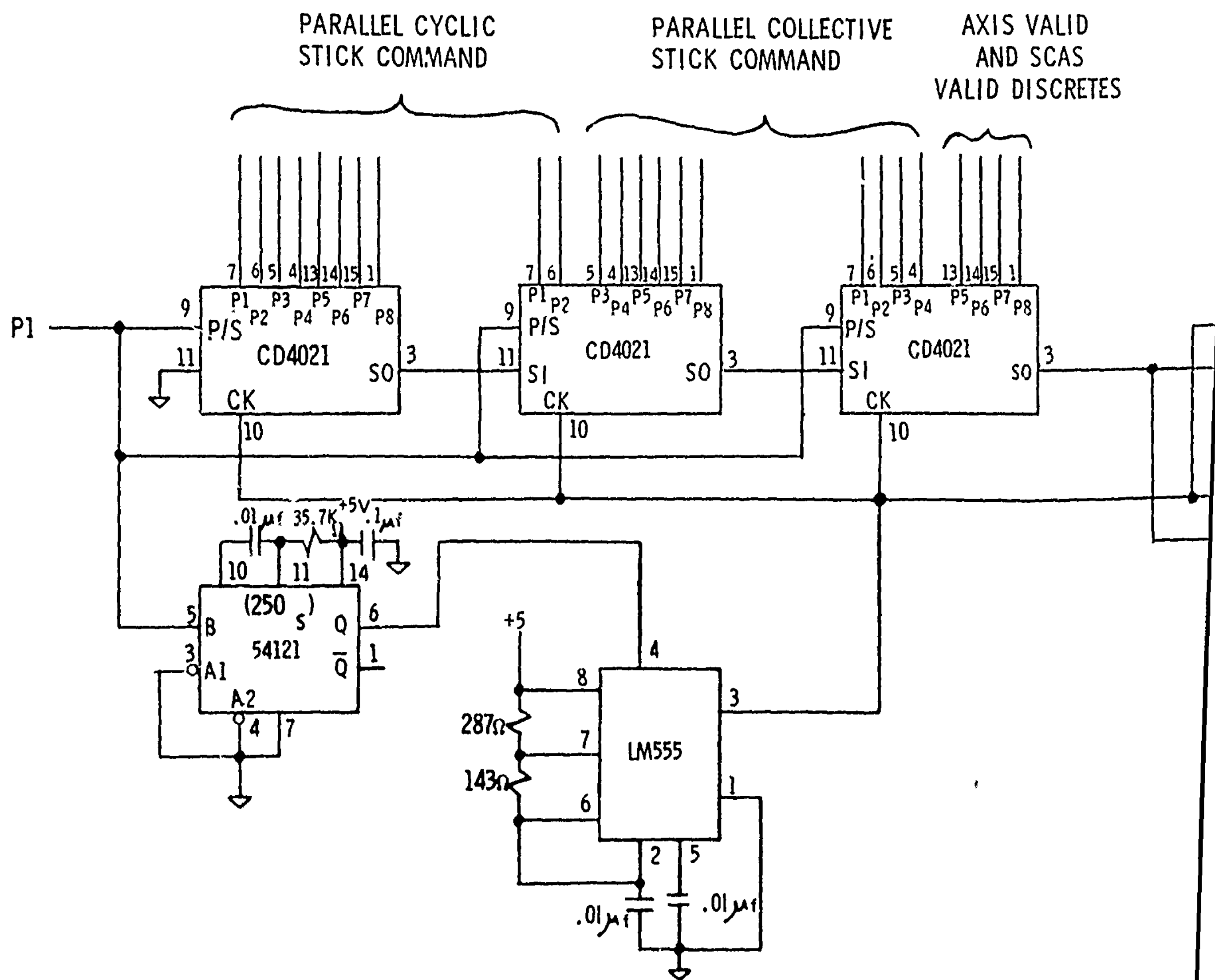
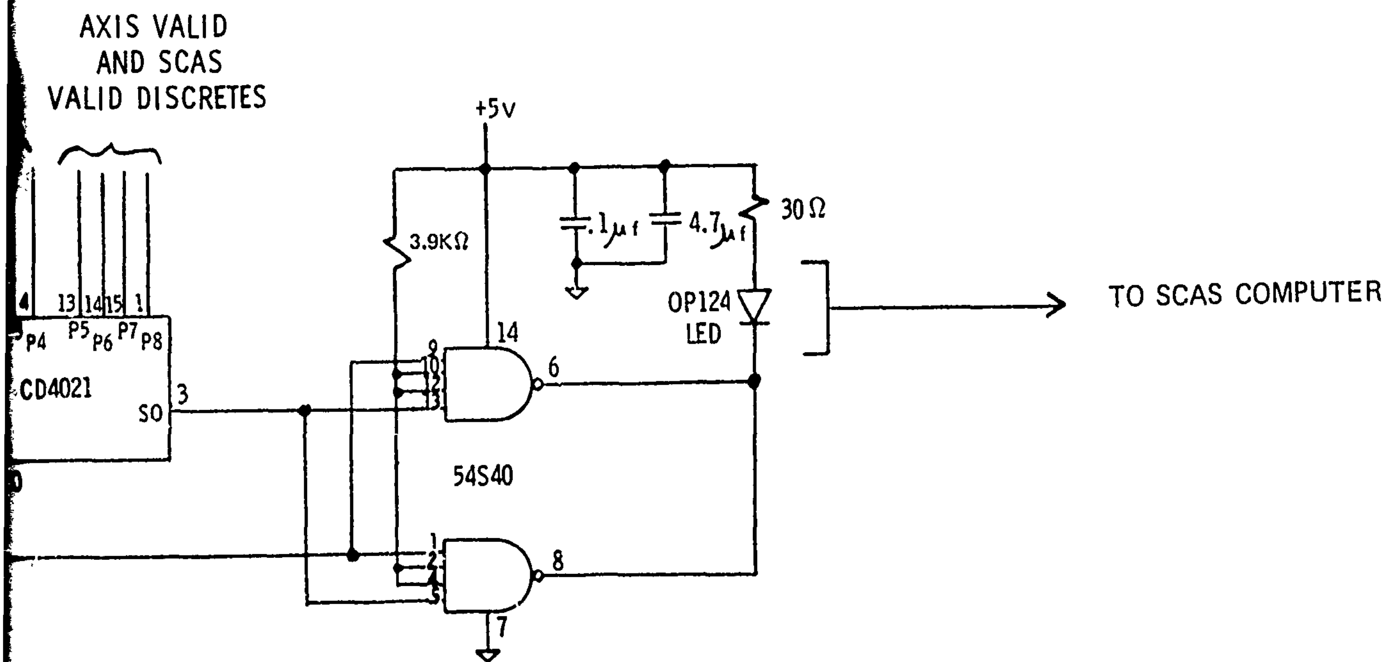


Figure 23. SCAS transmitter electronics - two of these per IAP.


AXIS VALID
AND SCAS
VALID DISCRETES
+5V
3.9KΩ
.1μf
4.7μf
30Ω
OP124
LED
TO SCAS COMPUTER
CD4021
P4
P5
P6
P7
P8
SO
54S40

IAP.

ALL CAPACITANCES IN FARADS
ALL RESISTANCES IN OHMS
ALL VOLTAGES IN VOLTS

14
B

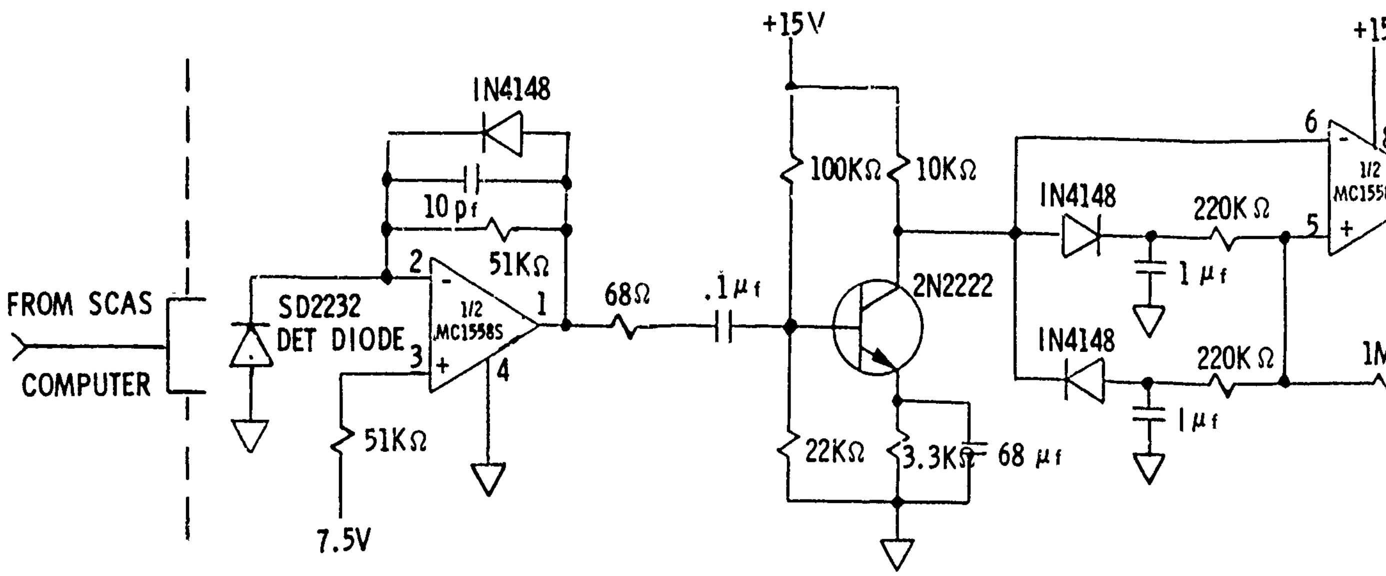
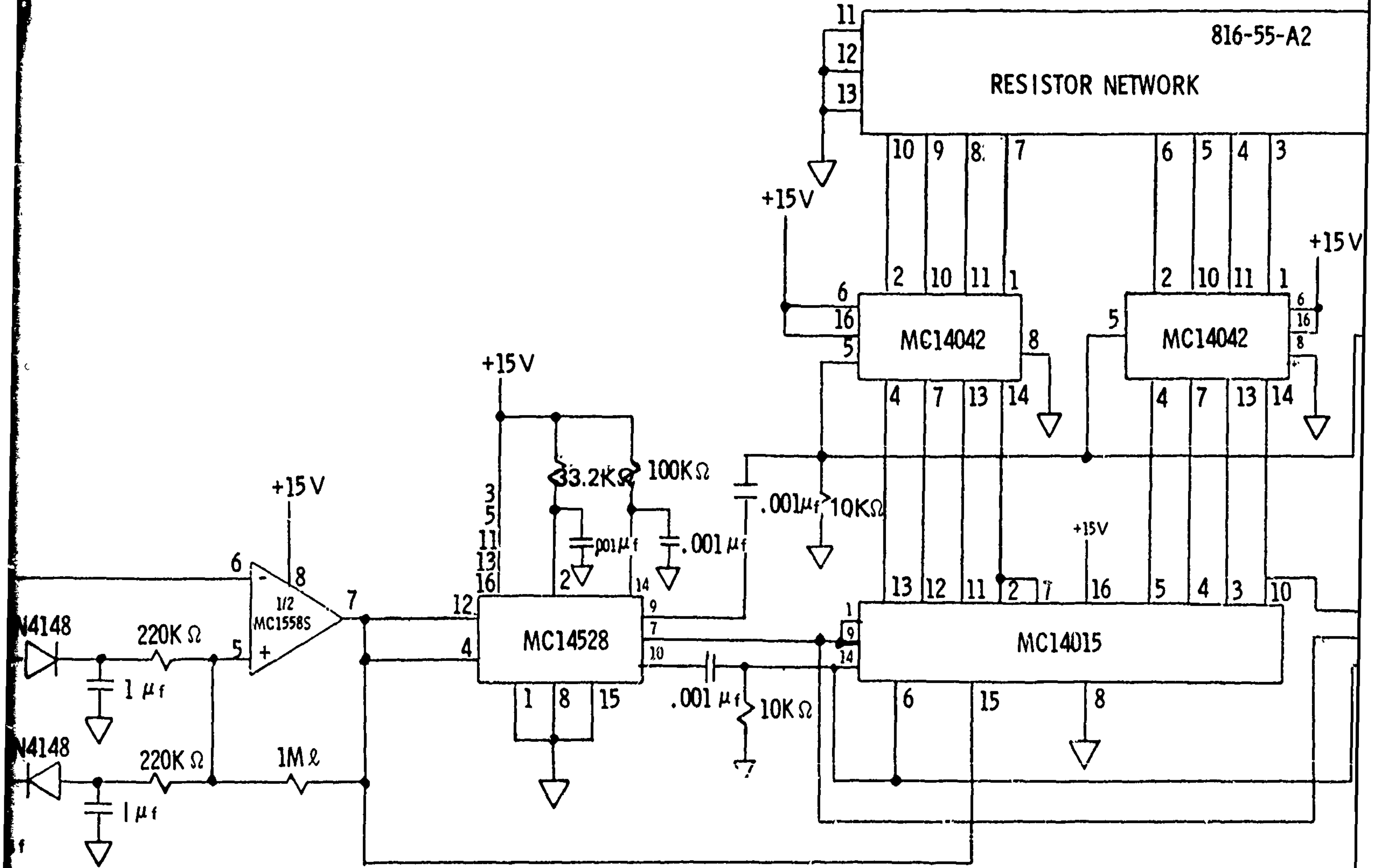
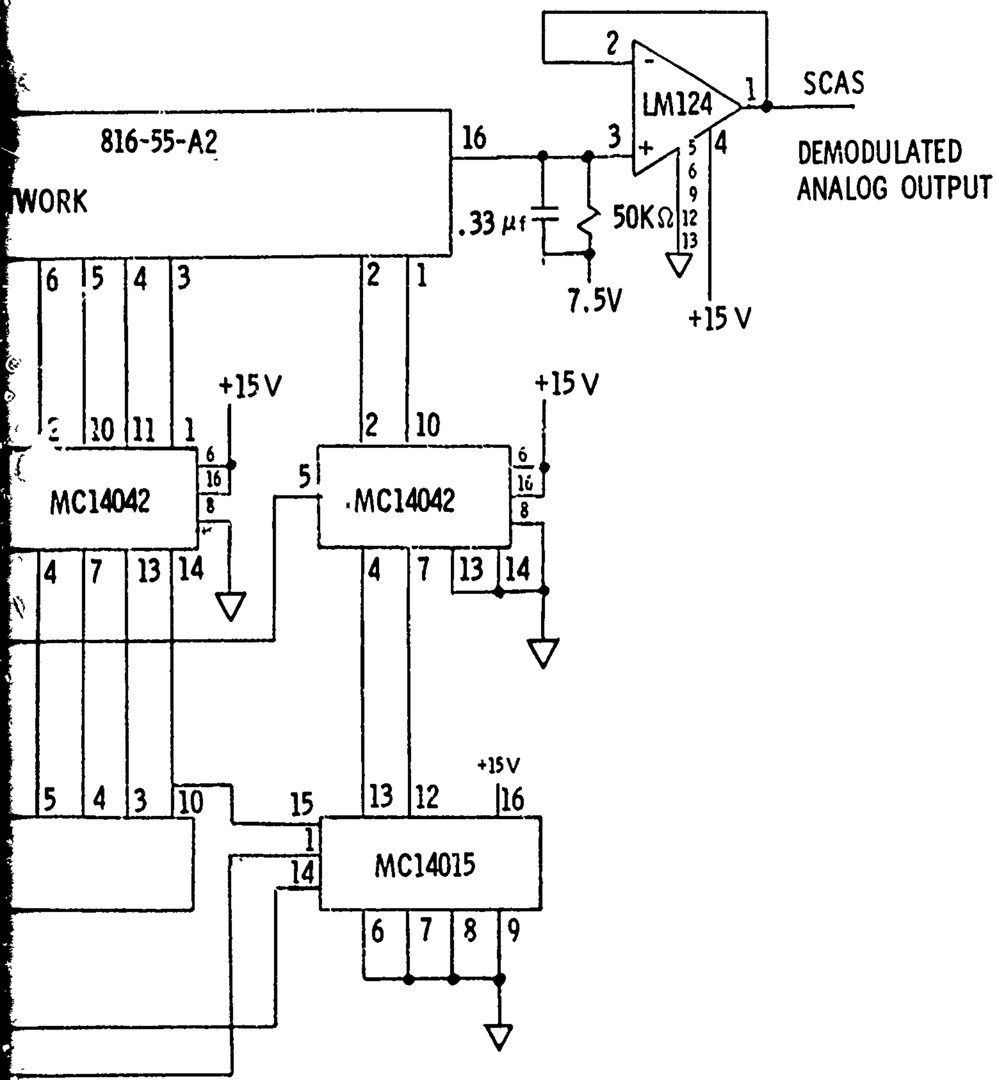


Figure 24. SCAS electronics receiver and decoder - two of these per IAP.


816-55-A2
RESISTOR NETWORK
MC14042
MC14042
MC14015
MC14528
1/2 MC1558S
+15V
33.2KΩ
100KΩ
.001μf
10KΩ
220KΩ
1MΩ
1μf
N4148

7



816-55-A2
WORK
LM124
SCAS
DEMODULATED
ANALOG OUTPUT
.33 µf
50KΩ
7.5V
+15 V
MC14042
MC14042
MC14015
+15V

3

ALL CAPACITANCES IN FARADS
ALL RESISTANCES IN OHMS
ALL VOLTAGES IN VOLTS

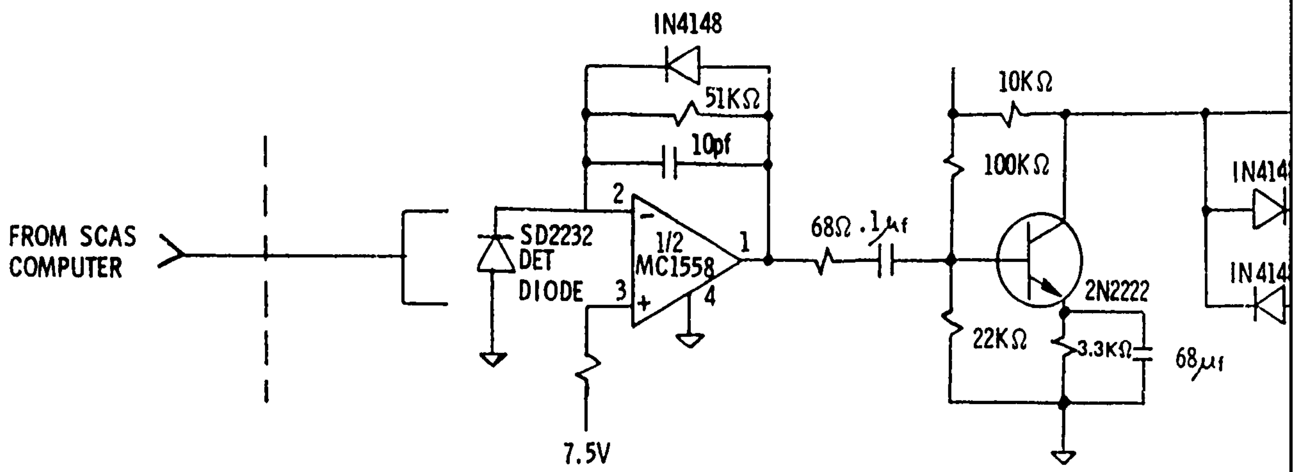
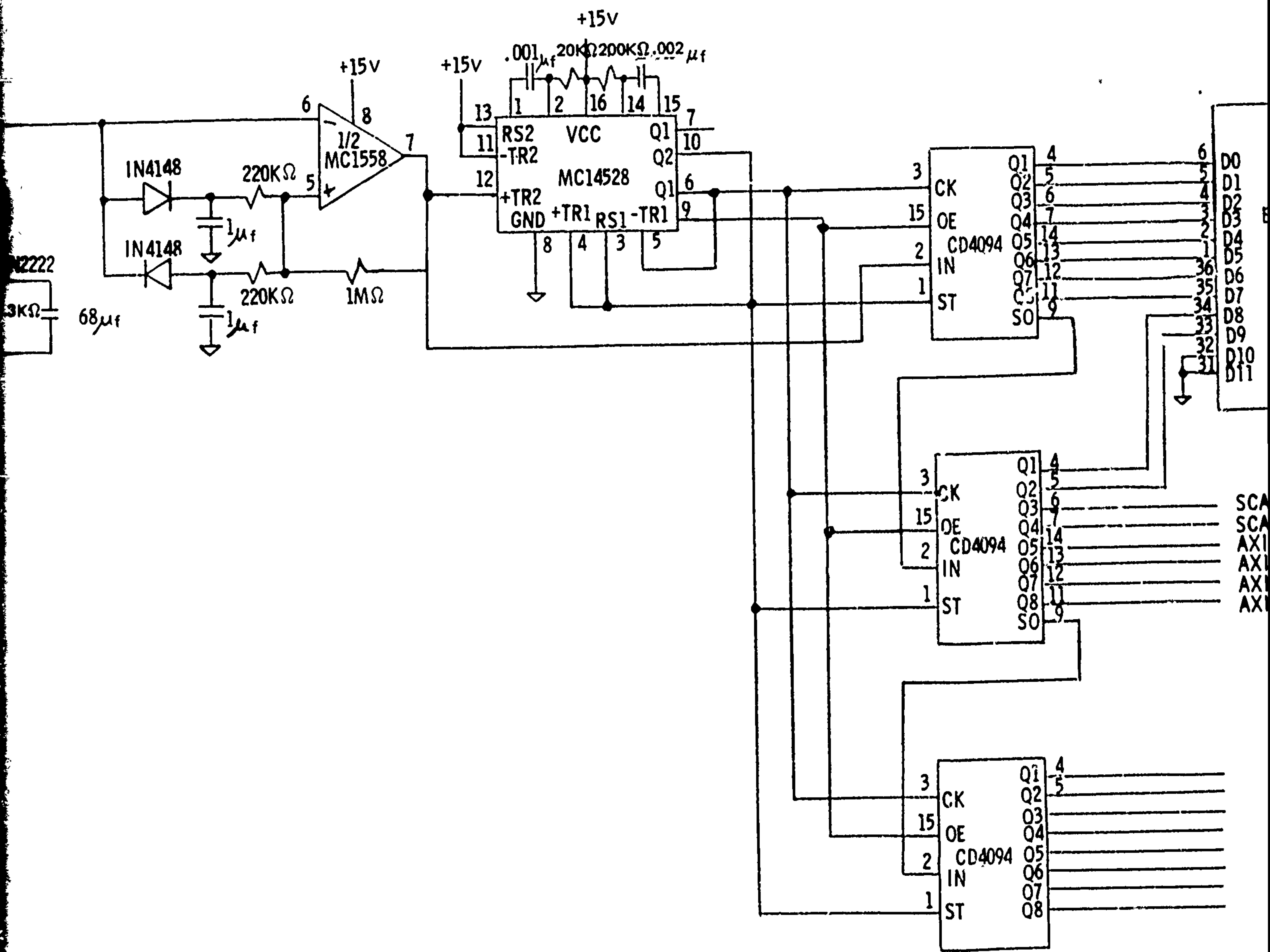
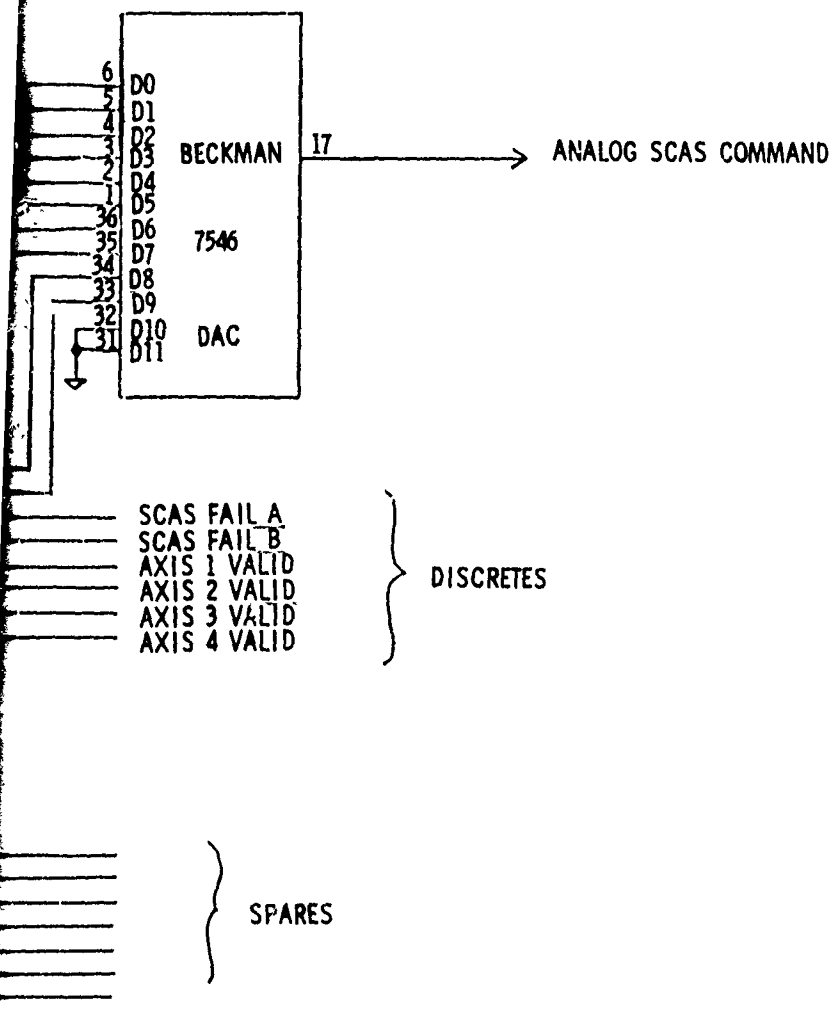
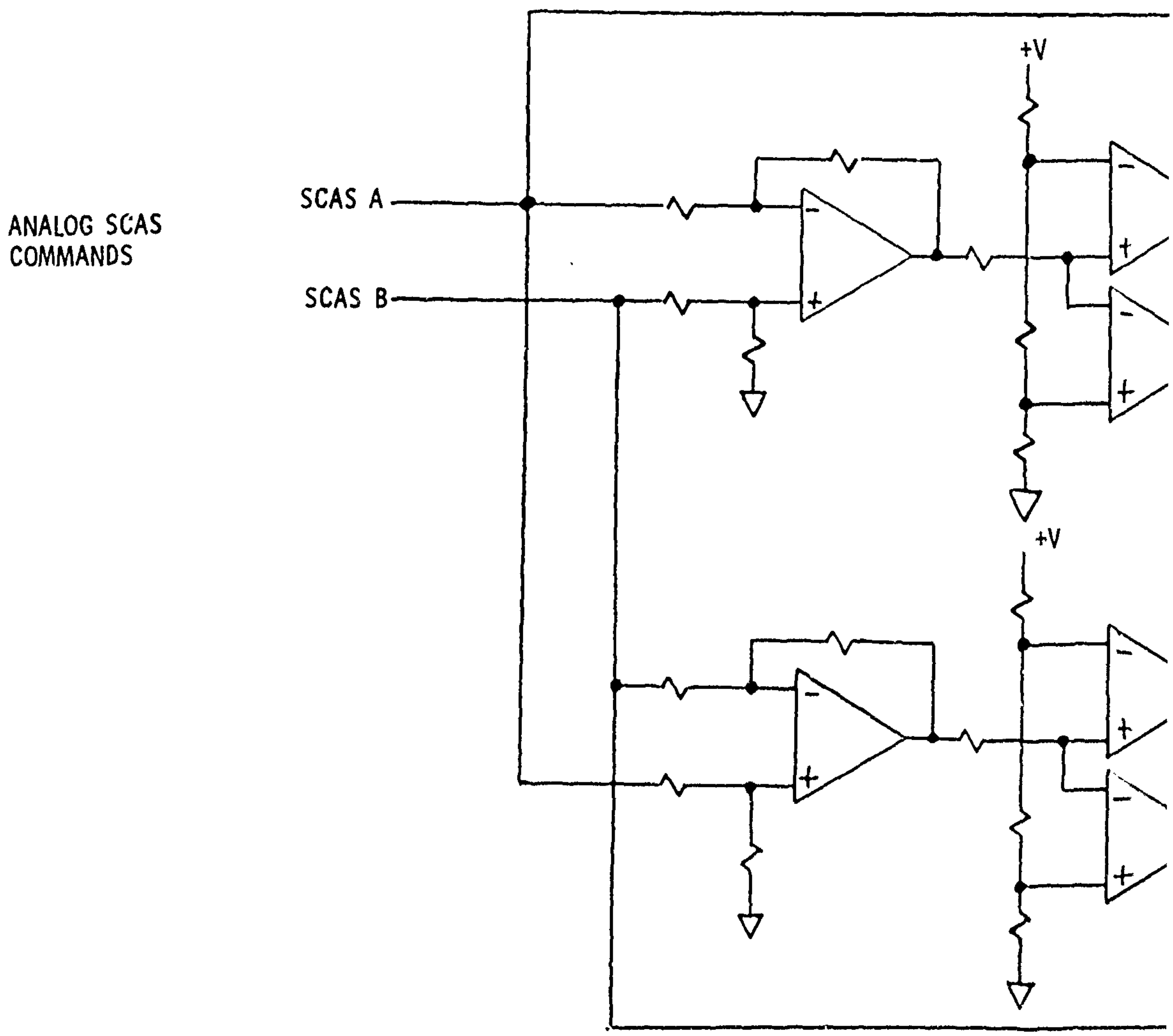


Figure 25. SCAS receiver electronics - two of these per IAP.


