NASA CR-152086

CONCEPTUAL DESIGN STUDY OF A HARRIER V/STOL RESEARCH AIRCRAFT

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FEB 1978

Prepared Under Contract No. NAS2-9748 by
McDonnell Douglas Corporation
Saint Louis, Missouri

for

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

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CONCEPTUAL DESIGN STUDY OF A HARRIER V/STOL RESEARCH AIRCRAFT

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TABLE OF CONTENTS

TABLE OF CONTENTS (CONT'D)

TABLE OF CONTENTS (CONT'D)

1. INTRODUCTION AND SUMMARY

MCAIR recently completed a conceptual design study to define modification approaches to, and derive planning prices for the conversion of a two place Harrier to a V/STOL control, display and guidance research aircraft. This study was performed for NASA Ames Research Center under the NASA contract "Conceptual Design Study of Modifications to Harrier G-VTOL" (NASA Contract NAS2-9748). The statement of work for this program is contained in Appendix A.

NASA's objective is the flight investigation of control systems that have been developed by analysis and simulation. Satisfactory handling qualities for all weather VTOL operation both shipborne and landbased, are to be sought in transition, approach, landing and takeoff. Control concepts such as rate damping, attitude stabilization, velocity command, and cockpit controllers are to be demonstrated. Display formats will also be investigated, and landing, navigation and guidance systems flight tested. And, of course, all this is to be done in a safe airplane.

This MCAIR study is an early step in a long term NASA program. The proposed test bed aircraft is to fly in 1981 so that NASA can spend the eighties developing the control and display technology for the future. The obvious application is to the U.S. Navy Type A and Type B V/STOL which have an IOC in the early nineties. Thus, MCAIR was asked to define and schedule modifications to a British civil registry Harrier (G-VTOL) that would permit this testing. The resulting program schedule is shown in Figure 1-1.

The modifications defined leave the front cockpit of the Harrier virtually untouched. Thus, the safety pilot, flying in this cockpit is flying an unchanged two place Harrier. The rear cockpit is modified such that it can be quickly adapted to faithfully simulate the controls, displays and handling qualities of a Type A or Type B V/STOL. The safety pilot always has take command capability.

The modifications studied fall into two categories: basic modifications and optional modifications. The basic modifications include a simplex parallel digital fly-by-wire flight control system, a data acquisition system, a headup display, a landing guidance system, simulation, system software, ground tests, and airworthiness tests. The optional modifications include a duplex series digital fly-by-wire flight control system, throttle and nozzle control systems (simplex parallel, duplex series, and triplex parallel), a low speed air data system, and a side arm controller.

Technical descriptions of the basic modifications are contained in Section 2. Technical descriptions of the optional modifications are contained in Section 3. The modification plan and schedule as well as the test plan and schedule are presented in Section 4. The failure mode and effects analysis, aircraft performance, aircraft weight, and aircraft support are discussed in Section 5.

The task descriptions contained in Appendix B were prepared to aid in deriving the planning prices needed by NASA. (Those cost estimates were provided to NASA under separate cover.) Appendix C presents the detailed failure mode and effects analysis that was performed. Appendix D summarizes the disposition of action items accumulated at four coordination meetings with NASA.

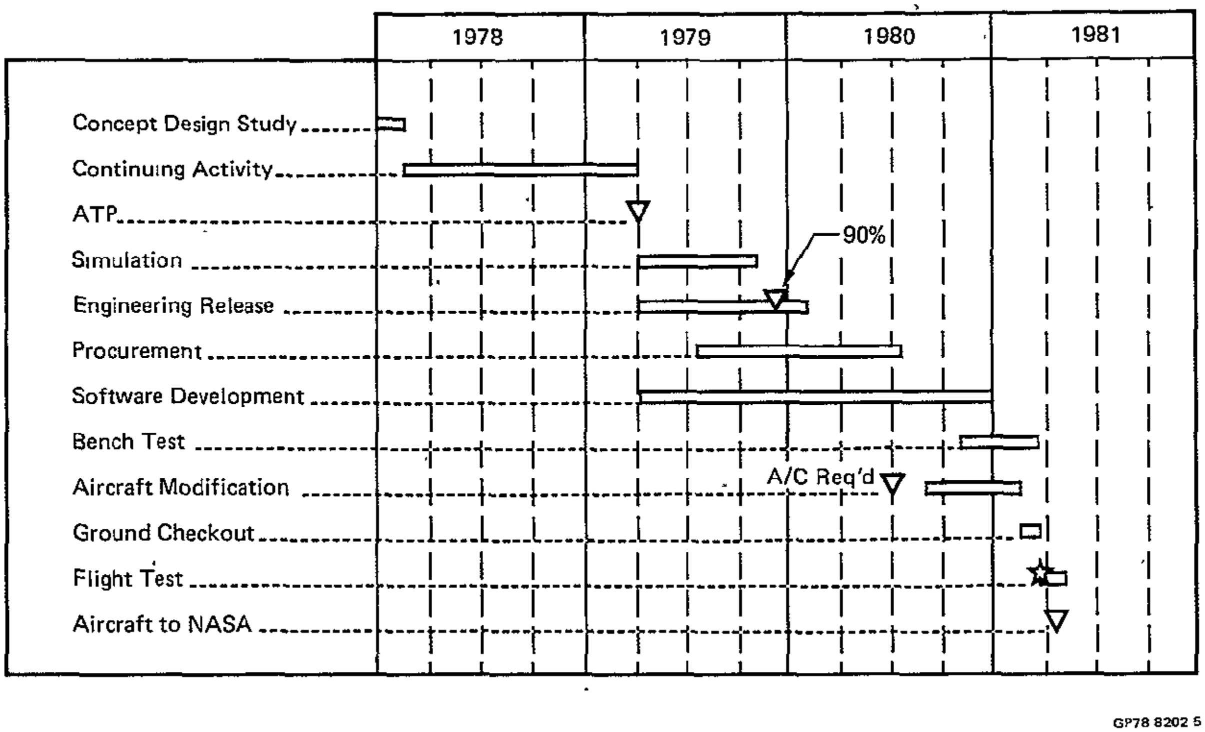

FIGURE 1-1 NASA RESEARCH AIRCRAFT

MCAIR strongly believes that the NASA Two Place V/STOL Research Aircraft will be an extremely valuable research tool for studying a variety of problems associated with V/STOL flight. We are anxious to participate with NASA in efforts aimed at making this research aircraft a reality.

2. BASIC MODIFICATIONS

The basic modifications to the two place Harrier which were investigated in this conceptual design study were simplex digital computer flight controls for the aft cockpit, a flight test data acquisition system, a programmable head up display, a landing guidance system, manned flight simulation, software for the onboard computer, ground tests of the modified aircraft and equipment, and airworthiness tests of the aircraft before delivery to NASA.

2.1 SIMPLEX FLIGHT CONTROL SYSTEM

The flight control system design modifications were established on the basis that the rear cockpit will be the evaluation pilot station and the forward cockpit will be the safety pilot and solo pilot station. Sufficient information will be provided in the front cockpit to enable the safety pilot to monitor the activity of the evaluation pilot and of the digital flight control systems and to disengage these systems should the necessity arise. The front seat pilot would then fly the aircraft using an essentially unmodified Harrier flight control system.

2.1.1 SIMPLEX FLIGHT CONTROL SYSTEM DESCRIPTION - The aft cockpit's control stick and rudder pedals are mechanically disconnected from the Harrier flight control system and electrical position and force transducers substituted to provide pilot command inputs. Electrical torque motors provide artificial feel. An onboard digital computer will be used to compute pitch, roll and yaw commands to the servo actuators which will be attached to the mechanical linkages of the Harrier longitudinal, lateral and directional flight control systems. It will also supply command signals to the aft control stick and rudder pedal variable feel systems.

Figure 2-1 is a schematic diagram of the longitudinal control system. An electromechanical servoactuator is attached to the longitudinal mechanical control mechanism at the tailplane compensator. This is a parallel servo since it moves all of the mechanical controls, including the front cockpit stick, as well as the valve on the stabilator actuator. The limited authority series servo in the Harrier stabilator actuator will be retained and an electromechanical series servo actuator will be added to the forward reaction control valve (RCV). The series servo design for this RCV will be based on that developed for the YAV-8B (the forward RCV of the production Harrier is not servoed). Note that the series servos will not move the forward cockpit control stick.

When the digital fly-by-wire control system is turned off, the safety pilot has the standard Harrier flight control system. (The drag of the parallel servo will be low and will not affect handling qualities.) He can then fly the aircraft with or without the production stability augmentation system.

When the digital flight control system is turned on, the evaluation pilot's stick force and stick position are measured by the transducers connected to the stick and applied to the digital computer. Using these signals and other information, such as aircraft angular rates, accelerations and air data, the digital computer calculates the signals to send to the parallel servo. In special situations, small-amplitude high-frequency signals could also be sent to the series servos. The computer would also use these signals to compute the signal to send to the aft cockpit stick's variable feel system.

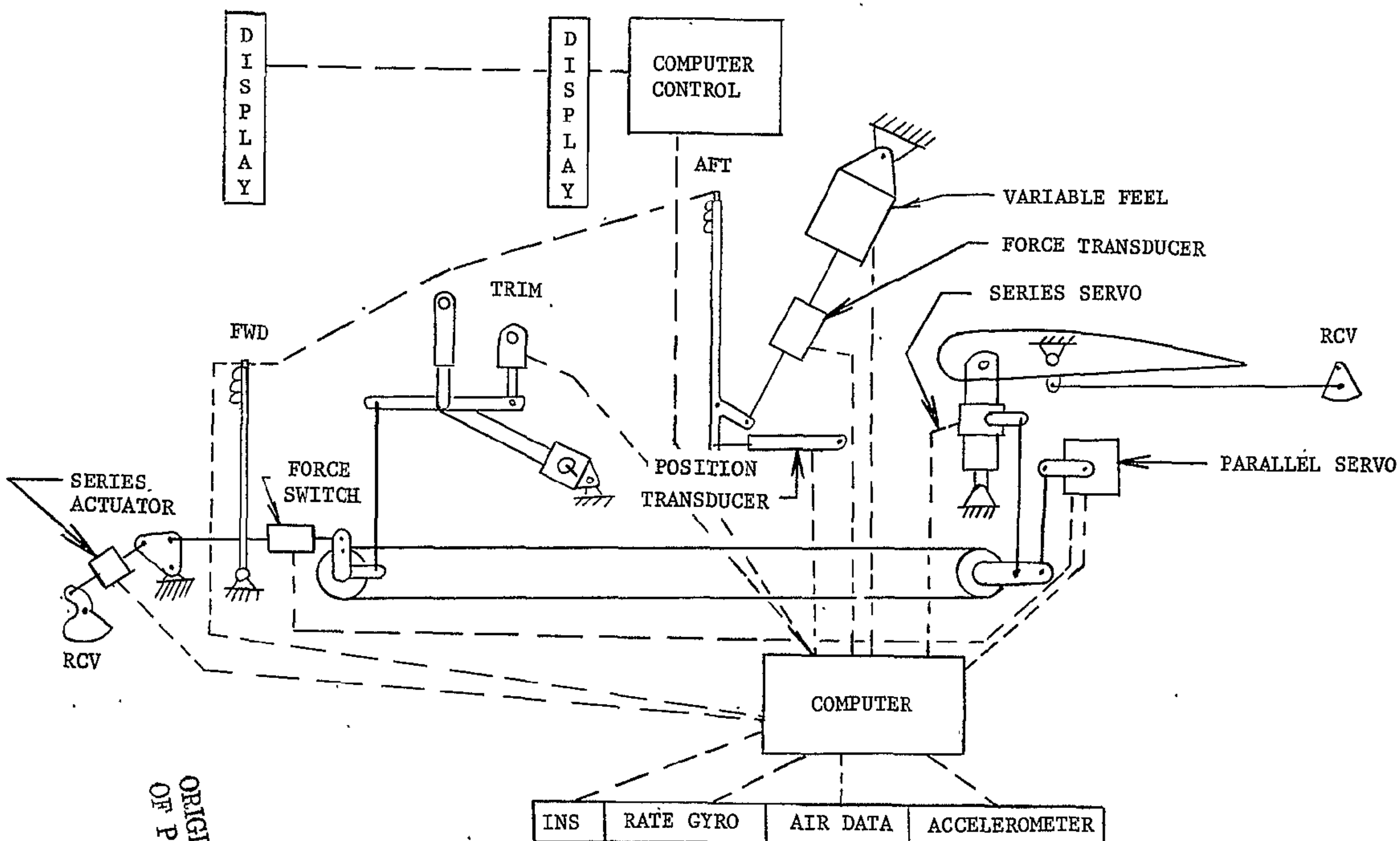

FIGURE 2-1 LONGITUDINAL CONTROL SYSTEM SIMPLEX PARALLEL

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The forward pilot can monitor the position of the stabilator and RCVs commanded by the parallel servo by monitoring the activity of his control stick. If he deems it necessary, he can press a button on the stick grip or throttle and immediately disengage the fly-by-wire flight control system. Alternatively, he can apply sufficient stick force to cause the force disengage switch attached to the stick to disengage the parallel servo, and if this force link fails, he can manually override the parallel servo. A master switch on his control panel can also be used to switch off the system.

Longitudinal trim is accomplished by feeding trim signals from the aft cockpit through the computer and into the parallel servo. The normal system trim actuator is slaved to the computer so that the trim position of the normal trim actuator is the same as the trim position of the parallel servo.

The technique for implementing the variable feel system for the aft stick is shown in Figure 2-2. The force transducer (strain gage) on the aft stick provides stick force information to the computer. The computer then computes the stick position which would give the desired stick-force stick-displacement characteristic. This desired stick position is compared to the actual stick position measured by the stick position transducer (RVDT) and the resulting error signal used to position the stick. This technique is widely used at MCAIR and throughout the industry to provide variable feel systems for manned simulators.

The main component of the electromechanical parallel servo actuator is a two-phase servo motor equipped with a gear train to reduce velocity and increase torque. An RVDT is used for position feedback and a tachometer is used to provide rate feedback for servo stabilization. Brakes are not used so that when the servo actuator is electrically disengaged it rotates freely. When the pilot overpowers the servo actuator, the force generated by the pilot is proportional to the servo motor stall force.

Since the parallel servo has to move the entire longitudinal mechanical control system, an analysis was performed to estimate the longitudinal servoactuator frequency response. Longitudinal mechanical control system dynamics data developed during the YAV-8B forward servo RCV design activities along with a nonlinear model of the electrohydraulic servoactuator were used in the frequency response calculations. The resulting stabilator valve position/servo command frequency responses are shown in Figure 2-3. It can be seen that the frequency response depends on the amplitude of the input signal. It was found that the bandpass was determined by the force level at which the pilot could override the servo. A pilot override force of 26 pounds was used for this study. Increasing the pilot override force level would increase the bandpass. If the frequency response capability of the parallel servo is found to be inadequate for certain closed loop control configurations, the computer can be programmed to use the series servos to increase the effective servo bandpass.

Figure 2-4 shows response to the forward cockpit stick/series servo input frequency response and the series servo output/series servo input frequency response. These frequency responses are almost identical to about 4 Hz. Therefore, the motion of the forward stick should give the safety pilot a good indication of the stabilator and RCV positions which the digital computer is commanding.

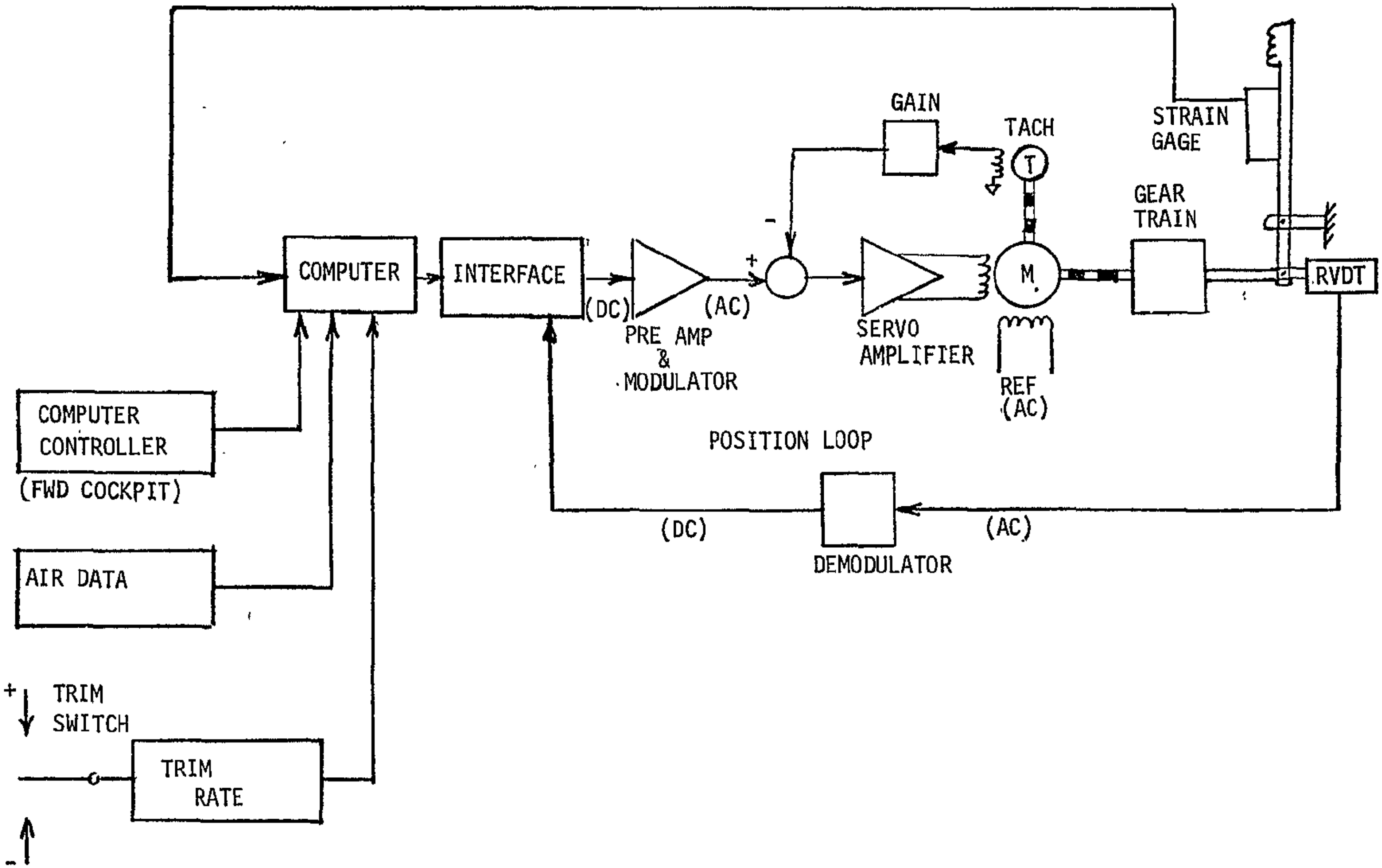

FIGURE 2-2 FEEL SYSTEM - AFT COCKPIT

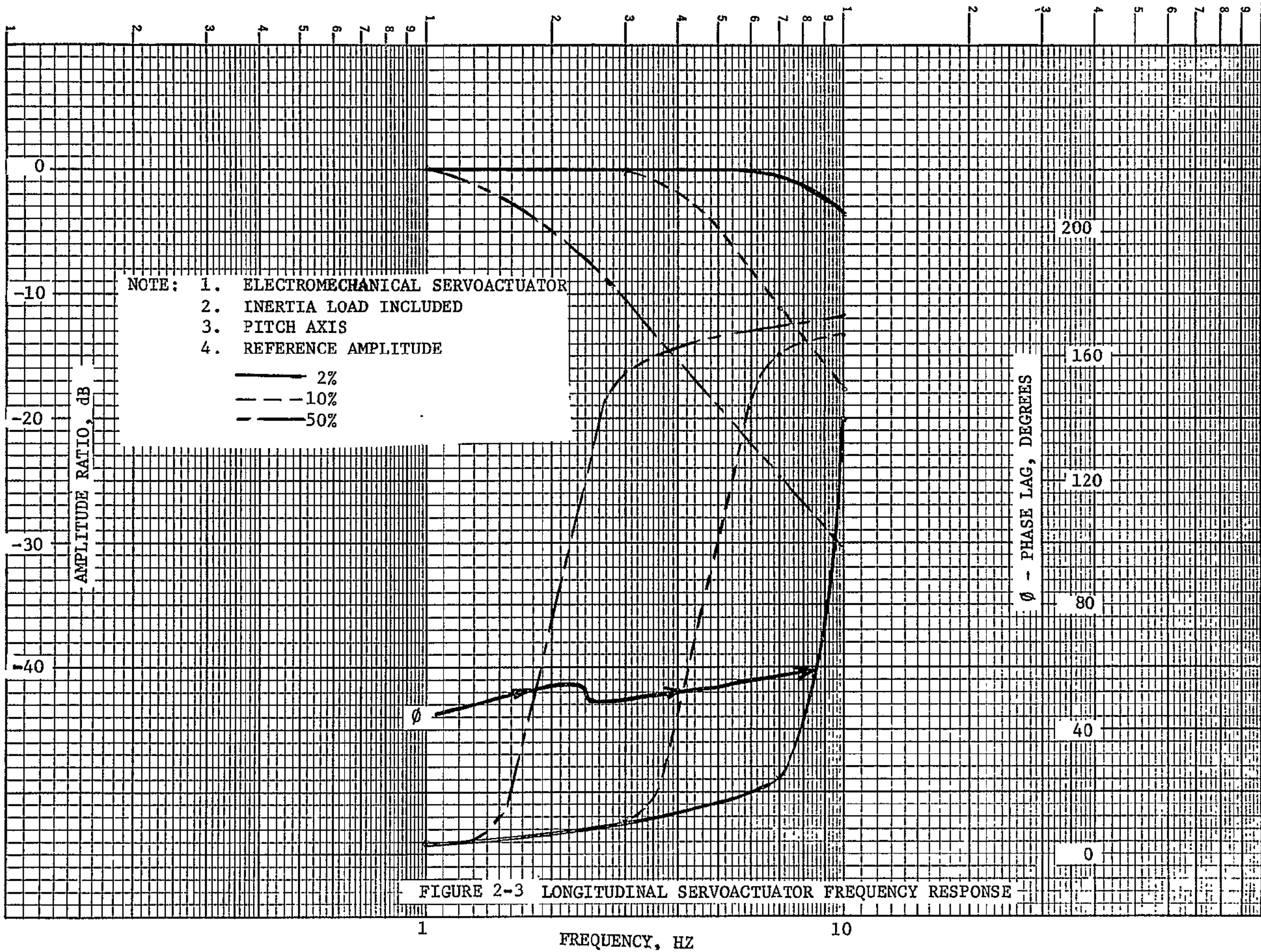

FIGURE 2-3 LONGITUDINAL SERVOACTUATOR FREQUENCY RESPONSE

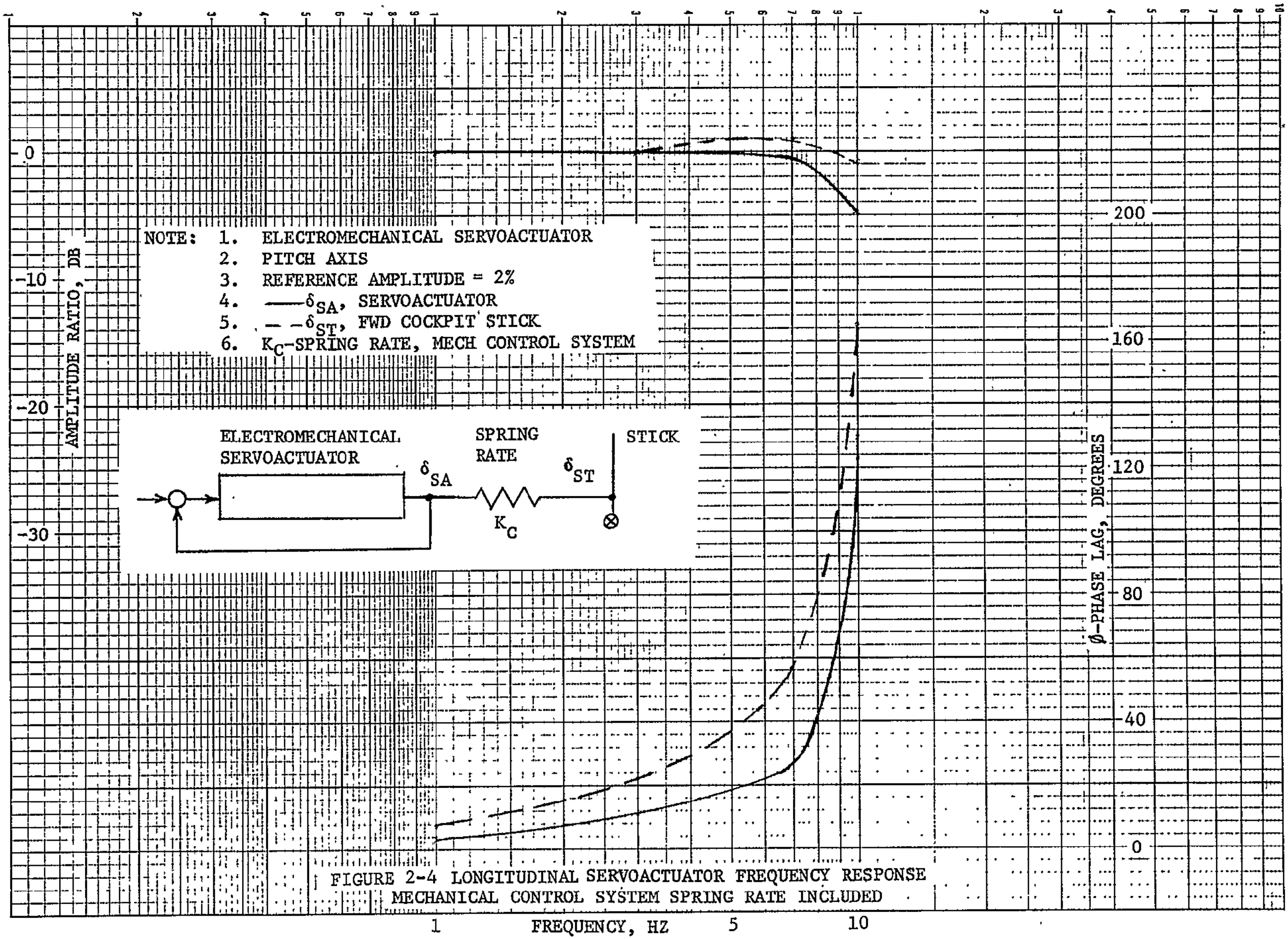

FIGURE 2-4 LONGITUDINAL SERVOACTUATOR FREQUENCY RESPONSE MECHANICAL CONTROL SYSTEM SPRING RATE INCLUDED

The lateral control system is shown in Figure 2-5. In concept, it is identical to the longitudinal system. Aft cockpit stick force and stick position signals are sent to the computer which computes commands for the parallel servo which has been added to the lateral system. The parallel servo drives the lateral control system mechanism, including the forward cockpit stick and servo valves on both ailerons. The roll RCVs are mechanically connected to the ailerons. Forward cockpit disengage includes stick and throttle disengage buttons, stick force electrical disengage, and force override of the parallel actuator. The limited authority series servos of the production Harrier aileron actuators can be used to augment the parallel servo's frequency response capability if necessary. Trim is accomplished in the same manner as in the longitudinal system.

The directional control system is shown in Figure 2-6. In concept it is similar to the longitudinal and lateral systems. However, the rudder of the Harrier is not powered. Therefore, the parallel servo will experience the rudder airloads. Since the pilot must be able to override the parallel servo driving the rudder, airloads will also be able to override the servo so the rudder will be hinge moment limited during fly-by-wire flight. Figure 2-7 gives the rudder hinge moment versus airspeed for rudder deflections of 5°, 10°, and 15°. Since a rudder force of 100 pounds is equivalent to 812.5 inch-pounds of rudder hinge moment, it can be seen that the selected override force value of 100 pounds will provide full rudder displacement of 15° for airspeeds to approximately 160 KEAS.

The yaw puffer of the production Harrier is servoed and the production yaw puffer servo will be available to augment the frequency response capabilities of the parallel servo. The method used for directional trim will be the same as used for longitudinal and lateral trim.

Figure 2-8 gives the pitch, roll and yaw parallel servo actuator override forces which the safety pilot must exert if his electrical force disengage does not function. The electrical disengage force loads, which will be well below these override force levels, will be determined in manned simulations.

The maximum rates of stabilator actuator and aileron actuators are 60 degrees/second and 80 degrees/second, respectively. The parallel actuators would be sized to have a maximum rate capability about 10% higher than these values. It is expected that the rudder actuator will have a rate capability of approximately 45 degrees/second.

Figures 2-9 thru 2-11 show the longitudinal, lateral and directional flight control systems installed in the aircraft. The installations of the longitudinal and directional parallel servos are shown in Figure 2-12. The longitudinal parallel servo is connected at the tailplane compensator. The directional parallel servo is connected at the rudder compensator. Spring cartridges are included so that the safety pilot will be able to fly the aircraft in the event of a parallel servo jam.

The installation of the lateral parallel servo is shown in Figure 2-13. This servo is attached to the front spar of the wing. A spring cartridge enables the safety pilot to fly the aircraft if the parallel servo jams.

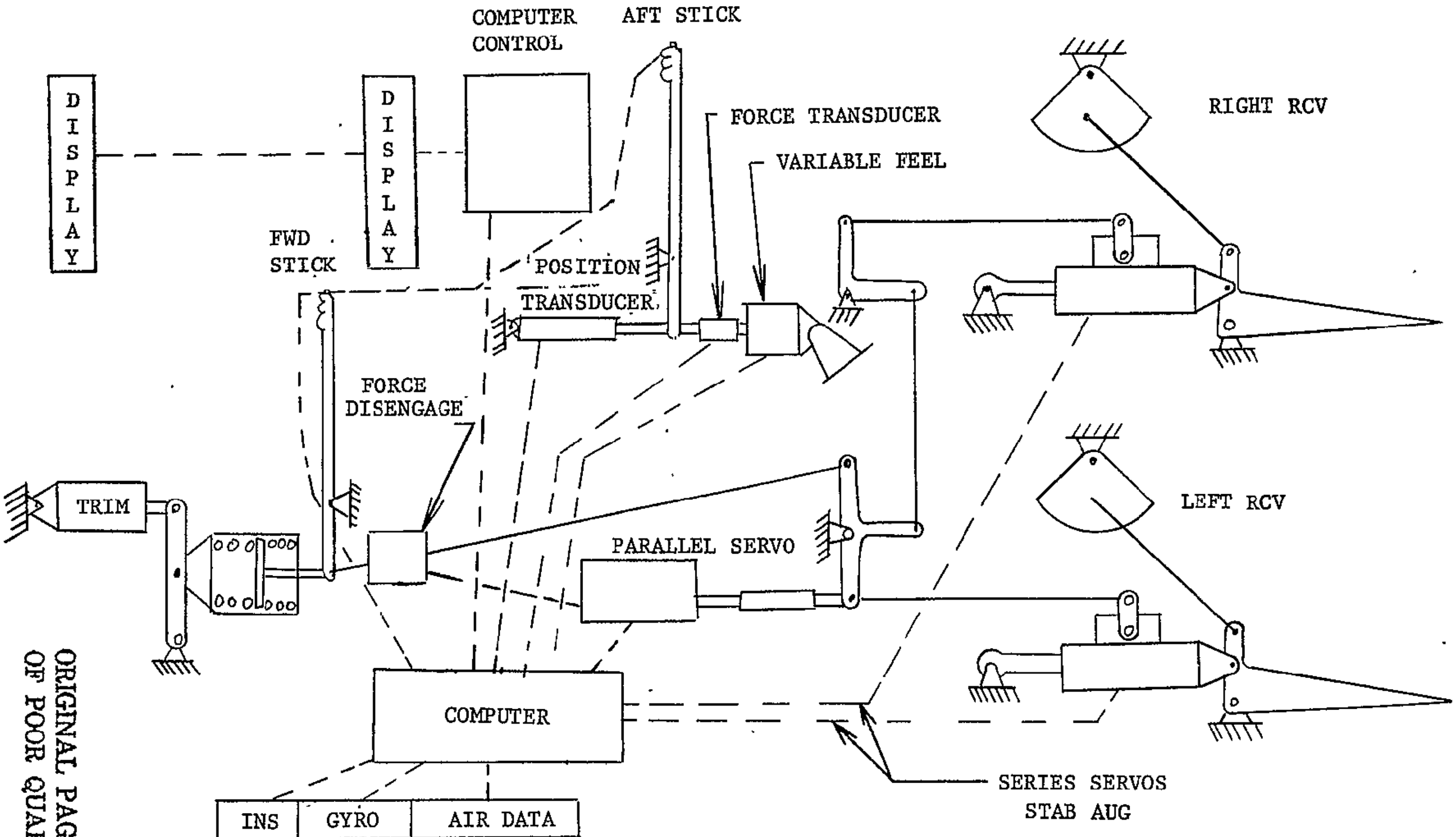

FIGURE 2-5 LATERAL CONTROL SYSTEM, SIMPLEX PARALLEL

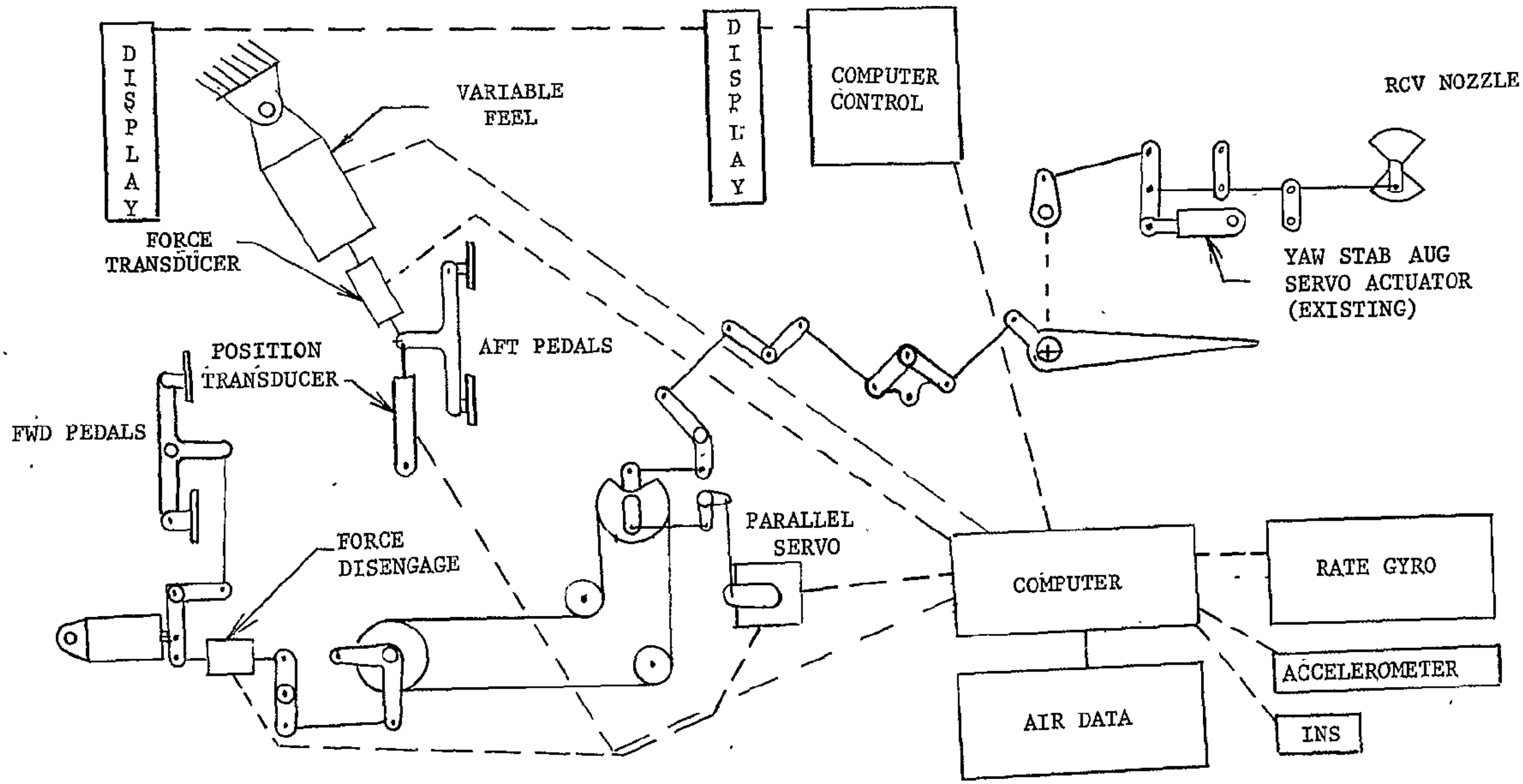

FIGURE 2-6 DIRECTIONAL CONTROL SYSTEM
SIMPLEX PARALLEL

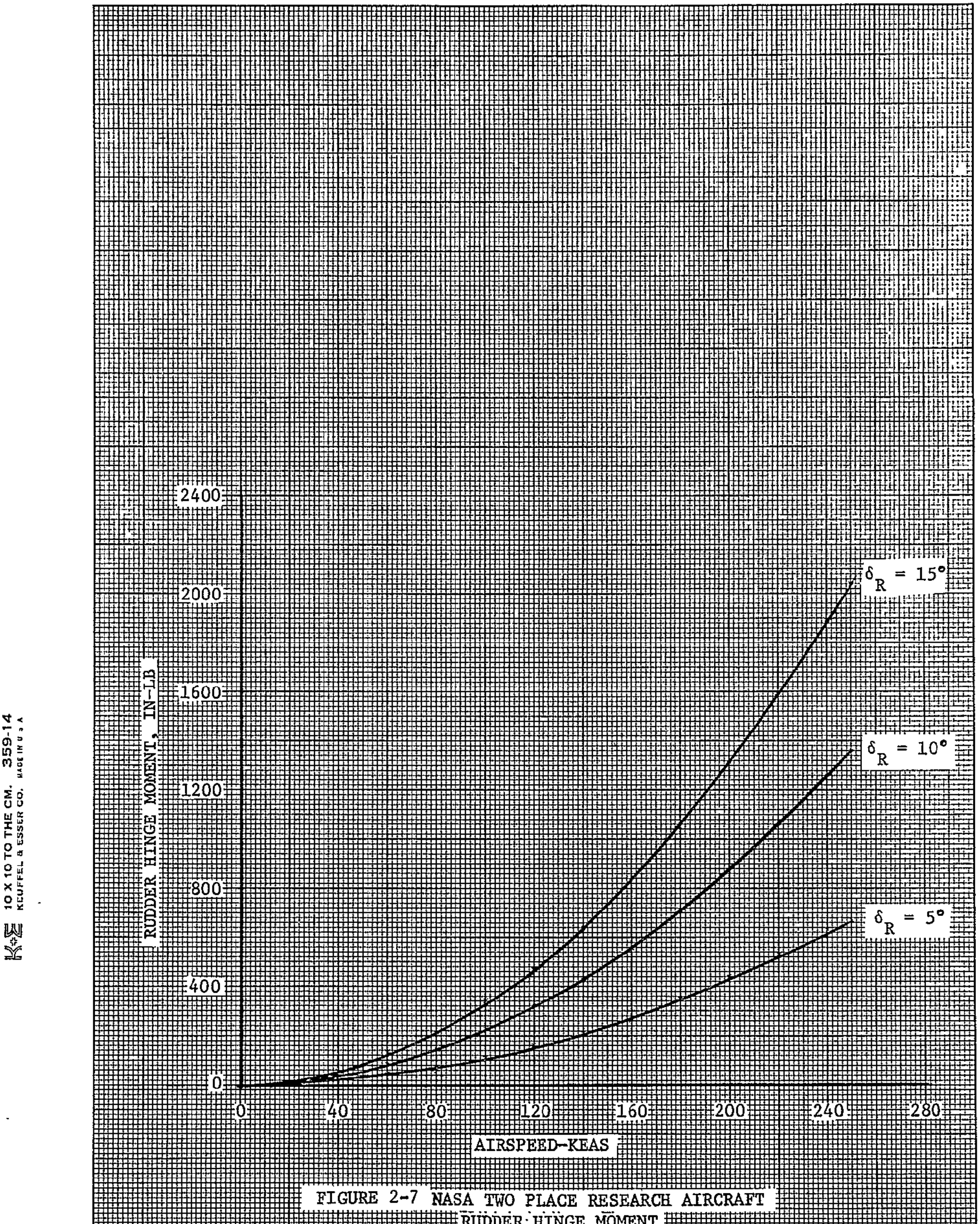

FIGURE 2-7 NASA TWO PLACE RESEARCH AIRCRAFT
RUDDER HINGE MOMENT

AXIS	MAXIMUM FEEL FORCES (TAV-8A)	OVERRIDE FORCE AT STICK	OVERRIDE FORCE AT RUDDER PEDALS	OVERRIDE FORCE AT ACTUATOR TIE IN
	LB	LB	LB	LB
PITCH	13	26	-	148
ROLL	7.5	15	-	63
YAW	45	-	100	200