+15v
+15v
+15v
.001μf
20KΩ
200KΩ
.002μf
1/2
MC1558
1N4148
1N4148
220KΩ
220KΩ
1MΩ
1μf
1μf
N2222
3KΩ
68μf
RS2
-TR2
+TR2
GND
VCC
MC14528
Q1
Q2
+TR1
RS1
-TR1
CK
OE
IN
ST
CD4094
Q1
Q2
Q3
Q4
Q5
Q6
Q7
Q8
SO
D0
D1
D2
D3
D4
D5
D6
D7
D8
D9
D10
D11

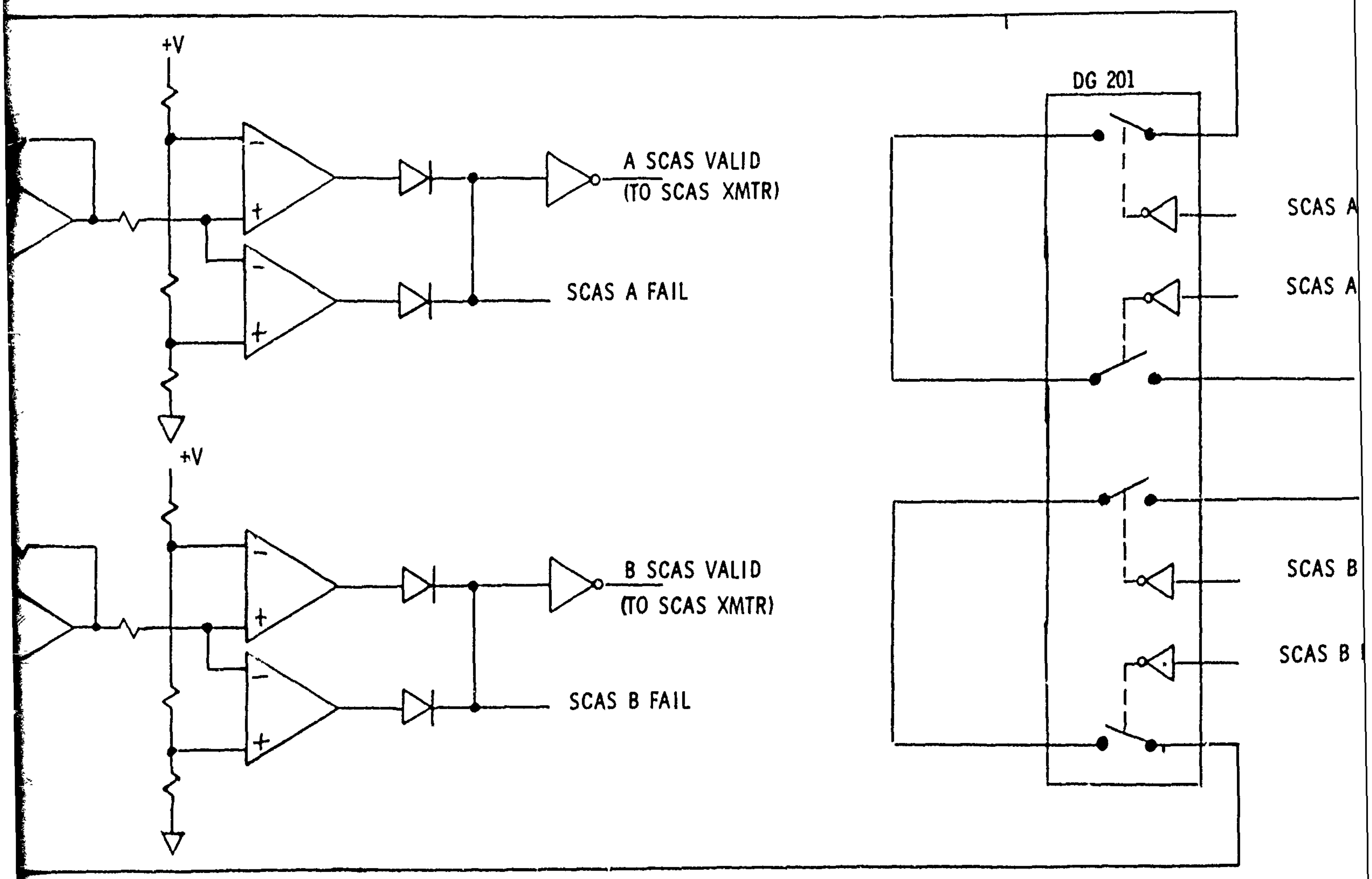

6
5
4
3
2
1
36
35
34
33
32
31
D0
D1
D2
D3
D4
D5
D6
D7
D8
D9
D10
D11
BECKMAN
7546
DAC
17
ANALOG SCAS COMMAND
SCAS FAIL A
SCAS FAIL B
AXIS 1 VALID
AXIS 2 VALID
AXIS 3 VALID
AXIS 4 VALID
DISCRETES
SPARES

2

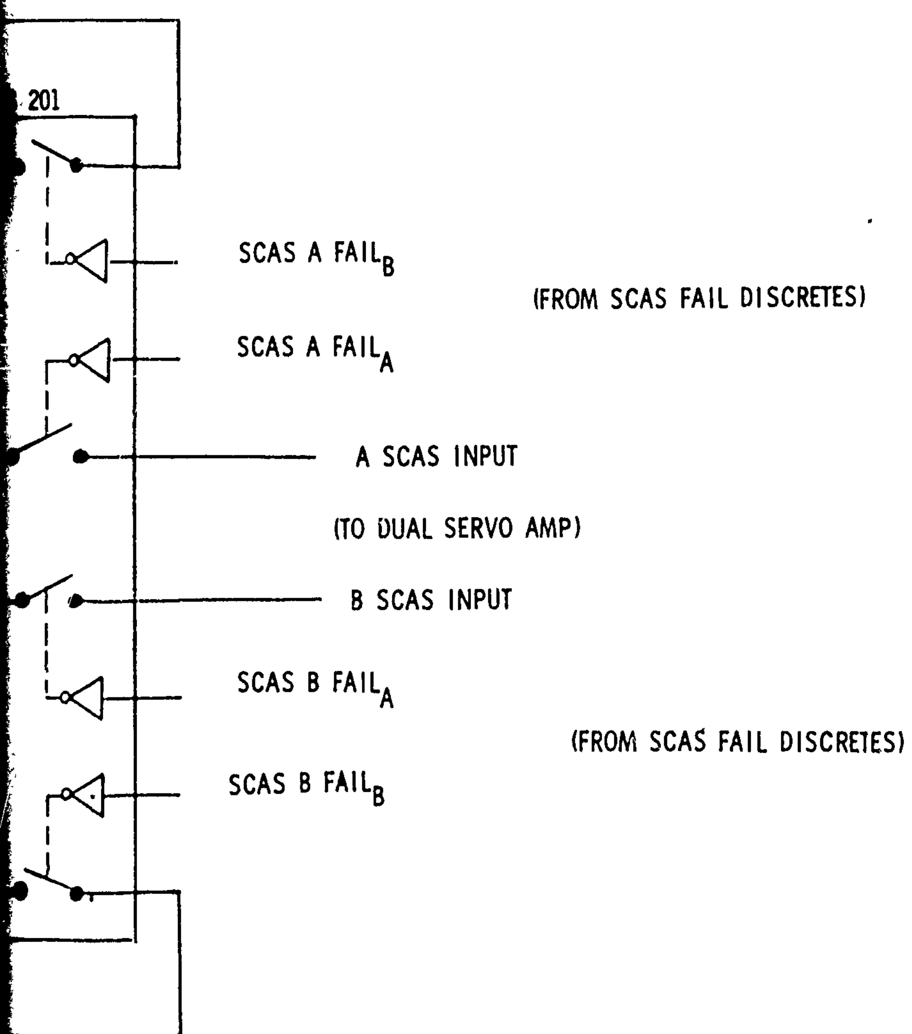


16
F

Figure 26. SCAS valid monitor one per IAP.


+V
DG 201
A SCAS VALID
(TO SCAS XMTR)
SCAS A FAIL
+V
B SCAS VALID
(TO SCAS XMTR)
SCAS B FAIL

12


201
SCAS A FAIL$_B$
(FROM SCAS FAIL DISCRETES)
SCAS A FAIL$_A$
A SCAS INPUT
(TO DUAL SERVO AMP)
B SCAS INPUT
SCAS B FAIL$_A$
(FROM SCAS FAIL DISCRETES)
SCAS B FAIL$_B$

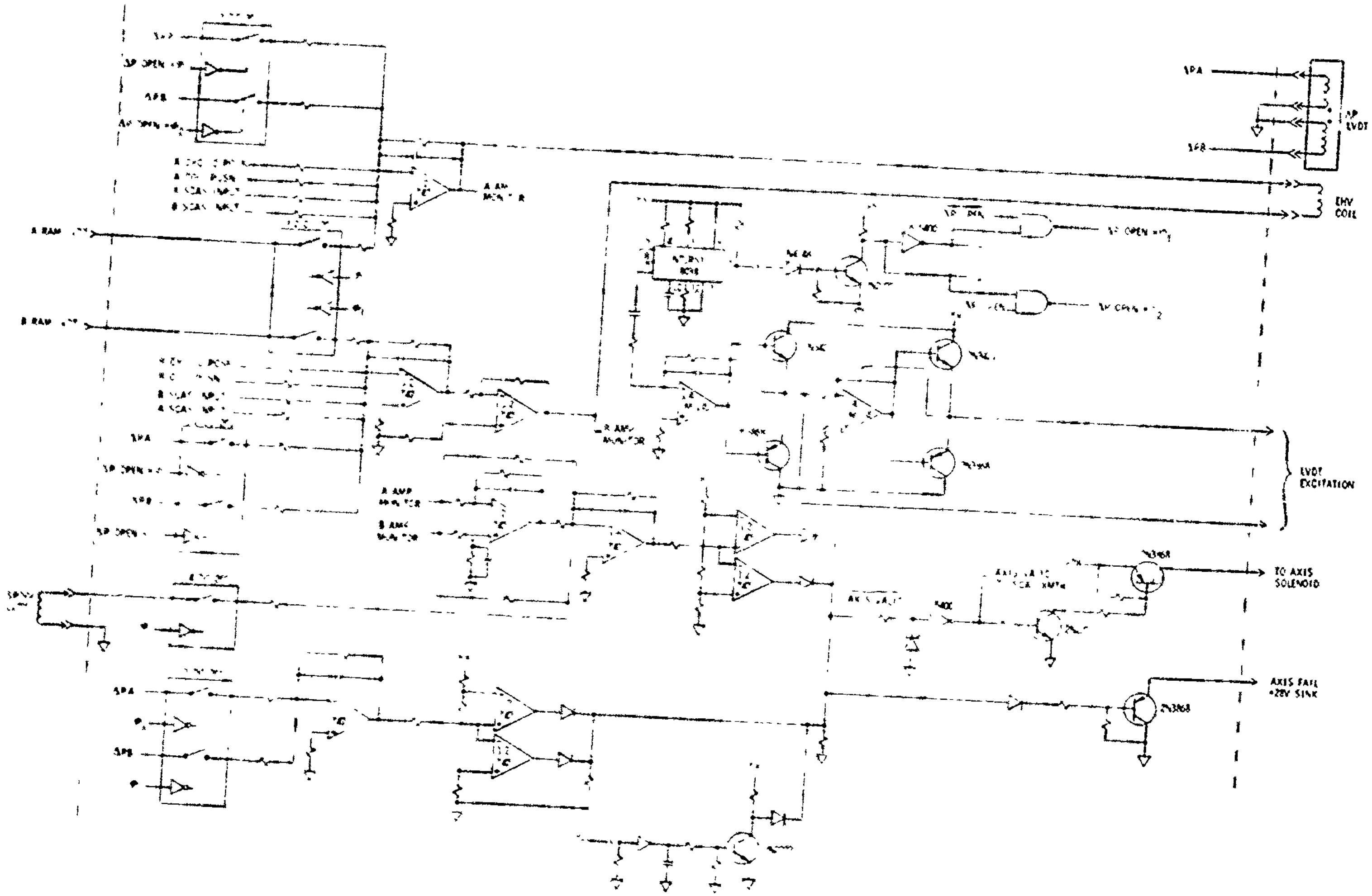


Figure 27. Dual servoamp - one per IAP.

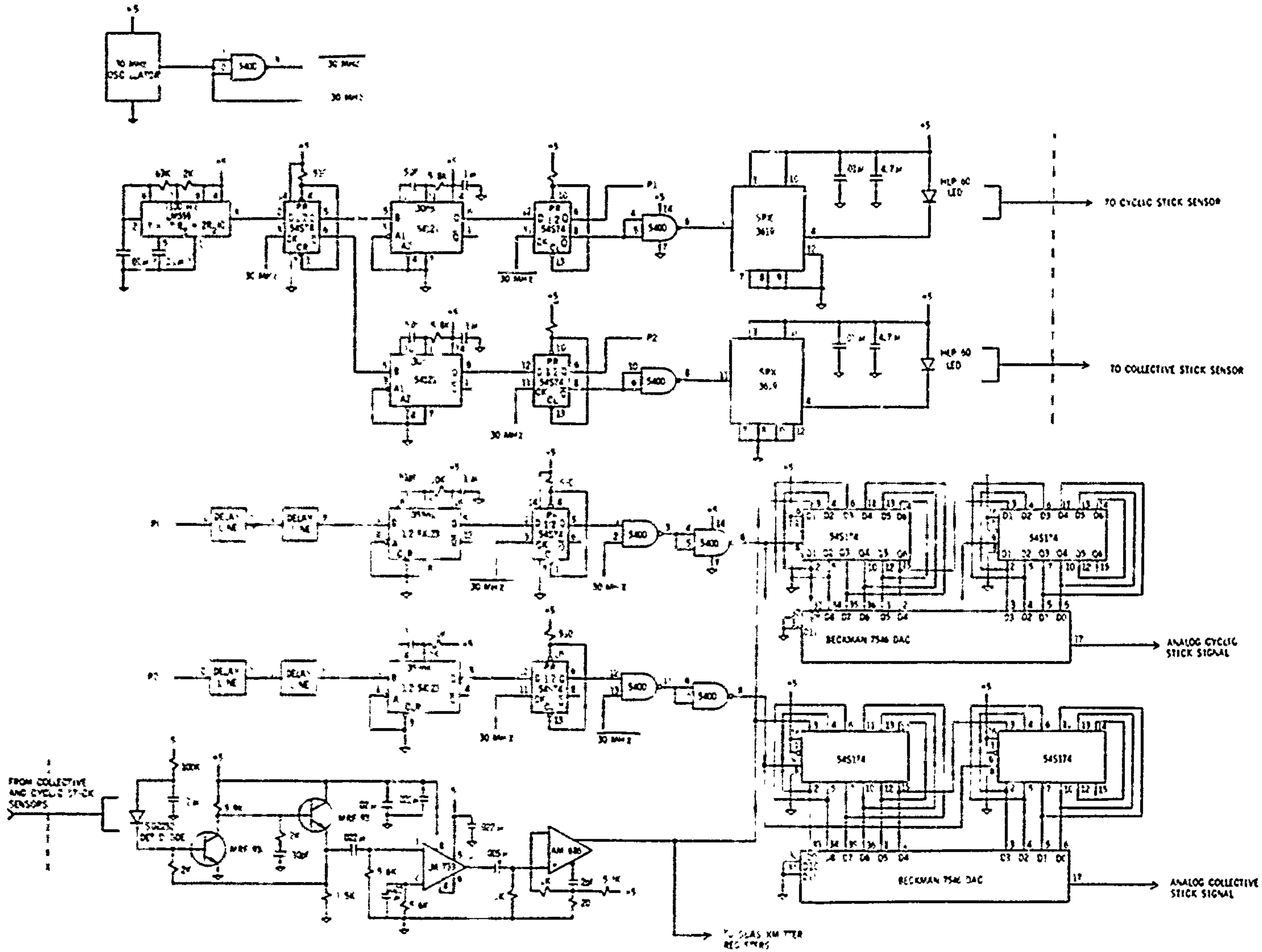


Figure 28. Typical sensor electronics channel - two per IAP.

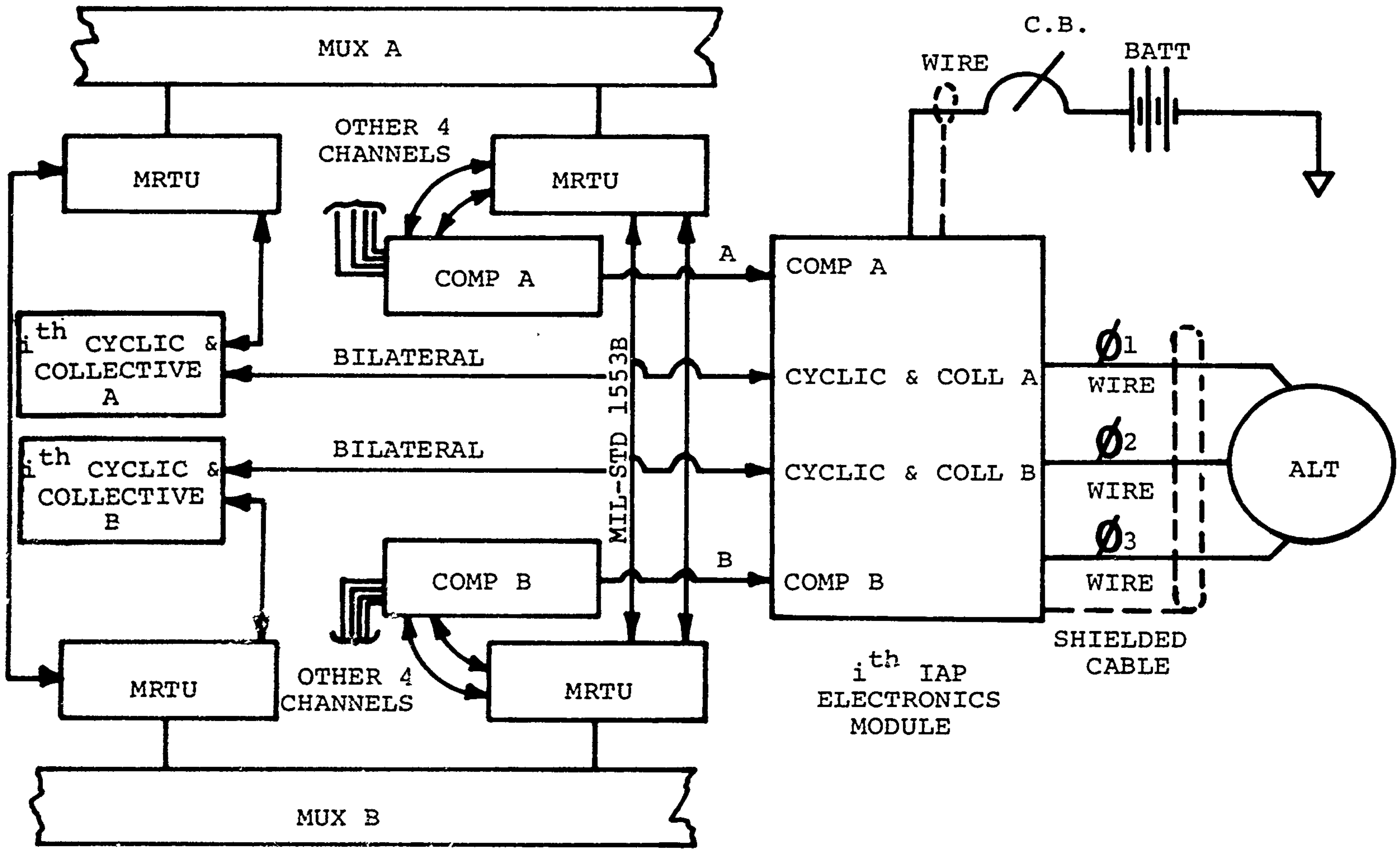


Figure 29. Typical STAR IAP electronics module interface.

A three-fiber connector is used for interfacing with the passive sensors to provide for transmission and reception of optic pulses. Sync pulses will be generated in the electronic module and transmitted through dual optic lines to the passive sensor. The sensor will then encode the pulse into serial digital data that will return to the electronic module through a third, time-shared, optic line. The electronics will then decode the data for use in the servoamp.

A two-fiber connector is used for interfacing the electronic module unit with the control augmentation computers to provide both a transmit and a receive line. A dual set of connectors is provided to interface with the dual control augmentation computers. The encoded digital data from the stick sensors and the actuator valid discretes are transmitted from the electronic module to the control augmentation computers over the optic lines. The signals received from the control augmentation computers by the electronic module include control augmentation serial data, control augmentation fail discretes from the control augmentation monitors, and interchannel logic data from the other STAR actuators.

These multifiber connectors will be so designed as to afford hermetic seals at the interface with the electronic module. The proposed connector is a form of a flat connector. The shell is standard, but the rubber or plastic insert is replaced with optically linear fiberglass. This type of glass would preclude dispersion of the optic signal in other than the desired direction. The LED or detection diode used for the fiber optic link will be mounted directly onto the glass window. Placement of the diode will be made for proper alignment with the mating fiber optic connector pins. Wires will be connected from the diodes to the substrate of the hybrid circuits where active coding or decoding are mounted. The hybrid circuits are in turn mounted on the circuit boards in the module so that the connectors can protrude from the case of the electronic module. The mating connector will have optic fiber sleeves mounted in the insert, rather than wire crimp pins. The fibers will protrude from the end of the sleeve and mate with the glass window aligned with the appropriate diode on the opposite side.

Control augmentation is maintained after a single computer failure. Control augmentation is lost after a second computer failure. This loss does not affect the pilot's direct control link.

2.2.6.3 Passive Sensor Design and Operation. For simplicity of description, a single channel of operation of the dual-passive optical sensor (DPOS) is first provided. Dual-channel operation is then covered.

2.2.6.3.1 Single-Channel Operation. With reference to Figure 30, two optical data lines, corresponding to the source signal from the EM and the return signal to the EM, are connected to the DPOS via two optical connectors. The source signal consists of a 22-nanosecond pulse sequence with a period of one millisecond. The return signal consists of a 10-bit digital word of 330 nanosecond duration, gray-code modulated in accordance with the DPOS shaft position.

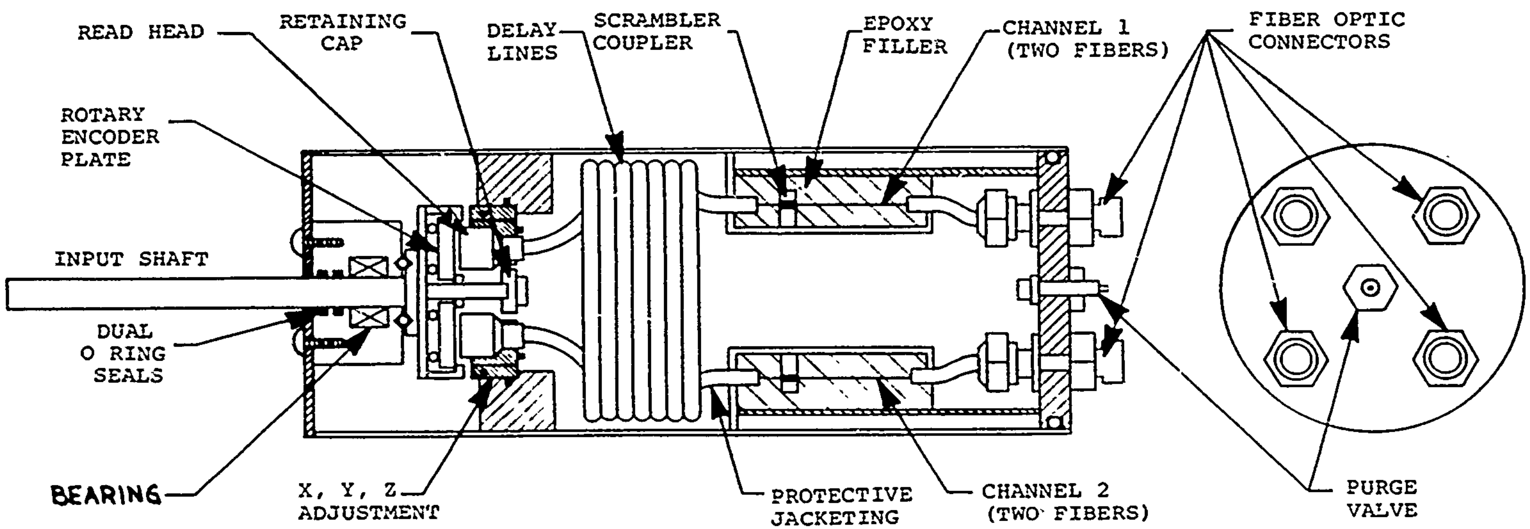
The source signal is optically coupled into ten parallel optical lines, which in turn connect to the read/write head for the encoder disk. Each of the lines has a different length, which results in different time delays in each line relative to the original source pulse. The delays are related by a constant factor, i.e., line 1 has zero nanosecond delay, line 2 has 16.5 nanosecond delay, line 3 has 33 nanosecond delay. Each successive line has an incremental 16.5 nanosecond delay. The tenth line has 148.5 nanosecond delay.

Encoding the incoming signal is achieved by reflecting the source illumination back into the 10 parallel data lines, each causing an additional delay to the reflected pulse. For example, total delay in line 1 is zero, that in line 2 is 33 nanosecond, line 3 delay is 66 nanoseconds, etc. In this manner, the pulse reflected to the tenth line is delayed 297 nanoseconds relative to the leading edge of the source pulse. When the pulses are recombined into a single data line, a 10-bit serial word is obtained. This data word is returned to the EM, decoded, and summed with augmentation and servofeedback signals to form a servoamplifier command.

2.2.6.3.2 Dual Channel Operation. The operation in each channel is identical to that described above, while certain functions are duplicated to provide dual outputs for a single shaft input. Dual elements consist of source/return connectors (4), source/return couplers (2), source/return parallel delay line sets (2), and read/write encoder heads (2).

With the nominal velocity of light of 2×10^8 meters/second in the fiber, an effective delay line length of 59.4 meters is required for the tenth line to achieve the 297 nanosecond maximum delay. As described above, this delay is actually achieved with a 29.7 meter line by using the same delay line for both incoming and reflected pulses. If a 1.7-inch coil diameter is assumed, 219 turns are required for the tenth line delay.