FIGURE 2-8 OVERRIDE FORCE LEVELS - FRONT COCKPIT

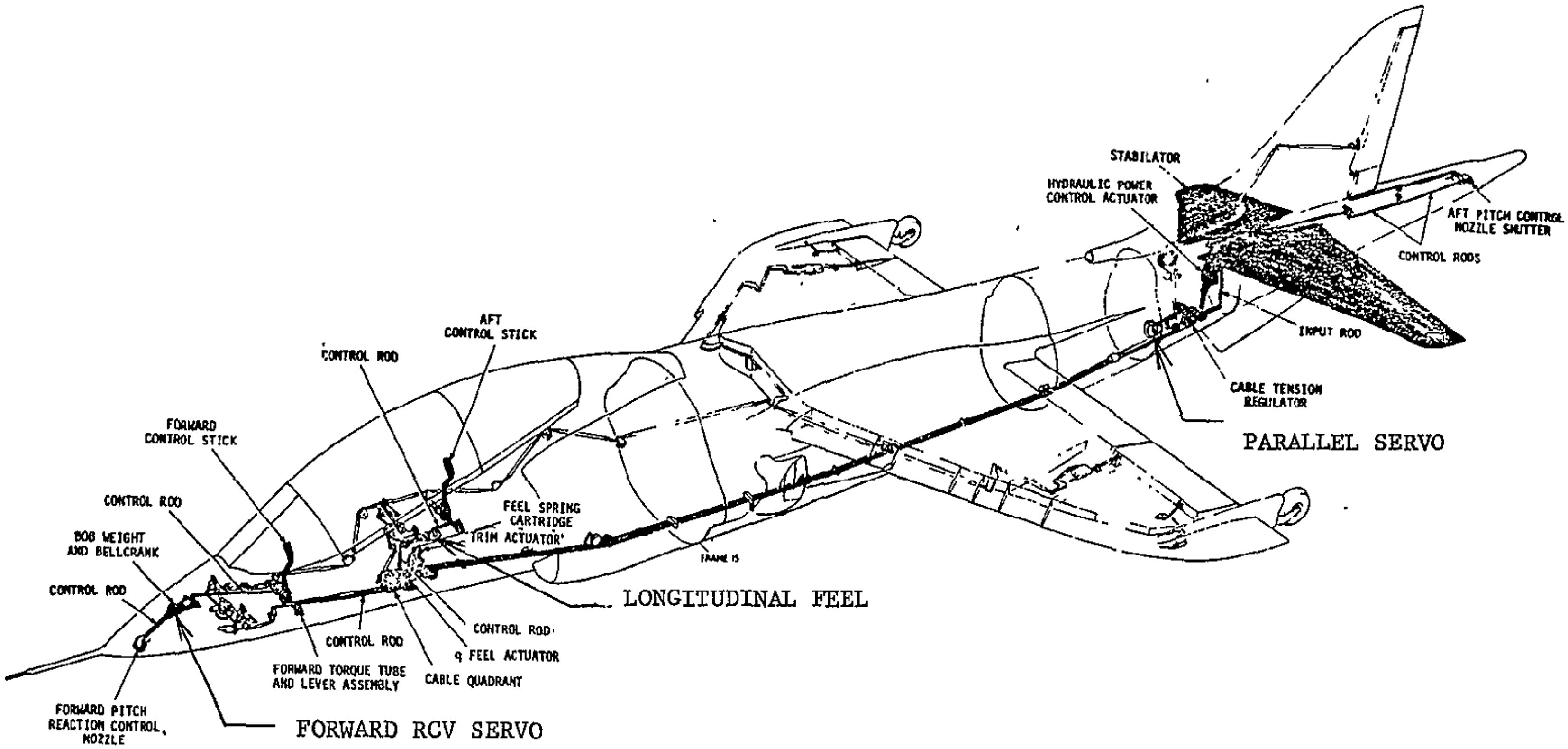

FIGURE 2-9 LONGITUDINAL CONTROLS

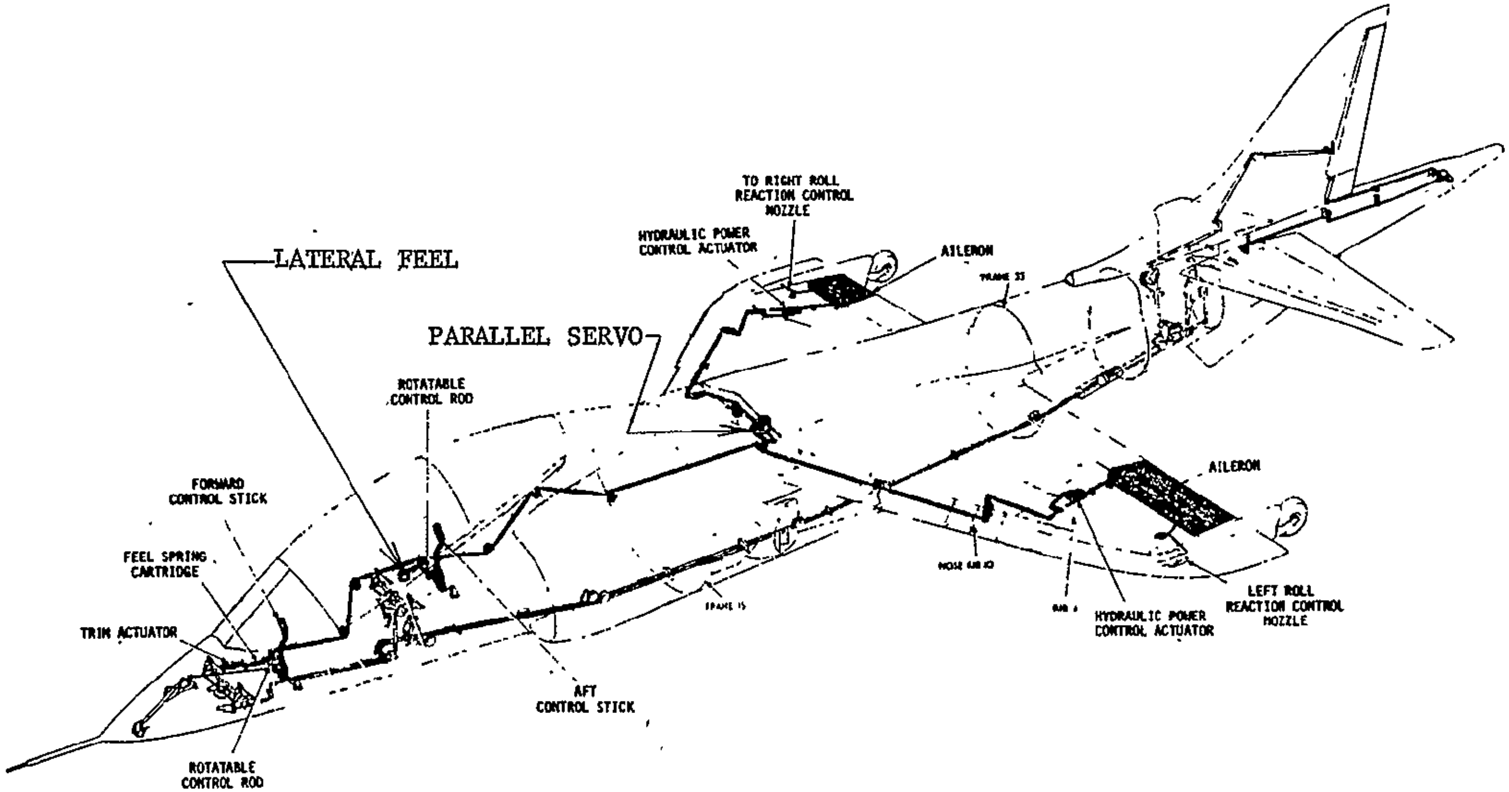

FIGURE 2-10 LATERAL CONTROLS

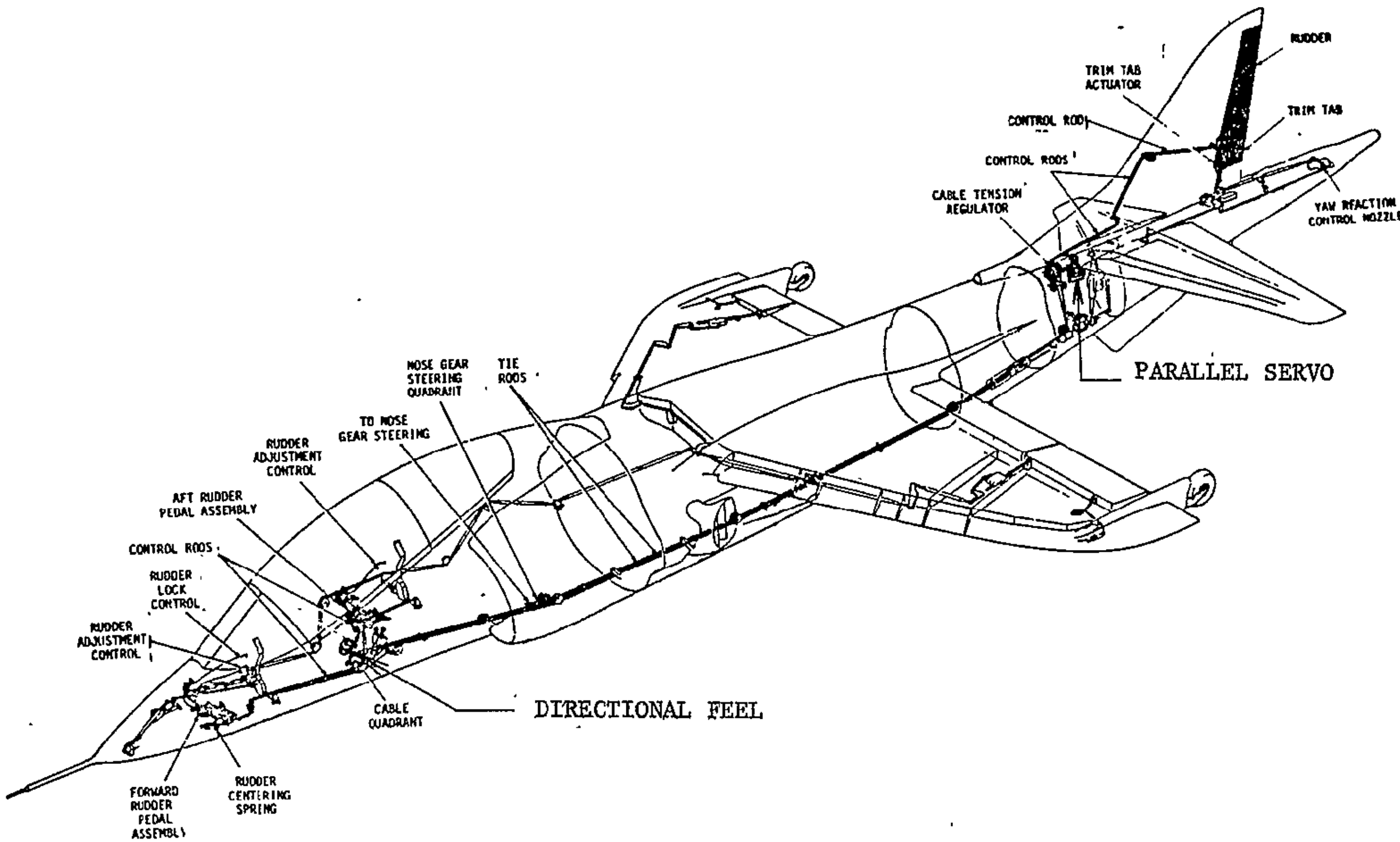

FIGURE 2-11 DIRECTIONAL CONTROLS

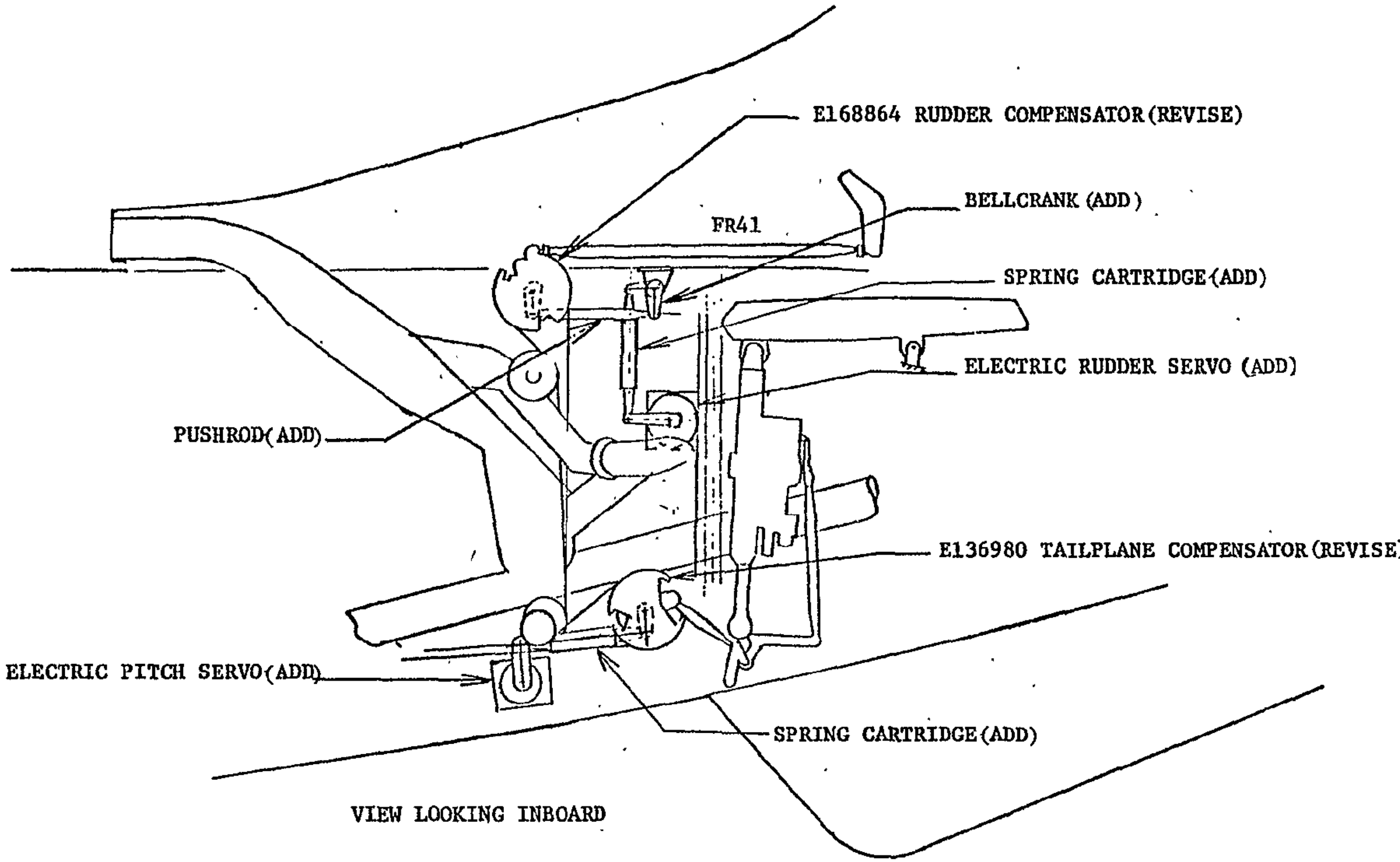

FIGURE 2-12 SERVO INSTALLATION , LONGITUDINAL AND DIRECTIONAL

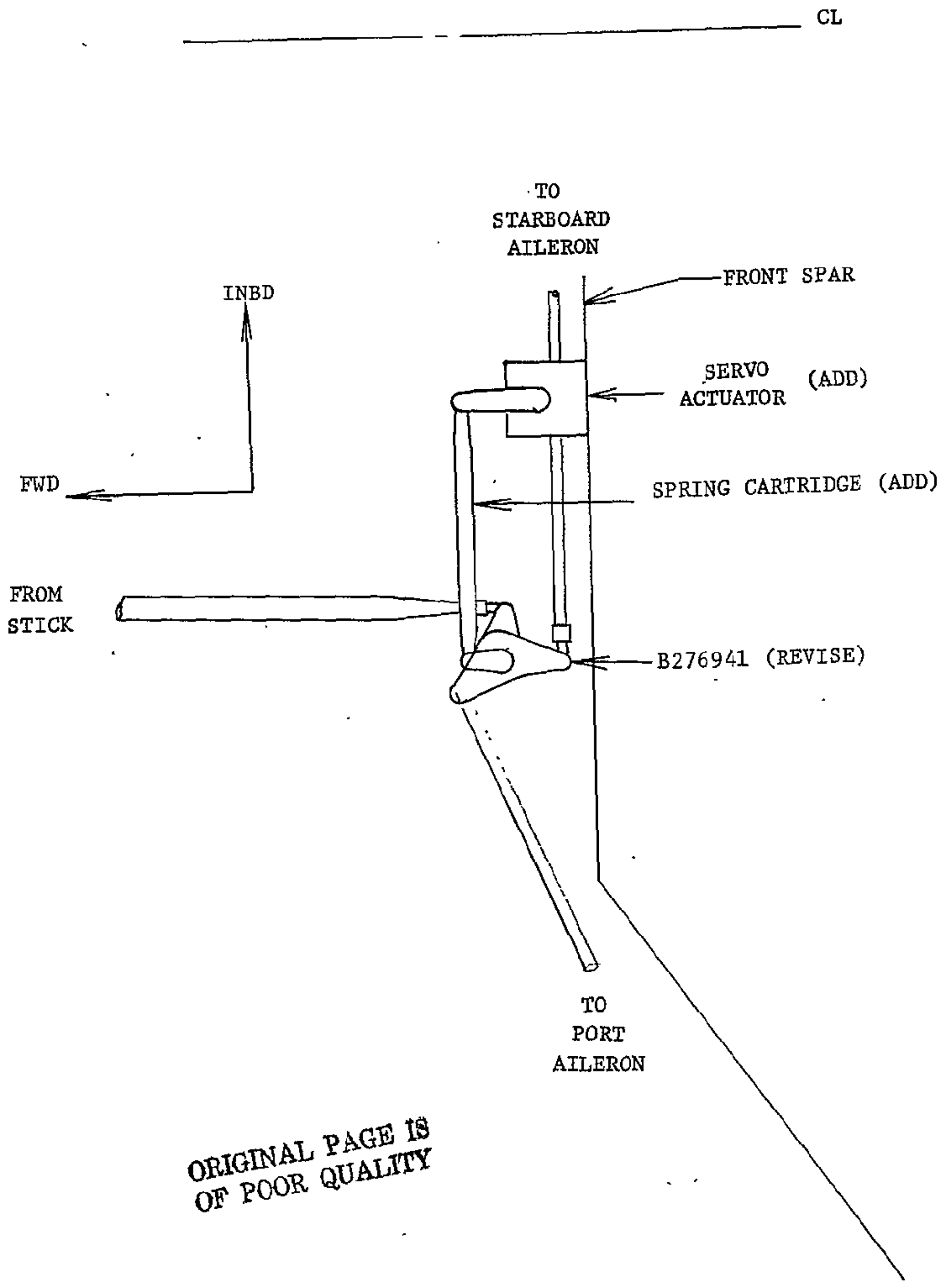

FIGURE 2-13 SERVO INSTALLATION , LATERAL-PARALLEL

Figures 2-14 through 2-16 show how the position transducers, force transducers and feel system torque motors will be attached to the aft cockpit stick and rudder pedals.

It is important to note that both electrohydraulic and electromechanical actuators were considered for mechanizing the parallel servos. Electromechanical actuators were chosen for two reasons. First, it is mandatory that the actuator be instantly disengaged and "float" while the safety pilot flies the aircraft. This could be done with off the shelf electromechanical actuators, but required development efforts if electrohydraulic actuators were used. Second, any appreciable increase in hydraulic loads would require significant changes in the Harrier hydraulic system.

2.1.2 COCKPIT INFORMATION AND DISENGAGE REQUIREMENTS - The special controls in the two place V/STOL Research Aircraft are pictured in Figures 2-17 and 2-18. The panels are further described in Figures 2-19 through 2-24. The pilot in the aft cockpit is the test conductor and has the necessary controls to engage the fly-by-wire system and change the computer program parameter values. The front cockpit pilot is the safety pilot and has the controls necessary to override or disconnect the fly-by-wire system and revert to the normal airplane control system. If a failure occurs in any axis of control, the safety pilot can take command by applying control force of a moderate level to activate a force disengage switch. The computer will then disengage the servo which is not operating properly. The force levels required for the safety pilot to override the pitch, roll, and yaw parallel servos are given in Figure 2-8. The safety pilot will also have emergency disengage buttons on his stick and throttle lever.

The controls for operating the digital flight control system are in the aft cockpit. They include a keyboard and single line readout for entering parameter values such as system gains into the computer. The individual axis gains are selectable and are shown on the aft cockpit panel and repeated on the forward cockpit control panel. The aft cockpit panel has the system engage switch which activates the fly-by-wire system if the safety pilot has activated the enable switch in the forward cockpit. The head-up display is used to show both pilots that there is a satisfactory range of mismatch between the forward and aft control position before the fly-by-wire system is engaged. Both cockpits have indicators showing channel selection, gain selection, built-in test results, and engage enable status, but only the aft cockpit has the controls to change the computer program parameters. A built-in safety included in the electronic design will limit a computer program parameter change to be made only when the system is not engaged. A summary of the cockpit displays, control selection available, and disengage requirements is given in Figure 2-25.

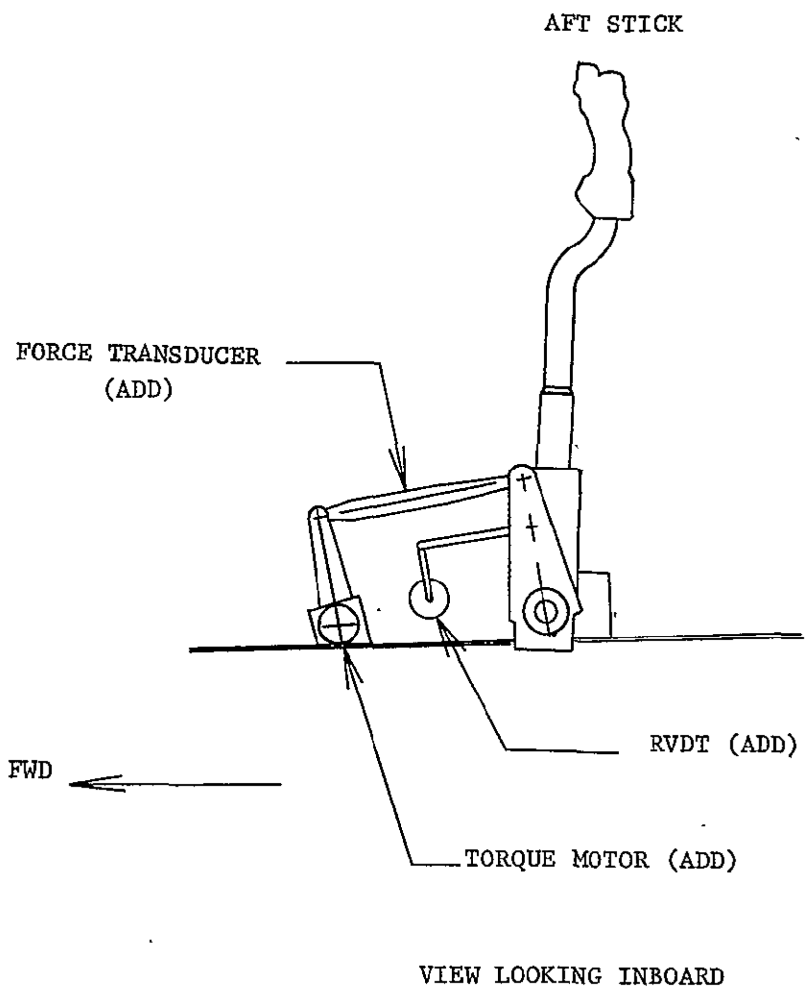

FIGURE 2-14 LONGITUDINAL FEEL SYSTEM INSTALLATION

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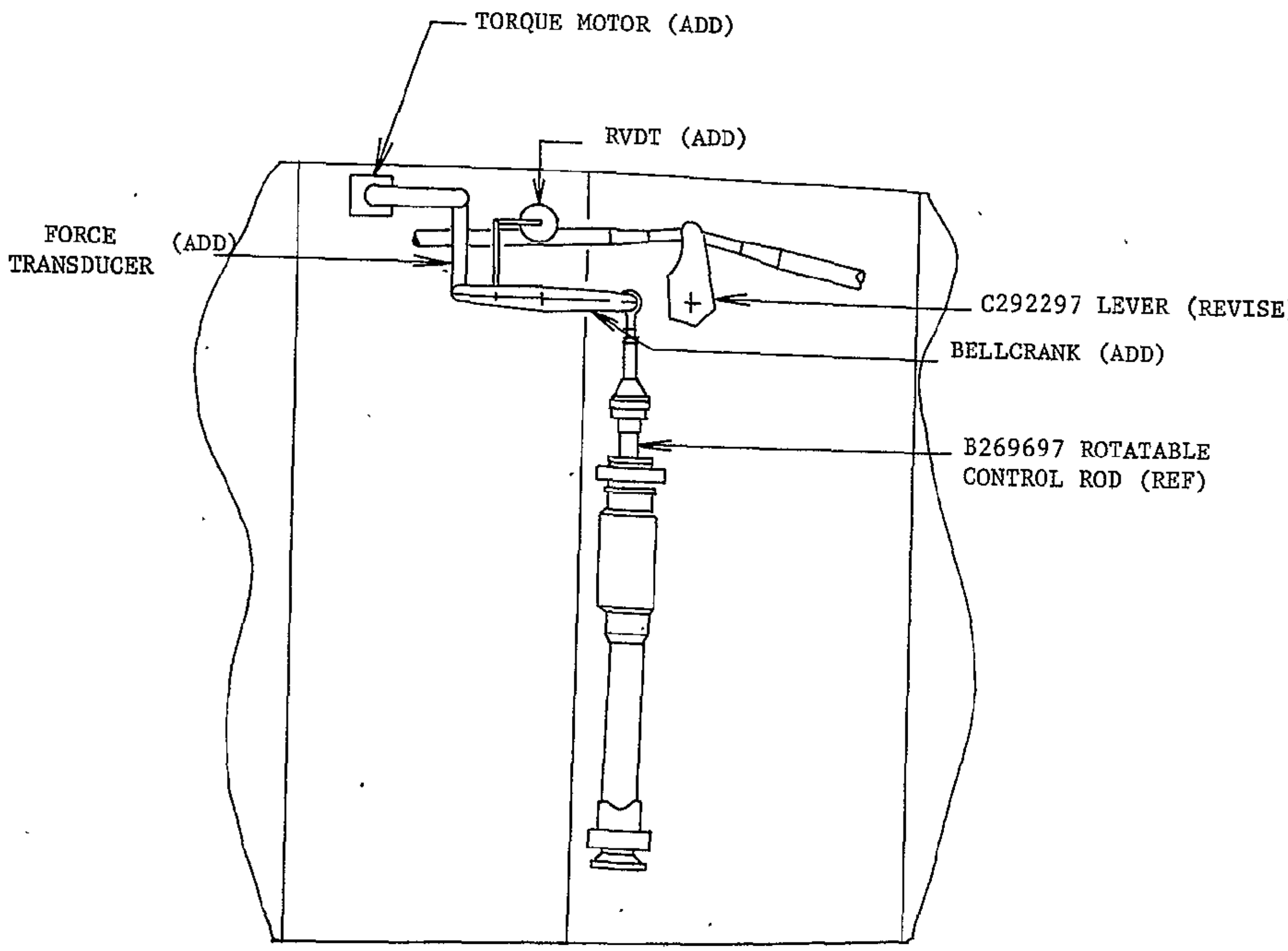

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FIGURE 2-15 LATERAL FEEL SYSTEM INSTALLATION

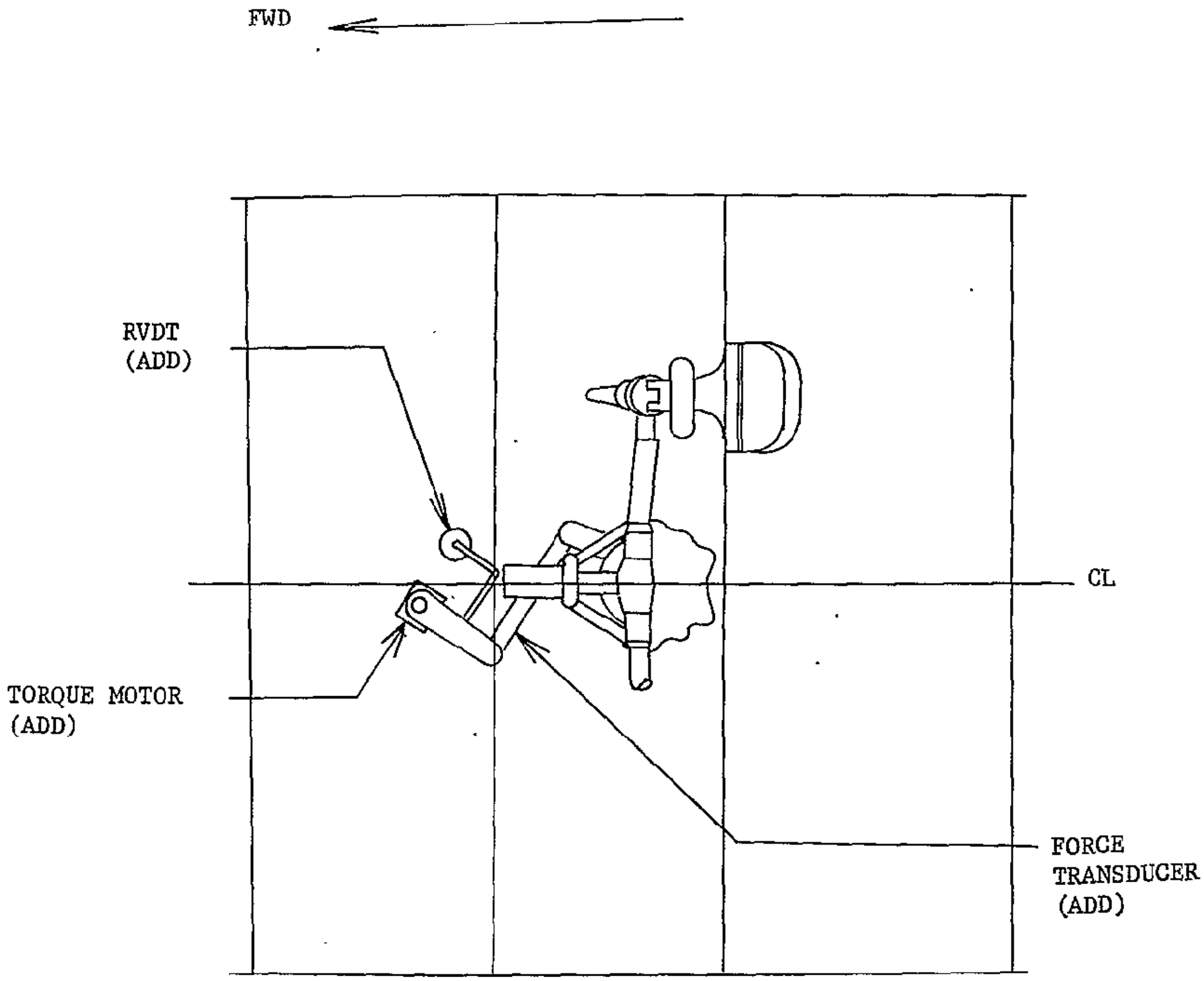

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FIGURE 2-16 DIRECTIONAL FEEL SYSTEM INSTALLATION

FIGURE 2-17

FORWARD COCKPIT
SPECIAL CONTROLS AND DISPLAYS

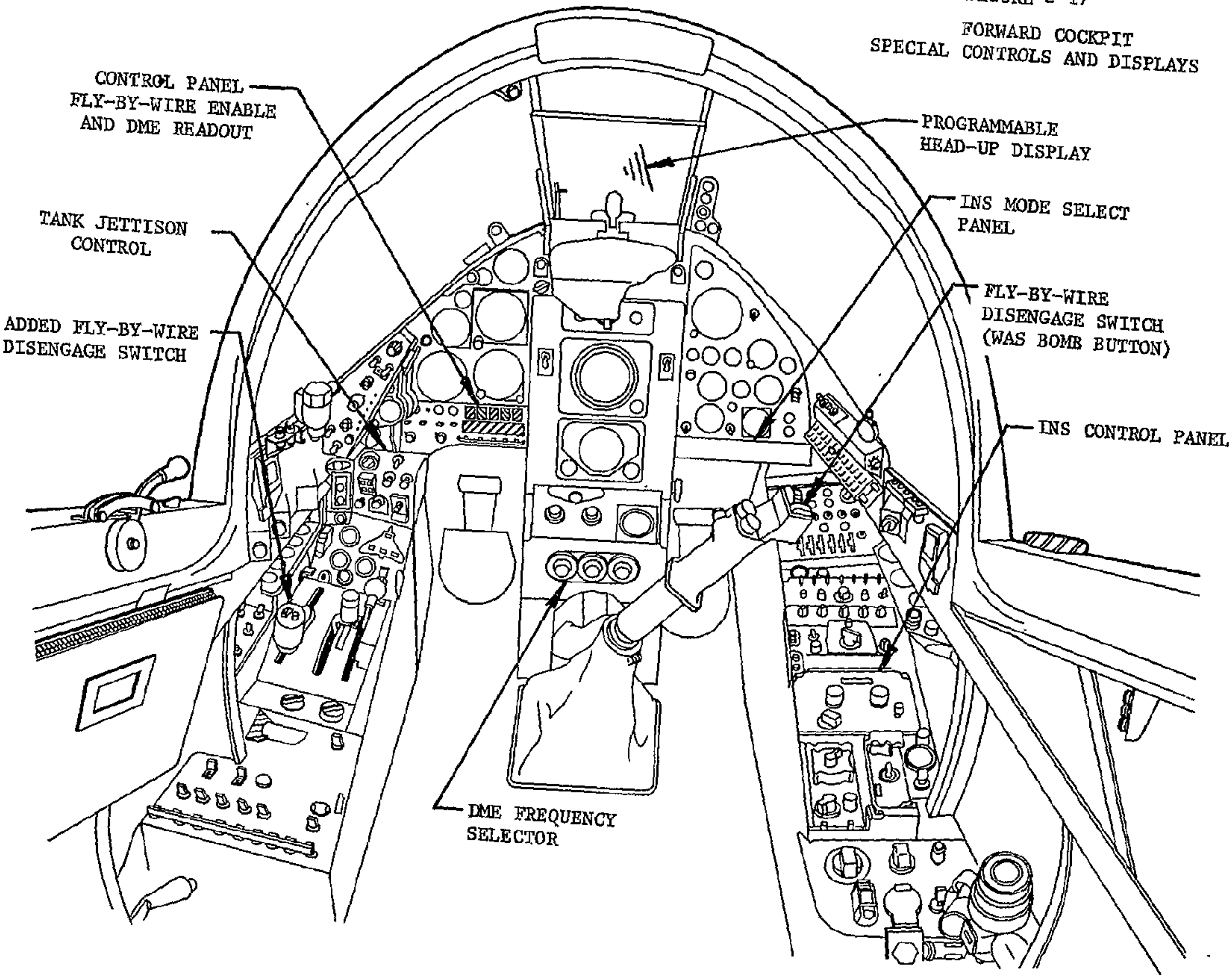
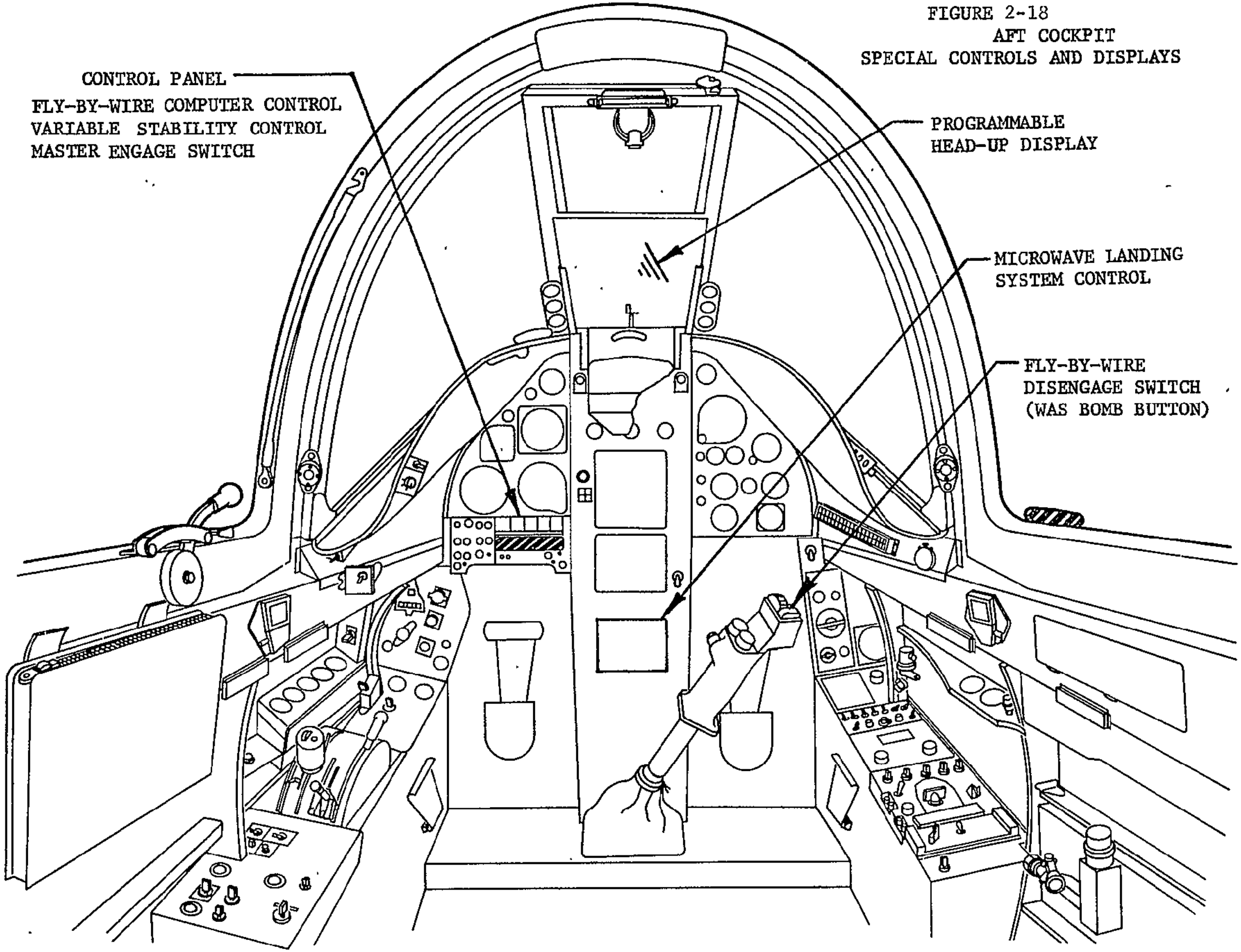


FIGURE 2-18
AFT COCKPIT
SPECIAL CONTROLS AND DISPLAYS



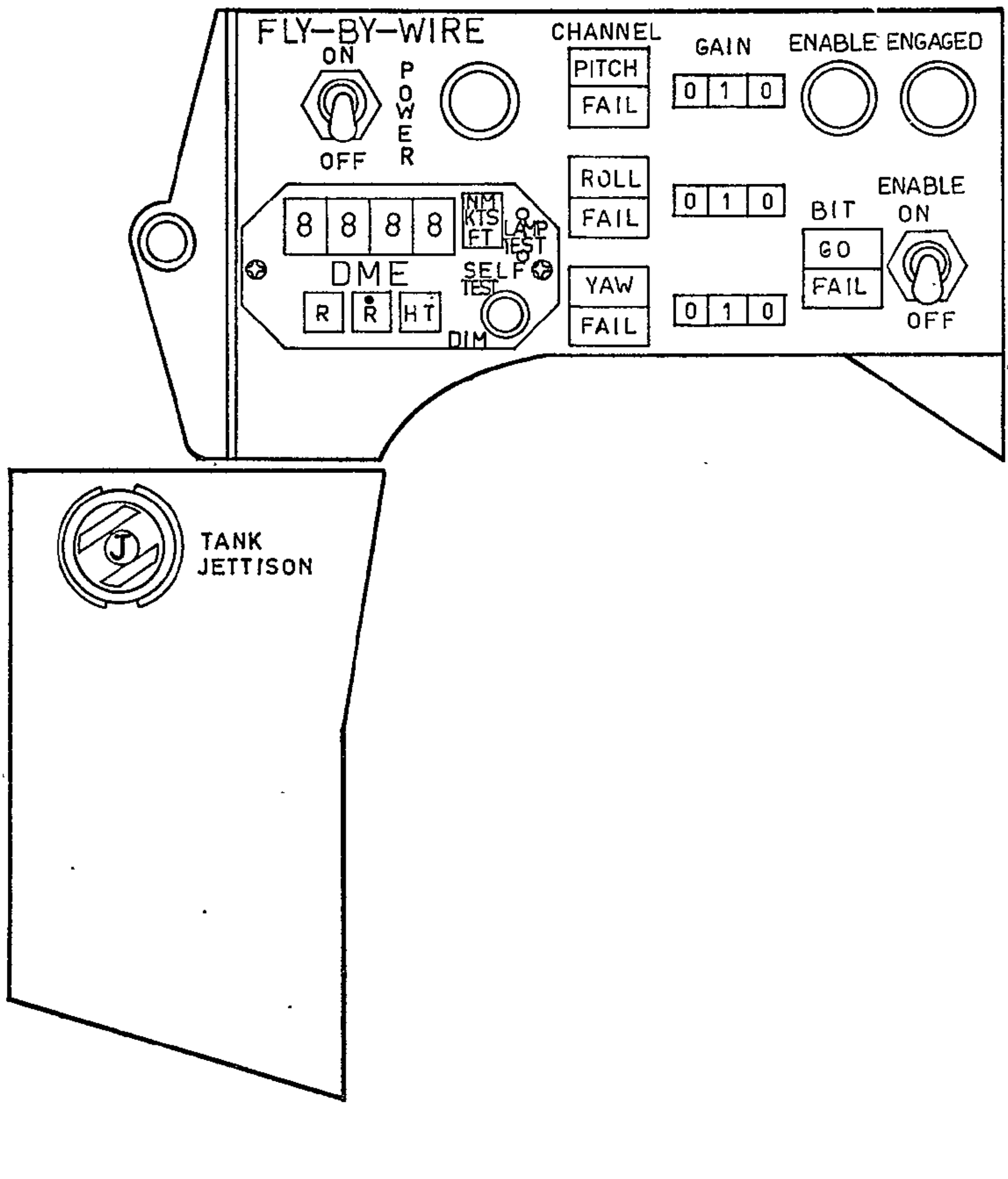


FIGURE 2-19 FLY BY WIRE ENABLE PANEL

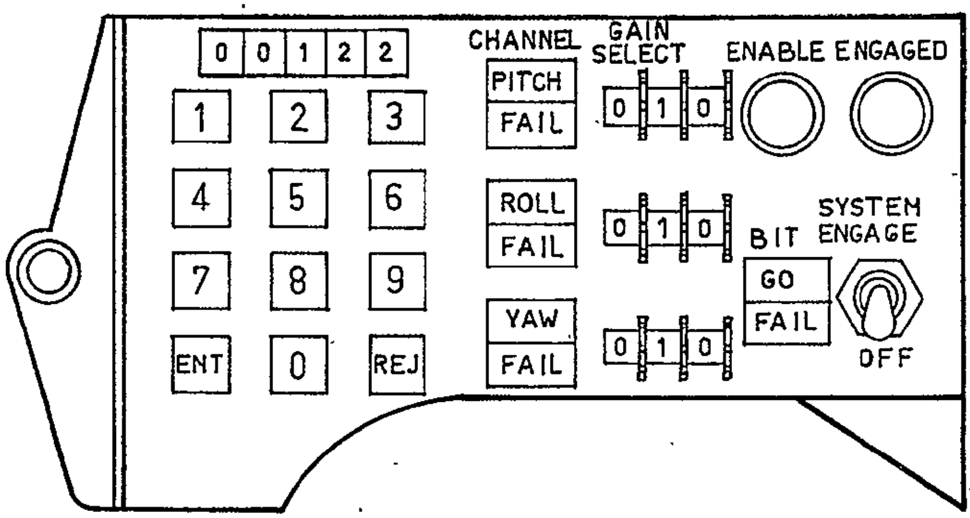


FIGURE 2-20 FLY BY WIRE SYSTEM ENGAGE PANEL

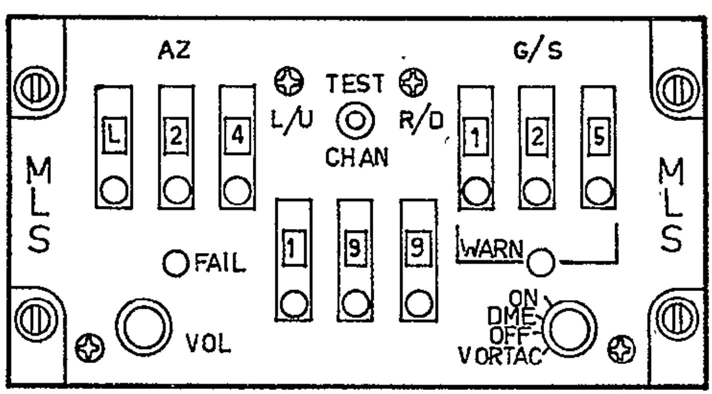


FIGURE 2-21 MICROWAVE LANDING SYSTEM CONTROL

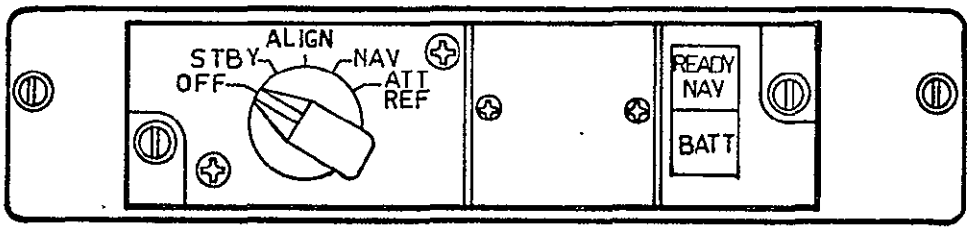


FIGURE 2-22 INS MODE SELECT

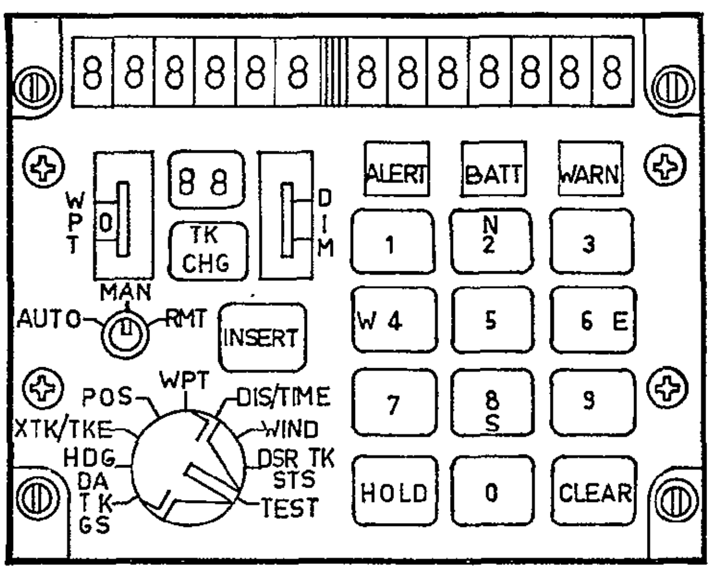


FIGURE 2-23 AVIONICS CONTROL INS PANEL

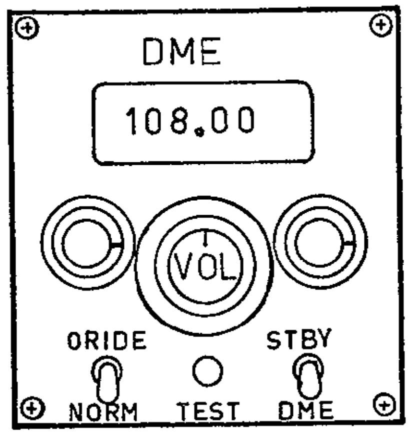


FIGURE 2-24 DME FREQUENCY SELECTOR

FIGURE 2-25 COCKPIT INFORMATION
AND DISENGAGE REQUIREMENTS

PROVISION	COCKPIT	FUNCTION	INDICATION
MASTER CONTROL PANEL	AFT		
o SYSTEM POWER		SWITCH SYSTEM POWER ON AND OFF	GREEN SYSTEM POWER LIGHT ON OR OFF (BOTH COCKPITS)
o BUILT IN TEST		PREFLIGHT INITIATION (WITH SAFETY INTERLOCKS) OF GROUND TEST FOR FBW SYSTEM	GO/NO GO (BOTH COCKPITS)
o MODE SELECTION		SELECTIVE ENABLEMENT OR DESELECTION OF EACH CONTROL SYSTEM BY AXIS (THROTTLE AND NOZZLE SYSTEMS IF APPLICABLE)	AMBER MODE LIGHT ON OR OFF FOR EACH CONTROL SYSTEM BY AXIS (THROTTLE AND NOZZLE SYSTEMS IF APPLICABLE) (BOTH COCKPITS)
o MASTER ENGAGE/ DISENGAGE		MASTER ENGAGEMENT/DISENGAGEMENT OF ALL CONTROL SYSTEM MODES SELECTED (THE FORWARD COCKPIT ENGAGE ENABLE SWITCH MUST BE ACTIVATED PRIOR TO AFT COCKPIT ENGAGEMENT)	GREEN MASTER ENGAGE/DISENGAGE LIGHT ON OR OFF (BOTH COCKPITS)
o INFLIGHT FAILURE MONITORING		ALERT AND DEFINE TO THE FLIGHT CREW THOSE FAILURES DETECTED BY THE INFLIGHT INTEGRITY MANAGEMENT SYSTEM AND THE AFFECT OF AUTOMATIC DISENGAGEMENT WHEN APPLICABLE	RED MASTER FAIL LIGHT ON OR OFF (BOTH COCKPITS) RED MODE LIGHT ON OR OFF FOR FAILED CONTROL SYSTEM BY AXIS AND/OR CHANNEL (THROTTLE AND NOZZLE SYSTEMS IF APPLICABLE) (BOTH COCKPITS) AMBER MODE LIGHT(S) AND/OR GREEN MASTER ENGAGE/DISENGAGE LIGHT ON OR OFF DEPENDING UPON DISENGAGE LOGIC (BOTH COCKPITS)

FIGURE 2-25 CONT'D
COCKPIT INFORMATION
AND DISENGAGE REQUIREMENTS

PROVISION	COCKPIT	FUNCTION	INDICATION
DIGITAL COMPUTER CONSOLE	AFT		
o GAIN SELECTION		SELECTIVE ENABLEMENT OF AIRCRAFT FBW DYNAMIC RESPONSE AND FEEL SYSTEM CHARACTERISTICS FOR EACH CONTROL SYSTEM AXIS (THROTTLE AND NOZZLE SYSTEMS IF APPLICABLE) WITHIN A PREDEFINED SAFE RANGE OF VALUES	RANGE OF VALUES AVAILABLE AND VALUE SELECTED (BOTH COCKPITS)
o GAIN ENGAGE		ENGAGE SELECTED GAINS (AVAILABLE ONLY WHEN NOT IN FBW MODE)	GREEN ENGAGE LIGHT ON OR OFF (BOTH COCKPITS)
DISENGAGE MECHANISMS			
o MASTER DISENGAGE	AFT	MASTER CONTROL PANEL DISENGAGEMENT OF ALL CONTROL SYSTEM MODES SELECTED	GREEN MASTER ENGAGE/DISENGAGE LIGHT ON OR OFF (MASTER CONTROL PANEL) (BOTH COCKPITS)
o SWITCH		DISENGAGEMENT OF ALL CONTROL SYSTEM MODES SELECTED	GREEN MASTER ENGAGE/DISENGAGE LIGHT ON OR OFF (MASTER CONTROL PANEL) (BOTH COCKPITS)
STICK AND THROTTLE	FWD		
STICK	AFT		
o FORCE LINK	FWD	CONTROL STICK AND RUDDER PEDALS - DISENGAGEMENT OF ALL CONTROL SYSTEM MODES SELECTED BY APPLICATION OF A SPECIFIED FORCE LEVEL	GREEN MASTER ENGAGE/DISENGAGE LIGHT ON OR OFF (MASTER CONTROL PANEL) (BOTH COCKPITS)
		(IF APPLICABLE) THROTTLE AND NOZZLE - DISENGAGEMENT OF THROTTLE AND NOZZLE CONTROL SYSTEMS BY APPLICATION OF A SPECIFIED FORCE LEVEL	AMBER THROTTLE AND NOZZLE MODE LIGHTS ON OR OFF (MASTER CONTROL PANEL) (BOTH COCKPITS)
o AUTOMATIC DISENGAGEMENT		INFLIGHT INTEGRITY MANAGEMENT AUTOMATIC DISENGAGEMENT OF FAILED CONTROL SYSTEM	RED MASTER FAIL LIGHT ON OR OFF (MASTER CONTROL PANEL) (BOTH COCKPITS)

FIGURE 2-25 CONT'D

COCKPIT INFORMATION

AND DISENGAGE REQUIREMENTS

PROVISION	COCKPIT	FUNCTION	INDICATION
			RED MODE LIGHT ON OR OFF FOR FAILED CONTROL SYSTEM BY AXIS AND/OR CHANNEL (THROTTLE AND NOZZLE SYSTEMS IF APPLICABLE) (MASTER CONTROL PANEL) (BOTH COCKPITS)
			AMBER MODE LIGHT(S) AND/OR GREEN MASTER ENGAGE/DISENGAGE LIGHT ON OR OFF DEPENDING UPON DISENGAGE LOGIC (MASTER CONTROL PANEL) (BOTH COCKPITS)
AIRCRAFT CONTROL TRANSFER MATCHING	BOTH	ENABLE MANUAL ALIGNMENT OF FORWARD AND AFT COCKPIT CONTROL DISPLACEMENTS FOR AIRCRAFT CONTROL TRANSFER BETWEEN COCKPITS	HUD DISPLAY DEPICTING CONTROL DISPLACEMENT ALIGNMENT FOR EACH CONTROL SYSTEM BY AXIS (THROTTLE AND NOZZLE SYSTEMS IF APPLICABLE) (BOTH COCKPITS)
COCKPIT CONTROLS DISPLACEMENT	FWD	DISPLACE COCKPIT CONTROL STICK AND RUDDER PEDALS (THROTTLE AND NOZZLE IF APPLICABLE)	DISPLACEMENT OF APPROPRIATE FORWARD COCKPIT CONTROLS DURING FBW MODE
MANUAL OVERRIDE (PARALLEL SERVO FBW CONFIGURATIONS ONLY)	FWD	MANUALLY OVERRIDE ANY FBW CONTROL SYSTEM COMMAND IF DISENGAGEMENT IS NOT POSSIBLE	

2.1.3 COMPUTER HARDWARE - A trade study was conducted to select a digital computer to perform flight control system and display system computations.