Failure rate of the DPOS is estimated to be 32 per million hours. Weight is estimated to be 2.2 pounds.



Scale ~ 1:1

Figure 30. Cross section of dual channel passive encoder.

2.2.6.4 Multiplex Bus. A multiplex system provides a structural architecture for data transfer between helicopter subsystems. Instead of each signal having its own dedicated communication path, all signals share a common communication link. Each subsystem line replaceable unit (LRU) is either connected directly to a data bus or wired to a multiplex remote terminal unit (MRTU) located nearby. Information from an LRU that is to be sent to other parts of the aircraft is assembled to form one or more digital words. These words are sent along a data bus that connects all the MRTUs and compatible LRUs together. As a result, the bulk of information transfer between LRUs travels over the data bus instead of a large number of individual wires. The high speed of the multiplex system provides for data sorting, storage, and manipulation without degradation of signal quality. Hence, the presence of the multiplex system is 'transparent' to the LRUs it services.

Data transfers between LRUs and MRTUs are controlled by one or more bus controllers. The bus controller is a minicomputer located inside one of the MRTUs. The controller issues a command to an LRU telling it to receive data, send required data back, or transfer data from one LRU to another. No data transfers take place on the data bus except by command of the bus controller.

A valuable feature of a multiplex system is its ability to perform the function of (and eliminate the need for) many subsystem LRUs. This benefit comes about because the bus controller is fast enough to see that all data is sent to and received by the proper LRU with plenty of time remaining to perform numerous computations. In addition, all the information needed to perform the computations is already in the multiplex system, ready for use by the bus controller.

For both the baseline and the fly-by-wire aircraft, the multiplex bus is expected to perform the following functions:

- Control Augmentation
- Stores Management
- Navigation
- Nav/Comm Radio Management
- Fault Detection and Isolation

Integration of the control augmentation functions into the multiplex bus system is readily accomplished, since most of the data required for these functions are already on the bus.

The integration of the flight control system with the multiplex bus during the initial system design ensures that the bus

structure will provide the iteration rate and data sample rate required for the control augmentation functions. This is a particularly important factor in the design of the flight control system, since the iteration rate selected must be high enough to obtain the system bandwidth necessary for the high-frequency, inner loop command augmentation computations.

The multiplex bus system provides a practical means of interconnecting digital systems in an organized and structured manner. The concept was originally conceived to save wiring weight in aircraft. Today, however, multiplexing is recognized as a powerful tool that offers a significant degree of integration and flexibility.

The use of multiplex dictates an integrated approach to avionics system design. Communication between avionics components is established in a centralized manner. Control and computation, on the other hand, may be distributed throughout the system. The distribution need not be functional, but may be topological, as best fits the aircraft configuration. These key considerations lead to significant advantages.

- Crew capabilities are enhanced - the multiplex system allows an integrated approach to control and display. This approach allows maximum use of human factor principles to reduce operator workload.

- Weight is reduced - an original goal of multiplex is achieved in reduced wiring weight and, more importantly, in reduced special-purpose panels and displays.

- Survivability is increased - the combination of centralized communication and distributed control permits the system to automatically reconfigure and select reversionary modes of operation in case of failures or battle damage. In many cases, the reconfiguration can be accomplished without any performance degradation.

- Flexibility is built in - the integrated systems design approach coupled with enforced communication standards will allow new equipment to be installed without major aircraft rewiring.

- Multiplex reduces development risk - the distributed [illegible]or approach to control changes many system para[illegible] from hardware consideration to software.

2.2.6.5 <u>Preflight Test of Main Rotor Controls Using the Multiplex Bus</u>. The multiplex bus can be used to perform preflight checkout of the IAP electronic module. A preflight test

can be broken down into two steps: verification of proper operation of the IAP and verification of proper operation of the IAP monitor functions. Since it is desirable to minimize the amount of hardware built into the IAP electronic module strictly for test purposes, an efficient self-test capability would utilize the built-in IAP monitor circuits wherever possible. The control augmentation inputs, therefore, should be used to verify the correct operation of the IAP monitor functions that, in turn, can be used to verify correct operation of the IAP control circuits.

With the incorporation of a few components into the digital decoder circuits, the IAP electronic module can be configured to accept a self-test command from the computers. This command can be used to inhibit the transmission of light pulses to one of the dual stick sensors, thus causing a mistrack to occur in the IAP computations. The mistrack will, in turn, cause the IAP monitor to indicate a failed condition. This type of check can be performed on both channels of the IAP using the computers. In addition, the computers can command an individual IAP to go into the pressure bypass mode through the transmission of the appropriate commands. Since the IAP status is transmitted back to the computers, a check on the receipt and proper execution of this type of command can be made. The ability of the IAP actuator model to detect improper motion in the actuator can also be checked by the computers. For this test, all the IAPs are commanded to the "on" state with no hydraulic bypass and then each actuator, in turn, is commanded to move in opposition to the remaining four. The resulting force fight should preclude any significant motion of the actuator under test, thus causing the actuator model and monitor circuits to indicate a failure. Again, the actuator status is transmitted back to the SCAS computers for evaluation by the preflight program.

2.2.6.6 EMI/EMP Considerations. The fly-by-wire system utilizes fiber optic transmission for all data paths. The dielectric nature of the light conductors provide distinct advantages in immunity to EMI caused by lightning or inductive fields, RFI caused by radar, and EMP caused by nuclear events. The elimination of all external wires through the use of the IAP package concept has been a driving factor in the design of the system. Nevertheless, the various electronic assemblies themselves are subject to some ECI and RFI effects. The major factors addressed in the study, therefore, were the minimization of crosstalk between internal wires and the use of circuit techniques to minimize the effects of noise.

The internal wiring design minimizes inductive and capacitive crosstalk by:

- Maximizing cable spacing between generator and susceptor circuits. Maximum efforts will be taken to separate digital, analog, power, and discretes.

- Minimizing cable spacing above chassis to enhance capacitance to chassis.

- Routing analog signals as twisted pairs wherever possible.

- Routing all primary and secondary power leads as twisted pairs with their respective returns to reduce H-field radiation.

- Routing all digital signals in a multilayer motherboard and placing a shield layer between digital traces and the analog back plane wire-wrap leads.

- Routing all leads point to point in the wire-wrap plane and keeping returns adjacent to their signal-carrying conductor.

Particular attention should be directed to the layout and partitioning of printed circuit cards. The following measures should be incorporated:

- Minimize trace lengths of high-frequency clocks. Locate high-frequency devices near the PC board connector.

- Employ a complete ground plane and locate high-frequency and power traces adjacent to plane to maximize capacitance to same.

- Provide as many pins as practical between each PC board ground plane and motherboard ground plane for all digital boards. A minimum of two is considered necessary.

- Where analog and digital devices have to be located on the same board, grouping should be employed to minimize interaction. During A/D conversion, if analog and digital references must be commoned, then the given analog signal should be returned through the digital plane. Digital returns should never be referenced to analog. As presently planned, only isolated A/D devices will be employed to prevent common referencing between analog and digital.

3. QUANTITATIVE COMPARISON

In order to provide information necessary for assessment of the desirability of a fly-by-wire/light flight control system in the ASH, the dual mechanical MUT, as configured for the ASH mission, is quantitatively compared with the proposed redundant FBW/L flight control system.

3.1 VULNERABILITY

In the following paragraphs, the vulnerability of the dual mechanical and fly-by-wire/light systems is analyzed.

3.1.1 Threat Definition

The 12.7mm API projectile is a kinetic energy penetrator. It is designed as a hard steel core surrounded by an outer metal jacket. An incendiary mixture is located in the nose of the projectile between the outer jacket and the steel core. The outer jacket is designed to strip away at first impact, igniting the incendiary mixture. The physical characteristics of the intact projectile are:

Length	2.54 inches
Diameter	.51 inch
Weight	745 grains

For this analysis, the 12.7mm API projectile is intact and fully aligned at first impact and for 12 inches after first impact. Impact velocity is 2000 feet per second. Beyond this, the projectile is intact and 30 degrees tumbled (yaw) giving a cutting length of 1.27 inches. This degree of tumble is considered to be realistic for the scope and purposes of this study.

The 23mm HEI-T projectile has four basic kill mechanisms; intact penetration, blast loading, fragment penetration and incendiary fires. Blast loading, fragment penetration and intact projectile penetration are the primary kill mechanisms for flight control components. The projectile is designed such that upon striking a target a fuse is activated, igniting an explosive charge that fragments the outer casing. These fragments and the blast loads from the explosive penetrate the target material.

The ground-to-air 23mm HEI-T projectile has an overall length of approximately 4.3 inches and a diameter of .9 inch. The

casing is made of 48 percent carbon steel with copper rotating bands. The fuse assembly, also made of carbon steel, contains a simple arming device that is actuated by acceleration and centrifugal forces. Also contained within this assembly is a simple self-destruct mechanism that detonates the projectile 6 to 10 seconds after firing. The explosive charge consists of two aluminized RDX pellets that have a combined weight of approximately 204 grains. This explosive has high-brisance characteristics that enhance projectile breakup.

The fragments act as a kill mechanism by impacting the target at high velocity and removing target material. The kinetic energy of the fragment at the time it strikes the target is one measure of its lethality. For projectiles like the 23mm HEI, the fragment mass will vary from less than one grain to over 100 grains. The striking velocity of the fragment on the target material will be the vector sum of the intact projectile's forward velocity, angular rotation at the time of detonation, and the static detonation velocity of the fragment. Fragment velocities from static detonation will range from about 200 to over 3500 feet per second.

Upon detonation of the explosive charge, the high pressure and temperature causes the explosive gases to expand, causing the case to swell until the failure point is reached. The case then fails and fragments are ejected at high velocity. The fragments obtain an initial velocity and form a burst pattern that depends primarily upon the physical shape and velocity of the casing before explosion. The burst pattern of the shell is considered symmetrical about the projectile's longitudinal axis. The total fragment pattern, for representative striking velocities, is contained in a cone forward of the projectile. The half angle of the cone is determined by the resultant fragment velocity. For typical striking velocities of projectiles, such as the 23mm HEI, the fragments will be contained in a cone with a half angle of approximately 50 degrees.

The second major damage mechanism from an HEI projectile comes from the shock or blast wave that propagates from the projectile explosion. At detonation, most of the explosive charge within the projectile is immediately converted to a gaseous form. These gases, at high temperatures and pressure, expand against the casing causing rupture. The gases also compress the surrounding air and then initiate a shock wave that propagates similar to a sound wave, except that the shock wave travels at supersonic velocity. This causes an abrupt increase in pressure, density, temperature and air particle velocity. When the blast wave strikes the target material, it is reflected, setting up another shock wave called the reflected shock wave, whose intensity is greater than the initial

wave. However, the target material will experience only a single shock because the reflected wave is formed instantaneously. Consequently, the value of the overpressure experienced by the target material is generally considered to be entirely reflected over-pressure (Reference 4). The reflected blast wave thus formed is capable of effecting considerable damage on components due to the overpressure. The magnitude of this overpressure decreases as an exponential function of the distance from the center of the explosive charge, and is typically measured in psi over atmospheric pressure as a function of distance from the center of a reference spherical charge.

3.1.2 Methodology

The candidate flight control systems are assessed for their vulnerability to the 12.7mm API and 23mm HEI-T threats. A quantitative analysis is performed on both flight control systems for the 12.7mm API threat; however, the scope of this analysis allows only a qualitative evaluation of the 23mm HEI-T threat. The vulnerability analysis first requires determination of those components within the flight control systems that are critical to the systems operation. This is a function of the damage states being analyzed, threat characteristics, and the operating condition of the helicopter. These components are classified as being singly vulnerable or multiple vulnerable to the threat. That is, the component is singly vulnerable if it can be disabled by a single projectile hit. An engine on a single-engine aircraft is an example of a component in this category. The multiple vulnerable class of components requires that a single projectile disable more than one component in order to render the system inoperable. The engines on a twin-engine aircraft are an example of this situation.

The major components in the flight control systems that are determined to be critical are listed as follows:

- Push-pull tubes and connectors
- Bellcranks, idlers, and rocker arms
- Mechanical mixers
- Actuators

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[4]Walther, Robert E., et al., AIRCRAFT VULNERABILITY, VOLUME V - FLIGHT CONTROL AND HYDRAULIC SYSTEMS, Ballistic Research Laboratory TR-02061, Aberdeen, Maryland, May 1978, Confidential.

- Jackshafts
- Transducers

These components may be either singly or multiply vulnerable depending on the system in which they are used.

The results are reported as a difference in the averaged vulnerability of the five viewing aspects of the fly-by-wire/light control system and the dual mechanical control system to prevent the report from containing classified data.

Within the areas where the components of the flight control systems are redundant, the analysis was concerned with two items. First, what is the probability that the multiple vulnerable components will be made inoperable from a single hit? Second, what is the probability that the system would fail due to a projectile jamming a component? This second case is of special interest to the dual mechanical control system.

To determine the first part, References 4 and 5 are used to obtain the required separation distance to minimize the possibility that redundant tubes would be rendered inoperable by a single projectile. In this analysis, Reference 6 is used to show that for typical boosted compression loads, flight control tubes can withstand a cut across the tube that causes loss of 61 percent of its circumference. In some cases, the tubes could take as high as 75 percent circumference loss and survive. Using this information and assuming an average tube diameter of 1.125 inches, the probability of a kill, given a hit, was determined to be .1 for a separation distance of 2.8 inches and .03 for 7.6 inches separation.

For the second part, each flight control system is checked for antijam devices located with the redundant paths. The absence of such devices would make the redundant components within that portion of the system singly vulnerable to jamming. Their condition probability of kill would be based on the probability that they may cause the system to jam.

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[5]Foulk, James B., A MEASURE OF THE EFFECT OF SEPARATION OF PAIRED REDUNDANT COMPONENTS ON THE VULNERABILITY OF AIRCRAFT TARGETS TO SINGLE PENETRATORS, Ballistic Research Laboratory, Report Number 229, Aberdeen Maryland, May 1974.

[6]Crist, J. David, and Blaser, Allen N., HIGH SURVIVABILITY FLIGHT CONTROLS DESIGN, VOLUME I - AH-1S HELICOPTER BALLISTIC-DAMAGE-TOLERANT SUBSYSTEMS PRELIMINARY DESIGN, USARTL TR-77-49A, Applied Technology Laboratory, U.S. Army Research and Technology Laboratories (AVRADCOM), Fort Eustis, Virginia, February 1979, ADBO36946L.

Singly vulnerable kill probabilities are estimated using:

- Data available from the U.S. Army (Fort Eustis VAT team and/or Ballistic Research Laboratories)

- Results from previous related contracts

- BHT, Sperry, and HRT analytical and empirical techniques

Once the conditional probabilities are quantified, the presented area of each component from a particular attack direction is determined. During this process of obtaining presented areas, it is necessary to consider the space orientation of the individual components and to evaluate any masking or shielding from major components onboard the aircraft.

The vulnerable area of a component can be interpreted as an equivalent area which, if subjected to the threat damage mechanism(s), results in a specified level of damage. More explicitly, the vulnerable area of a component is a weighted value of the presented area projected onto a plane normal to the trajectory of the projectile, where the weighting factor is the probability of a random hit defeating (killing) the component. For a given threat, striking velocity, damage state, and viewing aspect, the vulnerable area of a component can be calculated by:

$$A_V = A_p \cdot P_{k/h}$$

where

A_V = the vulnerable area of the component

A_p = the presented area of the component

$P_{k/h}$ = the conditional probability of damaging the component, given a hit

Vulnerable area values are computed for the front, rear, bottom, left and right side viewing aspects and then averaged.

This procedure is used to estimate the single-hit vulnerable areas of the fixed control systems for both the dual mechanical and dual fly-by-wire/light designs. Both systems are evaluated for attrition kill of the helicopter while in hover flight and 2/3 fuel capacity.

Typically, vulnerable area analysis for the high explosive projectiles are determined by mapping regions about the component where it is vulnerable. For selected viewing aspects,

these regions project areas that are usually larger than the component's presented area for that aspect. Development of this analysis is a task that is beyond the scope of this report. Thus, a qualitative analysis of the above described components are performed for the 23mm HEI-T threat.

3.1.3 Comparison of Dual Mechanical and Fly-By-Wire/Light

Both the dual mechanical and the dual fly-by-wire/light flight control systems have low vulnerable areas to the 12.7mm API threat for a helicopter in hover flight. The fly-by-wire/light control system has a five-aspect averaged vulnerable area that is .26 square foot less than the five-aspect averaged vulnerable area for the dual mechanical control system. This difference is primarily from the vulnerability of the main rotor actuators within the two systems.

To summarize, the dual mechanical flight control system has excellent survivability to the 12.7mm API projectile and good 23mm HEI-T threat survivability. The fly-by-wire/light control system also has excellent survivability to the 12.7mm API projectile (slightly better than the dual mechanical system), and good 23mm HEI-T threat survivability.*

3.2 CONTROL SYSTEM WEIGHT

The preliminary design effort resulted in a definition of the dual mechanical and fly-by-wire/light control systems. Figures 5 and 13 identify the various cranks, tubes, actuator, support, etc., of the dual mechanical system. These were tabulated and compared with similar parts from a comparable BHT production helicopter for weight estimation.

3.2.1 Methodology and Assumptions

To meet the ASH mission requirements, more electronics and control augmentation capability were added to the basic MUT control system.

The fly-by-wire concept required the modification of the main rotor transmission to permit the installation of five IAPs, along with their dedicated power supplies. Two additional horns for the nonrotating swashplate ring were also required to bring that total to five.

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*If more detailed vulnerability analysis information is required, qualified requestors may obtain the information from Bell Helicopter Textron with the approval of the Applied Technology Laboratory Security Manager.

Nonrotating control linkage between the cockpit and the main and tail rotors was removed. This was performed with no weight advantage being assessed for reduction of airframe structural stiffness that is no longer necessary for control backup.

The IAP weight was furnished by Hydraulics Research Textron. The dedicated power supplies contain off-the-shelf Abex hydraulic pumps and HTL Electro-Kinetics alternator components. Mission electronics weight is furnished by Sperry Flight Systems. Transmission drawing, Figure 9, shows the modification necessary to adapt to FBW. System ground check and other hydraulic plumbing required is shown in Figure 18.

The basic MUT dual mechanical system utilizes a fly-by-wire horizontal stabilizer that weighs 19 pounds. BHT uses a similar system on their Model 214ST and estimates its weight for the MUT to be essentially the same.

The tail rotor FBW arrangement is shown in Figure 10 with a total system weight of 28.4 pounds. This is apportioned to 17.4 pounds at the tail rotor, 6.5 pounds of wiring, and the remainder in the cockpit for sensors and control electronics.

3.2.2 Analysis

Conventional sticks, dual linkage including rotating controls, and antijam devices in the dual mechanical control system are 136 pounds heavier than the conventional sticks and rotating controls of the fly-by-wire system. An additional 15 to 20 pounds not counted here could be saved by implementing side-arm controllers in the cockpit.

The hydraulic actuators for both systems are designed for jam tolerance. A weight increase of only 27.5 pounds is required to replace four nonintegrated actuators in the dual mechanical system, with five integrated actuators in the FBW system.

The hydraulic power supply for the dual mechanical system consists of two independent expanded systems complete with separated pump cooler units, pump drives, reservoirs, filters, modules, manifolds, accumulators, switching and isolation values, and plumbing and associated hardware. The FBW hydraulic system, on the other hand, is integrated with the actuator that is located next to its dedicated pump. Components allocated to hydraulic supply are listed in Table 11. The integrated concept reduces overall hydraulic system weight by 56 pounds.

Both the dual mechanical and fly-by-wire control systems require electromechanical actuators for the autopilot and horizontal stabilizer. Only the dual mechanical controls use electromechanical actuators for control augmentation. Only the FBW control system uses electromechanical actuators for tail rotor control. In this category, the fly-by-wire system is 1.4 pounds heavier.

The FBW system uses a dedicated electrical alternator for each control channel of the main rotor. This power is in addition to the ship's normal electrical supply system. In addition, both the dual mechanical and the FBW systems have electrical supply processing equipment that rectifies, regulates, and/or interfaces the various electronic systems with their power supply source. These are listed and compared in Table 11. The FBW is 16.5 pounds heavier in this category than the dual mechanical system.

In the category of sensor and control signal electronics, the FBW is only 4.5 pounds heavier than the dual mechanical system. Part of this is allocated to the fact that separate SCAS electronics was used for the dual mechanical, whereas the function was incorporated into the computer system for the FBW system.

3.2.3 Weight Comparison of Dual Mechanical and FBW Systems

Table 11 lists the tabulated comparison of the control system weight for the dual mechanical system and the FBW system. From the table, it is seen that the FBW/L system is 143 pounds lighter than the dual mechanical control system. It is noted that conventional cockpit sticks are assumed here. Additional weight can be saved by adopting advanced cockpit controls. These are discussed in Section 5.

3.3 LIFE-CYCLE COSTS

Complete life-cycle cost analyses include every cost item associated with the system or subsystem under consideration. The costs are generally broken into the following categories:

- RDT&E
- Nonrecurring Initial Investment
- Recurring Initial Investment
- Operations and Maintenance
- Combat

TABLE 11. CONTROL SYSTEM WEIGHT COMPARISON

Category	Dual Mechanical (pounds)	FBW/L (pounds)
Sticks/linkages		
• Main Rotor Rotating	178.0	178.0
• Tail Rotor Rotating	12.5	12.5
• Nonrotating	231.2	95.0
Subtotal	421.7	285.5
Actuators, Hydraulic		
• Main Rotor	75.0	112.5
• Tail Rotor	10.0	0.0
Subtotal	85.0	112.5
Power Supply, Hydraulic		
• Pumps, misc.	131.0	37.8
• Mechanical drives for FBW/L	0.0	36.9
Subtotal	131.0	74.8
Actuators, Electromech.		
• SCAS	16.0	0.0
• Elevator	14.0	14.0
• Tail Rotor	0.0	17.4
• Autopilot	8.0	8.0
Subtotal	38.0	39.4
Electrical Cables, Hardware, & Alternators	16.0	32.5
Sensors, Signal Processing Components	84.4	88.9
Total	776.1	633.5

3.3.1 Methodology

The methodology used to provide a comparison between the FBW and dual mechanical flight control systems centers on identifying the major cost items in which differences are expected to occur. This system results in a cost difference but does not provide an estimate of the total life-cycle cost for each system. The method was chosen because of lack of detail in the description of the two systems and the means of developing them.

3.3.1.1 RDT&E. This cost category includes all the cost for research, engineering, design, analysis, development and testing related to the total system. Normally, these costs occur during the Advanced Development and Engineering Development Phase of R&D. For purposes of this analysis, it was assumed that both control systems would be through this phase of development before they were considered for incorporation into an Advanced Scout Helicopter.

3.3.1.2 Nonrecurring Initial Investment. This category includes all nonrecurring engineering design, tooling, production engineering, vendor development and qualification, and flight test. Only the costs directly associated with the flight control systems were included in this figure. Estimates were made by BHT pricing personnel and pricing personnel from the various vendors. The estimates are considered to be the best available for the level of engineering description at this time.

Each cost estimate was derived by describing the subtasks required and submitting them to the responsible groups for manpower and material estimates. These estimates were then priced in constant FY 1979 dollars. Comparisons to previous and existing programs were used to add greater validity to the estimates. Table 12 shows the different elements of the nonrecurring initial investment cost for the fly-by-wire and the dual mechanical control systems. The dual mechanical is 26 percent more costly.

3.3.1.3 Recurring Initial Investment. The recurring initial investment is divided into systems' parts and assembly cost. Table 13 shows the systems' parts cost for the major subsystems for each flight control system. A total buy of 1450 units was assumed. Actual quotes were obtained from vendors for the major items and BHT estimates were used for BHT produced items and smaller vendor items.

Table 13 combines the systems' parts and assembly cost for total recurring initial investment cost. The FBW system shows a 12-percent advantage in this category.

TABLE 12. NONRECURRING INITIAL INVESTMENT COST

Cost category	FBW	Dual mech
Nonrecurring Engineering Design	$3,964,225	$3,639,157
Nonrecurring Tooling and Production Engineering	130,064	3,415,580
Nonrecurring Vendor Development and Qualification	1,897,848	1,095,646
Flight Test	489,143	286,735
Total Nonrecurring Initial Investment	$6,481,280	$8,150,383

TABLE 13. RECURRING INITIAL INVESTMENT COST

Cost category	FBW	Dual mech
System Parts Cost*	$199,947	$219,970
Assembly Cost	3,148	6,952
Total Recurring Initial Investment per Aircraft	$203,095	$226,922

*Based on 1450 Units

3.3.1.4 Operations and Maintenance (O&M). The O&M costs that were identified as major cost contributors that would have significant differences are corrective maintenance labor cost and corrective maintenance parts cost. The organizational and intermediate maintenance man-hours per flight hour, as shown in Section 3.6, were used to determine labor costs shown in Table 14. The FBW system shows a 68-percent advantage in this area. The parts consumption was determined using the mean-time-between-failures (MTBF) values found in Section 3.5. The parts costs are shown in Table 15. The FBW system has a total O&M advantage of 35 percent.

TABLE 14. CORRECTIVE MAINTENANCE COST (DOLLARS PER FLIGHT HOUR)*

Subsystem	FBW		Dual mech	
	Labor	Parts	Labor	Parts
Stick, Cranks and Linkages	1.82	1.66	5.39	2.46
Hydraulic Actuators	1.50	9.45	2.59	14.73
Electromechanical Actuators	0.07	0.19	0.28	0.57
Hydraulics System	0.58	2.78	1.40	3.38
Electrical Supplies	3.94	6.86	3.85	6.75
Sensors and Signal Processors	0.12	4.11	0.28	2.95
Subtotal	$ 8.03	$25.05	$13.79	$30.81
Total Parts and Labor	$33.08		$44.60	

*Based on 40 flight hours per month for 20 years

TABLE 15. SYSTEM PARTS COST*

Subsystem	FBW	Dual mech
Sticks, Cranks and Linkages	$ 23,616	$ 43,143
Hydraulic Actuators/Packages	68,852	66,564
Electromechanical Actuators	9,903	25,204
Hydraulic Systems	26,162	24,076
Electrical Supplies	5,187	4,031
Sensors and Signal Processors	66,227	56,952
Total	$199,947	$219,970

*Based on 1450 units

3.3.1.5 Life-Cycle Cost Comparison. The total program life-cycle cost for each system is shown in Table 16. These figures are qualified by the notes shown at the bottom of the table and by the fact that all cost categories were not included in the analysis. The total program life-cycle cost advantage of the FBW system is $196 million in constant FY 1979 dollars.

TABLE 16. FLIGHT CONTROL LIFE-CYCLE COST (TOTAL PROGRAM)

Cost category	FBW	Dual mech
RDT&E*	0	0
Nonrecurring Initial Investment	6,481,280	8,150,383
Recurring Initial Investment**	294,690,845	329,036,900
Corrective Maintenance***	460,477,012	620,832,000
	$761,649,137	$958,019,283

*See Section 3.3.1.1
**Based on 1450 units
***Based on 40 flight hours per month for 20 years

Dollar inflation will alter and magnify this differential cost over the life of the program. Assuming fleet service initiation in 1983 and an inflation rate, for example, of 10 percent per annum, then the total program life-cycle cost advantage of the FBW system increases to $810 million.

3.4 POWER SUPPLY REQUIREMENTS

In order to provide continued operational capability after various possible system failures, redundancy must be provided not only in the signal transmission paths, but also in the electrical and hydraulic power supplies. For two-fail-operate capability, a FBW/L system must have at least three separate electrical and three separate hydraulic supplies.

The command augmentation system must function in order to complete the ASH mission. This requires that the electrical supply for the dual mechanical control system to be at least fail-operate. This is accomplished by the use of two AC generators and transformer rectifiers coupled through automatic switchover circuitry. For the FBW/L system, the electrical power supply is expanded to include five dedicated alternators to power the STAR main rotor control system. This electrical supply configuration is depicted in Figure 20.