The three candidate computers for the flight control system and display and control system were the IBM AP-101, the ROLM 1664 and the ROLM Ruggedized Eclipse. The criterion for selecting the three candidates was the availability of a HAL/S compiler. A search for additional candidates was made but no other suitable computers with an associated HAL/S compiler are available. The ROLM 1664 does not, in fact, have a compatible HAL/S compiler but Intermetrics Inc., (the company that developed the HAL/S compiler for the AP-101) states that a moderate change to an existing compiler for the Nova Computer will produce a somewhat limited ROLM 1664 compatibility.

The IBM AP-101 computer is being used in the Space Shuttle and an F-8 aircraft digital fly-by-wire system; 56 have been delivered to date. The ROLM 1664 has been available for about a year and has been used in small numbers on a NASA-Langley Helicopter Program, and on various U.S. Navy shipboard applications. An order for just under 200 computers has been placed by the U.S. Navy under a "Design to Price" contract. The Ruggedized Eclipse Computer is being designed for a U.S. Army application and its architecture is almost identical to that of the Data General Eclipse Commercial Computer. Since the Ruggedized Eclipse is only in the development stage, the data available for this trade study was very limited. Although the manufacturer will not furnish even budgetary cost data, preliminary information from ROLM Corp. indicates that the cost of the Reggedized Eclipse will be considerably more than that of the ROLM 1664.

A comparison of the important trade study parameters and the computer selected are shown in Figure 2-26. The following paragraphs present the tradeoff parameters in more detail and discuss the selection of the IBM AP-101 computer.

Software Support - An extensive support software library is available with each of the three candidate computers. The support software for the AP-101 includes program development software, which is used with the IBM 360/370 equipment, plus preflight and inflight self-test programs for fault detection and isolation. The AP-101 support software includes such HOL compilers as HAL/S, FORTRAN, JOVIAL and others.

The ROLM 1664 computer support software includes an extensive software development library and program debugging software which can be used for preflight software check. Compilers are available for FORTRAN, ALGOL and BASIC.

The ROLM 1664 Computer does not have an inflight self-test program which is an important component in the IFIM and Redundancy Management software modules. The AP-101 computer self-test program plus associated BIT hardware provide a 95% fault-detection capability.

The ROLM 1664 computer does not have a HAL/S compiler although a HAL/S compiler has been developed for a similar machine, the NOVA computer. A compiler could be developed for the ROLM 1664 by either retargeting the NOVA compiler or by generating a new code generator for the ROLM 1664. Disadvantages in retargeting the NOVA compiler are:

	IBM AP-101	ROLM 1664	ROLM RUG ECL
HAL/S COMPILER AVAILABLE	√		
INFLIGHT SELF TEST	√		
RELIABILITY (MTBF)	1050	NOT AVAILABLE	NOT AVAILABLE
WEIGHT - LB (32K)	46	85	80
SIZE - IN	7.62 X 10.12 X 19.56	7.62 X 13.12 X 24.31	7.62 X 13.12 X 19.56
MEMORY EXPANSION (16 BIT WDS)	160K	64K	NOT AVAILABLE
THROUGHPUT (KOPS)	405	490	NOT AVAILABLE
COST - NONRECURRING INCL TEST SET	$138,000	$233,100	NOT AVAILABLE
- RECURRING (2 COMPUTERS)	316,000	109,900	NOT AVAILABLE
- TOTAL	453,000	343,000	NOT AVAILABLE
	SELECTED		

FIGURE 2-26 COMPUTER TRADE STUDY SUMMARY

- o The HAL/S compiler used on NOVA does not implement all of the current HAL/S language.
- o The retargeted compiler would not be as efficient as a compiler generated specifically for the ROLM 1664.

The big advantages in using the retargeted compiler are a considerable cost saving and a shorter development time.

The AP-101 HAL/S computer was developed for the NASA Space Shuttle program. Early difficulties experienced with the compiler have been corrected and it is reported that the compiler efficiency is very good.

NASA/Houston has developed extensive software which was used to debug and validate the operational flight program for Space Shuttle. The software was written to operate on the IBM 360/370 and would be useful on this program if the AP-101 computer were used.

The ROLM Ruggedized Eclipse Computer will have a support software complement similar to that of the ROLM 1664.

A summary of support software available for the candidate computers is shown in Figure 2-27. The immediate availability of a very efficient HAL/S compiler and an inflight self-test program are important factors which favor selection of the AP-101.

Reliability - The IBM AP-101 computer has a specified MTBF of 1050 hours and is undergoing reliability tests for the Space Shuttle. It contains burned-in, high reliability parts similar to those incorporated in the AP-1 computer which was tested and qualified for the F-15 aircraft.

The ROLM 1664 computer does not have a specified MTBF and has not experienced reliability tests. It uses Class C parts which require no burn-in.

Therefore, the IBM AP-101 computer is the most reliable candidate for use in a flight control system.

Weight - The AP-101 computer has a definite weight advantage over the other candidates. As shown below, the uninstalled weight advantage over the ROLM 1664 in a triplex configuration is 117 lb.

	WEIGHT-LB (32K MEMORY)			WEIGHT PENALTY	
CONFIGURATION	AP-101	ROLM 1664	ROLM R.E.	ROLM 1664	ROLM R.E.
SIMPLEX	46	85	80	+39	+34
DUPLEX	92	170	160	+78	+68
TRIPLEX	138	255	240	+117	+102

Size - As shown below the AP-101 computer has a size advantage over the ROLM computers. This is most important in the duplex and triplex configurations. In the duplex configuration the 24.31 in length of the ROLM 1664 will not permit

PROGRAM	IBM AP-101	ROLM 1664	ROLM R.E.
Linkage Editor	X	X	X
Assembler	X	X	X
Utilities	X	X	X
Subroutine Library	X	X	X
Functional Simulator	X		
Self-Test Preflight	X	X	X
Self-Test Inflight	X		
HAL/S Compiler	X		
Fortran Compiler	X	X	X

FIGURE 2-27 SUPPORT SOFTWARE AVAILABILITY

its mounting on the lower aft access door. For the triplex configuration, in addition to the door mounting problem, the 13.12 in. width of the ROLM computers prevent their installation in the gun pod.

COMPUTER	HEIGHT	WIDTH	LENGTH
AP-101	7.62 in	10.12 in	19.56 in
ROLM 1664	7.62	13.12	24.31
ROLM R.E.	7.62	13.12	19.56

Memory - Program memory requirements for the single computer in the simplex system and the computers in the duplex or triplex system are shown in Figure 2-28. A more detailed breakdown of memory module size is presented in Section 2-7. Program estimates indicate that a 32K/16 bit memory size is sufficient for the operational program and adequate spare memory. The ROLM 1664 provides internal memory expansion to 64K/16 bit words, the IBM AP-101 to 160K/16 bit words.

The OFP memory requirements were estimated for programming in HAL/S language. The estimates were derived from comparisons with a MCAIR flight control system which were programmed in assembly language. An upward adjustment of 20% was made in the memory estimates to provide for the use of HAL/S language. All three computers have sufficient memory capacity to satisfy initial requirements, but the internal memory expansion capability of the IBM AP-101 and ROLM 1664 are an advantage for future growth.

Computation Speed - Computation speed is a very important consideration in this flight control system. MCAIR's experience has been that a computer with a throughput of at least 380 Kops is necessary to perform the computation task for a system with redundancy management or inflight integrity management (IFIM). This minimum value is based on a computer program iteration rate of 60 per second. Computer programming is assumed to be in HOL. Computation rates were calculated for an instruction mix of 85% add, 10% multiply and 5% divide.

The ROLM 1664 Computer is somewhat faster than the AP-101 and therefore, from a computation speed aspect, is somewhat superior. All three candidates are acceptable, however.

Cost Comparison - A comparison of non-recurring and recurring costs for the IBM AP-101 and ROLM 1664 computers is shown in Figure 2-29 for a buy of two, three and four computers. It is estimated that these quantities of computers would be required for the simplex, duplex and triplex computer configurations, respectively. This would provide one spare computer for each configuration which is considered adequate. The ROLM Corp. was unwilling to supply cost figures for the Ruggedized Eclipse computer due to its early stage of development.

Since the unit cost of the IBM AP-101 computer is significantly higher than the ROLM 1664, a comparison of the three configurations shows a cost advantage for the ROLM 1664. However, the extra software development required for the ROLM 1664 partially offsets that advantage for the simplex system and, to a lesser degree, the duplex and triplex configurations.

MODULE	SIMPLEX	DUPLEX	TRIPLEX
	— MEMORY WORDS (16 BITS) —		
COMMON FUNCTION			
CONTROL LAWS	2300	2300	2300
IFIM/REDUNDANCY MGT	6100	3300	3900
EXECUTIVE	2600	2600	2600
BIT	2300	2500	2500
PECULIAR FUNCTION			
MLS/FLIGHT DIRECTOR	2400	2400	2400
TOTALS			
1ST COMPUTER	15,700	13,100	13,700
2ND COMPUTER*	NONE	10,700	11,300
3RD COMPUTER*	NONE	NONE	11,300
TOTAL	15,700	23,800	36,300

*MLS/FLIGHT DIRECTOR NOT REQUIRED.

FIGURE 2-28 OPERATIONAL FLIGHT PROGRAM MEMORY REQUIREMENTS (HAL/S PROGRAMMING)

COMPUTERS PURCHASED	ROLM 1664 TWO	ROLM 1664 THREE	ROLM 1664 FOUR	IBM AP-101 TWO	IBM AP-101 THREE	IBM AP-101 FOUR
NON-RECURRING COST						
TEST SET	8,100	8,100	8,100	137,000	137,000	137,000
SOFTWARE						
- HAL/S COMPILER	130,000	130,000	130,000			
- MODIFY OS FOR HAL/S	20,000	20,000	20,000		NOT REQUIRED	
- COMPILER MAINTENANCE*	75,000	75,000	75,000			
	233,100	233,100	233,100	137,000	137,000	137,000
RECURRING COST						
32K COMPUTERS	109,900	164,900	219,900	316,000	474,000	632,000
TOTAL	343,000	398,000	453,000	453,000	611,000	769,000

*COMPILER MAINTENANCE IS ESTIMATED FOR 5 YEARS @ $15,000 PER YEAR.

FIGURE 2-29 COMPUTER COST COMPARISON

Costs shown in Figure 2-29 are vendor catalog prices or estimates and do not necessarily represent the actual current price of this equipment.

Conclusion - The IBM AP-101 computer was selected based on an evaluation of the significant trade-off parameters. The AP-101 design incorporates high-reliability burned-in parts. The computer has been designed to achieve high reliability and has been reliability tested. The AP-101 is lighter and smaller than the other candidates and has a greater memory expansion capability. An inflight self-test program is available with the AP-101 and an extensively reworked, efficient HAL/S compiler is immediately available. Although the throughput of the AP-101 is not as fast as the ROLM computers it is within the estimated requirement for this application. Thus, the advantages of the AP-101 are considered to justify the cost difference with the ROLM 1664 for this research application.

Selected Computer Description - The AP-101 computer (Figure 2-30) consists of:

- o Central Processor unit and input/output (CPU-I/O).
- o Main storage (MS).
- o Power supply (PS).

The electrical parts and assemblies, along with the repair status of each item, are listed in Figure 2-30. The internal items are accessible when the top and bottom covers are removed.

External electrical connections to the computer are made through the I/O connector J1, the aerospace ground equipment (AGE) connector J2, the dc power connector J3, and the rear mounted blower power connector J4. Connectors J1, J2 and J3 are mounted on the front panel along with the time totalizing meter M1 and the fault indicator DS1.

The computer is mounted in the aircraft shelf by means of two mounting hooks on the front of the unit and two mounting pins on the rear. There are two handles on the front panel.

Cooling air is drawn into the unit by a Retron 3501 centrifugal blower. It enters through two inlet ports located on the structure front panel, passes through the two side wall heat exchangers, flows into a common plenum and is exhausted to the environment by the blower. Heat generated in the electronic assemblies is conducted through the structure walls to heat exchanger fins where it is dissipated by the forced air that passes through the ducts.

CPU Packaging - The CPU I/O group consists of ten pluggable electronic assemblies (A3-A8, A10-A12 and A25), and a fault indicator (DS1) that are interconnected by a backpanel assembly (A23). (A01, A02, and A09 are spare locations.) The backpanel assembly consists of a multilayer printed circuit board bonded to a metal support plate. Page connector receptacles are soldered to plated holes in the board. The backpanel assembly is attached to two "tray rails" which are mounted to the heat exchanger rails with socket head cap screws. The ten pluggable electronic assemblies are plugged into the backpanel and secured to the tray rails by captive mounting hardware.

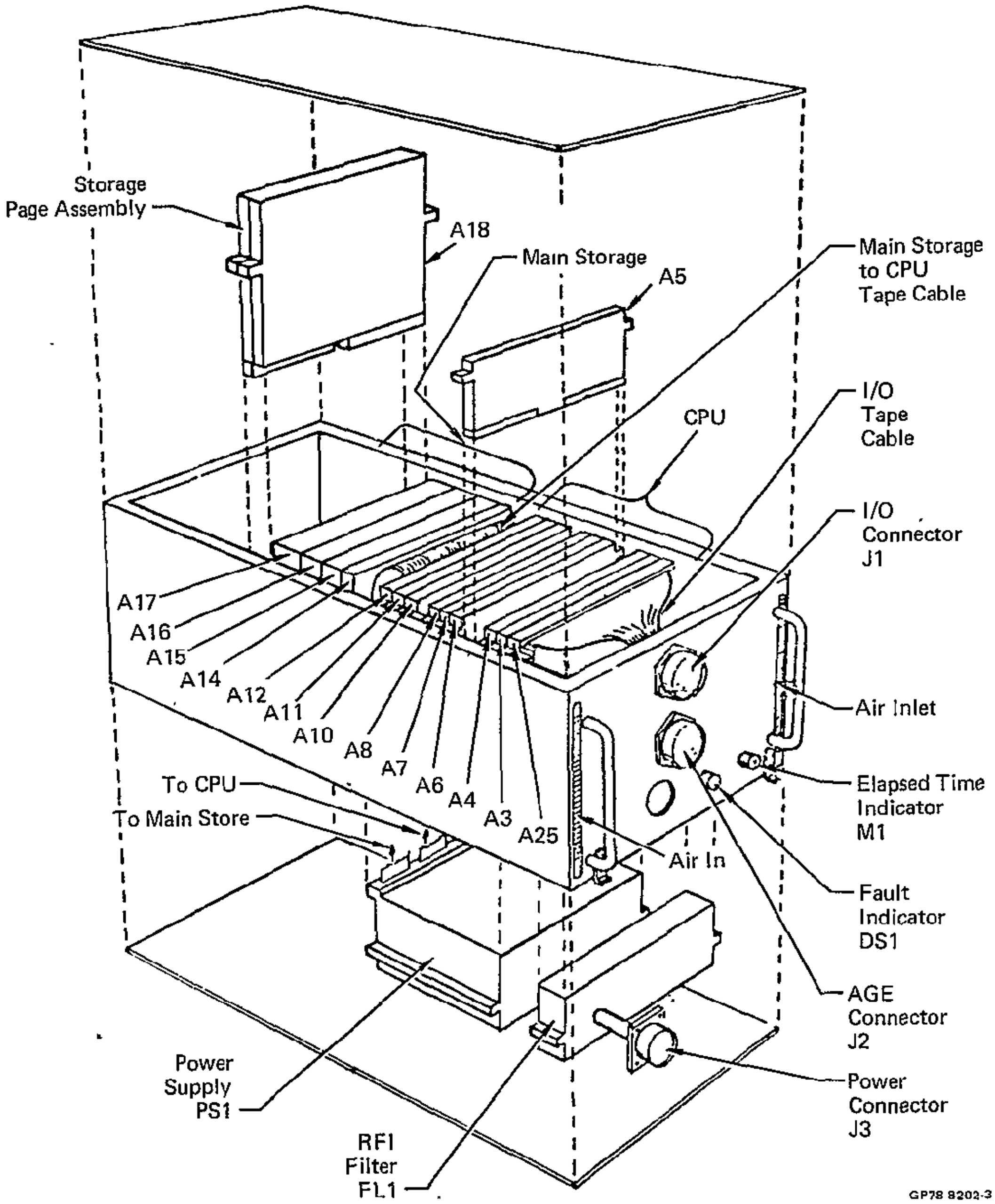


FIGURE 2-30 AP-101 LOCATION OF MAIN PARTS AND ASSEMBLIES

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OF POOR QUALITY

Main Storage Packaging - The Main Storage Group consists of five pluggable page assemblies (A14 through A18) interconnected by a backpanel assembly (A24). Four of the page assemblies are identical, interchangeable storage pages; the fifth is a timing page. A19 through A22 are spare storage page locations.

Power Supply Packaging - The power supply consists of a radio-frequency filter (FL1) which contains power connector J3, a modular power supply (PS1), and a time totalizing meter (M1). The power supply consists of eight printed circuit board assemblies and an RFI filter assembly. Each printed circuit board is bonded to a metal backing plate which provides structural support. Interconnections between the various printed circuit board assemblies are made through a backpanel Multilayer Interconnect Board (MIB) which is mounted perpendicular to the board assemblies. Each assembly is plugged to the backpanel through 49-pin connectors. Discrete wires are used to connect the high power, frame-mounted components to the backpanel. Input and output connections are made through three 50-pin connectors mounted along the edge of the backpanel.

2.1.4 HYDRAULICS - The simplex flight control system requires no modifications to the existing hydraulic system. Electromechanical actuators have been chosen for the fly-by-wire mode so there is no increase in hydraulic requirements.

2.1.5 ELECTRICAL - The existing power system in the Harrier G-VTOL consists of two 4 KVA AC generators and two 2 KW transformer rectifiers. A basic assumption was made that the electrical load will be similar to the post Mod 800 AV-8A. Based on this assumption, the load summary shown in Figure 2-31 indicates that the additional equipment increases the load beyond the capability of the existing electrical system for each of the flight control configurations. It will therefore be necessary to update the aircraft to 12 KVA AC and 5 KW DC. However, this is the standard system for the TAV-8A and will be installed before the aircraft is delivered to MCAIR for modification.

The wiring for the systems that are being removed will be capped and tied back except where it can be used for the new systems or if the available space dictates that the wire be removed. New wiring will consist of compact wire bundles with Kapton insulated wire.

2.1.6 EQUIPMENT INSTALLATION - Figure 2-32 shows the locations of the added avionics equipment. With the exception of the low range airspeed system and the microwave landing system mounted in the nose equipment bay, all the avionic equipment for the simplex fly-by-wire control system is located in the aft equipment bay. The equipment shelf was designed specifically for this test airplane and is built to accept the large INS platform oriented fore and aft in the airplane. It is attached to the same pickup points as the replaced shelf, and includes cooling ducts that will deliver the required cooling air from the aft environmental control unit. (See Figure 2-33). The equipment is accessible through two large structural doors located on the sides of the fuselage between Frames 33 and 36.

2.1.7 INCREASED ROLL POWER - A number of methods for increasing control power were investigated. It was found that the roll control power could be increased by about 12.5% by changing the wing tip reaction nozzle to the unit developed for the Sea Harrier. Other methods for improving control power require major aircraft design changes.

	TOTAL A/C LOAD		
SIMPLEX PARALLEL	115V AC	28V DC	EXISTING GVTOL POWER SOURCE
ELECTRO - MECH	8130 VA	4320 W	o TWO 4 KVA GENERATORS
			o TWO 2 KW TRUs
DUAL SERIES			
ELECTRO - HYD	8330 VA	4495 W	
TRIPLEX PARALLEL			
ELECTRO - MECH	9435 VA	5045 W	

CONCLUSION:

- o THE EXISTING POWER SYSTEM IS INADEQUATE FOR ALL OPTIONS.

RECOMMENDATION:

- o INCORPORATE ECP 580 WHICH INCREASES THE POWER SOURCE TO ONE 12 KVA GENERATOR AND ONE 5 KW TRU.

FIGURE 2-31 ELECTRICAL LOAD SUMMARY

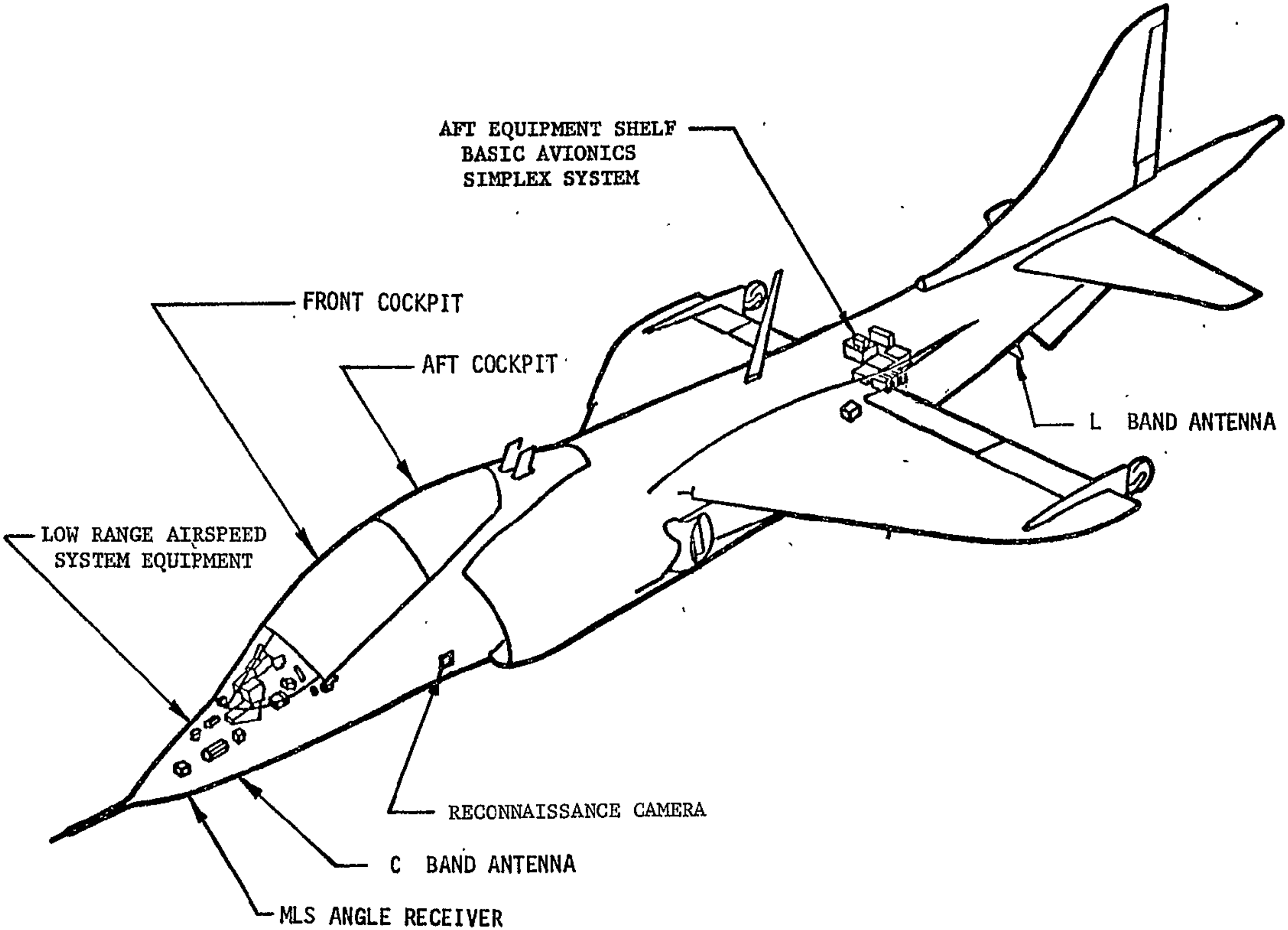

FIGURE 2-32 AVIONICS INSTALLATION SIMPLEX SYSTEM

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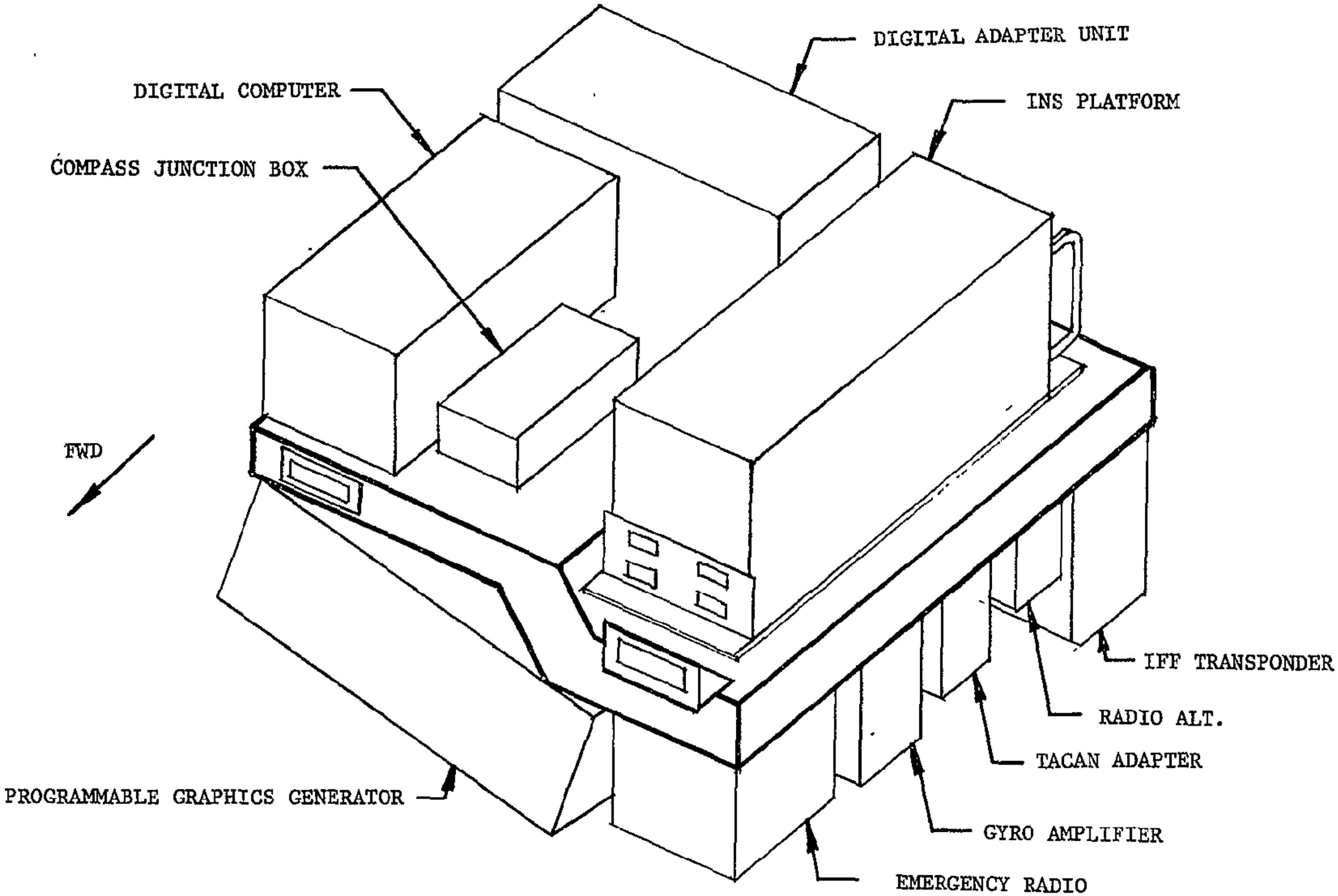


FIGURE 2-33 AVIONICS INSTALLATION EQUIPMENT BAY SHELF

Figure 2-34 shows the present wing tip reaction control valve design and the Sea Harrier design. In the present design, upblowing does not start until the aileron has traveled 6° from neutral. In the Sea Harrier design, upblowing begins when the aileron starts to move from 0°. Also, the upblowing force increases by approximately 80 pounds at maximum deflection due to an increase in the bucket area of the nozzle. The combination of the upblowing reaction control on one wing tip and the downblowing reaction control on the other produces greater roll control moments for the Sea Harrier.

Figure 2-35 compares the reaction control rolling moments of the present nozzle design and the Sea Harrier design. The Sea Harrier design produces higher rolling moments over the entire range of deflections. It can be seen that at 6° of aileron deflection the ratio of the change in rolling moment to change in aileron deflection increases for the present design. This is where upblowing starts. At 7°, both the downblowing shutters are fully open for both the present system and Sea Harrier.

2.2 DATA ACQUISITION SYSTEM

The aircraft will be instrumented for NASA use as a V/STOL control, display and guidance research aircraft. The data system is anticipated to reach a maximum of 90 measurands. The current measurand list is given in Figure 2-36.

2.2.1 MEASURAND REQUIREMENTS - Because the aircraft will be used to obtain research data, NASA participation is required in the definition of measurands to be acquired. To ensure the development of an integrated meaningful measurand list, an early series of design coordination meetings will be held. Emphasis will be placed on the definition and correlation of evaluation pilot (rear seat) commands and reactions. Control surface and control system feedbacks are of primary concern and will be measured with sufficient frequency response to be meaningful.

2.2.2 METHOD OF IMPLEMENTATION - The nucleus of the data acquisition system will be the Teledyne AIFTDS-4000 data acquisition/control system. This system, consisting of a Remote Multiplexer/Demultiplexer Unit (RMDU) and associated transducer power supply, will be provided as GFE through NASA-Ames. MCAIR will specify, procure and install the measurand sensors. The data system will consist of an onboard tape recorder, provided as GFE and a telemetry system consisting of signal conditioning equipment, transmitter, power divider and telemetry antennas. The telemetry system and an airborne time code generator will also be provided as GFE. The aircraft will be modified for upper and lower telemetry antennas to maximize telemetry reception during VTOL transition and conventional flying.

2.2.3 CONFIGURATION - The RMDU, tape recorder, time code generator and telemetry transmitter will be housed in a centerline pod. The loading density in the equipment and avionics bay will preclude the installation of any instrumentation hardware other than sensors in these areas. The configuration of the centerline pod is currently being evaluated but in all probability, a configuration that is already cleared for the Harrier aircraft will be chosen. Due to power dissipation loads which are currently being evaluated, it may be necessary to send cool air to the centerline pod under certain flight conditions. Existing pods (GFE) will be modified or new pods built of suitable structural sheet metal will be manufactured (GFE) based on airworthy configurations. Design installation drawings utilizing British Aerospace and MCAIR design data will be utilized whenever possible.

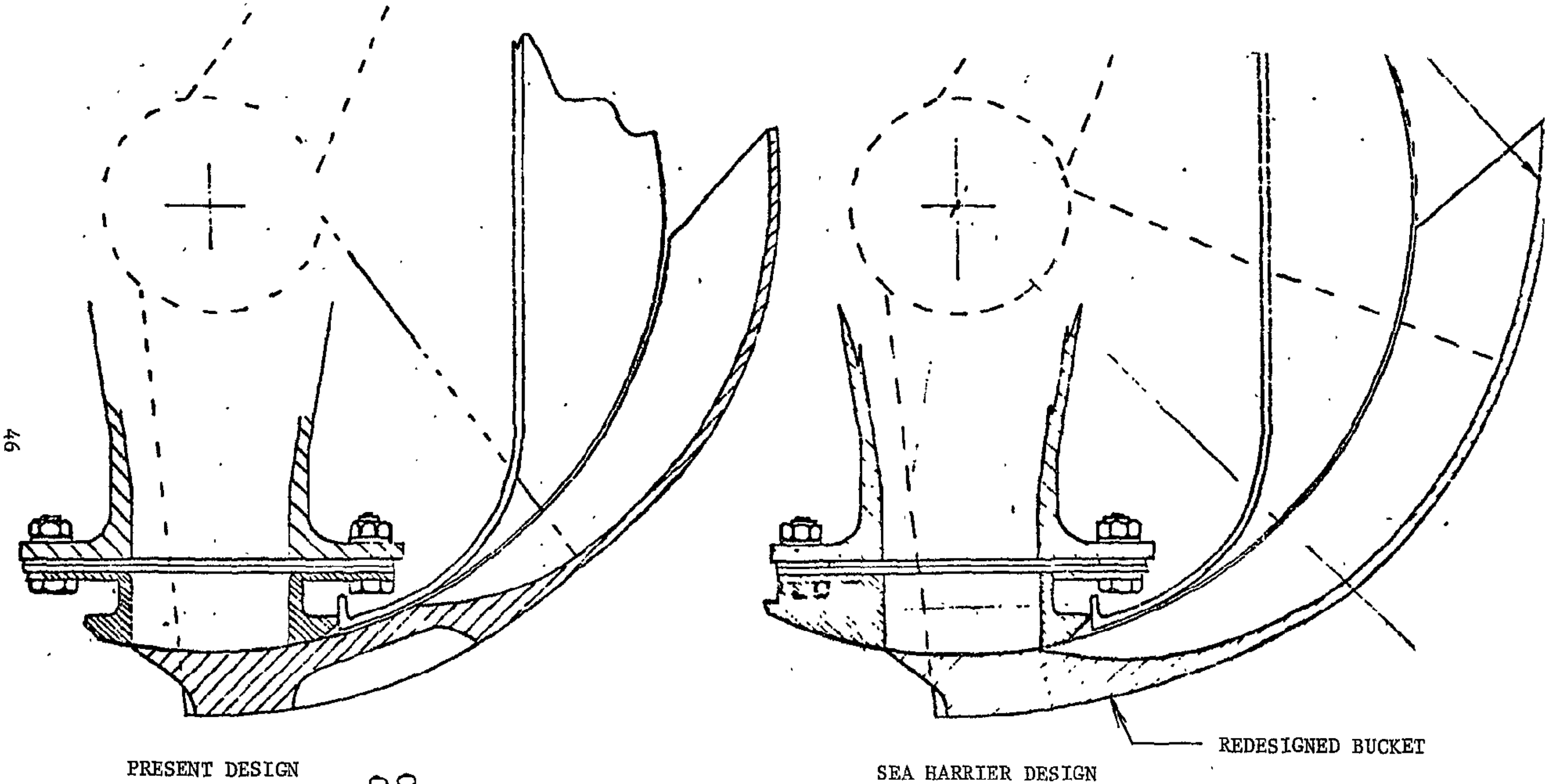


FIGURE 2-34 INCREASED ROLL POWER
REDESIGNED WING TIP REACTION CONTROL VALVE

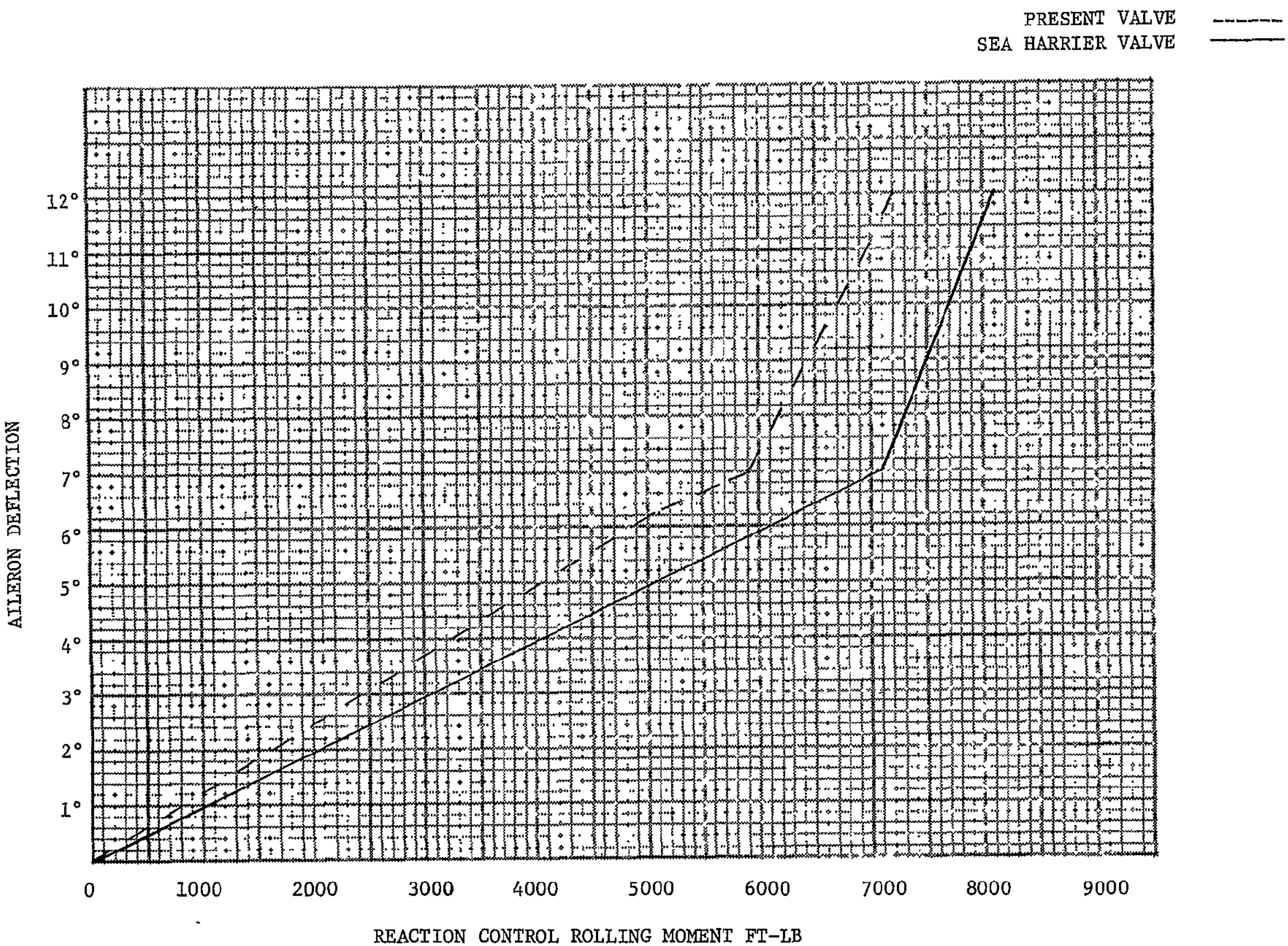


FIGURE 2-35 REACTION CONTROL ROLLING MOMENT VS AILERON DEFLECTION

FIGURE 2-36 DATA ACQUISITION PRELIMINARY MEASURAND LIST

1. AIRSPEED PRESSURE
2. ALTITUDE (PRESS)
3. Nz @ CG
4. Nx @ CH
5. Ny @ CG
6. ANGLE OF ATTACK
7. ANGLE OF SIDESLIP F.T. NOSEBOOM
8. FLT PATH ACCEL
9. TOTAL TEMPERATURE
10. ROLL ANGLE
11. ROLL RATE
12. PITCH RATE
13. YAW RATE
14. PITCH ANGLE
15. FUEL QUANTITY
16. ALTITUDE (RADAR)
17. SINK SPEED
18. BEACON
19. RUDDER POSITION
20. TAILPLANE POSITION
21. TAILPLANE TRIM POSITION
22. L AILERON POSITION
23. R AILERON POSITION
24. FLAP POSITION
25. SPEEDBRAKE POSITION
26. SAS ON/OFF
27. PILOT VOICE
28. TIME CODE GENERATOR
29. RUDDER PEDAL FORCE
30. LONGITUDINAL STICK FORCE
31. LATERAL STICK FORCE
32. LONG STICK POSITION
33. LATERAL STICK POSITION
34. RUDDER PEDAL POSITION
35. LANDING GEAR POSITION
36. MAIN LANDING GEAR - LOADS - SIDE
37. MAIN LANDING GEAR - LOADS - DRAG
38. MAIN LANDING GEAR - LOADS - TORQUE
39. MAIN LANDING GEAR - LOADS - VERTICAL BENDING - LEFT
40. MAIN LANDING GEAR - LOADS - VERTICAL BENDING - RIGHT
41. NOSE LANDING GEAR - LOADS - SIDE
42. NOSE LANDING GEAR - LOADS - DRAG
43. NOSE LANDING GEAR - LOADS - TORQUE
44. NOSE LANDING GEAR - LOADS - VERTICAL BENDING - LEFT
45. NOSE LANDING GEAR - LOADS - VERTICAL BENDING - RIGHT
46. MAIN LG LIFTOFF
47. NLG LIFTOFF
48. HYD NO. 1 PRESS
49. HYD NO. 2 PRESS
50. HYD NOS. 1 AND 2 TEMP
51. THROTTLE ANGLE
52. NOZZLE ANGLE
53. NOZZLE LEVER ANGLE
54. JET PIPE TEMP
55. LP RPM
56. HP RPM
57. WATER QUANTITY
58. WATER RUN ON/OFF
59. FIRE WARN MONITOR
60. ENG FUEL FLOW
61. ENG INLT FUEL PRESS
62. PRIMARY CONTROLLER POSITION (VERTICAL VELOCITY COMMAND)
63. PROPORTIONAL CONTROLLER (HORIZONTAL VELOCITY (COMMAND)
64. SIDE ARM CONTROLLER (PITCH COMMAND)
65. SIDE ARM CONTROLLER (ROLL COMMAND)
66. SIDE ARM CONTROLLER (YAW COMMAND)
67. THROTTLE COMMAND
68. THRUST VECTOR ANGLE COMMAND
69. EVALUATION SYSTEM DISENGAGE
70. SERVO PITCH COMMAND
71. SERVO ROLL COMMAND
72. SERVO ENGINE THROTTLE COMMAND
73. SERVO ENGINE NOZZLE COMMAND
74. SERVO RUDDER COMMAND
75. SERVO RUDDER CONTROL
76. INS RCVR GLIDESLOPE ANGLE
77. INS RCVR AZIMUTH ANGLE
78. INS INTEGRATOR - SLANT RANGE
79. TRANSVERSE SHAFT ANGLE
80. ROLL THRUSTER POSITION - LEFT
81. ROLL THRUSTER POSITION - RIGHT
82. YAW THRUSTER POSITION - FORWARD
83. PITCH THRUSTER POSITION - FORWARD
84. PITCH THRUSTER POSITION - AFT
85. INLET GUIDE VANE POSITION
86. COMPRESSOR DISCHARGE PRESSURE
87. NOZZLE STATIC PRESS - LEFT AFT
88. NOZZLE STATIC PRESS - RIGHT AFT
89. NOZZLE STATIC PRESS - LEFT FORWARD
90. NOZZLE STATIC PRESS - RIGHT FORWARD

2.2.4 VERIFICATION - Due to the highly flexible nature of the RMDU and of a lack of similarity with existing MCAIR data systems, the checkout and verification of the installed instrumentation system is going to be highly dependent on Instrumentation Ground Equipment (IGE) being provided as GFE. If simulated entries, either physical or software controlled, are to be applied as part of the preflight checkout, the instrumentation system must be responsive and capable of being diagnostically evaluated to assure a high degree of confidence in flight environment.

2.3 PROGRAMMABLE HEAD-UP DISPLAY (HUD)

The programmable HUD is based on technology being developed for the YAV-8B. Programmable symbology is provided by replacing the GFE HUD Display Waveform Generator (DWG) with a CFE form fit replacement Programmable Display Processor (PDP). Figure 2-37 is a block diagram of the PDP.

The direct replacement PDP is an all digital calligraphics generator under the control of a microprocessor. It can communicate with both analog and digital equipment and can simultaneously drive the two Pilot Display Units (PDUs) with an almost unlimited variety of symbology. The PDP has a Programmable Read Only Memory (PROM) enabling it to perform the HUD display function independently of the aircraft digital computer. This feature reserves the computer time for the flight control functions and provides the pilot with head-up primary flight data in case of computer failure.

Reprogramming is accomplished by replacing a plug-in PROM element with one having the desired new code. It is estimated that no more than 2 of the 12 PROM elements will be replaced for any but the most drastic symbology changes. PROM elements are estimated to cost less than $100 in small lots, making this an economical technique.