Since the helicopter is uncontrollable without hydraulic pressure, redundancy of hydraulic pumps is provided. The dual mechanical mechanization utilizes two independent hydraulic systems, with an electrically powered auxiliary system. These are discussed in Section 2.1.1. One system is used to pressurize one of the two pistons of each dual boost actuator; the other system is used to pressurize the other piston. If both systems fail simultaneously, the auxiliary system is hydraulically switched to provide pressure. Of concern, however, is the possibility that the switchover logic may itself fail and result in a "domino" failure effect.

The FBW/L control system considered in this study provides a unique approach toward the hydraulic supply redundancy. The system utilizes electromechanical actuation for horizontal stabilizer and tail rotor control; thus, no hydraulics are needed there. However, hydraulic actuation is utilized for main rotor control. The main rotor STAR control utilizes five IAPs, five small dedicated hydraulic pumps, and one electrically driven auxiliary hydraulic pump that is used only for fill-and-bleed procedures and for ground check of the FBW/L system without the engine running. With this arrangement, no hydraulic switchover is necessary in the redundancy management concept.

Components of the dual mechanical and FBW/L system are tabulated in Table 17.

3.5 RELIABILITY

3.5.1 Methodology and Definitions

3.5.1.1 Methodology. An analysis was performed to compare a dual mechanical flight control system to the BHT proposed ASH FBW configuration and to customer-supplied goals. The analysis was made to determine whether the ASH Helicopter system, mission, or flight safety reliability would be improved or degraded as a result of using a FBW configuration rather than a conventional dual mechanical configuration.

TABLE 17. POWER SUPPLY CONFIGURATION

Dual mechanical	Qty	Fly-By-Wire/Light	Qty
AC Generator/Controller	(2)	AC Generator/Controller	(2)
Transformer/Rectifiers	(2)	Transformer/Rectifiers	(2)
Battery	(1)	Battery	(1)
Switchover Circuitry for Elevator and Command Augmentation		Alternator, IAP Dedicated	(5)
		Switchover Circuitry Elevator, Tail Rotor, and Command Augmentation	
Hydraulic Pump/Cooler	(2)		
Auxiliary Pump, Electrically Driven	(1)	Dedicated Hydraulic Pump	(5)
Accumulators	(3)	Auxiliary Pump, Electrically Driven	(1)

The reliability analysis covers the following:

- An estimate of the system reliability

    System reliability

    50 percent of the system reliability

    200 percent of the system reliability

- An estimate of the mission reliability

    Mission reliability

    50 percent of the mission reliability

    200 percent of the mission reliability

- An estimate of the flight safety reliability

    Flight safety reliability

    50 percent of the flight safety reliability

    200 percent of the flight safety reliability

The data sources used for the analysis were:

- Navy Maintenance and Material Management (3-M) data for Models UH-1N and AH-1J

- U.S. Air Force AFM 66-1 data for Model UH-1N

- BHT field service reports for Models 214A and 222

- U.S. Army Safety Center (USASC) for Model UH-1H

- MIL-HDBK-217B, Section 3

3.5.1.2 <u>Definitions</u>. The following definitions were used to perform the analysis.

- Failure - The inability of an item to perform within its specified limits. For those items subject to progressive deterioration (i.e., corrosion, leaks), a failure will be considered to have occurred when the condition:

  Creates a flight hazard

  Can reasonably be expected to abort a mission

  Will require repair or replacement before flying another mission

- Inherent Failure - A failure caused by a physical condition or phenomenon internal to the failed item.

- Independent Failures - Those failures that occur or can occur without being related to the malfunctioning of associated items.

- Time (for reliability values in the analysis) - flight hours measured from aircraft lift-off until aircraft touchdown.

- Mean-Time-Between-Failures (MTBF) - The average operational flight hours between independent failures.

- Mission - A time period measured from aircraft lift-off until aircraft touchdown. A mission starts only after preflight checkout has been completed and the system is determined to be operationally ready.

- System Reliability - The probability that an operationally ready, mission-configured system of the aircraft will complete a one-hour mission without a failure that would require corrective maintenance.

- Mission Reliability - The probability that an operationally ready, mission-configured system of the aircraft will perform all mission-necessary functions successfully during a one-hour mission.

- Flight Safety Reliability - The probability that an operationally ready, mission-configured system of the aircraft will operate for a one-hour mission without the occurrence of an in-flight failure that would result in injury to the crew (that would preclude them from performing their mission task) or which would prevent performance of a controlled landing.

3.5.2 Reliability Goals

The specified reliability goals for the flight control system (including electrical and hydraulic power supplies) are defined below. The goals assume a constant hazard function and a mature system.

- System failure (λ_s) goal of one failure per 2500 hours of operation

$$\begin{aligned} \lambda_s &= 1/\text{MTBF} \\ &= 1/2500 \text{ hours} \\ &= 0.000400 \text{ failure per flight hour} \end{aligned}$$

- Mission failure (λ_m) goal of one failure per 10,000 hours of operation

$$\begin{aligned} \lambda_m &= 1/\text{MTBF} \\ &= 1/10{,}000 \text{ hours} \\ &= .000100 \text{ failure per flight hour} \end{aligned}$$

- Flight safety (λ_c) failure goal of one failure per 10,000,000 hours of operation

$$\begin{aligned} \lambda_c &= 1/\text{MTBF} \\ &= 1/10{,}000{,}000 \text{ hours} \\ &= .0000001 \text{ failure per flight hour} \end{aligned}$$

3.5.3 Analysis

3.5.3.1 System Reliability Failure Rates. The system failure rate analysis was performed using the inherent failure rates from the data sources of Paragraph 3.5.1.1. A failure rate was predicted for each component in the flight controls, electrical, and hydraulic systems. The failure rates were based on the historical data of the same component or a similar component from one of the data sources. These data were used to obtain the system failure rates and, with modification, the 50- and the 200-percent system failure rates.

The system block diagrams shown in Figures 31 and 32 were used in the analysis procedure. Figure 31 is the dual mechanical system block diagram. Figure 32 is the FBW system block diagram. Since the failure of any component will create unscheduled maintenance, the various components are considered (for system reliability purposes) to be in series and independent. The failure rate of the system λ_s was thus computed as:

$$\lambda_s = \sum_{i=1}^{n} \lambda_i \qquad (1)$$

where,

λ_i = failure rate (failures per flight hour) of each ith system element

n = number of elements in the system

The results of the system reliability analysis are shown in Tables 18 and 19. Table 18 presents the mechanical system and the FBW system predicted reliability for each major system/component grouping and the allocated system reliability goal for the total flight control system. Table 19 presents the system failure rates for each component of the mechanical system and the FBW system. The dual mechanical flight control system has a predicted system failure rate of 0.039759 failure per flight hour. The FBW flight control system has a predicted system failure rate of 0.030914 failure per flight hour. The FBW MTBF is thus better than the mechanical MTBF by 28 percent. Both systems are, however, worse than the specified goal by about two orders of magnitude.

The results of the system reliability analysis using 50 percent component failure rate values are shown in Table 20. The table

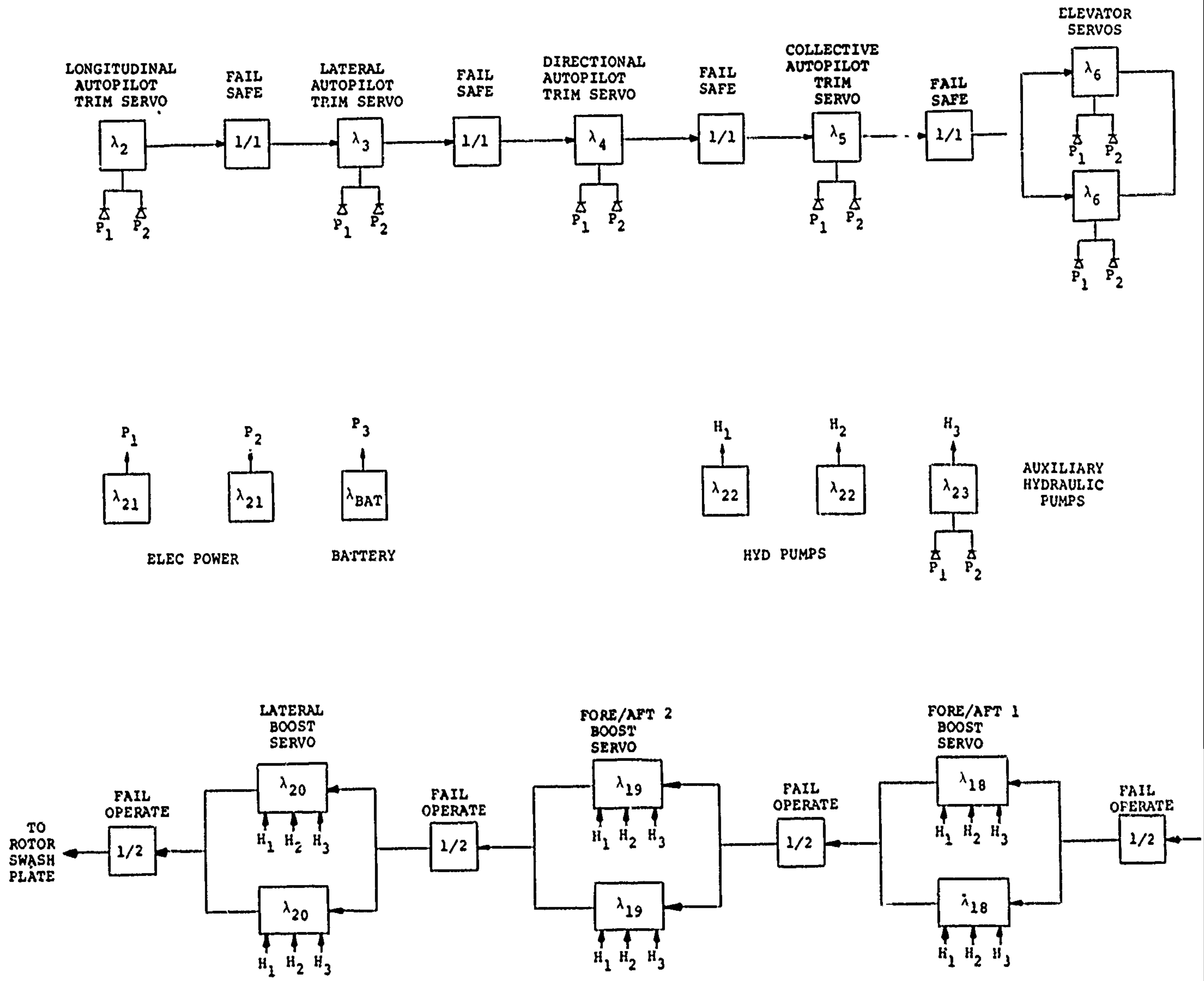
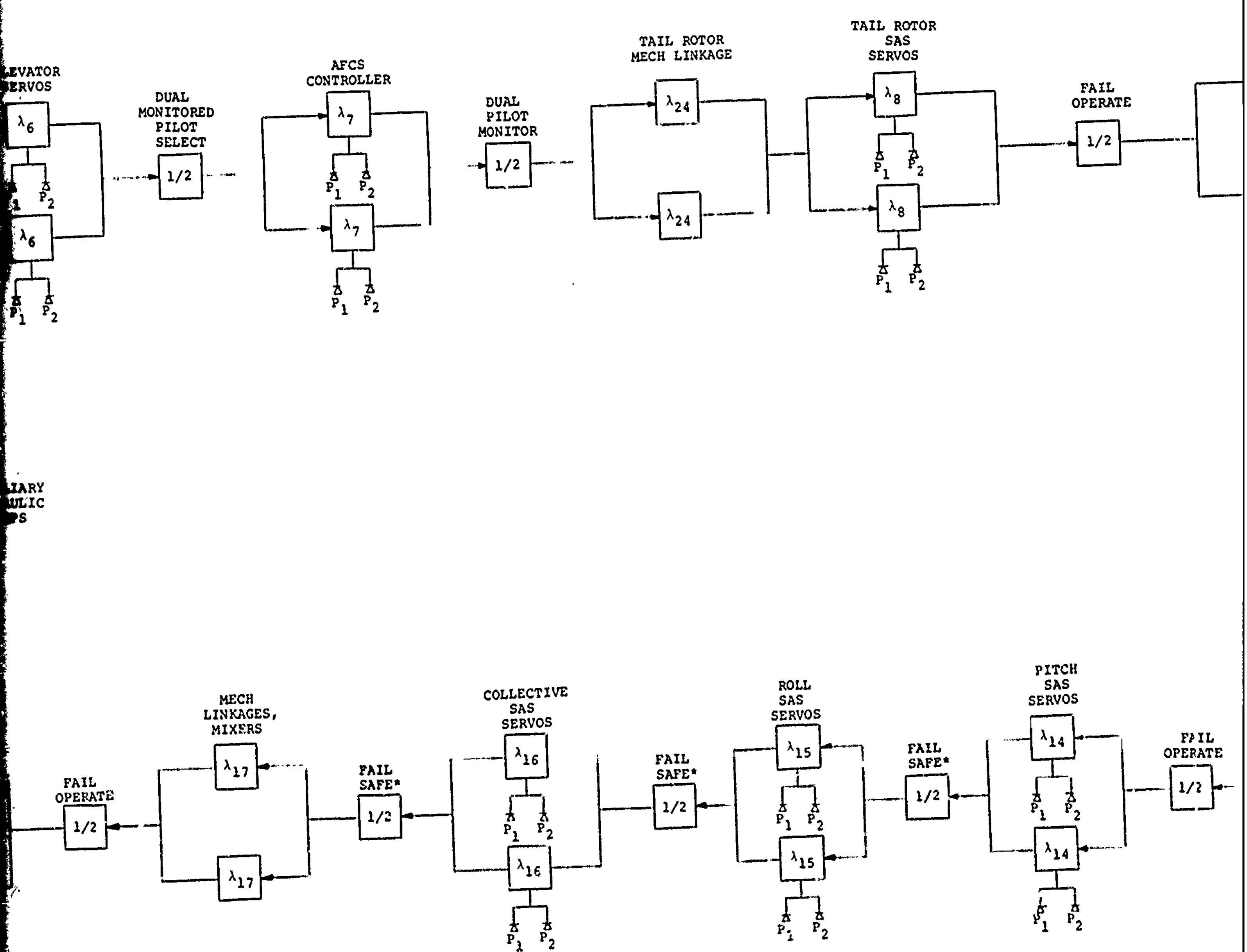
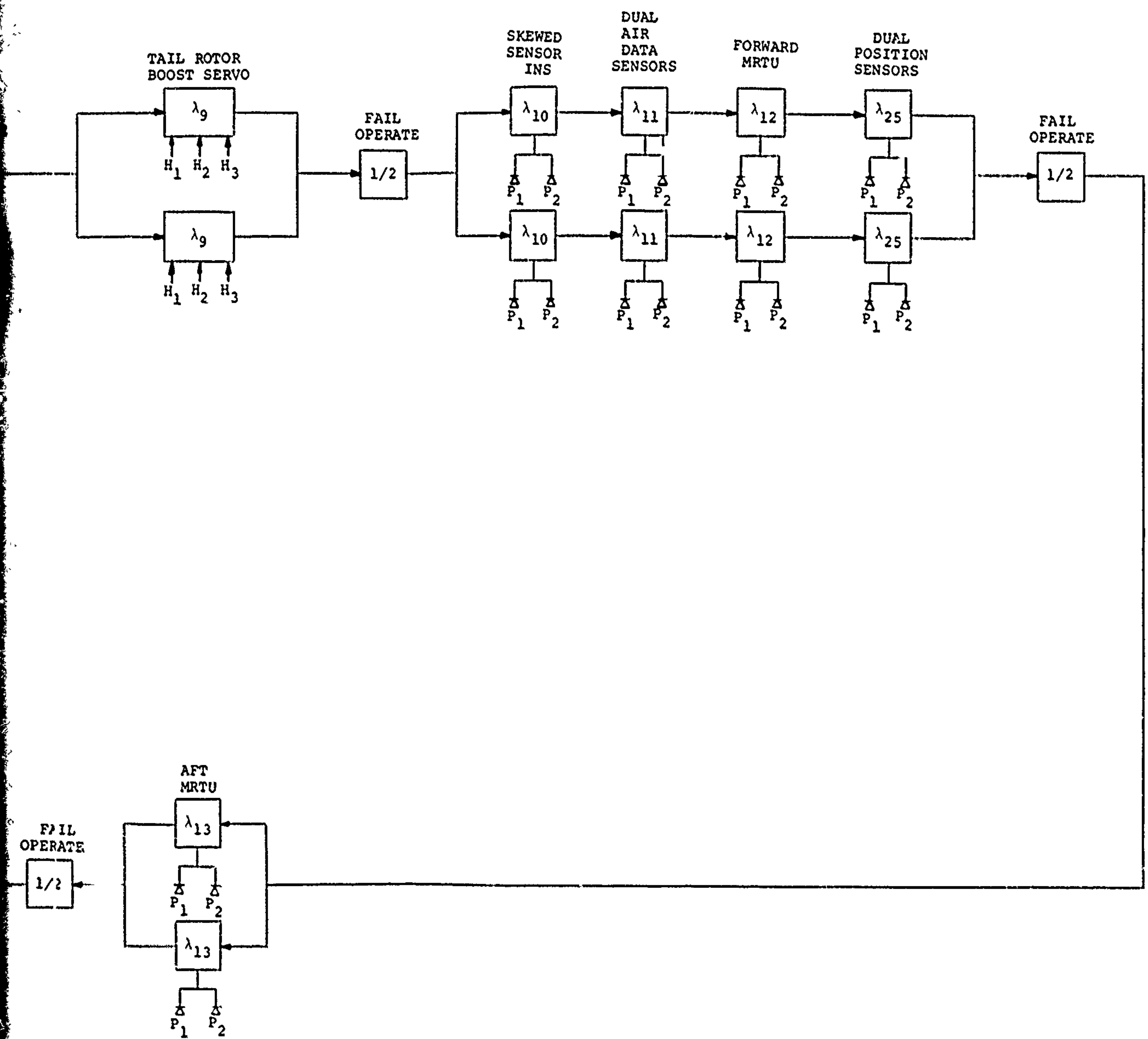


Figure 31. Dual mechanical system reliability block diagram.



* PILOT RETRIM, RESET



TAIL ROTOR BOOST SERVO
λ_9
H_1 H_2 H_3
FAIL OPERATE
1/2
SKEWED SENSOR INS
λ_{10}
DUAL AIR DATA SENSORS
λ_{11}
FORWARD MRTU
λ_{12}
DUAL POSITION SENSORS
λ_{25}
P_1 P_2
FAIL OPERATE
1/2
AFT MRTU
λ_{13}
FAIL OPERATE
1/2

2

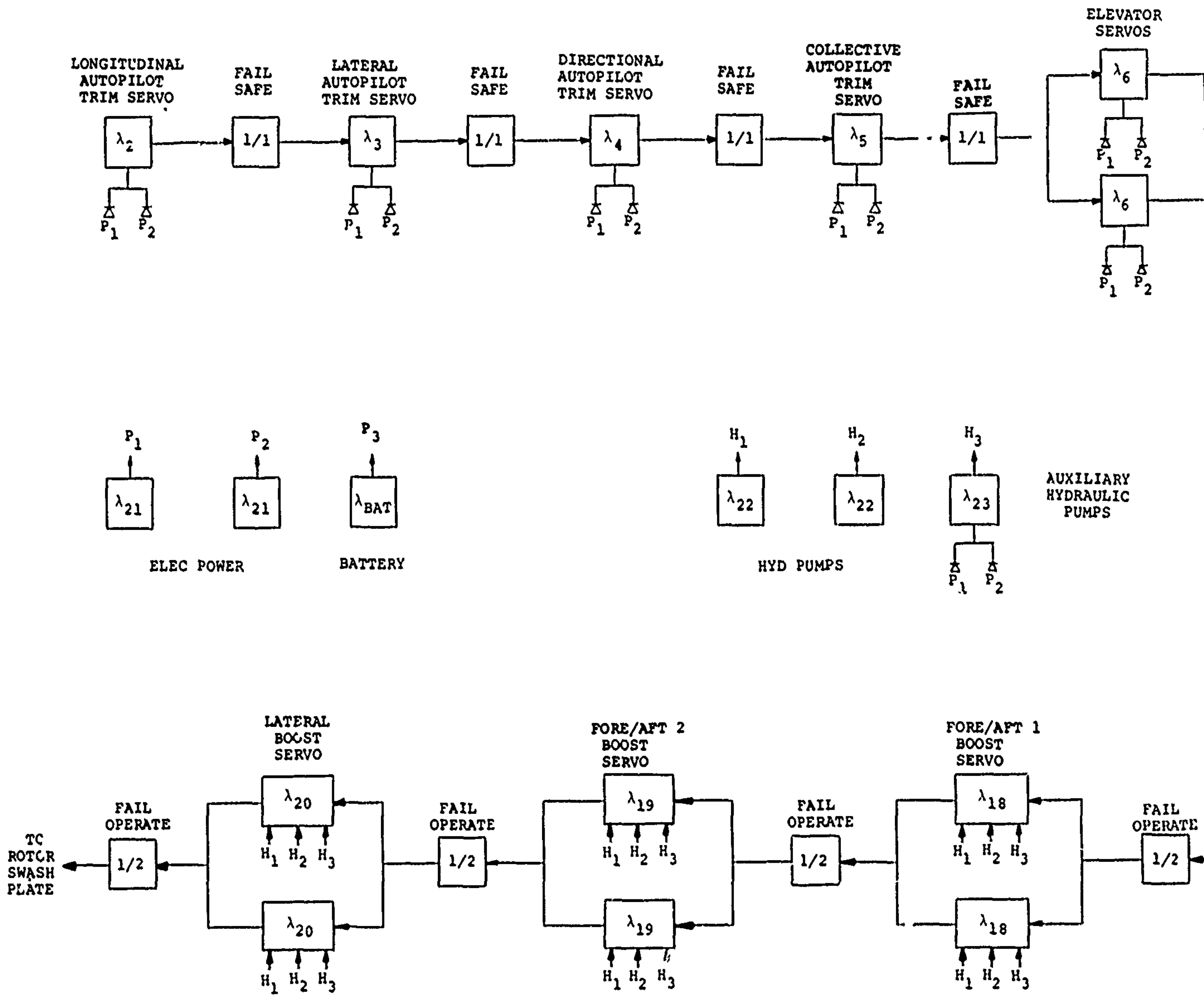
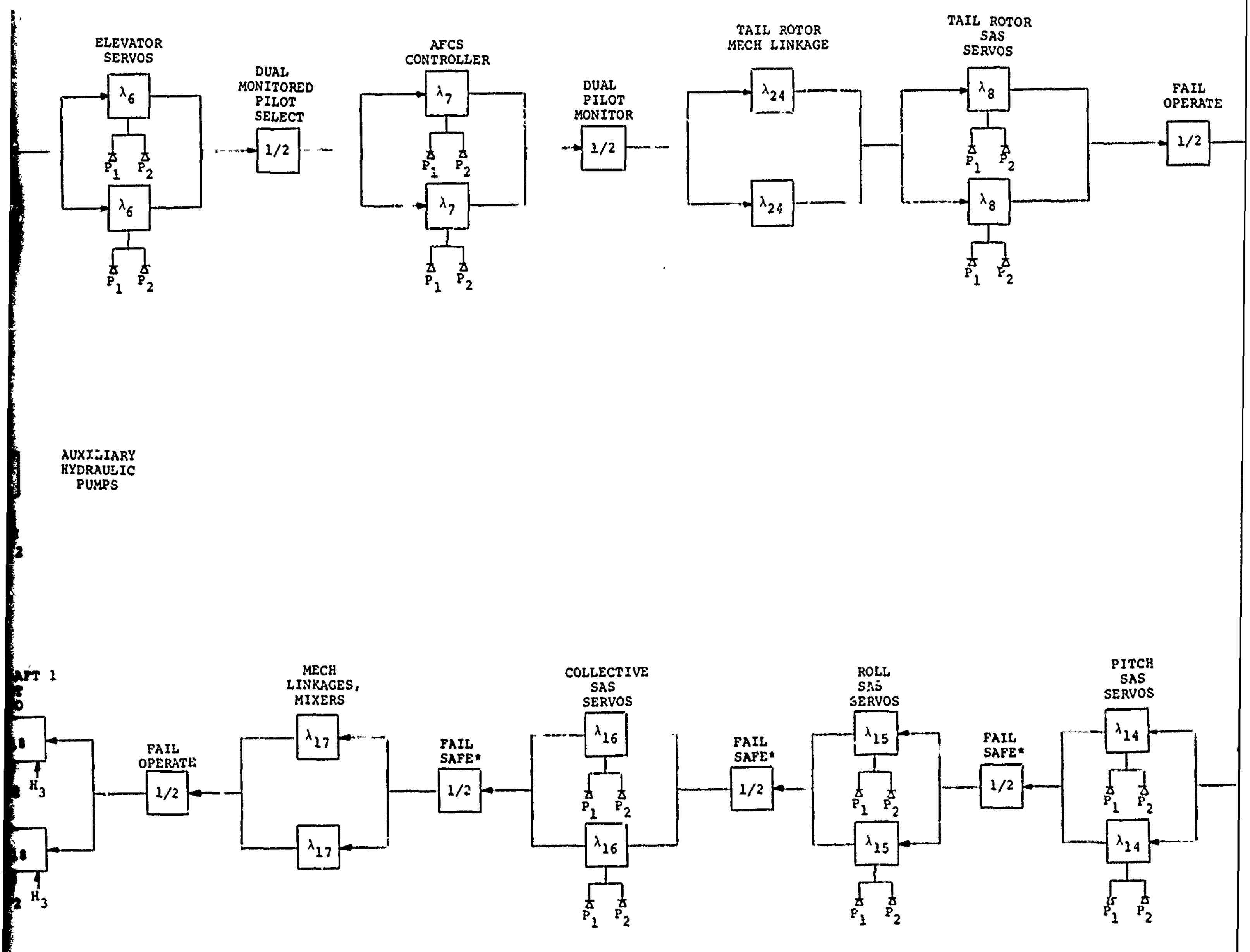
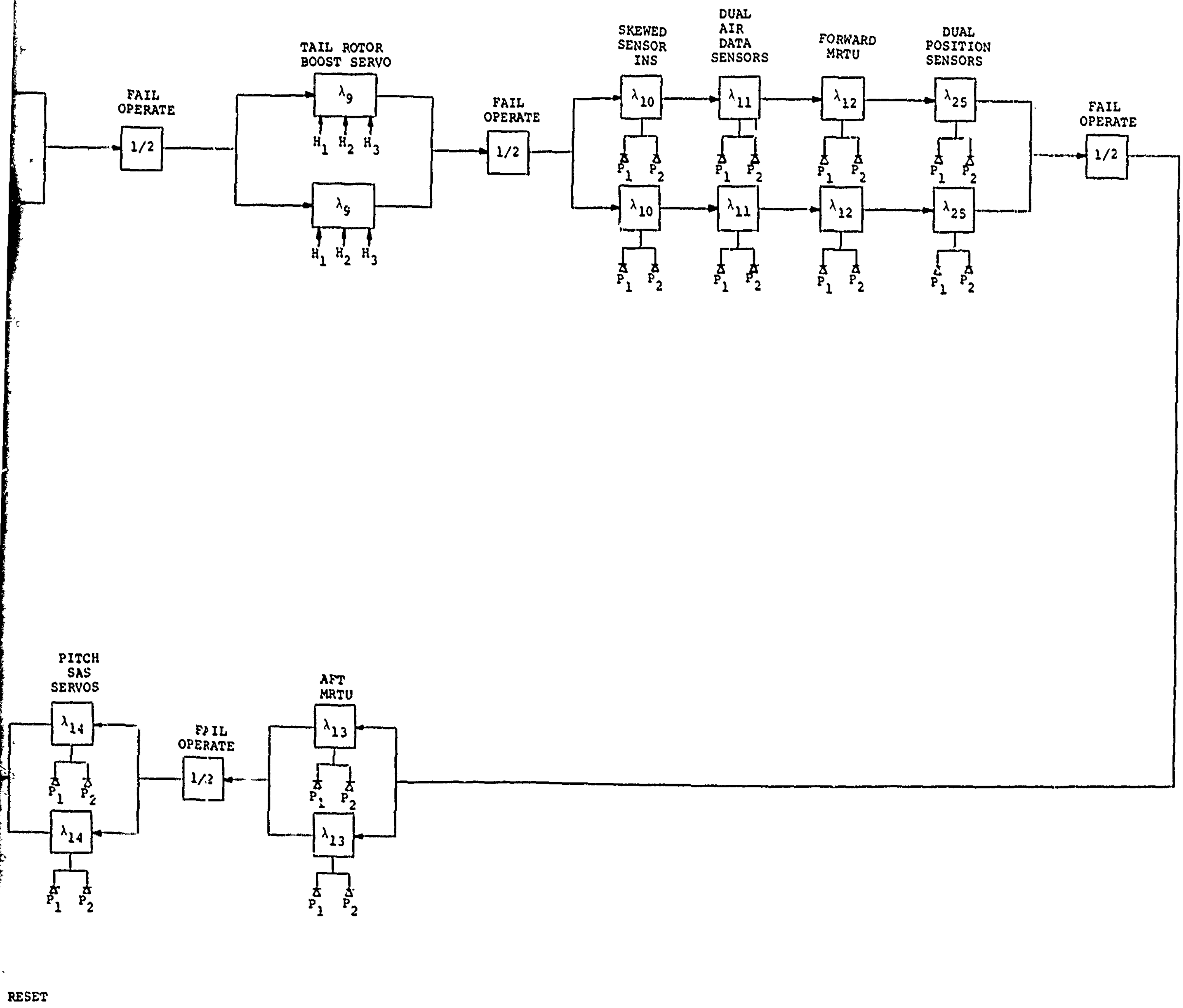


Figure 31. Dual mechanical system reliability block diagram.