PROM programming, which uses a programming adapter, consists of the following steps:

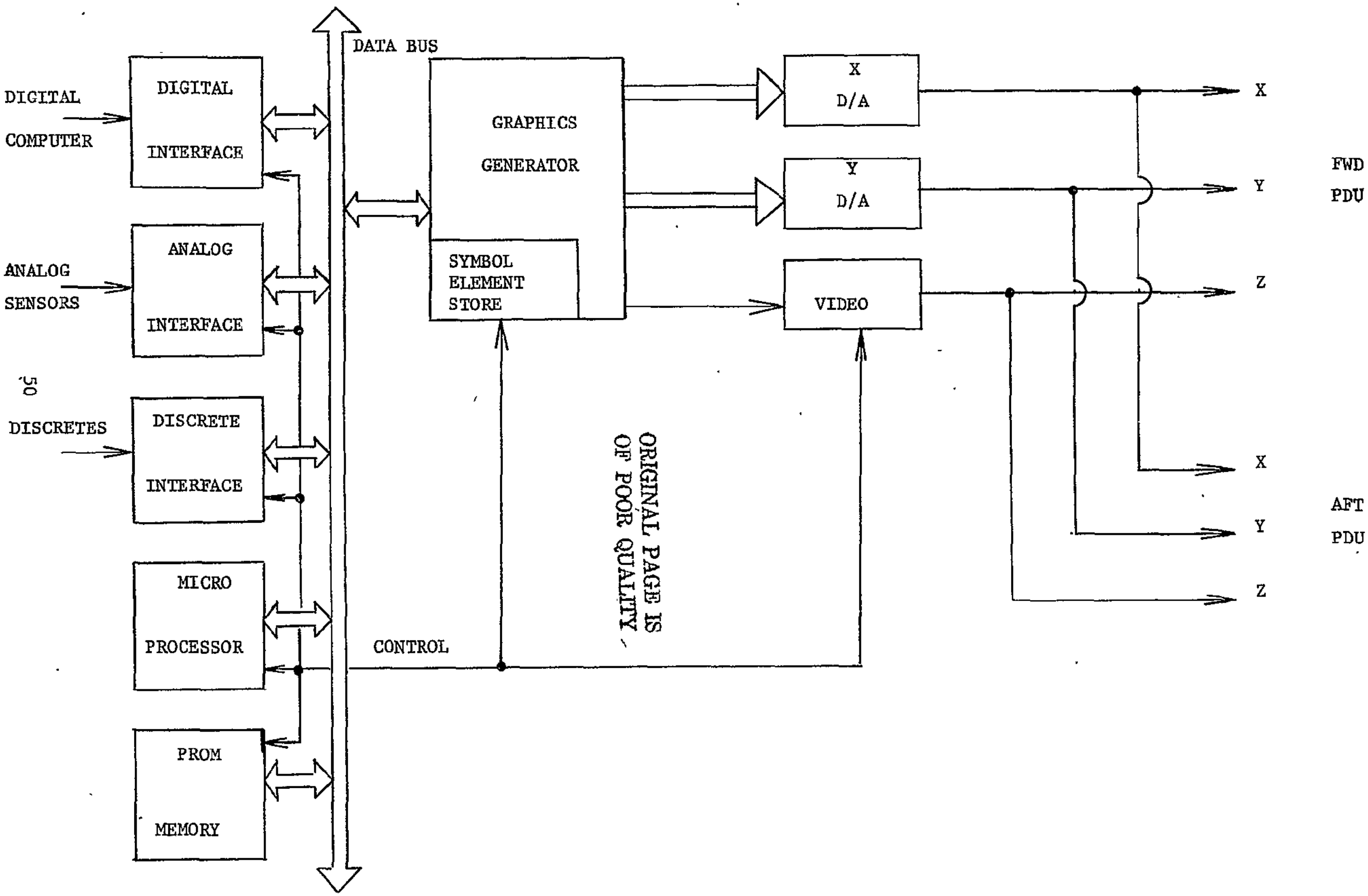
- Develop and debug the desired program (FORTRAN) using a host computer tied into the programming adapter or insert desired program manually via the adapter keyboard.
- Replace adapter PROM simulator with new PROM element and transfer program into PROM element.
- Install new PROM in PDP and verify symbology.

The fact that the PDP has an analog interface permits deletion of the Interface/Weapon Aiming Computer (IWAC), for a weight savings of 14.5 pounds.

The aircraft modifications required to install the PDP in the NASA Research aircraft include:

- Delete the HUD Display Waveform Generator (DWG) and Interface/Weapon Aiming Computer (I/WAC).
- Install Programmable Display Processor (PDP).
- Delete the wiring that presently exists between the DWG and I/WAC.

FIGURE 2-37 PROGRAMMABLE DISPLAY PROCESSOR BLOCK DIAGRAM



- Add approximately 10 new wires from the PDP to various aircraft sensors.
- Add digital interface wiring (2 shielded pairs) from the aircraft digital computer to the PDP.
- Reterminate the existing I/WAC wiring, as required, to be compatible with the PDP.

2.4 LANDING GUIDANCE SYSTEM

The Microwave Landing System (MLS) installation provides steering information to both cockpits for the landing guidance function. The Bendix landing guidance system (GFE) consists of the following Weapon Replaceable Assemblies (WRAs):

- MLS Angle Receiver
- Receiver Mount
- MLS Control Panel
- C-Band Antenna
- DME Interrogator
- DME Indicator
- L-Band Antenna

The mechanization includes the flight director computation within the airborne computer. Steering information is presented both head-up on the HUD and head-down on the ADI.

An automatic landing interface with the digital fly-by-wire system is not provided. The TACAN system is retained as a separate mechanization. A Digital Adapter Unit provides buffering and scaling of interface parameters. A trimetric view showing WRA locations is presented in Figure 2-38. Physical descriptions are presented in Figure 2-39.

2.5 INERTIAL NAVIGATION SYSTEM

The GFE Litton LTN-51 Inertial Navigation System will provide data to the airborne computer and pilot displays, both head-up and head-down. The system consists of an LTN-51 Inertial Navigation Unit, INS Control Display Unit, and INS Mode Select Unit.

The Inertial Navigation Unit will be mounted on the aft equipment bay shelf. The INS Control Panel will be mounted in the forward cockpit on the right hand console where the ARC-114 radio is now located. The INS Mode Select Panel will be mounted in the forward cockpit on the right hand main instrument panel. The LTN-51 will be loaded with the appropriate INS program prior to delivery to MCAIR.

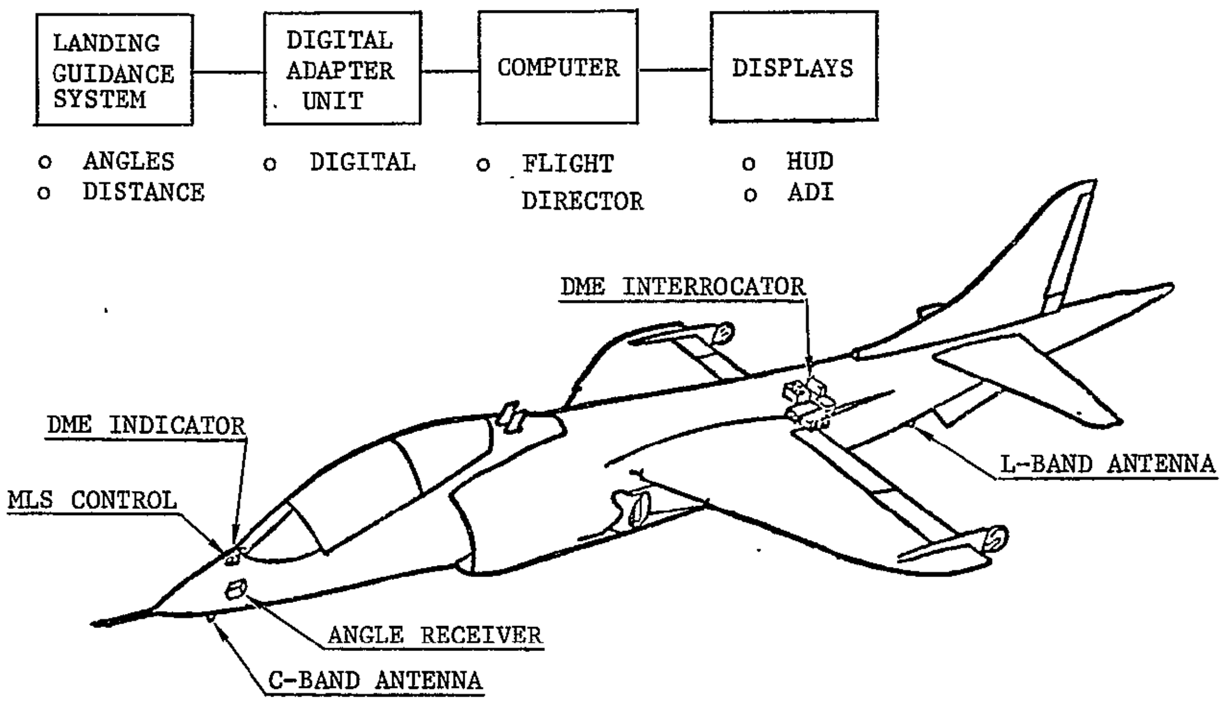


FIGURE 2-38 MLS WEAPON REPLACEABLE ASSEMBLY LOCATIONS

WRA	SIZE			VOL	WT
	H	W	D		
ANGLE RECEIVER	7.84	3.73	12.62	369	9.5
MOUNT	3.31	3.63	14.34	IRREG	1.8
MLS CONTROL	3.0	5.75	4.17	72	1.8
C-BAND STUB	2.0	2.12	5.1	IRREG	0.3
DME INTERROGATOR	7.625	5.25	12.5	500	10.5
DME FREQ SELECT	3.4	2.25	5.2	40	2.2
DME INDICATOR	1.5	3.21	8.5	41	1.1
L-BAND ANTENNA	3.23	0.92	3.97	IRREG	0.4

FIGURE 2-39 WEAPON REPLACEABLE ASSEMBLY PHYSICAL DESCRIPTIONS

2.6 SIMULATION

Simulation studies associated with the NASA Two Place V/STOL Research Aircraft will include the evaluation of the flight control system modifications, associated flight safety aspects and failure effects. These studies will be conducted on the NASA Ames Research Center's Flight Simulator for Advanced Aircraft (FSAA). The FSAA simulation studies will be conducted by NASA with MCAIR technical assistance. Manned flight simulations at MCAIR will be limited to verification of the two place Harrier simulation model, checkout of digital control laws which will be provided by MCAIR, and familiarizing NASA pilots with the two place aircraft flight characteristics.

The simulation program will consist of three phases as indicated in Figure 2-40.

2.6.1 PHASE I - During the first phase of the simulation program MCAIR will prepare a simulation data package which will include aerodynamics, propulsion, flight control and landing gear characteristics and HUD displays in the latest Harrier format. MCAIR will flight test the two place Harrier model using MCAIR manned simulator facilities.

After the aircraft simulation has been verified, it will be used to familiarize NASA pilots with the handling qualities and flight characteristics of the aircraft. A three day flight simulation familiarization program for two Harrier qualified NASA pilots is suggested. Familiarization will include VTO, hover, transition, short takeoff, short landing, and high speed maneuvers. Ground effects, winds and turbulence will be simulated. Flights will be conducted with stability augmentation on and off.

The first phase of the simulation program will require approximately nine weeks to complete.

2.6.2 PHASE II - During the second phase of the simulation program the NASA Ames Research Center's FSAA simulation of the two place Harrier will be developed. At the start of the second phase MCAIR will deliver the simulation data to NASA. The package will include sufficient check cases to permit verification of the simulation. The programming of the FSAA Sigma 8 digital computer will be performed by NASA. MCAIR will provide technical assistance for:

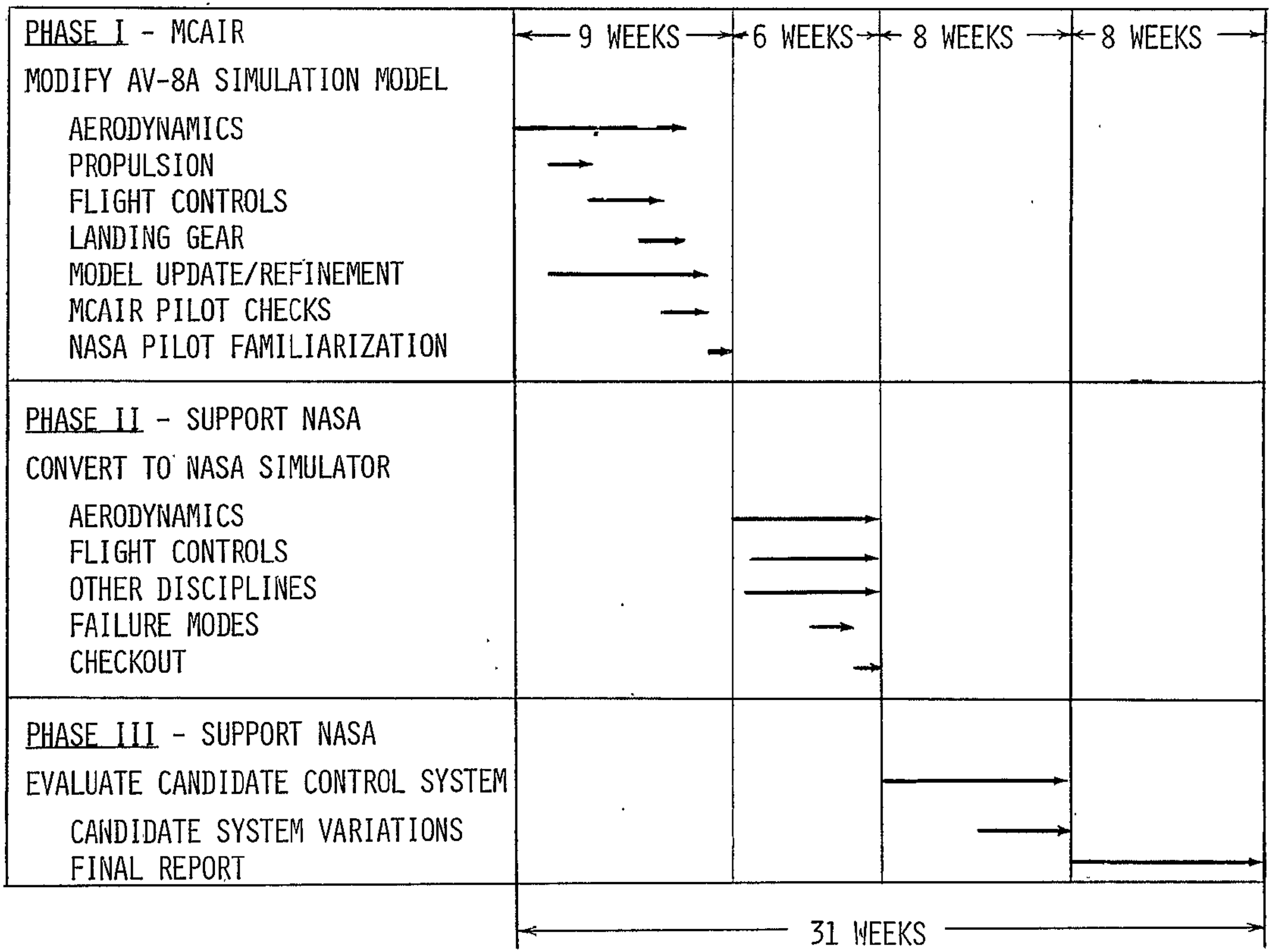
- o Aircraft Simulation Generation.
- o Control System Failure Mode Simulation.
- o Software/Hardware Checkout.
- o Data Requirements, Formats and Units.

The two cockpits required to generate the FSAA simulation will be provided by NASA. One cockpit will be used to simulate the two place Harrier front (safety pilot) cockpit, the other will be used to simulate the aft (evaluation pilot) cockpit. All hardware for the two cockpits will be furnished by NASA. Check cases provided by MCAIR will be used to verify the software conversion including the control system failure effects. Use of a spare flight control research computer for onboard computations within the simulation is suggested.

The second phase of the simulation program will require approximately six weeks to complete.

FIGURE 2-40 NASA V/STOL RESEARCH AIRCRAFT

SIMULATION SCHEDULE



2.6.3 PHASE III - During the third phase of the simulation program the FSAA simulation will be used to evaluate candidate systems, and to examine flight safety implications and flight system failure effects. At the start of the third phase MCAIR will provide a simulation test plan for NASA use. The test plan, balanced insofar as possible for the effects of pilot learning, pilot fatigue, time of day, day of test and candidate order of appearance, will provide for a sufficient number of flight operating points so that sufficient data of quality will be generated to permit the determination of the statistical significance of results. NASA Ames will furnish Harrier qualified pilots, operating personnel and technicians. NASA Ames will generate and analyze the data. MCAIR will provide technical assistance for the simulation, reduction and analysis of data and for report preparation.

Phase III may be conducted as a continuous eight week simulation or two four week simulations with sufficient time to analyze data and modify the control system.

2.7 SYSTEM SOFTWARE

The onboard computer will implement the control law equations, inflight integrity management and system BIT. It will also generate steering commands for the display system. The computer program will be organized in a modular design in which program functional responsibilities will be partitioned into well-defined modules. Figure 2-41 shows each of the major software modules that make up the computer program and their respective submodules. An estimate of the computer memory requirement for the simplex configuration is also shown.

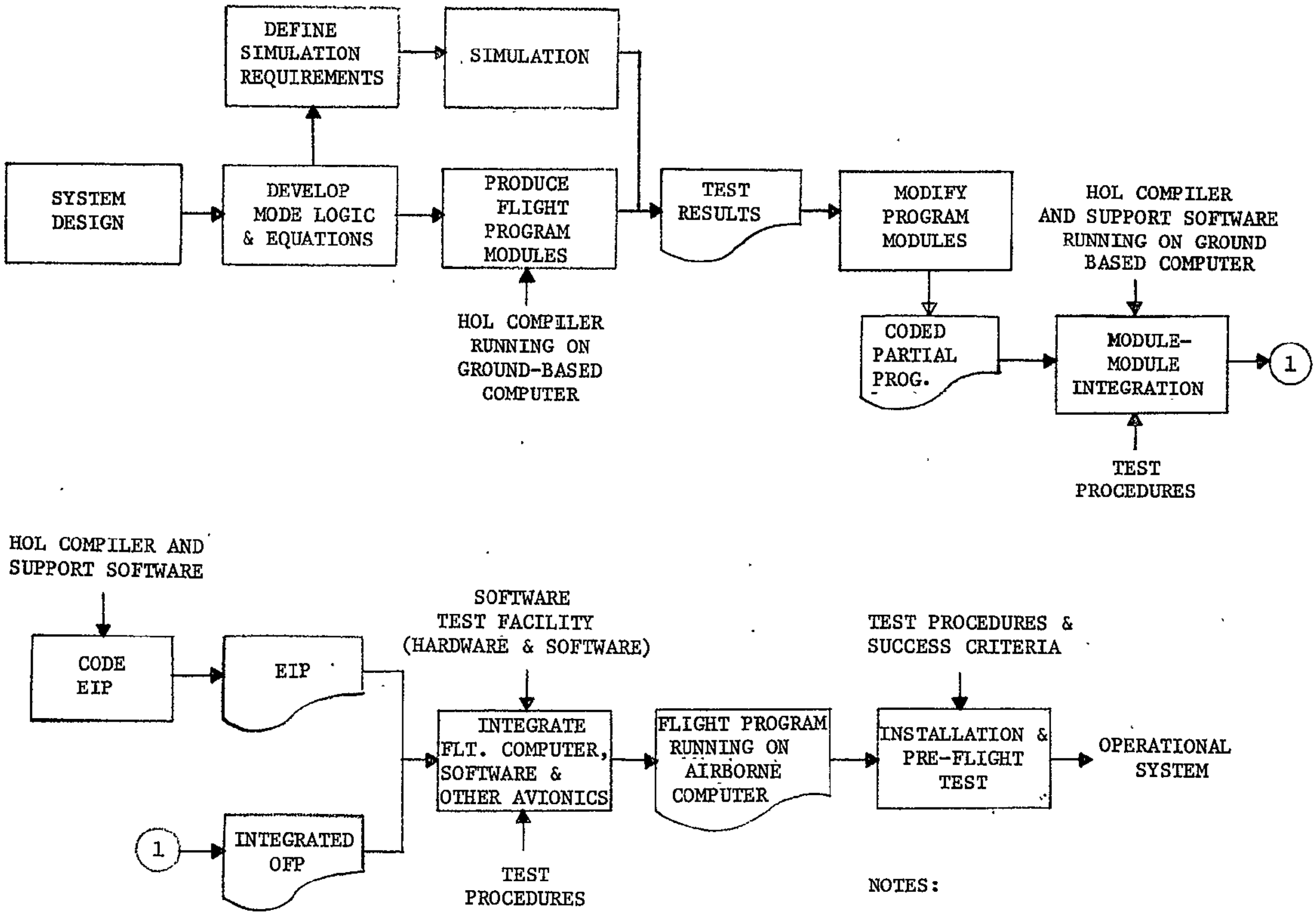
2.7.1 COMPUTER SOFTWARE DEVELOPMENT - Development of the computer software requires that NASA furnish MCAIR with the latest version of the HAL/S compiler. Required from the computer vendor will be the support software needed in program development (linkage editor, functional simulator, and utilities) as well as preflight and inflight computer self-test program modules.

An overview of the software development process is presented in Figure 2-42. As shown, the Flight Control System software will evolve during the System Design Phase which will be devoted to problem analysis, planning and establishing standards for subsequent software activities. Specific tasks to be accomplished during this phase include:

- Generation of Integration Block Diagrams (IBDs).
- Extraction of major computation tasks from IBDs.
- Establishment of timing and program operation.
- Determination of modes and mode switching requirements.
- Resolution of logic for selection of secondary modes.
- Establishment of variables, iteration rates range scaling, format and engineering units for computer interface signals.
- Organization of software standards.
- Generation of test specifications and software design verification.

SOFTWARE MODULE	STORAGE (16 BIT WORDS)	
Control Laws		
- Solution of Control Law Equations & Algorithms	2000	
- Digital Filtering	300	
		2300
Inflight Integrity Mgmt		
- Data Reasonableness Check	2100	
- Aircraft Departure Limit Check	1400	
- Predetermine Hazard Situation Check	1600	
- Computer Self-Test	1000	
		6100
BIT		
- Power Supply Tests	100	
- Sensor Tests	600	
- Actuator Tests	400	
- Digital Interface Tests	700	
- Switch Tests	200	
- BIT Subexecutive & Display Routine	300	
		2300
Executive		
- Program Initialization	500	
- Interrupt Processing	700	
- Program Scheduling	200	
- Input/Output Processing	500	
- Mode Switching Logic	700	
		2600
Display		
- MLS/Flight Director		2400
TOTAL		15,700

FIGURE 2-41 COMPUTER MEMORY REQUIREMENTS - SIMPLEX



NOTES:

OFP - OPERATIONAL FLIGHT PROGRAM
EIP - ELECTRICAL INTERFACE PROGRAM
HOL - HIGHER ORDER LANGUAGE

FIGURE 2-42 SOFTWARE DEVELOPMENT (HOL)

(a) Development of Program Modules - After the system design has been completed and the software requirements defined, the actual programming phases of the software development will begin by generating top-level flow diagrams from the IBDs. Equations to be solved will then be formulated and modified to a form suitable for a digital computer. Detailed math flows will be created, the flight program will be organized into functional modules and will be programmed in the HAL/S language. The program will be compiled and the program modules debugged.

Each individual software module will be tested in an isolated environment before combining it with other modules. The program module will be tested on an IBM 360/370 with a test driver which is a special test routine that provides the proper test environment by simulating inputs and outputs to the module.

The individual modules will then be tested on a host computer (IBM 360/370) using a functional simulator. This simulator is a computer program that provides a bit-by-bit simulation of the airborne computer instruction set while executing on the host computer. The functional simulator will be used in conjunction with a User's Control Program, a Pathfind program and Dynamic Statistic Program, all of which were developed by MCAIR for the F-15 Program. These programs will be modified for use on this program.

The software modules will then be modified as a result of the functional simulator tests and as a result of MCAIR manned flight simulation.

(b) Module - Module Integration - As groups of modules are completed and tested they will be integrated. The module integration will be completed when the individual module source decks or tapes have been combined by the linkage editor into a single object deck or tape. This tape will include the inflight diagnostic support software which is furnished by the computer vendor.

The integrated modules will then be tested much as the individual modules were tested, i.e. using test drivers, and performing tests with the functional simulator and associated programs. The integration and testing will be continued at ever-higher program levels until the complete operational program is debugged and verified. An OFP tape will then be prepared in object computer language for insertion into the AP-101 computer for on-line testing.

(c) On-Line Software Test - The flight computer program which was validated off-line will next be tested on-line in a Software Test Facility. The Software Test Facility will contain:

- A flight computer loaded with the flight program.
- An auxiliary computer loaded with the software test facility operating system program.
- Tapes containing software utility routines and an assembler for the auxiliary computer.
- An interface unit which interfaces the flight computer and auxiliary computer.
- Peripheral equipment which interfaces with the auxiliary computer.