ELEVATOR SERVOS
λ_6
λ_6
P_1 P_2
DUAL MONITORED PILOT SELECT
1/2
AFCS CONTROLLER
λ_7
λ_7
DUAL PILOT MONITOR
1/2
TAIL ROTOR MECH LINKAGE
λ_{24}
λ_{24}
TAIL ROTOR SAS SERVOS
λ_8
λ_8
FAIL OPERATE
1/2
AUXILIARY HYDRAULIC PUMPS
APT 1
H_3
FAIL OPERATE
1/2
MECH LINKAGES, MIXERS
λ_{17}
λ_{17}
FAIL SAFE*
1/2
COLLECTIVE SAS SERVOS
λ_{16}
λ_{16}
FAIL SAFE*
1/2
ROLL SAS SERVOS
λ_{15}
λ_{15}
FAIL SAFE*
1/2
PITCH SAS SERVOS
λ_{14}
λ_{14}
* PILOT RETRIM, RESET

m.



FAIL OPERATE
1/2
TAIL ROTOR BOOST SERVO
λ_9
H_1 H_2 H_3
SKEWED SENSOR INS
λ_{10}
DUAL AIR DATA SENSORS
λ_{11}
FORWARD MRTU
λ_{12}
DUAL POSITION SENSORS
λ_{25}
P_1 P_2
PITCH SAS SERVOS
λ_{14}
AFT MRTU
λ_{13}
RESET

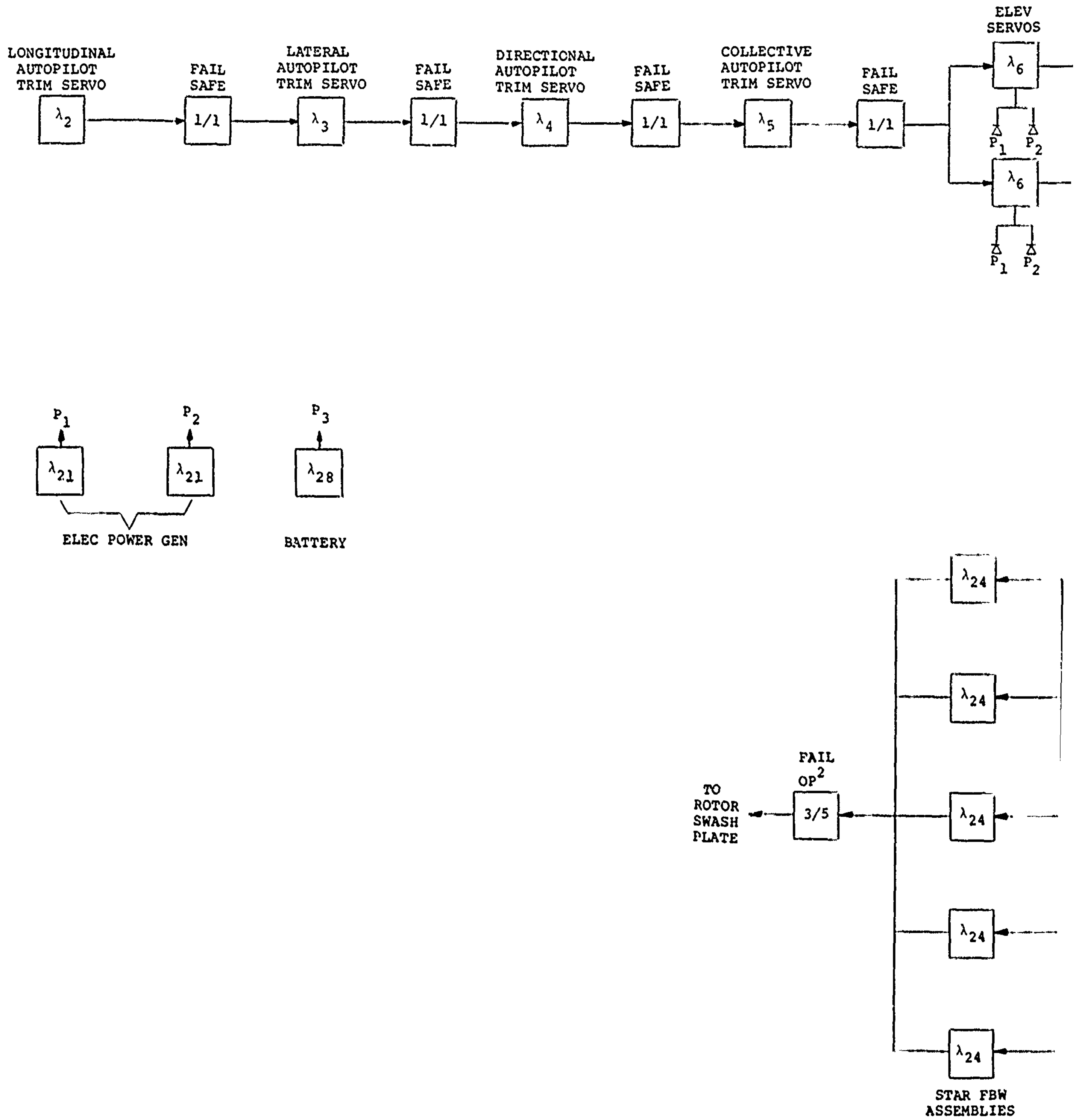
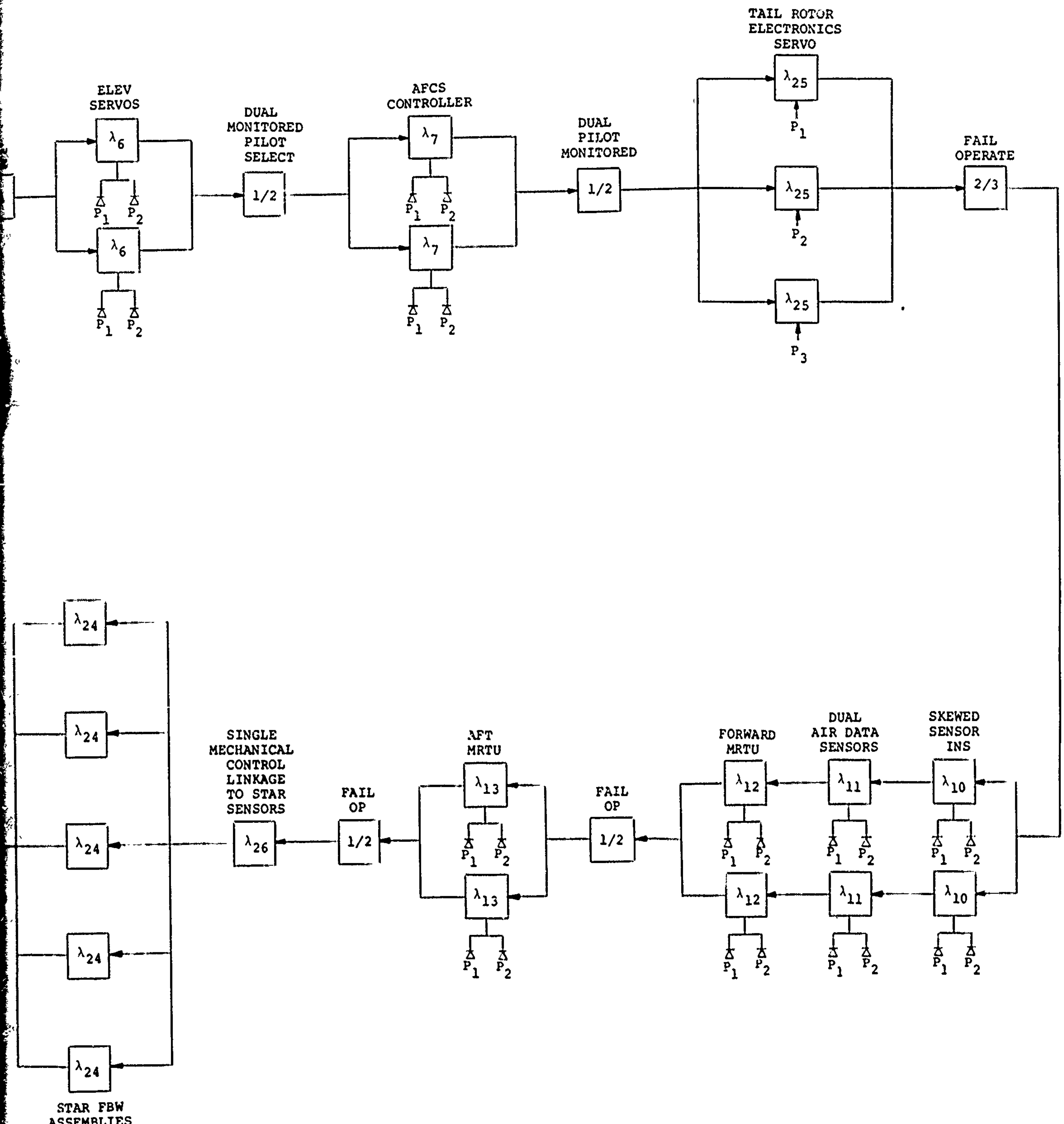


Figure 32. Fly-by-wire/light system reliability block diagram.



ELEV SERVOS
λ6
P1 P2
DUAL MONITORED PILOT SELECT
1/2
AFCS CONTROLLER
λ7
DUAL PILOT MONITORED
TAIL ROTOR ELECTRONICS SERVO
λ25
P1
P2
P3
FAIL OPERATE
2/3
SKEWED SENSOR INS
λ10
DUAL AIR DATA SENSORS
λ11
FORWARD MRTU
λ12
FAIL OP
AFT MRTU
λ13
SINGLE MECHANICAL CONTROL LINKAGE TO STAR SENSORS
λ26
λ24
STAR FBW ASSEMBLIES

TABLE 18. ASH HELICOPTER CONTROL SYSTEM/COMPONENT SYSTEM - FAILURE RATE SUMMARY

SYSTEM	Dual Mechanical		Fly-by-wire	
	Failure rate (per 10^6 flt hrs)	MTBF (flt hrs)	Failure Rate (per 10^6 flt hrs)	MTBF (flt hrs)
Sticks/Linkages	15,154	65.9	7017	142.5
Hydraulic Actuator	3920	255.1	3840	260.4
Electromechanical Actuators	1858	538.2	2689	371.9
Hydraulic Supplies	7621	131.2	3084	324.2
Electrical Supplies	7856	127.2	8856	112.9
Sensor and Processing Components	3350	298.5	5428	184.2
Totals	39,759	25.2	30,914	32.3
Allocated Goal Failure rate (per 10^6 hrs) = 400 MTBF (flt hrs) = 2500 Failures per 2500 flt hrs = 1	99.2		77.4	

TABLE 19. ASH HELICOPTER CONTROL SYSTEM/COMPONENT SYSTEM - FAILURE RATES

	Dual Mechanical		Fly-by-wire	
	Qty	Part failure rate (Per 10^6 flt hrs)	Qty	Part failure rate (Per 10^6 flt hrs)
Sticks/Cranks/Linkages				
Stick Assy - Collective	2	860	2	860
Jackshaft Assy - Collective	1	300	1	300
Force Gradient - Collective	1	150	1	150
Bellcrank & Support - Collective	2	65	1	65
Tube Assy - Collective	6	34	3	34
Miscellaneous Collective	-	865	-	387
Stick Assy - Cyclic	2	550	2	550
Jackshaft - Cyclie F/A	1	84	1	84
Force Gradient - Cyclic	2	150	2	150
Lever & Support - Cyclie F/A	2	65	-	-
Idler & Support - Cyclie Lat	2	65	-	-
Bellcrank & Support - Cyclic F/A	4	65	-	-
Bellcrank & Support - Cyclic Lat	8	65	2	65
Tube Assy - Cyclic Lat	10	34	4	34
Jam Override Spring Pkg	1	92	-	-
Tube Assy - Cyclic F/A	6	34	1	34
Miscellaneous Cyclic	-	992	-	136
Jam Override Spring Pkg, Mixer	6	92	-	-
Bellcranks, Drive Support & Mixer	14	34	-	-
Mixer Link	5	34	5	65
Mixer Assy	2	285	1	65

TABLE 19. (Continued)

	Dual Mechanical Qty	Dual Mechanical Part failure rate (Per 10^6 flt hrs)	Fly-by-wire Qty	Fly-by-wire Part failure rate (Per 10^6 flt hrs)
Sticks/Cranks/Linkages				
Tube Assy, Mixed Control	8	34	-	-
Elevator Horn	1	132	1	132
Miscellaneous Mixed Controls	-	506	1	118
Pedals Assy - Antitorque	2	102	1	102
Pedal Adjust - Antitorque	2	198	2	198
Force Gradient - Antitorque	1	150	1	150
Jam Override Spring - Antitorque	1	92	-	-
Bellcrank, Input - Antitorque	2	65	2	65
Bellcranks, Levers - Fwd Cabin Antitorque	4	65	2	65
Bellcranks, Idlers - Aft Cabin Antitorque	10	65	-	-
Bellcranks/Idlers - Tailboom - Antitorque	6	65	-	-
Bellcranks - Upper Fin - Antitorque	1	65	-	-
Tubes, Fwd - Cabin - Antitorque	4	34	3	34
Tubes, Aft Cabin	12	34	-	-
Tubes, Tailboom - Antitorque	4	34	-	-
Tubes, Fin - Antitorque	2	34	-	-
Input Link, T. R.	1	34	1	34
Link, Sensor Drive	-	-	3	34
Miscellaneous Antitorque	-	715	-	343
Subtotal		15,154		7,017

TABLE 19. (Continued)

	Dual Mechanical		Fly-by-Wire	
	Qty	Part failure rate (Per 10^6 flt hrs)	Qty	Part failure rate (Per 10^6 flt hrs)
Hydraulic Actuators				
Int Act Pkg (IAP)	-	-	5	768
Hydraulic Actuators (M/R)	3	980	-	-
Hydraulic Actuators (T/R)	1	980	-	-
Subtotal		3,920		3,840
Electromechanical Actuators				
FBW T/R Actuators	-	-	3	563
Actuator, Elevator, Dual	1	672	1	672
Actuator, Longitudinal, Trim	1	82	1	82
Actuator, Lateral, Trim	1	82	1	82
Actuator, Collective, Trim	1	82	1	82
Actuator, Directional, Trim	1	82	1	82
Actuators, Longitudinal, SAS	2	143	-	-
Actuators, Lateral, SAS	2	143	-	-
Actuators, Directional, SAS	2	143	-	-
Subtotal		1,858		2,689
Hydraulic System				
Pump, FBW	-	-	5	160
Pump/Cooler	2	917	-	-
Accumulator	3	37	-	-
Module, Hydraulic	3	1,110	1	1,110
Motor Pump Unit	1	290	1	290
Miscellaneous Hydraulics	-	2,056	-	519
Subtotal		7,621		3,084

TABLE 19. (Continued)

	Dual Mechanical		Fly-by-wire	
	Qty	Part failure rate (Per 10^6 flt hrs)	Qty	Part failure rate (Per 10^6 flt hrs)
Electrical Supplies				
AC Generators/Controllers	2	1095	2	1095
Transformers/Rectifiers	2	199	2	199
Battery	1	5268	1	5268
Alternators, IAP Dedicated	-	-	5	200
Subtotal		7,856		8,856
Sensors & Signal Processing Components				
Buss/Controls/Computers	2	385	2	385
Fwd MRTU	2	280	2	280
Aft MRTU/Computer	2	380	2	380
AFCS Controller	1	70	1	70
Inertial Reference Sensors	2	200	2	200
Air Data Sensor	2	25	2	25
Stick Position Sensors (LVDT's)	10	10	-	-
Electronics, Elevator	2	314	2	314
Control Panel	2	16	2	16
IAP Electronics	-	-	5	330
STAR Sensors (Dual Passive)	-	-	5	32
T/R FBW Electronics	-	-	3	96
T/R FBW Sensors (Single Passive)	-	-	3	20
Subtotal		3,350		5,428
Total		39,759		30,914

TABLE 20. ASH HELICOPTER CONTROL SYSTEM/50 PERCENT SYSTEM - FAILURE RATE SUMMARY

SYSTEM	Dual Mechanical		Fly-by-wire	
	Failure rate (Per 10^6 flt hrs)	MTBF (Flt hrs)	Failure rate (Per 10^6 flt hrs)	MTBF (Flt hrs)
Sticks/Linkages	7577	131.9	3508	285
Hydraulic Actuator	1960	510.2	1920	520.8
Electromechanical Actuators	929	1076.4	1345	743.8
Hydraulic Supplies	3810	262.4	1542	648.5
Electrical Supplies	3928	254.4	4428	225.8
Sensor and Processing Components	1675	597.0	2714	368.5
Totals	19,879	50.3	15,457	64.7
Allocated Goal Failure rate (per 10^6 hrs) = 400 MTBF (flt hrs) = 2500 Failures per 2500 flt hrs = 1	49.7		38.6	

presents the mechanical system and FBW system predicted reliability for each major system/component grouping and the allocated system reliability goal. From this analysis, the dual mechanical system has a failure rate of 0.019879 failure per flight hour. The FBW system has a failure rate of 0.015457 failure per flight hour. Similarly, the FBW MTBF is better than the dual mechanical MTBF by 29 percent. Both systems at 50 percent failure rate levels are still worse than the specified system reliability goal.

The results of the system reliability analysis using 200 percent component failure rate values are shown in Table 21. The table presents the mechanical system and FBW system predicted reliability for each major system/component grouping and the allocated system reliability goal. From this analysis, the dual mechanical system has a failure rate of 0.079518 failure per flight hour. The FBW system has a failure rate of 0.061828 failure per flight hour. The FBW MTBF is still better than the mechanical system, but worse than the specified system reliability goal.

3.5.3.2 Mission Reliability Failure Rates. The mission analysis was performed using the data sources of Paragraph 3.5.1.1. A limited failure modes and effects analysis was performed for each of the components of the flight controls, the electrical system, and the hydraulic system to obtain the mission failure rates. The components were analyzed to determine which type of failure modes would cause a loss of a function that would affect the mission and thereby cause a mission abort. For mission consideration, control of the main rotor, tail rotor, and command augmentation is required. This defines the following mission critical functions for the control system:

- Sticks, pedals, and mechanical linkages/mixers
- Main rotor and tail rotor boost actuators
- Hydraulic system
- Electrical system to power command augmentation
- Command augmentation sensors/electromechanical actuators/ computers

Tables 22 and 23 provide a summary of failure rates for various items depicted in Figures 31 and 32, respectively.

TABLE 21. ASH HELICOPTER CONTROL SYSTEM/200 PERCENT SYSTEM - FAILURE RATE SUMMARY

	Dual Mechanical		Fly-by-wire	
	Failure rate (Per 10^6 flt hrs)	MTBF (Flt hrs)	Failure rate (Per 10^6 flt hrs)	MTBF (Flt hrs)
Sticks/Linkages	30,308	32.9	14,034	71.2
Hydraulic Actuator	7840	127.6	7680	130.2
Electromechanical Actuators	3716	269.1	5378	185.9
Hydraulic Supplies	15,242	65.6	6168	162.1
Electric Supplies	15,712	63.6	17,712	56.4
Sensor and Processing Components	6700	149.3	10,856	92.1
Totals	79,518	12.6	61,828	16.2
Allocated Goal Failure rate (per 10^6 hrs) = 400 MTBF (flt hrs) = 2500 Failures per 2500 flt hrs = 1	198.4		154.3	

TABLE 22. DUAL MECHANICAL CONTROL SYSTEM
FAILURE RATES X 10^{-6}

Item	Description	Mission Rate	Flight Safety
9	Tail Rotor Boost Servo	269	-
10	Skewed Sensor Ins	200	-
11	Dual Air Data Sensors	25	-
12	Forward MRTU	280	-
13	Aft MRTU	380	-
14	SAS Pitch Servo	82	-
15	SAS Roll Servo	82	-
16	SAS Collective Servo	82	-
17	Mech Links, Mixer	1912	0.474
18	Fore/Aft 1 Boost Servo	269	0.237
19	Fore/Aft 2 Boost Servo	269	0.237
20	Lateral Boost Servo	269	0.237
21	Elec Power	1294	259.0
22	Hydraulic Supply	1775	355.0
23	Electrically Driven Pump Supply	1400	280.0
24	Tail Rotor Associated Mechanical Linkage	793	-
25	Dual Position Sensors	10	-

TABLE 23. FLY-BY-WIRE CONTROL SYSTEM
FAILURE RATES X 10^{-6}

Item	Description	Mission Rate	Flight Safety
10	Skewed Sensor Ins	200	-
11	Dual Air Data Sensors	25	-
12	Forward MRTU	280	-
13	Aft MRTU	380	-
21	Electrical Power (non-dedicated)	1294	-
24	STAR System*	8.29	0.0071
25	Tail Rotor Electronic Servos	679	—
26	Mechanical Linkage from Cockpit Controls to Motion Transducers	1553	0.038 0.038

* See Section 4.3.2

The mission failure probability for the dual mechanical system may be expressed

$$P_{M_m} \cong \lambda_9^2 + (\lambda_{10} + \lambda_{11} + \lambda_{12} + \lambda_{25})^2 + \lambda_{13}^2 + \lambda_{14}^2 + \lambda_{15}^2 + \lambda_{16}^2 + \lambda_{17} + \lambda_{18} + \lambda_{19} + \lambda_{20} + \lambda_{21}^2 + \lambda_{22} + \lambda_{24} \quad (2)$$

Substituting the failure rates from Table 22 yields

$$P_{M_m} \cong 5287 \times 10^{-6} \quad (3)$$

and

$$\text{Mission MTBF}_{(\text{Dual Mechanical})} \cong \frac{10^6}{5287} = 189 \text{ hours}$$

For the fly-by-wire system, the mission failure probability may be expressed

$$P_{M_{fbw}} \cong \lambda_{M_{STAR}} + (\lambda_{10} + \lambda_{11} + \lambda_{12})^2 + \lambda_{13}^{\,2} + \lambda_{21}^{\,2} + 3\lambda_{25}^{\,2} + \lambda_{26} \quad (4)$$

Substituting the failure rates from Table 23 yields

$$P_{M_{fbw}} = 11.74 \times 10^{-6} + \lambda_{26} \quad (5)$$

where λ_{26} is the mission failure rate estimated for the cockpit sticks/pedals and associated linkage to the fly-by-wire control motion transducer. Excluding the contribution of λ_{26}, the fly-by-wire mission failure MTBF is

$$\text{Mission MTBF}_{(\text{FBW without } \lambda_{26})} \cong \frac{10^6}{11.74} = 85{,}179 \text{ hours}$$

This exceeds the mission failure goal of 10,000 hours.

However, when the contribution of λ_{26} is included, the MTBF becomes

$$\text{Mission MTBF}_{(\text{FBW})} \cong \frac{10^6}{1565} = 640 \text{ hours}$$

A significant reduction in predicted mission MTBF is obtained when this factor is considered. This results from the fact that the conventional cockpit controls utilized here have, by nature, a number of possible failure anomalies. It is expected that human factor aspects and experience will enable a modification of the typical cockpit in the future so that this problem is substantially improved.

Fifty percent and 200 percent component failure rates may be used to compute additional mission reliability numbers. Results are tabulated in Table 24. In all cases, the fly-by-wire system is better than the dual mechanical.

3.5.3.3 Flight Safety Failure Rates. The flight safety analysis was performed using Sperry Flight System data or MIL-HDBK-217B for the electronic components and USASC accident data for the UH-1H helicopter for the mechanical, electrical, and hydraulic power components. The USASC data were examined to see what type of failure modes have caused the accidents in the past and to see if the modes had been eliminated by design

TABLE 24. ASH HELICOPTER CONTROL SYSTEM MISSION FAILURE SUMMARY

System	50% Rates		Normal		200% Rates	
	λ (Per 10^6)	MTBF (Flt hrs)	λ (Per 10^6)	MTBF (Flt hrs)	λ (Per 10^6)	MTBF (Flt hrs)
Dual Mechanical	2593	386	5287	189	10,574	95
Fly-by-wire	782	1280	1565	640	3136	319
Fly-by-wire Excluding sticks & Mechanical Linkage Failures	5.0	199,200	11.7	85,180	30.4	32,864

λ Goal $\leq$ 100 per 10^6 hours

MTBF Goal $\geq$ 10,000 hours

of the ASH system. Class identifications for the data examined were:

- Total loss
- Major
- Minor
- Incident
- Forced landing

Only components that are critical to the flight control function, i.e., where failure results in loss of controlled flight, were considered in the flight safety analysis.

For flight safety, control of the main rotor is required. It is assumed that loss of tail rotor control or command augmentation results in an abort of the mission such that critical flight conditions are avoided.

For the dual mechanical control system, control of the main rotor boost control actuators is required for flight safety. Two hydraulic systems and a third electrically powered auxiliary system are available for hydraulic power. The flight safety failure probability may be written

$$P_{C_m} = \lambda_{17}^2 + \lambda_{18}^2 + \lambda_{19}^2 + \lambda_{20}^2 + \lambda_{22}^2 (\lambda_{23} + \lambda_{21}^2) \qquad (6)$$

Substituting the flight safety failure rates from Table 22 yields

$$P_{C_m} = 3.57 \times 10^{-11} \qquad (7)$$

with the 50 and 200 percent component failure rate conditions yielding 4.52×10^{-12} and 2.84×10^{-10}, respectively.

For the fly-by-wire control system, control of the main rotor swashplate by the fly-by-wire implementation must be assured for flight safety. The various combinations of IAP states under which control is maintained are discussed in Section 4. Except for the single mechanical portion of the control system that connects the cockpit controls to the fly-by-wire control motion transducers, the system is quintuplexed. The flight safety failure probability may be written

$$P_{C_{fbw}} = \lambda_{26} + \lambda_{C_{STAR}}$$

Using the flight safety failure rates from Table 23, the flight safety failure probability may be approximated as

$$P_{C_{fbw}} \cong 4.51 \times 10^{-8} \tag{8}$$

The contribution to flight safety failures made by the mechanical linkage to the control motion transducers may be reduced to insignificance (compared to the redundant STAR channels) by using advanced cockpit controls (e.g., direct connections of transducers to side-arm controller) or by dualizing the linkage. In these cases,

$$P_{C_{fbw}} \text{ (neglecting } \lambda_{26}) = 7.1 \times 10^{-9} \tag{9}$$

The flight safety reliability for the dual mechanical and fly-by-wire control systems are summarized in Table 25. In all cases, the dual mechanical system is better than the fly-by-wire system.

3.6 MAINTAINABILITY AND AVAILABILITY

The following paragraphs compare the maintainability and availability features of the dual mechanical and the fly-by-wire/light control systems. The analysis includes the entire system from the cockpit controls to the main rotor and tail rotor control systems.

3.6.1 Methodology and Definitions

The following assumptions are used in the analysis:

- 720 hours/month
- 40 flight hours/month (FH/MO)
- 1.3 flight hours/flight
- 1.8 flights/day (average) for days flown
- 300 flight hours/periodic inspection
- Preflight inspection is accomplished prior to the first flight of the day. Its intended purpose is to assure that nothing has occurred during the prior period of unattended idleness that could jeopardize safety of flight or mission accomplishment.

TABLE 25. ASH HELICOPTER CONTROL SYSTEM FLIGHT SAFETY FAILURE RATE SUMMARY

System	50% Rates		Normal		200% Rates	
	λ (per hr)	MTBF (flt hrs)	λ (per hr)	MTBF (flt hrs)	λ (per hr)	MTBF (flt hrs)
Dual Mechanical	4.5×10^{-12}	2.2×10^{11}	3.57×10^{-11}	2.80×10^{10}	2.84×10^{-10}	3.52×10^{9}
Fly-By-Wire	2.0×10^{-8}	5.0×10^{7}	4.51×10^{-8}	2.22×10^{7}	1.33×10^{-7}	7.52×10^{6}
Fly-By-Wire (Neglecting linkage to transducers)	1.0×10^{-9}	1.0×10^{9}	7.1×10^{-9}	1.41×10^{8}	55.0×10^{-9}	1.82×10^{7}

λ Goal ≤ 1.0 per 10^7 hours (1.0×10^{-7} per hour)

MTBF Goal $\geq 1.0 \times 10^7$ hours

Model UH-1N 3-M data are used as the basis in calculating corrective maintenance factors such as mean-time-to-repair (MTTR), elapsed time (ET) per task, number of men required per task, and maintenance man-hours per flight hour (MH/FH) for the MUT dual mechanical system. These data were logged during 189,251 flight hours with an average flight time of 1.3 hours. The UH-1N helicopter is used for a baseline because the flight control system closely resembles one channel of the MUT dual mechanical system and an abundance of data were available.

Calculations of fly-by-wire (FBW) corrective maintenance are based on comparisons with the dual mechanical system where similar systems exist and on previous in-house FBW reference studies where components are unique.

The tasks, task times, and the number of men to rig the control system of the Model UH-1N are used as a baseline for rigging the MUT dual mechanical system. These times were factored to account for redundant components. The rigging times are included in the corrective MMH/FH. Rigging times for the FBW systems are estimated based on tasks defined in previous in-house studies.

Daily and periodic inspection requirements for the Model UH-1N are used as a basis for the MUT dual mechanical system. Inspection times are factored for differences in complexities. The factors are used for both daily and periodic inspections. Access times for both inspections are deleted because the daily does not require access, and periodic requires all accesses to be opened before the inspection can start. FBW inspection requirements and times are based on comparison with the dual mechanical system with factors for complexity and from results of other in-house studies.

3.6.2 Corrective Maintenance

Corrective maintenance is performed on a nonscheduled basis to restore equipment to satisfactory condition by providing correction of a malfunction that has caused degradation of the item below the specified performance.

Organizational and intermediate man-hours per flight hour for the dual mechanical and fly-by-wire/light control systems have been tabulated and are compared below.

3.6.2.1 Dual Mechanical System

	MH/FH
Organizational level	0.318
Intermediate level	0.099
	0.417

Rigging (end-to-end) except elevator

9.49 elapsed hours (EH) 17.86 MH/occurrence	Time included in organizational level

3.6.2.2 Fly-By-Wire System

	MH/FH
Organizational level	0.142
Intermediate level	0.096
	0.238

Rigging (end-to-end) except elevator

2.4 elapsed hours 3.63 MH/occurrence	Time included in organizational level

3.6.3 Preventive Maintenance

Preventive maintenance is the care and servicing by personnel for the purpose of maintaining equipment in satisfactory condition by providing for systematic inspection, detection, and correction of incipient failures, either before they occur, or before they develop into major defects.

Daily inspection is one that is performed after the last flight of the day, or prior to the first flight on the next day the aircraft is flown. By previous assumptions, 1.3 flight hours/flight and 1.8 flights/day yields 2.34 flight hours/day. Thus, we have an inspection frequency of

$$\frac{1}{2.34} = 0.43 \text{ (inspection frequency)}$$

Periodic inspection is a comprehensive inspection performed each 300 hours. Thus, we have an inspection frequency of

$$\frac{1}{300} = 0.00333$$

Comparisons of the preventive maintenance man-hours per flight hour for the dual mechanical and fly-by-wire/light control systems are made below.

3.6.3.1 <u>Dual Mechanical System</u>

Daily inspection: 0.43 x 0.44 MH = 0.189 MH/FH

Periodic inspection: 0.00333 x 3.8 MH = 0.013 MH/FH

3.6.3.2 <u>Fly-By-Wire System</u>

Daily inspection: 0.43 x .22 MH = 0.095 MH/FH

Periodic inspection: 0.00333 x 1.9 MH = 0.006 MH/FH

3.6.4 <u>Availability</u>

The effects of the two systems on the availability of the helicopter is computed from the system failure and scheduled and unscheduled maintenance. The following formula is used

$$A = \frac{\text{Calendar Time - Downtime}}{\text{Calendar Time}}$$

where downtime is any time the helicopter is down for either scheduled or unscheduled maintenance.

In the following paragraphs, the availability of the dual mechanical system and fly-by-wire/light system are computed:

3.6.4.1 <u>Dual Mechanical System</u>

<u>Corrective (organizational level)</u> .318 MH/FH

1.97 men/task (average)
.318 MH/FH ÷ 1.97 men/task = .162 elapsed time (MTTR)
.162 x 40 FH/MO = <u>6.48 EH/Mo</u> (elapsed hours/month)

<u>Preventive</u>

Daily: .43 FH (frequency) x .44 MH = .189 MH/FH

1.0 man/task
.189 MH/FH ÷ 1 man/task = .189 EH/FH
.189 EH/FH x 40 FH/Mo = <u>7.56 EH/Mo</u>

Periodic: .00333 FH (frequency) x 3.8 MH = .013 MH/FH

1.5 men/task average
.013 MH/FH ÷ 1.5 men/task = .008 EH/FH
.008 EH/FH x 40 FH/Mo = .320 EH/Mo

6.48 EH/MO + 7.56 EH/Mo + .320 EH/Mo = 14.36 EH/Mo

$A = \frac{720-14.36}{720}$ = 98.01 percent availability for the flight control system (dual MUT)

3.6.4.2 Fly-By-Wire System

Corrective (organizational level) .142 MH/FH
1.97 men/task (average)

.142 MH/FH ÷ 1.97 men/task = .072 elapsed time (MTTR)
.072 x 40 FH/Mo = 2.88 EH/MO (elapsed hours/mo)

Preventive

Daily: .43 FH (frequency) x .22 MH = .095 MH/FH

1.0 man/task
.095 MH/FH ÷ 1 man/task = .095 EH/FH
.095 EH/FH x 40 FH/Mo = 3.8 EH/Mo

Periodic: .00333 FH (frequency) x 1.9 = .006 MH/FH

1.5 men/task average
.006 MH/FH ÷ 1.5 men/task = .004 EH/FH
.004 EH/FH x 40 FH/Mo = .160 EH/Mo

2.88 EH/Mo + 3.8 EH/Mo + .160 EH/Mo = 6.84 EH/Mo

$A = \frac{720-6.84}{720}$ = 99.05 percent availability (for the flight control system - ASH FBW)

The maintainability and availability comparisons may be summarized in Table 26.

3.7 PREDICTED HANDLING QUALITIES

In comparing the FBW/L flight control system with the dual mechanical flight control system, the difference results primarily from the added capability derived from the more sophisticated, flexible command augmentation system implemented in the FBW/L system. The unaugmented FBW/L control system is simply a replacement for the conventional push-pull tube, bellcrank,

TABLE 26. MAINTAINABILITY AND AVAILABILITY COMPARISON

System	Corrective Maintenance MH/FH		Preventive Maintenance MH/FH		Availability
	Org	Int	Daily	Periodic	
Dual Mechanical	.318	.099	.188	.013	98%
Fly-by-Wire	.142	.096	.096	.006	99%

etc. The unaugmented FBW/L system operates in a similar manner as the conventional control system. The conventional system, however, will exhibit some control free play, friction, inertia, and/or wind-up that will prevent the pilot from having as responsive a control as is available with the FBW/L control system. This limits the capability of the command augmentation function of the conventional mechanical control where authority and rate-limited secondary actuators mechanically sum the command augmentation input to the pilot's input. The limitation results basically from the nonlinearities and mechanical impedances that constrain the closed-loop gains and compensation techniques from a stability and response standpoint. In the FBW/L control system, high gain feedback is used to attenuate gust responses, and good model-following characteristics are determined by the command augmentation signal shaping.

Another area of comparison between the dual mechanical and the FBW/L control systems is in the area of control mode and adaptive flight capability. In the case where the control sticks are mechanically connected to the helicopter swashplate, the type of control mode is fixed. A displacement of the stick establishes a particular position of the swashplate. In the case of FBW/L control, on the other hand, the control signal derived from the control stick (which could be generated from either displacement or force) may be electronically processed to provide rate and/or displacement of the swashplate and may be scheduled by other parameters, such as airspeed, altitude, etc. Great flexibility is available in the FBW/L control law implementation so that the pilot may concentrate his effort more on the tactical situation and less on

the basic flight requirements. This is of significant importance during such diverse tasks as nap-of-the-earth, night flying, precision hover, and formation flight.

Typically, in hover or low-speed flight, the pilot must control the helicopter's velocity or the integral of velocity (position). However, velocity is proportional to the integral of attitude, which is the integral of attitude rate. Most basic aircraft present essentially an attitude rate response to a control input. Therefore, the pilot must mentally perform three integrations in order to control position. Studies of pilot performance in a compensatory task have indicated that, ideally, the pilot should be represented as a first-order lag of about one second and a transport delay of about one-half second. Thus, the pilot should not be expected to provide any lead.

An ideal system to reduce the pilot's hovering work load is one that provides a velocity response to a control input. This has been provided in the FBW/L control system using accelerometers to derive velocity signals. This velocity maneuvering system relieves the pilot of 180 degrees of lead or anticipation required, thus significantly reducing his work load.

4. REDUNDANCY MANAGEMENT

Redundancy management techniques recommended for the FBW/L flight control system are those that provide the capability for continuous flight control even after certain failures have occurred.

4.1 REDUNDANCY PHILOSOPHY

Deciding upon a level of redundancy for the various systems, subsystems, circuits, and assemblies requires the trade-off of two opposing considerations. On one hand, the simple systems have fewer parts, lower cost, and typically lower weight. These systems may have unacceptable flight safety reliability. On the other hand, more complex systems that have the multiplicity of subsystems, circuits, and assemblies, in order to meet the flight safety reliability, may have an unacceptable unscheduled maintenance reliability and may be too costly and too heavy. Thus, some compromise may be required. However, since loss of life is considered unacceptable, the flight safety reliability tends to predominate the philosophy of design typically with the result of an undesirably high unscheduled maintenance (system) failure rate.

Reference 7 considered three different approaches to fly-by-wire implementation on the main rotor. The first approach utilizes a quad-redundant electronic channel in each of the three control axes (roll, pitch, and collective) and controls a triple-piston actuator and electronic actuator model. The second approach uses a dual piston actuator on each of the three axes with each piston section controlled by dual electronics and electrohydraulic servo valves. The third approach utilizes a five-arm swashplate concept described previously. The last system was selected as the least complex, the most reliable, and the lightest.

A detailed FBW actuation study is presently underway at BHT under NASA Ames contract entitled "A Study of the Reliability of Present and Proposed Helicopter and Aircraft Actuation Systems." This study is considering the following schemes:

- Secondary electrohydraulic actuators.
- Primary hydraulic actuators.
- All-electric actuators.

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[7] Carlock, G., and Guinn, K., STATUS REPORT OF FLY-BY-WIRE RESEARCH PROJECT FOR 1978 FLIGHT EVALUATION SYSTEM, Bell Helicopter Textron Report 599-303-001, December 1976.

- Vibration control through actuators.
- Control for helicopters without swashplates.

The particular system configurations for both the dual mechanical control system and the FBW/L control system were presented in Section 2. The corresponding reliability and maintainability features were then compared in Section 3.

4.2 DUAL MECHANICAL CONTROL SYSTEM

The dual mechanical control system includes dual boost servos for pitch, roll, collective, and tail rotor control functions, the use of two hydraulic systems, plus a utility backup and two electric power supplies.

The dual boost actuator design utilizes spring cartridges at the input of the mechanical actuator valves so that if one side of the dual mechanical system was to become jammed, the other side could still valve the actuator. The boost actuator itself is designed so that, in theory, either of the two pistons may break a jam of the other. This requires that the break-free force of a jam be less than the design load capability of either side of the dual actuator.

The hydraulic and electrical supplies are configured so that automatic switchover from one supply to another is achieved after the first failure. In the case of the hydraulic supplies, a third supply, the utility, may provide hydraulic pressure after the second main hydraulic system failure. Of concern, however, is the possibility that the switchover logic may itself fail and result in a "domino" failure effect as the other systems are switched.

4.3 FLY-BY-WIRE/LIGHT CONTROL SYSTEM

The fly-by-wire/light control system utilizes the same horizontal stabilizer control system that is configured for the dual mechanical system (originally specified on the MUT as FBW). The control is designed to be dual and fail-safe, and is not to be mission or flight essential for the ASH mission.

The tail rotor control is considered mission essential and the main rotor control is considered both mission and flight essential.

4.3.1 STAR Channel Status Definition

The five actuator STAR main rotor control system utilizes independent control paths from the pilot's input (as measured

by a passive sensor) to the output of the IAP (connected to the swashplate). Each channel operates independently of the other channels using a dedicated hydraulic pump and a dedicated electrical alternator. Functionally, each channel may be represented as shown in Figure 33.

The helicopter is controlled in collective, pitch, and roll by the position of the swashplate; thus, loss of control of the swashplate constitutes a failure condition of the STAR control system. Since the swashplate is positioned by redundant IAPs, their status contributes to the failure modes.

The status of an IAP can be categorized by one of the following states: normal, bypassed, nulled, hardover, or hydraulically locked. Since the IAP is electronically controlled and monitored, the failure of the associated electronics, as well as certain failures of the IAP itself, will contribute to the determination of the IAP status at any particular time. Functionally, each channel has a power, control, monitor, bypass, pressure check, and load-relief characteristic.

4.3.1.1 Function Group Failure Model. Figure 33 identifies the functional component groupings of the integrated actuator package. These component groups are labeled A through H, with three states defined for each group. The first state in each group is the normal state and the other two are failure states. The failure state and failure rate of each group are listed in Tables 27 and 28, respectively.

The failure model used to establish the group failure rates in Table 28 is depicted in Figure 34. Each of the groups, A through H, has an identical model where the individual components together establish the associated failure rates. Switches S_1 and S_2 have failure rates λ_1 and λ_2, respectively, determined by reliability analyses of the functional components.

The model is in the normal state, State 1, when switches S_1 and S_2 are open (no failures). State 2 occurs when S_1 is closed and S_2 is open. State 3 occurs when S_2 is closed.

Mathematically, the probabilities may be written

$$P[\text{State 1}] = e^{-\lambda_1} e^{-\lambda_2} = e^{-(\lambda_1 + \lambda_2)}$$

$$P[\text{State 2}] = (1-e^{-\lambda_1})e^{-\lambda_2}$$

$$P[\text{State } 3] = 1-e^{-\lambda_2}$$

Since the model must exist in one of the above states, the mathematical sum of the three probabilities must sum to unity. This can be shown to be the case.

In the above equations, $e^{-\lambda_i}$ represents a reliability expression, R_i. Correspondingly, a failure expression can be written

$$Q_i = 1-R_i = 1-e^{-\lambda_i}$$

In the case where λ_i is a very small number, the above mathematical expressions can be approximated:

$$R_i = e^{-\lambda_i} \cong 1-\lambda_i$$

and

$$Q_i = 1-e^{-\lambda_i} \cong \lambda_i$$

Thus, the probability expressions for States 1, 2, and 3 may be approximated as

$$P[\text{State } 1] \cong 1-\lambda_1-\lambda_2$$

$$P[\text{State } 2] \cong \lambda_1 (1-\lambda_2) \cong \lambda_1 = \lambda_{\text{State } 2}$$

$$P[\text{State } 3] \cong \lambda_2 = \lambda_{\text{State } 3}$$

4.3.1.2 IAP State Probabilities. Using the failure models and group failure rates tabulated above, the various IAP state conditions and state probabilities may be derived. Using Boolean algebra, functional group states may be aggregated into logic equations, which represent the various states of the IAP.

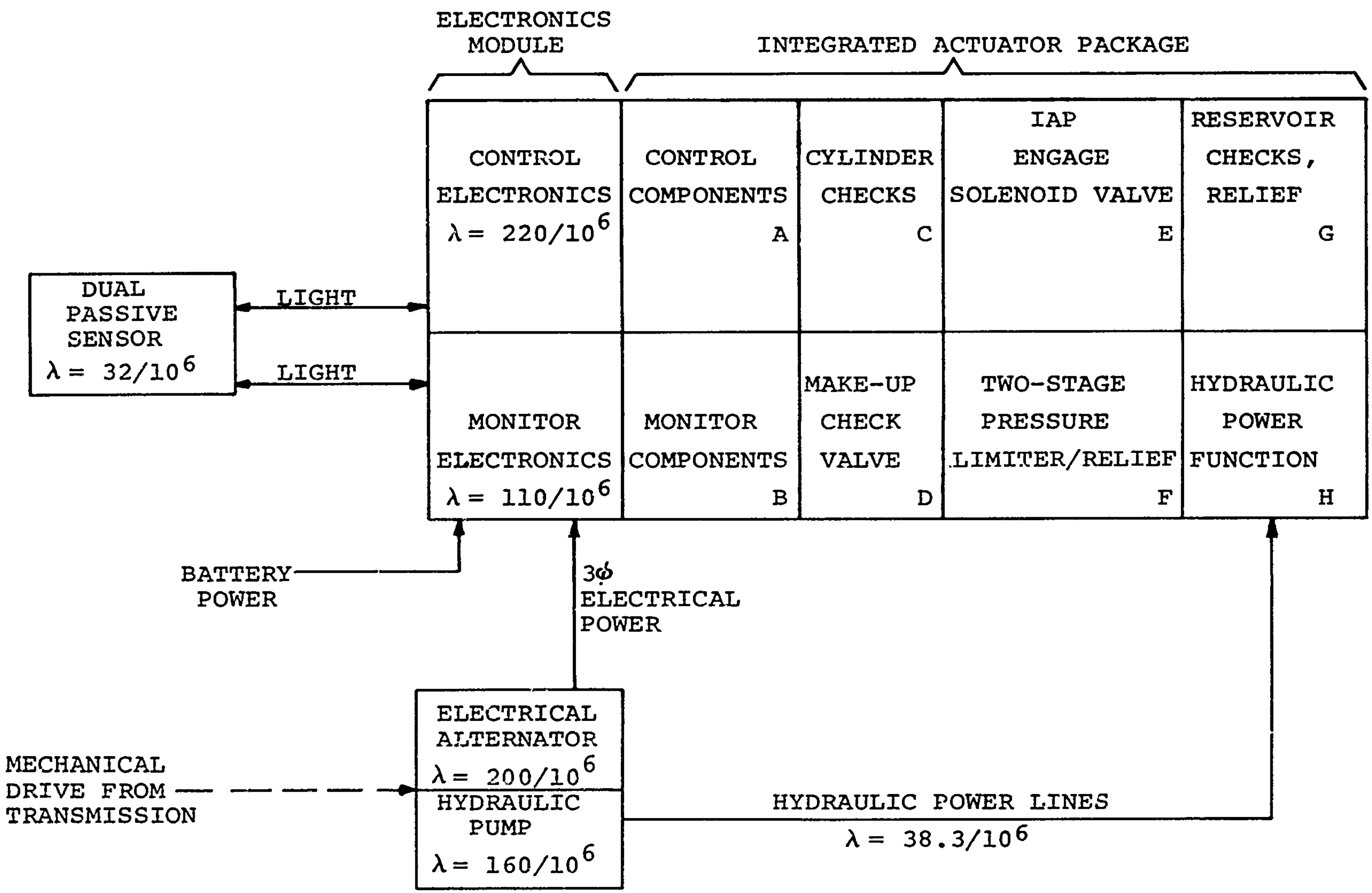


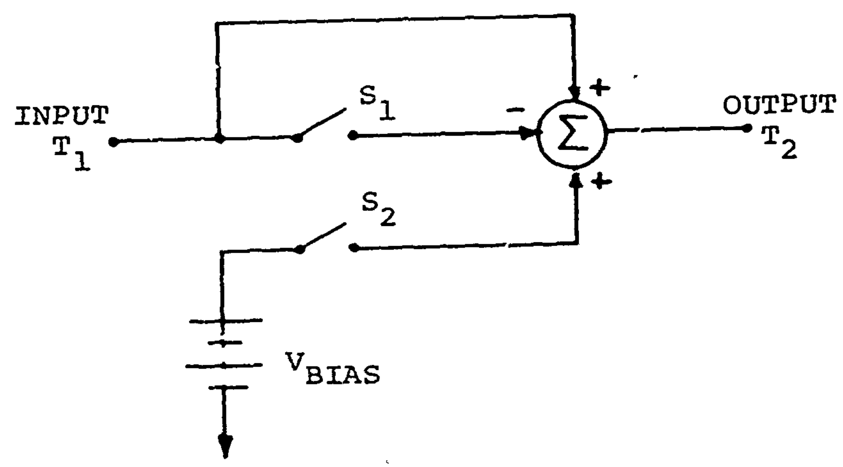
Figure 33. Functional block diagram of one STAR channel.

TABLE 27. FUNCTIONAL GROUPINGS OF IAP COMPONENTS

Group	Title	States	Comments
A	Control components	1 - Normal 2 - Failed null 3 - Failed hardover	Consists of EHSV, LVDT, and ΔP XDCR A2 - static $\Delta P < 350$ psi; dynamic $\Delta P > 250$ psi A3 - spool hardover - P = 3500+ psi
B	Monitor components	1 - Normal 2 - Failed-no indication 3 - Failed-false indication	Consists of EHSV, LVDT, and actuator LVDT
C	Cylinder checks	1 - Normal 2 - Failed shut 3 - Failed open	C2 - prevents bypass and load relief C3 - causes A3
D	Makeup check valve	1 - Normal 2 - Failed shut 3 - Failed open	D2 - prevents bypass and load relief D3 - allows return filter backflush during bypass/load relief
E	Engage solenoid valve	1 - Normal 2 - Failed on 3 - Failed off	E2 - IAP engaged; E1 or E3 required for bypass
F	Two-stage pressure/limiter relief	1 - Normal 2 - Failed shut 3 - Failed open	F2 - causes bypass
G	Reservoir pistons return checks and low pressure relief	1 - Normal 2 - Failed-blocked 3 - Failed bypass	G2 - caused by 1 of 4 pistons jam or return check/relief jam
H	Hydraulic power	1 - Normal 2 - Failed without fluid 3 - Failed with fluid	H2 - external leaks H3 - pressure filter clogged

TABLE 28. JAP GROUP FAILURE RATES

Group/state	Failure rate x 10^6 hours
A2	$\lambda_{A2} = 24.3$
A3	$\lambda_{A3} = 38.4$
B2	$\lambda_{B2} = 4.7$
B3	$\lambda_{B3} = 4.7$
C2	$\lambda_{C2} = 10.0$
C3	$\lambda_{C3} = 20.4$
D2	$\lambda_{D2} = 2.5$
D3	$\lambda_{D3} = 5.6$
E2	$\lambda_{E2} = 7.5$
E3	$\lambda_{E3} = 9.6$
F2	$\lambda_{F2} = 12.5$
F3	$\lambda_{F3} = 16.7$
G2	$\lambda_{G2} = 9.0$
G3	$\lambda_{G3} = 14.8$
H2	$\lambda_{H2} = 32.0$
H3	$\lambda_{H3} = 0.5$



TRUTH TABLE

S_1	S_2	T_2	STATE
0	0	T_1	1
1	0	0	2
0	1	$V_{BIAS} + T_1$	3
1	1	V_{BIAS}	

1. Switches S_1 and S_2 are normally open. When they fail, they close and affect the output, T_2.

2. Switches S_1 and S_2 have failure rates λ_1 and λ_2, respectfully.

Figure 34. Tristate failure model with state definition.

One of the failure states of the IAP is the Bypass state, which occurs when one or more of the following conditions exist, as expressed in Boolean algebra:

$$(A_2 + A_3 + H_3)\ B_1\ \bar{C}_2\ \bar{D}_2\ \bar{E}_2\ \bar{F}_2\ \bar{G}_2$$

$$B_3\ \bar{C}_2\ \bar{D}_2\ \bar{E}_2\ \bar{F}_2\ \bar{G}_2$$

$$E_3\ \bar{C}_2\ \bar{D}_2\ \bar{F}_2\ \bar{G}_2$$

$$H_2$$

$$F_3\ \bar{C}_2\ \bar{D}_2\ \bar{G}_2$$

If these expressions are interpreted as events, then the probability of an event may be determined from the functional group failure rates by substituting probability expressions from the failure model and adding the "or" terms and multiplying the "and" terms of the Boolean expression. When this is performed, the following expression is obtained:

$$P_{BP} \cong \lambda_{A2} + \lambda_{A3} + \lambda_{B3} + \lambda_{E3} + \lambda_{F3} + \lambda_{H2} + \lambda_{H3}$$

In addition to these IAP group failure rates, the failure rates of the other components of the STAR channel (as depicted in Figure 33) must be considered. When the appropriate values from Table 29 are added, the following overall Bypass state probability is obtained:

$$P_{BP_T} \cong \lambda_{A2} + \lambda_{A3} + \lambda_{B3} + \lambda_{E3} + \lambda_{F3} + \lambda_{H2} + \lambda_{H3}$$

$$+ \lambda_{PS} + \lambda_{ALT} + \lambda_{CE} + \lambda_{ME} + \lambda_{HP} + \lambda_{HF}$$

$$= 886 \times 10^{-6}$$

Another failure state of the IAP is the Null state, which occurs under the following condition:

$$(A2 + H3)\ \bar{G}_2\ (B2 + C2 + D2 + E2 + F2)$$

TABLE 29. NON-IAP STAR CHANNEL COMPONENT FAILURE RATES

Component	Failure rate - hours
PS - Passive sensor	$\lambda_{PS} = 32 \times 10^{-6}$
ALT - Alternator	$\lambda_{ALT} = 200 \times 10^{-6}$
CE - Control electronics	$\lambda_{CE} = 220 \times 10^{-6}$
ME - Monitor electronics	$\lambda_{ME} = 110 \times 10^{-6}$
HP - Hydraulic pump	$\lambda_{HP} = 160 \times 10^{-6}$
HF - Hydraulic fittings	$\lambda_{HF} = 38 \times 10^{-6}$

Manipulating the probability expressions, the following equation is derived:

$$P_N \cong (\lambda_{A2} + \lambda_{H3})(\lambda_{B2} + \lambda_{C2} + \lambda_{D2} + \lambda_{E2} + \lambda_{F2})$$

Adding the additional non-IAP failure rates, the overall Null state probability is obtained:

$$P_{N_T} \cong (\lambda_{A2} + \lambda_{PS} + \lambda_{CE} + \lambda_{H3} + \lambda_{HP})(\lambda_{B2} + \lambda_{ME} + \lambda_{C2}$$

$$+ \lambda_{D2} + \lambda_{E2} + \lambda_{F2})$$

$$= 6.42 \times 10^{-8}$$

Another failure state of the IAP is the Hardover state, which occurs under the following condition:

$$A3(B2 + D2 + E2 + F2)\ \bar{G}_2 + C3(A2 + A3)\ \bar{G}_2$$

Manipulating the probability expressions, the following equation is derived:

$$P_{HO} \cong \lambda_{A3}\ (\lambda_{B2} + \lambda_{D2} + \lambda_{E2} + \lambda_{F2}) + \lambda_{C3}\ (\lambda_{A2} + \lambda_{A3})$$

Adding the additional non-IAP failure rates, the overall Hard-over state probability is obtained:

$$P_{HO_T} \cong (\lambda_{A3} + \lambda_{PS} + \lambda_{CE})(\lambda_{B2} + \lambda_{ME} + \lambda_{D2} + \lambda_{E2} + \lambda_{F2})$$

$$+ \lambda_{C3} (\lambda_{A2} + \lambda_{PS} + \lambda_{CE})$$

$$= 4.53 \times 10^{-8}$$

The last failure state of the IAP is the hydraulic lock state. This represents a potential condition when the return check valve and the low pressure relief valve simultaneously jam or if the main reservoir piston or boot strap pistons jam. However, the return and reservoir pressures are nominally 60 psi. Under the potential jam condition, this pressure would approach 2200 psi (due to the unbalanced extend/retract piston areas used in this design). The reservoir end cap is designed to rupture at 1000 psi to prevent this possible hydraulic lock condition. Hydraulic lock would occur only if this rupture failed to occur. Using 2.5×10^{-6} as the failure-to-rupture probability, the probability of an actual hydraulic lock state may be written

$$P_{HL} \cong 2.5 \times 10^{-6} \lambda_{G_2} = 0.0225 \times 10^{-9}$$

The probability of the IAP being in a particular state is summarized in Table 30.

4.3.2 STAR System Contribution to Mission Abort

Since each channel has multiple states, the overall Swashplate Control state is determined by a number of possible combinations of the five channels. Five channel failure combinations and event probability expressions are listed in Tables 31 and 32. As presented in Table 4, the main rotor operational criteria are classified into five cases. These cases are tabulated from the design goal of controlling the swashplate after two possible channel failures.

The following ground rules would be applied for the STAR system:

- Following a first failure, maintain vehicle velocity less than or equal to 0.6 V_H. (This allows full maneuver loads for ASH mission capability.)

TABLE 30. IAP STATE PROBABILITY

State	Probability
Normal	$P_{NORMAL} = 0.999114$
Bypass	$P_{BP_T} = 0.000886$
Null	$P_{N_T} = 0.642 \times 10^{-7}$
Hardover	$P_{HO_T} = 0.453 \times 10^{-7}$
Hydraulic lock	$P_{HL} = 0.0225 \times 10^{-9}$

- Following a second channel failure, abort this mission. (However, maneuver loads at 0.6 V_H and cruise loads at V_H are still controllable after two channels have been switched to bypass.)

Pilot decision to abort results from system status information provided from the computer/multiplex, bus/display links from the IAP electronics. Normally, the monitor electronics would detect a channel failure, automatically switch the channel to bypass, and provide pilot warning.

Pilot warning is implemented through dualized computer/MRTU/ multiplex bus channels to the display unit. Failure of this system can provide either failure indication when there is no failure or no failure indication when there is a failure. The first would contribute to the mission abort rate and the latter would contribute to decreased flight safety. Instead of aborting after a second-channel failure (due to no warning), the pilot would continue to fly, thus increasing his probability of crash. It will be shown, however, that the pilot warning system is quite reliable and the probability of simultaneous failure of three control channels and the pilot warning system is quite low.

TABLE 31. FIVE-CHANNEL FAILURE COMBINATIONS

Case No.	Channel					No. of Failures						Combination Probability
	1	2	3	4	5	0	1	2	3	4	5	
1	0	0	0	0	0	X						$(1-Q)^5$
2	0	0	0	0	1		X					$Q(1-Q)^4$
3	0	0	0	1	0		X					$Q(1-Q)^4$
4	0	0	0	1	1			X				$Q^2(1-Q)^3$
5	0	0	1	0	0		X					$Q(1-Q)^4$
6	0	0	1	0	1			X				$Q^2(1-Q)^3$
7	0	0	1	1	0			X				$Q^2(1-Q)^3$
8	0	0	1	1	1				X			$Q^3(1-Q)^2$
9	0	1	0	0	0		X					$Q(1-Q)^4$
10	0	1	0	0	1			X				$Q^2(1-Q)^3$
11	0	1	0	1	0			X				$Q^2(1-Q)^3$
12	0	1	0	1	1				X			$Q^3(1-Q)^2$
13	0	1	1	0	0			X				$Q^2(1-Q)^3$
14	0	1	1	0	1				X			$Q^3(1-Q)^2$
15	0	1	1	1	0				X			$Q^3(1-Q)^2$
16	0	1	1	1	1					X		$Q^4(1-Q)$
17	1	0	0	0	0		X					$Q(1-Q)^4$
18	1	0	0	0	1			X				$Q^2(1-Q)^3$
19	1	0	0	1	0			X				$Q^2(1-Q)^3$
20	1	0	0	1	1				X			$Q^3(1-Q)^2$
21	1	0	1	0	0			X				$Q^2(1-Q)^3$
22	1	0	1	0	1				X			$Q^3(1-Q)^2$
23	1	0	1	1	0				X			$Q^3(1-Q)^2$
24	1	0	1	1	1					X		$Q^4(1-Q)$
25	1	1	0	0	0			X				$Q^2(1-Q)^3$
26	1	1	0	0	1				X			$Q^3(1-Q)^2$
27	1	1	0	1	0				X			$Q^3(1-Q)^2$
28	1	1	0	1	1					X		$Q^4(1-Q)$
29	1	1	1	0	0				X			$Q^3(1-Q)^2$
30	1	1	1	0	1					X		$Q^4(1-Q)$
31	1	1	1	1	0					X		$Q^4(1-Q)$
32	1	1	1	1	1						X	Q^5
		Total				1	5	10	10	5	1	

TABLE 32. FIVE-CHANNEL EVENT PROBABILITIES

Event	Event probability
T_{3M} = (3 or more channels failed)	$10Q^3(1-Q)^2 + 5Q^4(1-Q) + Q^5$ $= 6Q^5 - 15Q^4 + 10Q^3 \simeq 10Q^3$
T_{2M} = (2 or more channels failed)	$10Q^2(1-Q)^3 + T_{3M}$ $= -4Q^5 + 15Q^4 - 20Q^3 + 10Q^2 \approxeq 10Q^2$
T_{1M} = (1 or more channels failed)	$5Q(1-Q)^4 + T_{2M}$ $= Q^5 - 5Q^4 + 10Q^3 - 10Q^2 + 5Q \approxeq 5Q$

The failure rate of a single computer, MRTUs, and multiplex bus is 660 per 10^6 hours. The reliability of this single pilot warning channel is

$$R_{SC} = e^{-\lambda_{SC}}$$

where

$$\lambda_{SC} = 660 \times 10^{-6}$$

Thus, the dual system pilot warning system failure rate can be expressed

$$Q_{PWS} = (1-e^{-\lambda_{SC}})^2 \cong \lambda_{SC}{}^2 = 0.44 \times 10^{-6}$$

The mission abort failure rate resulting from main rotor control system failure is established from the following conditions:

- Two channels become bypassed and the pilot is properly warned.

- The pilot warning system provides false indication of bypass condition.

Indicating the first condition probability as P_1 and the second as P_2, the probability of mission abort may be expressed as

$$P_{MA} = P_1 + P_2$$

From Table 30, the probability of two channels in bypass can be computed to be

$$10\ Q^2(1-Q)^3$$

where

$$Q = P_{BP_T} = 0.000886$$

Thus,

$$P_1 = [10\ P_{BP_T}{}^2\ (1-P_{BP_T})^3][R_{SC}{}^2]$$

$$\cong 10\ P_{BP_T}{}^2 = 7.85 \times 10^{-6}$$

and

$$P_2 = Q_{PWS} \cong 0.44 \times 10^{-6}$$

Combining the above probabilities, the mission abort probability becomes

$$P_{MA} = P_1 + P_2 = 8.29 \times 10^{-6} \cong \lambda_{MA}$$

Thus, the mean-time-between-aborts (MTBA) time interval resulting from the STAR system failure can be expressed

$$MTBA = \frac{1}{\lambda_{MA}} = 121{,}000 \text{ hours}$$

4.3.3 STAR System Contribution to Flight Safety

Loss of control of the main rotor swashplate can result in loss of flight safety, i.e., a possible catastrophic condition. This situation would result from any of the following conditions:

- Three or more IAPs in Bypass state
- Two or more IAPs in Hardover or Null state

- One or more IAPs in Hardover or Null state simultaneous with a load relief failure (F2) in the failed channel

- One or more IAPs hydraulically locked

Indicating the probability of these conditions as P_1, P_2, P_3, and P_4, respectively, the catastrophic failure probability may be expressed as

$$P_C = P_1 + P_2 + P_3 + P_4$$

Using Tables 30 and 32, the following probabilities are obtained

$$P_1 \cong 10\ P_{BP_T}{}^3 = 10(.886 \times 10^{-3})^3 = 7.0 \times 10^{-9}$$

$$P_2 \cong 10(P_{HO_T} + P_{N_T})^2 = 0.12 \times 10^{-12}$$

$$P_3 \cong 5(P_{HO_T} + P_{N_T})\lambda_{F2} = 6.8 \times 10^{-12}$$

$$P_4 \cong 5\ P_{HL} = 1.125 \times 10^{-10}$$

Thus, the catastrophic failure probability becomes

$$P_C = (7.0 + 0.00012 + 0.0068 + 0.11) \times 10^{-9}$$
$$= 7.1 \times 10^{-9} \cong \lambda_C$$

and the mean-time-between-crashes (MTBC) time interval resulting from the STAR system failure can be expressed

$$MTBC = \frac{1}{\lambda_C} = 1.41 \times 10^8 \text{ hours}$$

4.3.4 Power Supply Configuration

The hydraulic power supply consists of five mechanically driven, dedicated hydraulic pumps individually plumbed to a specific IAP and an electrically driven utility hydraulic system. The utility system is used for ground checkout of the FBW system and certain other auxiliary functions. This is discussed in Section 2.2.5.3.

Only the main rotor requires hydraulic pressure. The redundancy management consists of simply including a pump in each of the five parallel control channels. If a pump fails, that particular channel is switched into bypass. (It is noted that the possibility exists for redesigning the redundancy management logic so as to switch the failed pump and use the utility supply.)

The electrical power supplies are configured as presented in Paragraph 2.2.6.1. As is the case with the hydraulic pumps, the dedicated STAR alternators are made an integral part of each parallel channel. If the alternator fails, that particular channel is switched into bypass. (It is again noted that the possibility exists for utilizing the battery for backup.) Since the loss of electrical power disables the IAP electronic monitors (the IAP automatically switches to bypass with loss of voltage to the engage solenoid), the computers (which perform command augmentation, pilot warning functions, etc.) detect the anomaly of the data code from the IAP and provide appropriate warning to the pilot.

4.3.5 Hardover and Jam Considerations

As derived in Paragraph 4.3.1, the probability of a hardover condition in a STAR channel is extremely low:

$$\text{Prob[Hardover]} = 4.53 \times 10^{-8}$$

Since there are five channels, the system probability is five times as large. However, a design provision for such an extremely rare event allows the other four actuators to override the hardover channel through a pressure load-relief valve. (An example of the type of load reaction capability that the IAPs exhibit in such a condition is presented in Table 8.)

As in the case of the dual mechanical system, a jammed actuator could be catastrophic. The IAP is designed in an attempt to provide a jam-proof actuator (see Paragraph 2.2.5.2). One advantage of the STAR configuration over the dual mechanical system is the force advantage that the configuration offers to break a possible jam. In fact, a force of 2.2 times the design load of the IAP will be available from the resultant of the other four normal IAPs to break any potential jam.

5. ADVANCED COCKPIT CONTROLS AND DISPLAYS

The use of fly-by-wire controls for a helicopter allows improvements in cockpit design by the use of new controls and displays. With these improved controls and displays, the pilot can improve performance with greater comfort and reduced workload.

In addition, the use of redundant electronic channels in a fly-by-wire system makes cockpit situation displays and failure management control desirable. These control and display concerns are discussed below.

5.1 Cockpit Control Considerations

A limited effort was directed at two aspects of cockpit controls that are of prime importance in the consideration of fly-by-wire implementations. One aspect is the option of center-mounted versus side-arm-mounted controls. Another aspect is the option of force control versus displacement control.

Table 33 is a summary of trade-offs using center-mounted versus side-arm cyclic control. This summary indicates that many advantages are afforded by the side-arm implementation.

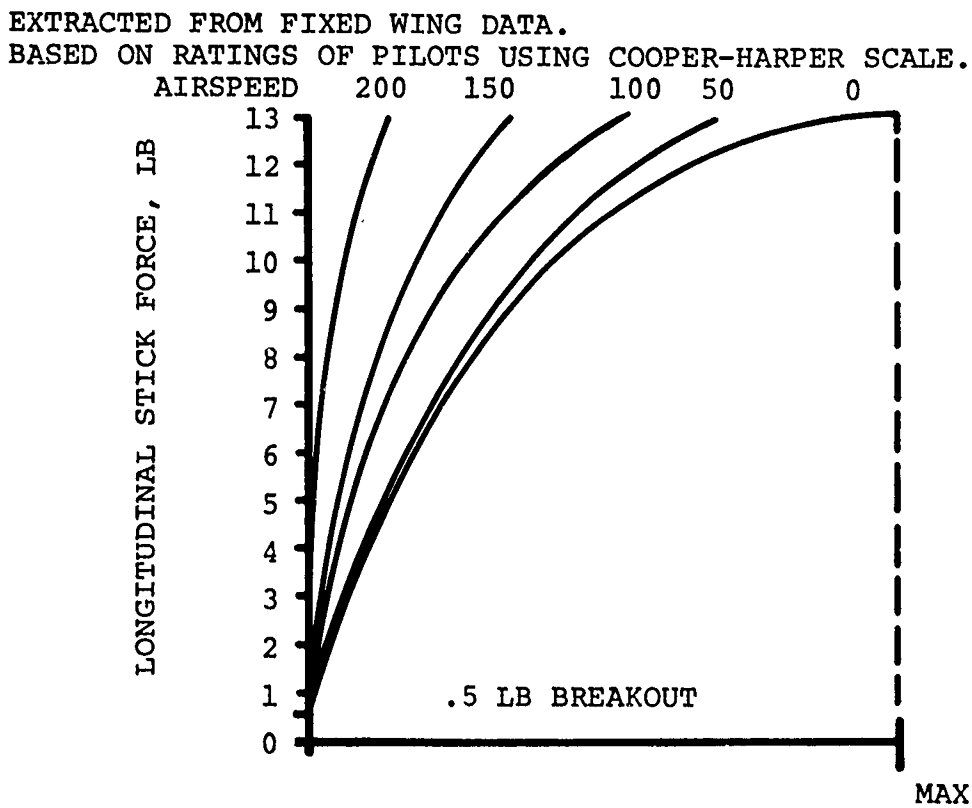
Table 34 is a summary of trade-offs using a force control versus displacement control for cyclic. This summary indicates that a force cyclic control with displacement may offer the most advantages. The displacement could be small (on the order of 25 to 30 degrees total travel) using a 5- to 6-inch-high stick. Figures 35 and 36 indicate desired relationships for control. The displacement would be for control position feedback only and not be part of the actual control loop. The necessity for a copilot control to have the displacement feature has not been established; however, it seems feasible in a system such as an ASH that the copilot, since he is not the primary pilot, could operate from force response only without the displacement feedback. This type of control could be integrated with his system controllers to save space and to be immediately available for use. The final configuration and mechanization of the force rate cyclic would have to be determined through simulator or flight test. In the development of the fly-by-wire control system for the USAF Model F-16, one of the enhancing characteristics of the system was the ability to quickly and easily change the response characteristics of the force stick.

TABLE 33. SUMMARY OF TRADE-OFFS ON CYCLIC STICK POSITION

Center mount (between the legs)	Side arm (with arm rest)
ADVANTAGES	
1. Established standardization.	1. Improved comfort, through reduced fatigue (assumes arm rest required). Less shoulder stretch.
2. Can use both hands.	2. Does not obstruct view to instrument panel.
3. Usually allows easier installation.	3. Less sensitive to "G" loading, turbulence, and vibration.
	4. Stays out of crew-occupied area during crash, thus reducing stick hazards.
	5. Reduced size of displacement envelope allows better cockpit integration.
	6. Aids ingress/egress.
	7. Grip can have more switch functions without causing visual obstruction.
DISADVANTAGES	
1. Hard to design arm rest and gain any comfort. Any adjustments for comfort involve complex mechanical devices.	1. For comfort of various percentile size requires adjustable arm rest.
2. Due to its physical placement, creates problems with head interference in crash attenuating seats.	2. Difficult for left-hand operation.
3. Always have to swing leg over or around for ingress/egress.	3. Limited displacement due to wrist action.
4. Obstructs reach and view to lower center panel area.	4. Arm rest restricts use of right console.

TABLE 34. SUMMARY OF TRADE-OFFS ON USING FORCE VS. POSITION CYCLIC CONTROL

Position - Conventional	Force/Response (Assumes some displacement for controls position feedback)
ADVANTAGES	
1. Historically proven through handling qualities requirements. 2. Provides built-in feedback of controls position.	1. Allows use of side arm controls with reduced arm motion, requires less space. 2. Simplifies force/feel, and trim system. 3. Allows for easier controls response, sensitivity shaping. Allows better controls force feedback for controls limiting on such things as "G" limiting, airspeed limiting, maneuver limiting. 4. Less dead band area or undesirable friction. 5. Less susceptible to accidental movement and to effect of shock and vibration. 6. May allow copilot control to be force stick without position feedback and no interconnect mechanics or electrical between pilot and copilot cyclic controls. 7. Simplifies synchronization of pilot's/copilot's controls.
DISADVANTAGES	
1. Requires large displacement to insure proper controls sensitivity. 2. Require interconnect either through electric motors or mechanical between sticks. 3. Requires artificial force feel trim system for handling qualities improvements on items such as airspeed, gradients, 'g' loading, and maneuver rates.	1. Past experience shows they have been susceptible to PIO (pilot induced oscillation). 2. If required, displacement or position feedback complicates design. 3. Handling qualities requirements do not contain good data base for force/response controls, and flight/simulation data necessary for development systems.



(a) PITCH (RATE)

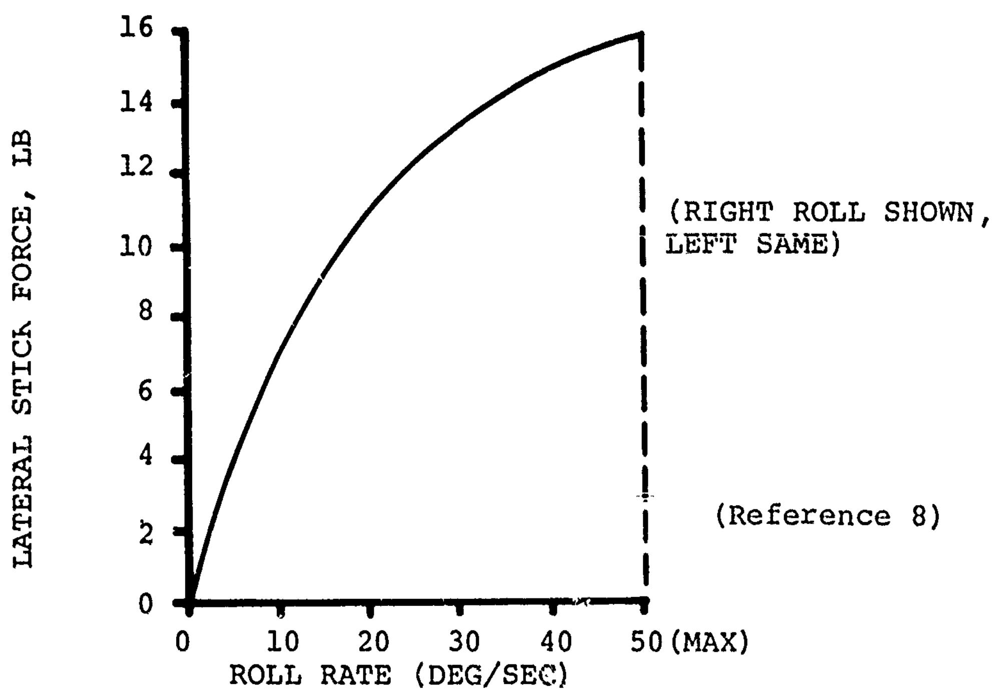


Figure 35. Force characteristics desired for cockpit controls.

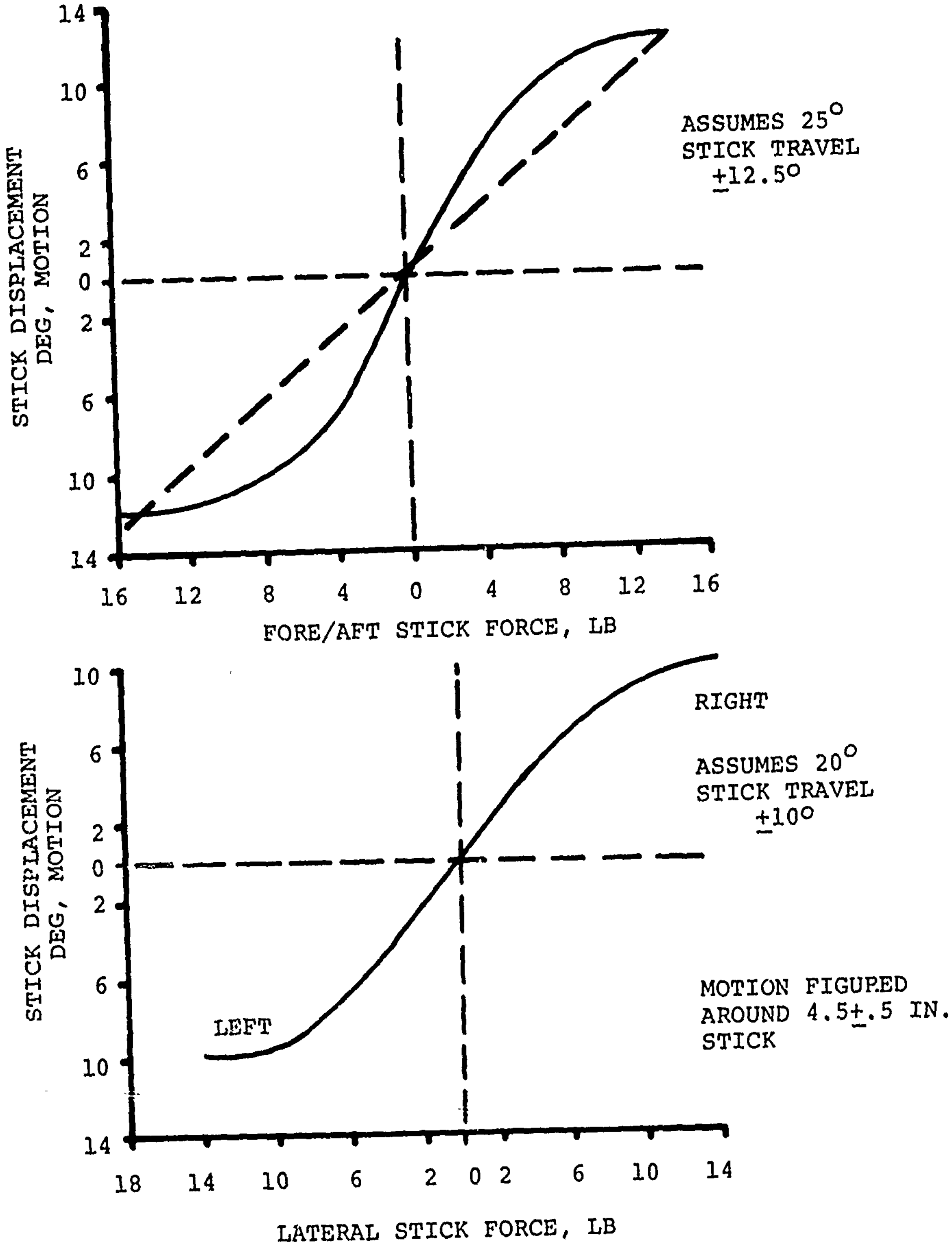


Figure 36. Cockpit controls desired displacement characteristics.

5.1.1 Consideration of Integrated Controls

In the consideration of fly-by-wire cockpit controls, consideration was given to integrated controls, i.e., cyclic and collective operation with one hand.

Many of these types of controls have been built and tested to various degrees of substantiation, but a review of these designs points out that they have either been used for limited, special-purpose control, or they create potentially severe physical coupling problems.

One integrated control concept under consideration in BHT research is shown in Figure 37. Pitch and roll are controlled by conventional displacements; collective is controlled by rotating the "T"-handle. Increased collective pitch is provided by clockwise rotation.

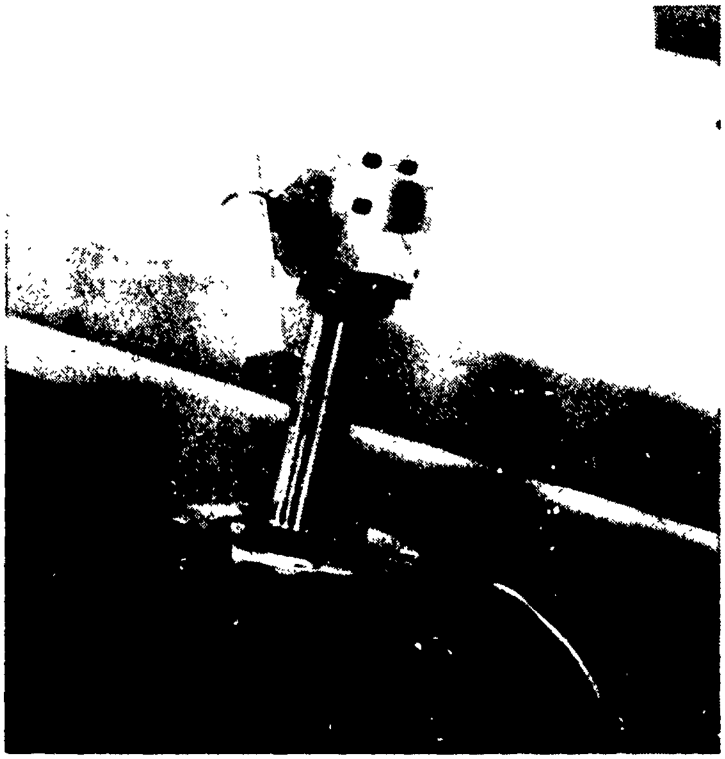
With full authority control for a highly maneuverable air vehicle, such as ASH, the operator coupling problem should be thoroughly studied in simulator or flight test vehicles. If an integrated control becomes accepted and proven, it could easily be adapted to the FBW implementation without major redesign, thus, again pointing up another advantage of FBW/Light control.

5.1.2 Consideration of Side-Arm Controls

Consideration was given to a side-arm controller with conventional left-hand-operated, displacement-type collective input. The controller was configured to implement the desired control response, as shown in Figures 35 and 36.

The side-arm controller is a gimballed short stick approximately six inches long with a standard grip arrangement. The control signals are a function of force on the stick, with the force output gradient varied with velocity to conform to the Army handling qualities specification. The position of the stick will continuously trim as a function of swashplate angle.

The controller utilizes an assembly with five force transducers. The five transducers will be positioned around the stick at azimuth angles corresponding to the five swashplate actuators that will be controlled by the transducer outputs. Each of the five transducers will provide signals to one of the five channels of STAR. The system will be configured to modify the sensor outputs to give a force gradient proportional to velocity. At the low end, the signal per force increment will be large. At the high velocity end, the signal per force increment will be small causing the pilot to be aware of approaching control limits by increased stick reaction.



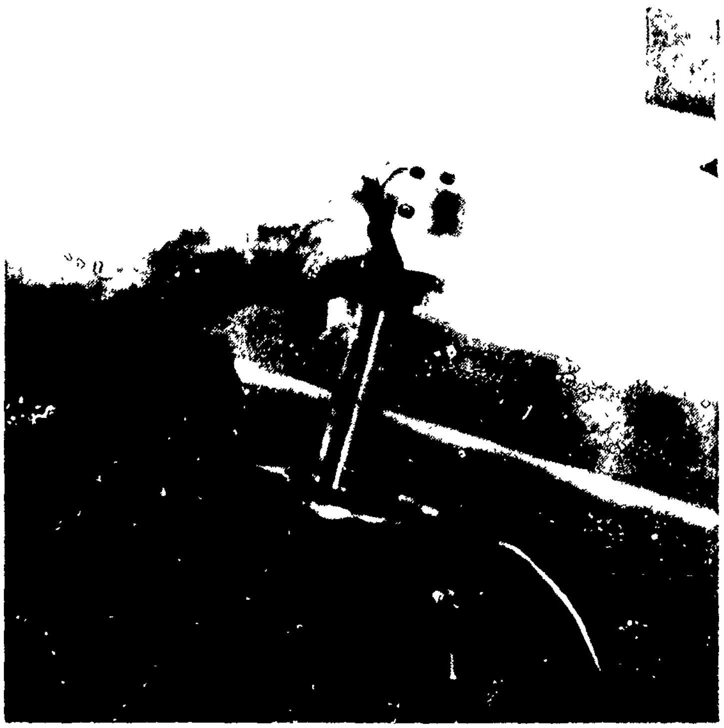


Figure 37. Research projects integrated control stick.

An automatic trim will be accomplished by two rate-limited trim actuators in the pitch-and-roll axis. The trim displacement of the stick will be stick displacement proportional to swashplate angle. The result will be that the pilot will be alerted to the present flight situation and control limits by both displacement and force feedback from the sticks.

Pilot and copilot sticks each have a complete set of sensors and trim actuators. There is no mechanical link between sticks. Force and displacement registration of the sticks are accomplished electronically.

5.1.3 Selection of Control Stick Configuration for Study

The final selection of the control stick configuration for the baseline FBW system for ASH was influenced by the requirement that the technology level used be applicable for a 1980 contract go-ahead. The technology necessary for side-arm displacement sticks would permit its use in this time frame. There are, however, certain questions about the exact implementation of such sticks that should be answered by simulator and/or flight tests. Since it is not known whether time for such evaluations would exist, it was determined that conventional sticks would be used for the comparison study. Tradeoff considerations show that the side-arm force sticks are really the most promising concept and, given time for refinement, would be the preferred stick design.

Figure 13 shows the stick design. Conventional cyclic sticks are used with five passive optical sensors attached. Each of the five sensors are mounted at the same azimuth angle as the corresponding integrated actuator package to which it serves as a signal source. The two cyclic sticks are mechanically coupled to move together. The collective sticks that are also mechanically interconnected have an input to a mechanical mixer that mixes the collective signal with cyclic for each sensor so that the output of each passive optical sensor is the summed cylic and collective for that channel.

The antitorque pedal controls are conventional. The interconnected pilot and copilot controls are attached to three passive optical sensors. The output of each of the three sensors goes to one of the antitorque fly-by-wire channels.

5.2 DISPLAYS

The crewstation display associated with the FBW/L system is a result of information requirements for the preflight and safe in-flight operations of the system. Although the system will have extensive self-test and diagnostic capability, along with its unique fail-safe, multiple-channel operation, certain cockpit information will be required. The requirements are

based on a Hazard Modes and Effects Analysis (HMEA) and the need for certain information to insure proper preflight checkout. An indication of IAP status is necessary in order to ascertain whether an abort situation is occurring. Without such indication, flight safety is significantly reduced.

Actual control display for the FBW/L control system is considered to be integrated with the multifunction display system and coordinates under dual multiplex bus/computer supervision, as discussed in Paragraphs 2.2.6.4 and 2.2.6.5. This approach will provide much more flexibility for test checkout and diagnostics. The multifunction, or CRT-type, display will be programmed to allow checkout and status as a function of keyboard access data. Procedures for checkout, as well as actual test data, will be easily displayed to the flight crew or maintenance crew for preflight, in-flight, or postflight use. A typical example of how a system of this type could be used for a preflight checkout is as follows:

- Crew power-up displays and bus system
- Access data key for "control system check"
- This results in paging of data showing all channels in "bypass" and why. (This is due to no hydraulic power to enable second-stage EHSV spool, thus, tripping monitor.) If all channels do not indicate bypass, the monitor system has a fault.
- Selects "AUX PUMP PWR" and "SEQUENCE IAP" on the display page. This powers the auxiliary hydraulic pump and develops hydraulic pressure. The sequence then commands each IAP into an auxiliary pump mode.
- The IAP in auxiliary pump mode will remove the bypass condition if the "Sequence IAP" is successful.
- The system is now operable for starting, and the other aircraft preflight procedures can be continued.

After the rotor is turning, the system initiates a powered-up check and will indicate any bypass condition for the primary hydraulic actuators.

6. WEIGHT IMPACT EVALUATION

The performance of a helicopter is determined to a large degree by its gross weight. Gross weight is determined by its constituent weights - empty weight, fixed useful load, payload, and fuel. Various combinations of these constituent weights can obviously result in the same gross weight. Performance for the subject 9544 gross weight vehicle is depicted in Figures 38 through 41. From Figure 42, it is seen that the mission range for a sea level, standard day condition is approximately 100 nautical miles.

It was shown in Paragraph 3.2 that the FBW/L flight control system was 143 pounds lighter than the dual mechanical flight control system. Advantage of this weight savings may be taken by increasing fuel capacity or by increasing the payload. Empty weight of the ASH mission dual mechanical MUT was increased by 83 pounds over the original MUT. To maintain the original gross weight and fuel capacity, the payload was reduced from 960 pounds to 877 pounds. The trade-off of the 143 pounds weight savings of the FBW/L system mechanization can be visualized from Figure 42. If the same fuel capacity is maintained, then 143 pounds additional payload may be carried for the 100-nautical-mile mission. If the payload is maintained, then the added fuel allows a 15-percent increase in mission range.

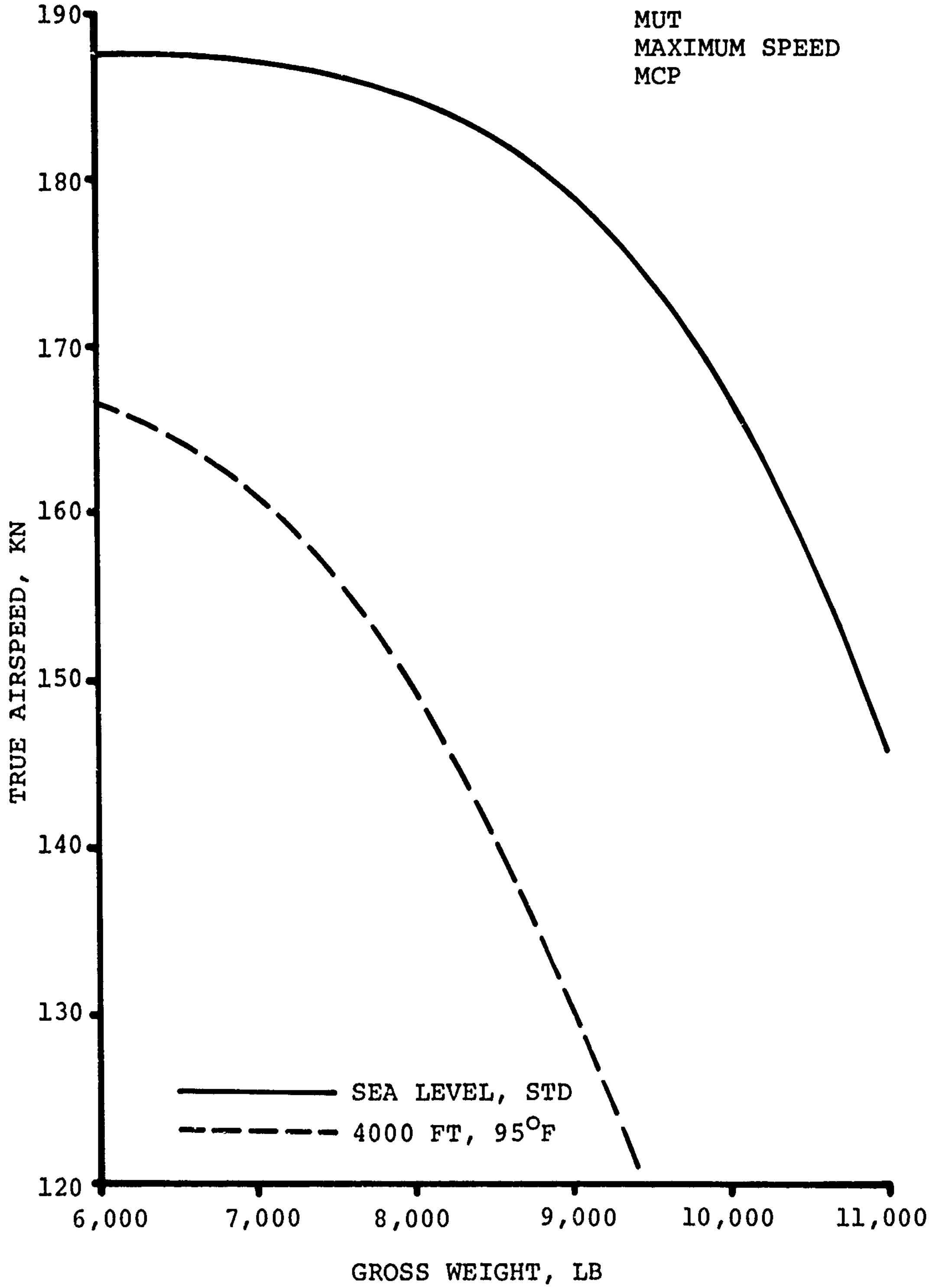


Figure 38. MUT maximum speed vs GW.

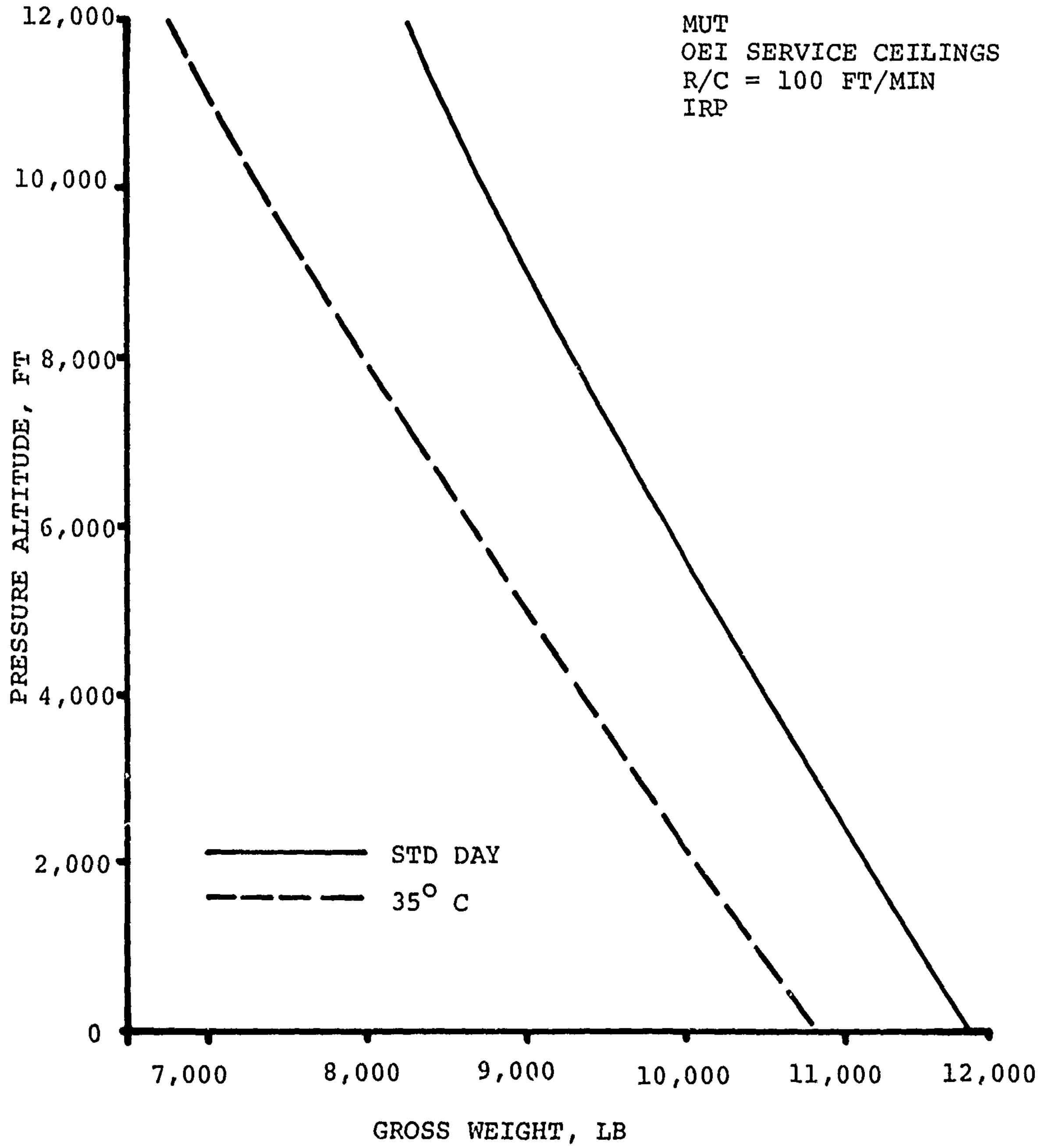


Figure 39. MUT service ceiling vs GW.

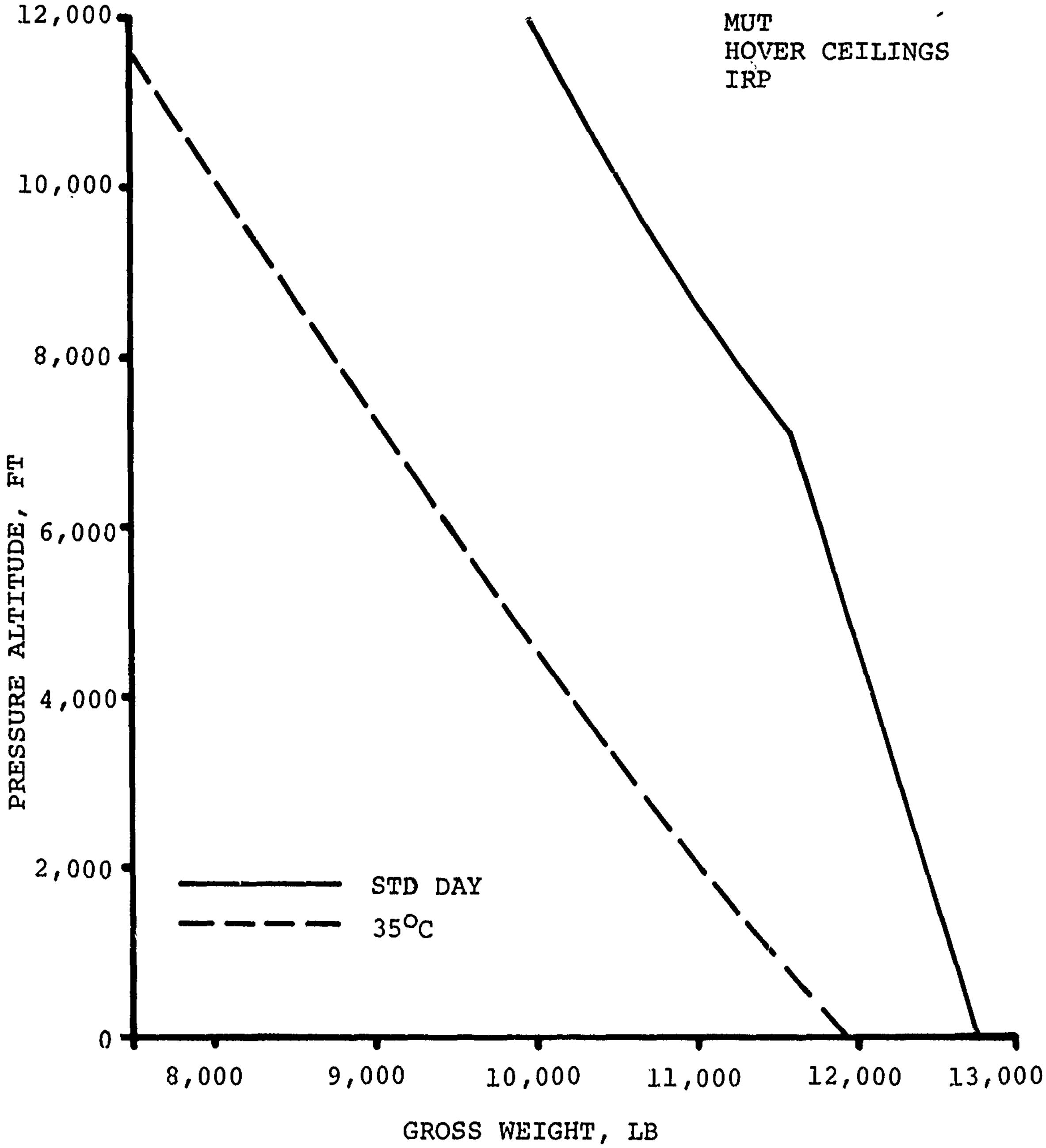


Figure 40. MUT hover ceiling vs GW.

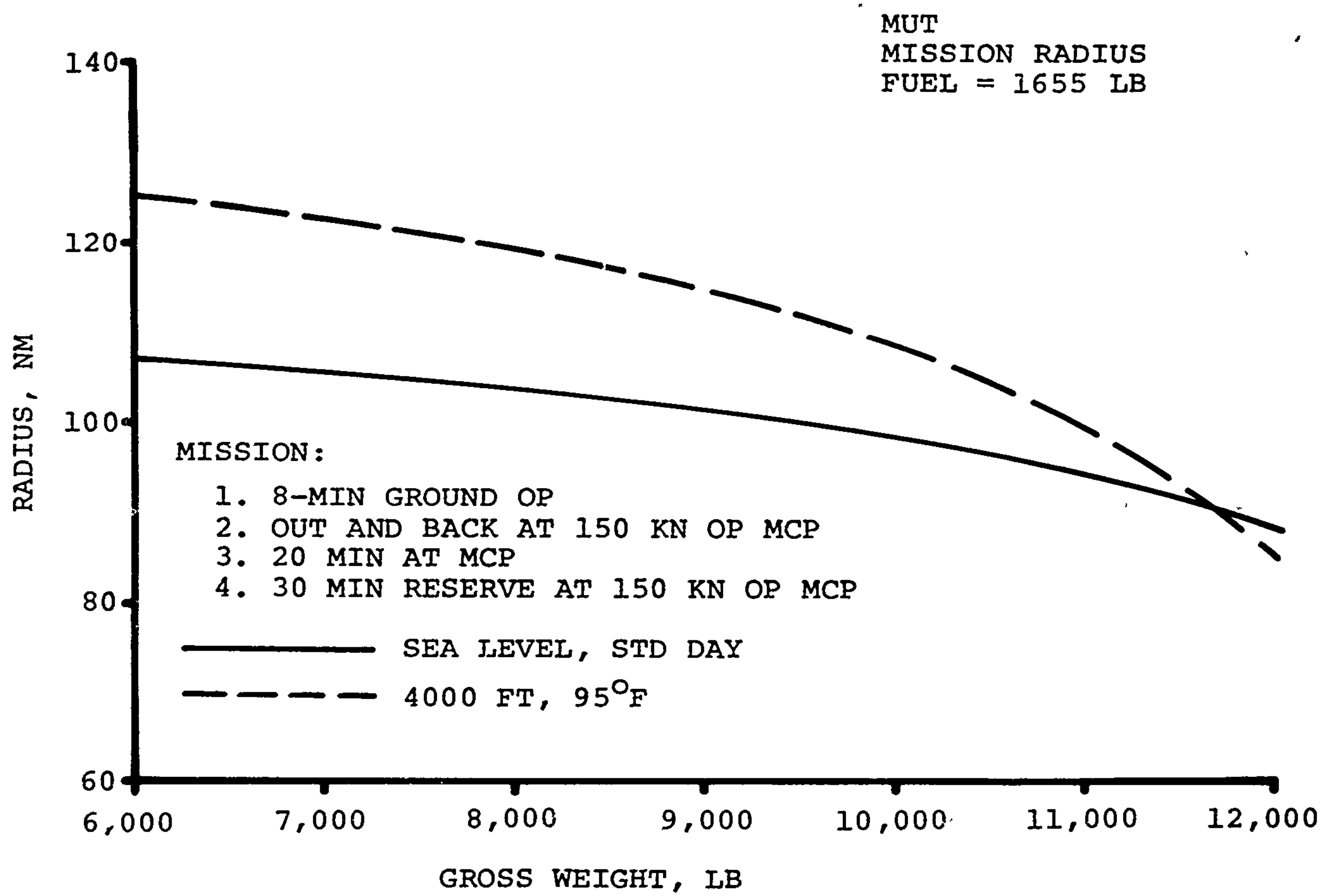


Figure 41. MUT mission radius vs GW.

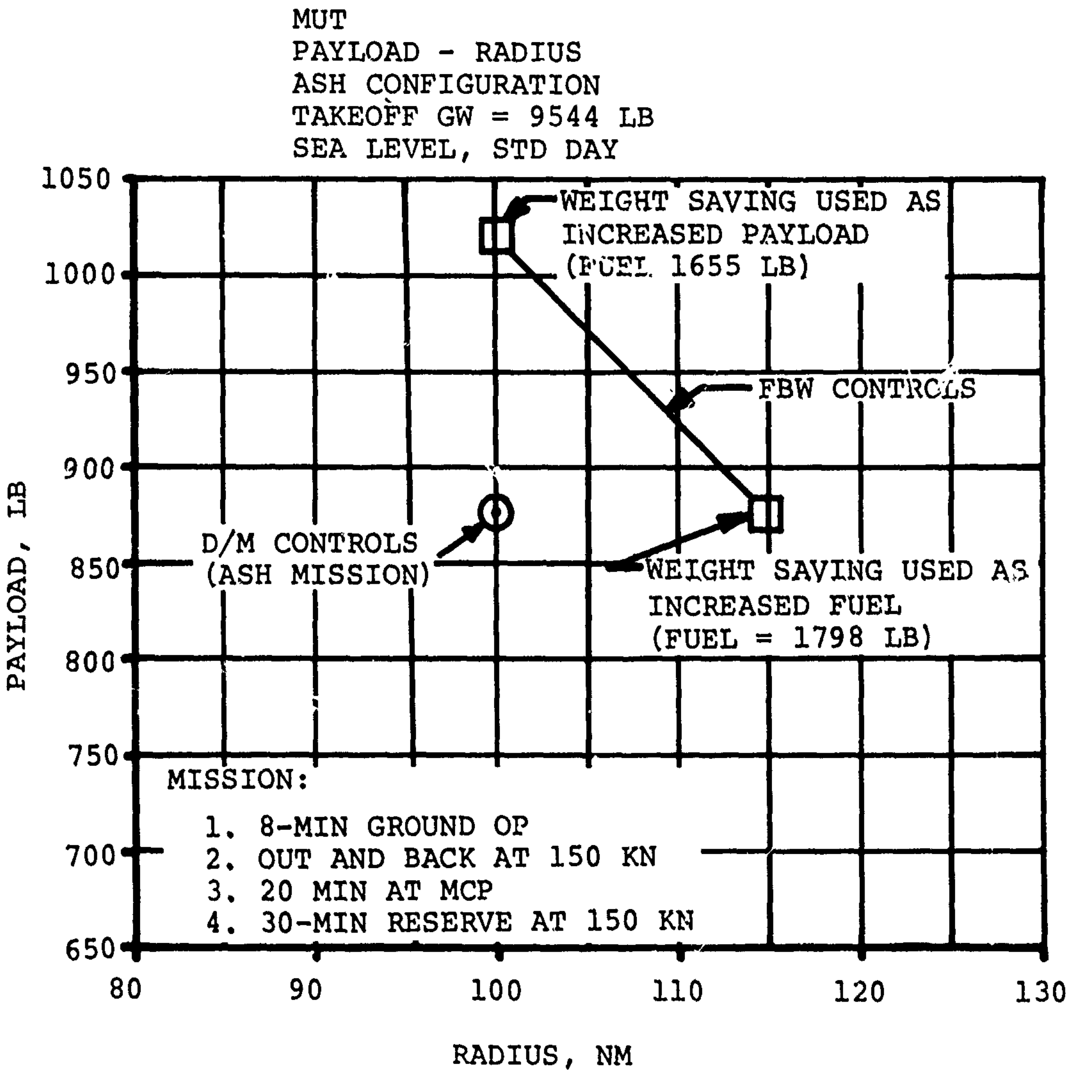


Figure 42. MUT payload versus radius.

7. CONCLUSIONS

Conclusions drawn from this study comparing a FBW/L control system with a dual mechanical control are:

- Significant payoffs are predicted when applying fly-by-wire/light technology to helicopter flight control systems.

- The flight safety goal (no more than one catastrophic failure per 10^7 hours of operation) can be achieved by both the dual mechanical system and the FBW/L control system. The dual mechanical system is predicted to be approximately three orders of magnitude better than the FBW/L control system. (Although not detailed here, it can be shown that the FBW/L control system is predicted to be approximately two orders of magnitude better than a single mechanical control system.)

- The mission failure goal (no more than one failure per 10^4 hours of flight) can be achieved by the FBW/L control system (provided the cockpit control linkage to the control transducers is so designed that its failure rate contribution to mission failure is negligible). This goal cannot be met by the dual mechanical control system.

- The system MTBF goal of 2500 hours cannot be met by either the dual mechanical or the FBW/L control system. However, the FBW/L system is approximately 28 percent better than the dual mechanical system.

- Both dual mechanical and FBW/L control systems have low vulnerable areas to the 12.7mm API threat with the FBW/L system exhibiting 0.26 square feet less area. Both systems have good survivability to the 23mm HEI-T but the FBW/L control system is slightly worse due to the vulnerability of the control motion transducers to blast and fragmentation damage. (This could be eliminated by adding 5 to 10 pounds of armor plate or by the use of side-arm force controllers.)

- Maintainability characteristics of the FBW/L control system are superior to those of the dual mechanical system. Organizational corrective maintenance for the FBW/L system is approximately one-third that of the dual mechanical system. Daily preventive maintenance is approximately one-half. The FBW/L system is, thus, more available.

- The weight of the FBW/L control system (with conventional cockpit controls) is 143 pounds lighter than the dual mechanical system. This weight differential may contribute to either increased mission range (fuel) or increased payload.

- FBW/L control system life-cycle cost savings in constant 1979 dollars amounts to approximately $0.2 Billion. The larger nonrecurring tooling and production engineering cost for the dual mechanical system offsets the extra cost of nonrecurring vendor development and qualification tests and flight testing of the FBW/L system. The other significant cost savings result from the reduced maintenance cost associated with the modularized FBW/L system with built-in test capability.

- Maximum benefit from FBW/L technology is not obtained unless advanced cockpit control and display concepts are utilized.

- The 5-arm rise/fall swashplate main rotor swashplate control system (STAR) is applicable to any helicopter.

- FBW/L systems for helicopters are considered within the state-of-the-art for a development program. Much of the system presented here has undergone laboratory and Iron Bird testing.

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