The auxiliary computer program will simulate the aircraft dynamic environment and airborne computer peripheral avionic subsystems and will exercise the airborne computer to operate the flight program in real time.

(d) Hardware/Software Bench Integration - After the OFP has been tested with simulated peripherals the AP-101 computer will be connected with avionic system peripheral systems and tested one step at a time until the airborne avionics system has been tested. In some cases where avionics systems cannot, from a practical viewpoint, be connected into the avionic system, simulated inputs will be used to substitute for these systems. A summary of software test requirements is shown in Figure 2-43

(e) Support Software Modification - The support software which is used in flight program generation and verification and which must be modified is shown in Figure 2-44.

2.8 GROUND TESTS

The ground test program will demonstrate that the new or modified components, subsystems and systems are flightworthy. In the main, these activities will be conducted prior to first flight and can be categorized as follows:

- o Component Airworthiness Tests.
- o Ramp Operations.
- o Aircraft/Software Verification.

2.8.1 COMPONENT AIRWORTHINESS TESTS - Modified or newly designed components will be qualified during Vendor Airworthiness Tests. During system buildup and integration, additional tests of components may be required to further evaluate operational or functional characteristics.

2.8.2 RAMP OPERATIONS - The weight and center of gravity of the aircraft will be determined. A ground vibration test of the rudder will be performed. Free play and rigidity and frequency response of the aileron, flap, horizontal tail and rudder will be tested. Accelerometers will be mounted on the equipment shelf and in the gunpod. Their outputs will be recorded during engine ground runs to check out equipment installations.

A complete control system checkout of the modified control system will be performed and this functional check will include items such as the digital computer, inertial platform, sensors and software.

2.8.3 AIRCRAFT/SOFTWARE VERIFICATION - MCAIR will design and build a flight line analyzer (FLA) to permit rapid flight control system checkout on the aircraft. It will provide open loop stimuli to the flight control system to the extent necessary to assess the integrity of the system for flight. System software checks performed by the FLA will be simplified tests such as memory scan checks. This equipment will be delivered with the aircraft.

FIGURE 2-43 SOFTWARE TEST REQUIREMENTS

TEST LEVEL	TEST FACILITY	TEST EQUIPMENT	TEST & SUPPORT SOFTWARE	TEST DOCUMENTATION
OFF- LINE TEST	GENERAL COMPUTING FACILITY	o HOST COMPUTER (IBM 360) AND PERIPHERALS	HAL/S COMPILER, LINK-EDITOR FUNCTIONAL SIMULATOR, USER CONTROL PROGRAM, PATHFIND, UTILITY PROGRAMS FOR HOST COMPUTER, DATA BASE CATALOG.	NO FORMAL DOCUMENTATION
ON-LINE TEST	SOFTWARE TEST FACILITY	o AUX COMPUTER – DATACRAFT 6024 o FLT. COMPUTER AP-101 o INTERFACE UNITS o AUX. COMP. PERIPHERALS o COMPUTER TEST SET – LOADER VERIFIER	STF OPERATING SYSTEM SOFTWARE, AUX. COMP. ASSEMBLER, STF UTILITY PROGRAMS	NO FORMAL DOCUMENTATION
HARDWARE/SOFTWARE BENCH TEST	SOFTWARE TEST FACILITY	o SAME AS ON-LINE TEST PLUS – SENSORS, ACTUATORS	SAME AS ON-LINE TEST PLUS ELECTRICAL INTERFACE PROGRAM.	HARDWARE/SOFTWARE TEST SPEC.
INSTALLATION AND PRE-FLIGHT	AIRCRAFT	o COMPUTER TEST SET – LOADER VERIFIER o PRE-FLIGHT TEST SET	o COMPUTER LOAD ROUTINE	ACCEPTANCE TEST SPEC.

	Function	Program Size	Per Cent Rework
STF Operating System	Provides real-time Inputs to Airborne Computer from modeled aircraft peripheral systems previously recorded on mag tapes in time history format. Debugging facilities for airborne computer program.	8000 words (24 BIT)	30
User Control Program (UCP)	Airborne Computer Functional Simulator Interface with Environment Program.	56K Bytes	50
Pathfind	Preprocessor to Compiler to source cards to floating point coding to enable path analysis	16.5K Bytes	50
Dynamic Statistics	Analysis of Trace Lines Output from UCP and Functional Simulator to give dynamic statistics.	29K Bytes	50
Data Base Catalog	Bookkeeping program for maintenance and sorting of OFP Data Base.	29K Bytes	20
Electrical Interface Program	Checks Airborne Computer I/O Interface with all peripherals	119K Bytes	80

FIGURE 2-44 SUPPORT SOFTWARE MODIFICATION

2.9 AIRWORTHINESS TESTS

The functional test flights will be conducted at the MCAIR, St. Louis facility. Five flights will be performed over a one month period to check out the basic aircraft control system and the airborne data acquisition system. The digital fly-by-wire flight control system will not be turned on during the MCAIR flight tests. The instrumentation ground support equipment, to be supplied GFE, will be required at St. Louis during the flight test period and also during the preceding ground test of the fly-by-wire control system.

The functional checkout of the basic airplane will be accomplished in accordance with NAVAIR 01-AV8A-1F, NATOPS Functional Checkflight Checklist, modified as required to accommodate the aircraft configuration. Additional flights, over and above those normally required to complete the NATOPS checklist, will be required to evaluate the frictional effects of the fly-by-wire system servos attached to the basic flight control system. Maneuvers will be performed throughout the conventional flight envelope and in the V/STOL mode.

3. OPTIONAL MODIFICATIONS

Seven optional modifications to the two place Harrier were investigated in this conceptual design study:

- o A duplex digital flight control system.
- o A triplex digital flight control system.
- o Simplex throttle and nozzle control systems.
- o Duplex throttle and nozzle control systems.
- o Triplex throttle and nozzle control systems.
- o A low speed air data system.
- o A side arm controller.

If either the duplex or triplex flight control system is selected rather than the simplex system the basic modifications described in Sections 2.2 through 2.9 are still required.

3.1 DUPLEX FLIGHT CONTROL SYSTEM

The modifications to produce the duplex flight control system were established on the basis that the rear cockpit will be the evaluation pilot station and the forward cockpit will be the safety pilot and solo pilot station. Sufficient information will be provided in the front cockpit to enable the safety pilot to monitor the activity of the evaluation pilot and of the digital flight systems and to disengage these systems should the necessity arise. The front seat pilot can then fly the aircraft using an essentially unmodified Harrier flight control system.

3.1.1 CONTROL SYSTEM MODIFICATIONS - The design approach for implementing the duplex flight control system differs significantly from the simplex design approach in two respects. First, the aft cockpit's control stick and rudder pedals remain mechanically connected to the Harrier flight controls. Second, dual channel electrohydraulic series servos are used for system implementation rather than single channel parallel electromechanical servos.

A schematic diagram of the dual series longitudinal flight control system is given in Figure 3-1. Since the aft cockpit control stick remains mechanically connected to the Harrier flight control system, the stick in the forward cockpit will move in unison with the aft cockpit stick. This means that the safety pilot could monitor the evaluation pilot's control stick movements by monitoring the motion of the forward control stick. However, aft stick motions will not necessarily be closely correlated with the stabilator and RCV inputs due to the full authority series servos.

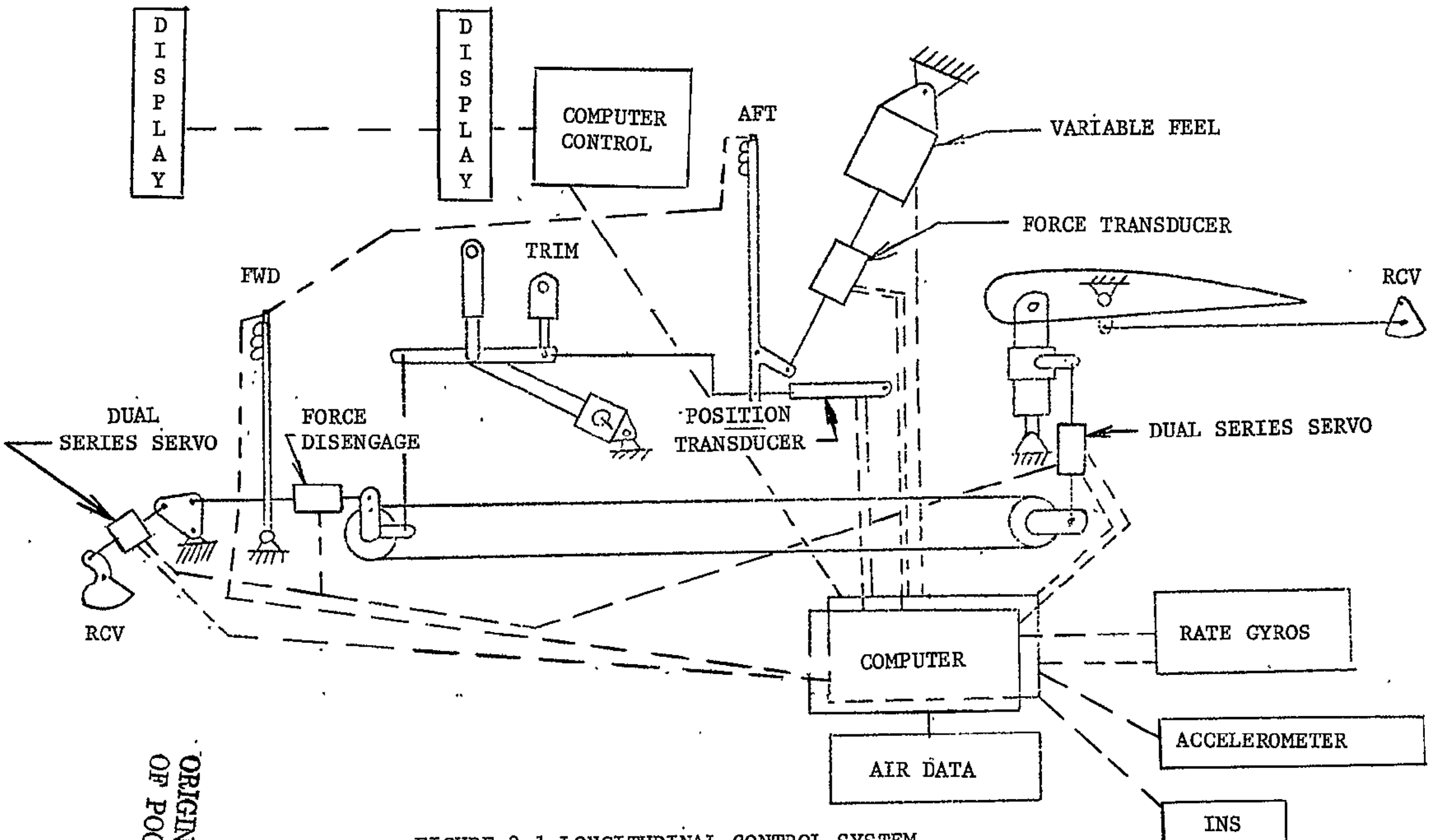


FIGURE 3-1 LONGITUDINAL CONTROL SYSTEM
DUAL SERIES

Dual position and force transducers are attached to the aft control stick. The digital computers, the pitch rate gyros, the series servo added at the tailplane, and the series servo driving the forward RCV are also dual. All other elements, including the normal accelerometer, are simplex.

Signals from the aft control stick position and force transducers are processed by the digital computers along with accelerometer and rate gyro signals and, when needed, with information from the inertial navigation and air data systems. The digital computers compute command signals for the forward RCV dual series servo and the dual series servo at the tailplane. These servo motions are then combined with the mechanical motion of the Harrier's longitudinal control system to provide the appropriate positioning of the stabilator and RCVs. If required, the digital computers can also send command signals to the limited authority series servo built into the production stabilator actuator in order to augment the frequency response capability of the dual series servos. Trim is accomplished through the production trim system.

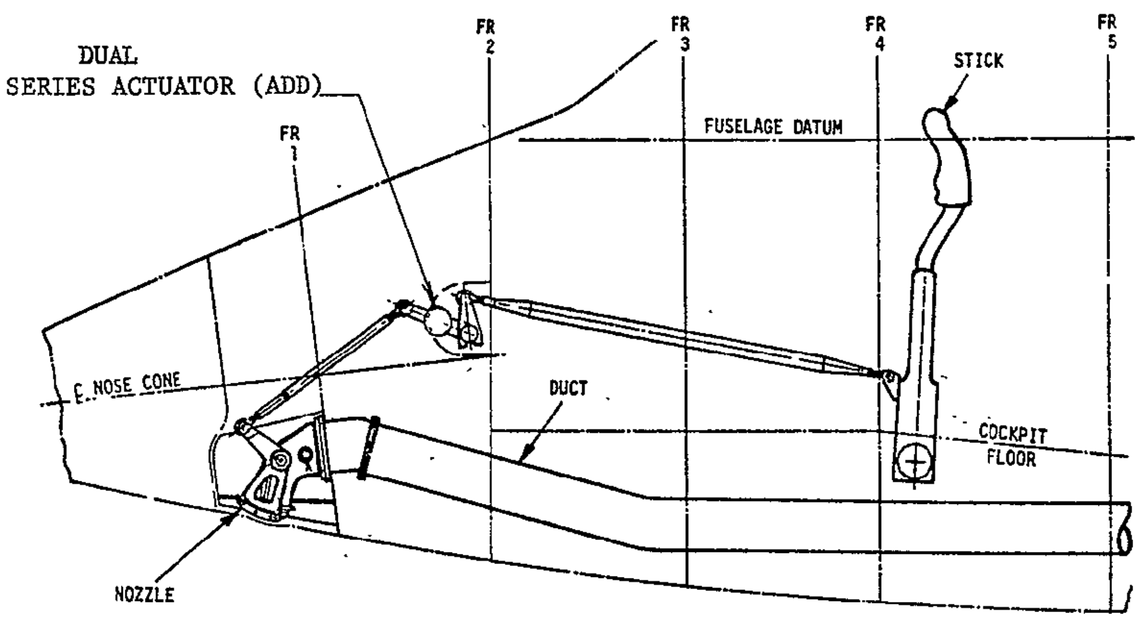
The dual series actuator installation for the forward RCV is shown in Figure 3-2. This installation is similar to the forward RCV series servo developed for the YAV-8B except a dual electromechanical servo actuator is used. Dual solenoids provide fast recentering of the servo when the digital fly-by-wire flight control system is disengaged.

The dual series servo installation at the tailplane is shown in Figure 3-3. The servo actuator consists of two F-4 lateral series servos which have had the orifices in their recentering circuits opened up in order to obtain near instantaneous recentering when the fly-by-wire system is disengaged. The servo actuator is connected to the actuator valve and the production Harrier flight control system through a "walking beam bellcrank." A typical walking beam bellcrank is shown in Figure 3-4. As shown in Figure 3-5, the mechanical input from the pilot and the mechanical input from the servo actuator are added to obtain a mechanical motion for the stabilator actuator valve.

A schematic diagram of the dual series lateral control system is given in Figure 3-6. Dual position and force transducers attached to the aft control stick provide pilot command signals to the two computers. Signals from the two roll rate gyros and possibly air data and signals from the inertial navigation systems are processed by the digital computer to generate a command signal for the dual series servo actuator. The dual series servo output motion combines with the mechanical motion generated through the pilot's stick motion to provide the actuator valve motion for the port and starboard aileron actuators. Trim is provided by the production trim system.

The aileron dual series servo installation is shown in Figure 3-7. A walking beam bellcrank type mechanism combines the motion from the pilot's stick and the motion from the servo to generate the control motions sent to the aileron actuators. The actuator is similar to the series electrohydraulic actuator used for the dual series longitudinal control system.

A schematic diagram of the dual series directional flight control system is shown in Figure 3-8. Position and force transducers attached to the aft rudder pedals provide pilot input signals to the two digital computers. These signals are processed with signals from the two rate gryos and the accelerometer, and possibly with inertial navigation and air data signals, to generate commands for



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VIEW LOOKING INBOARD

FIGURE 3-2 SERIES ACTUATOR INSTALLATION , FORWARD PITCH RCV

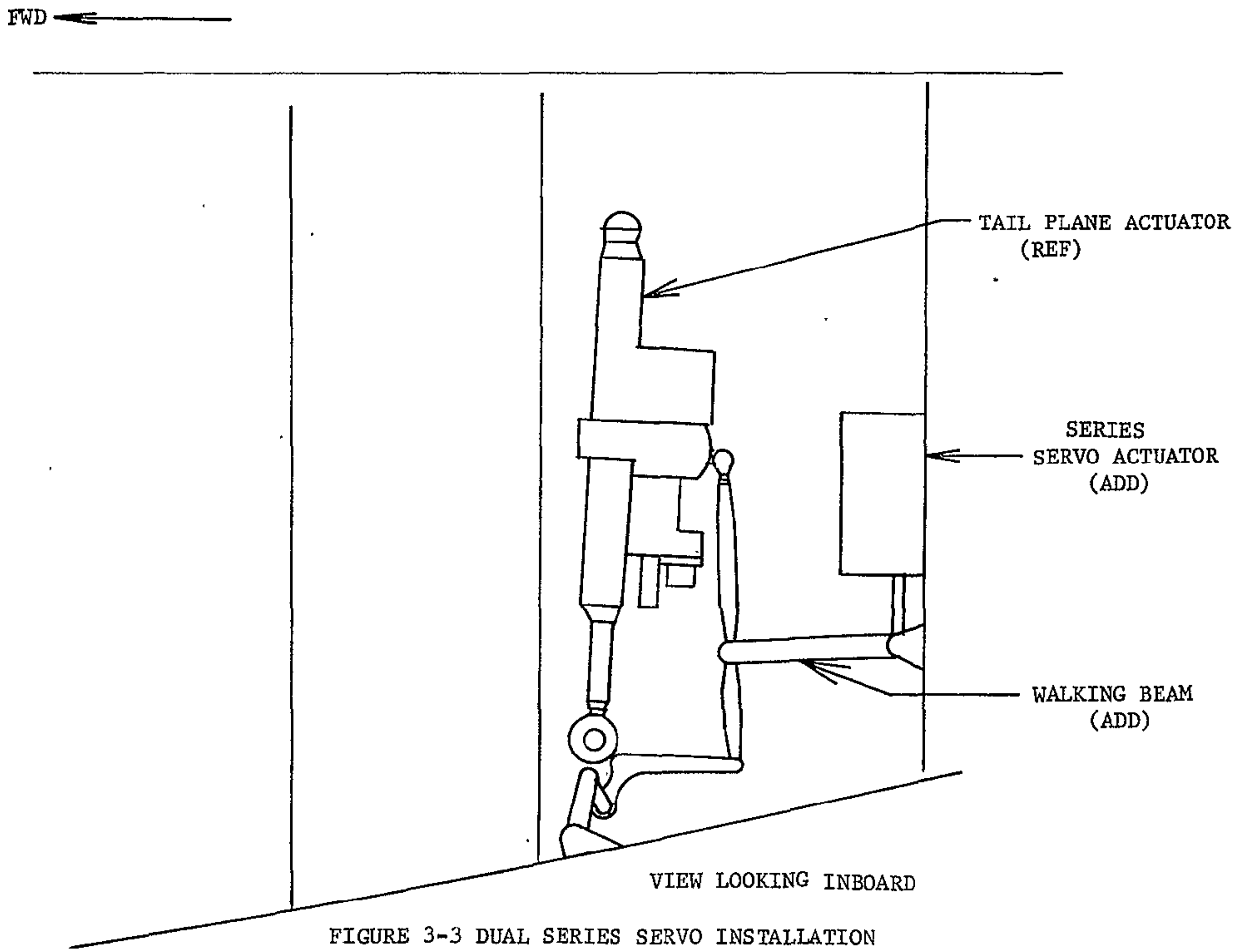

FIGURE 3-3 DUAL SERIES SERVO INSTALLATION
LONGITUDINAL SYSTEM

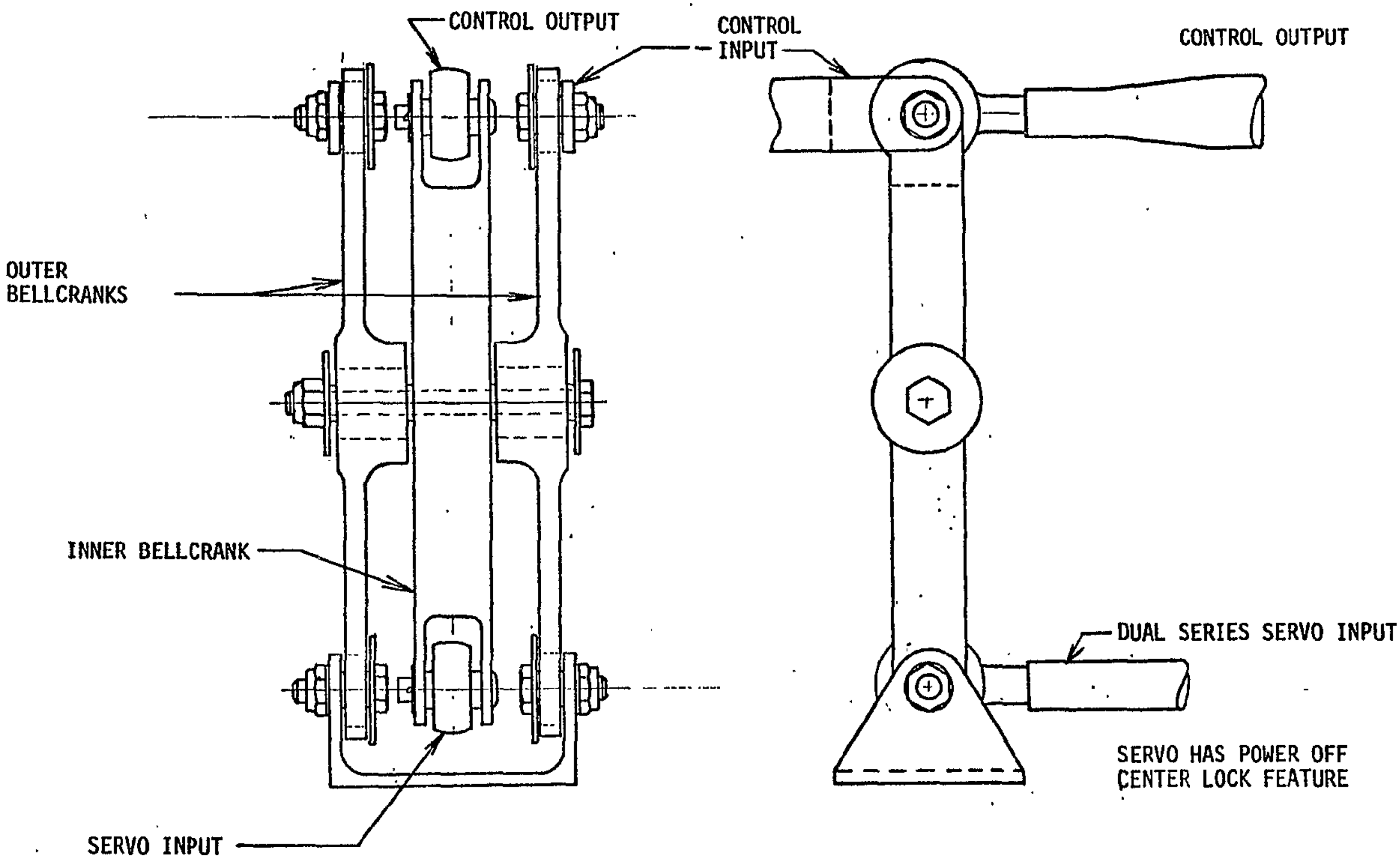

FIGURE 3-4 WALKING BEAM BELLCRANK OPERATION

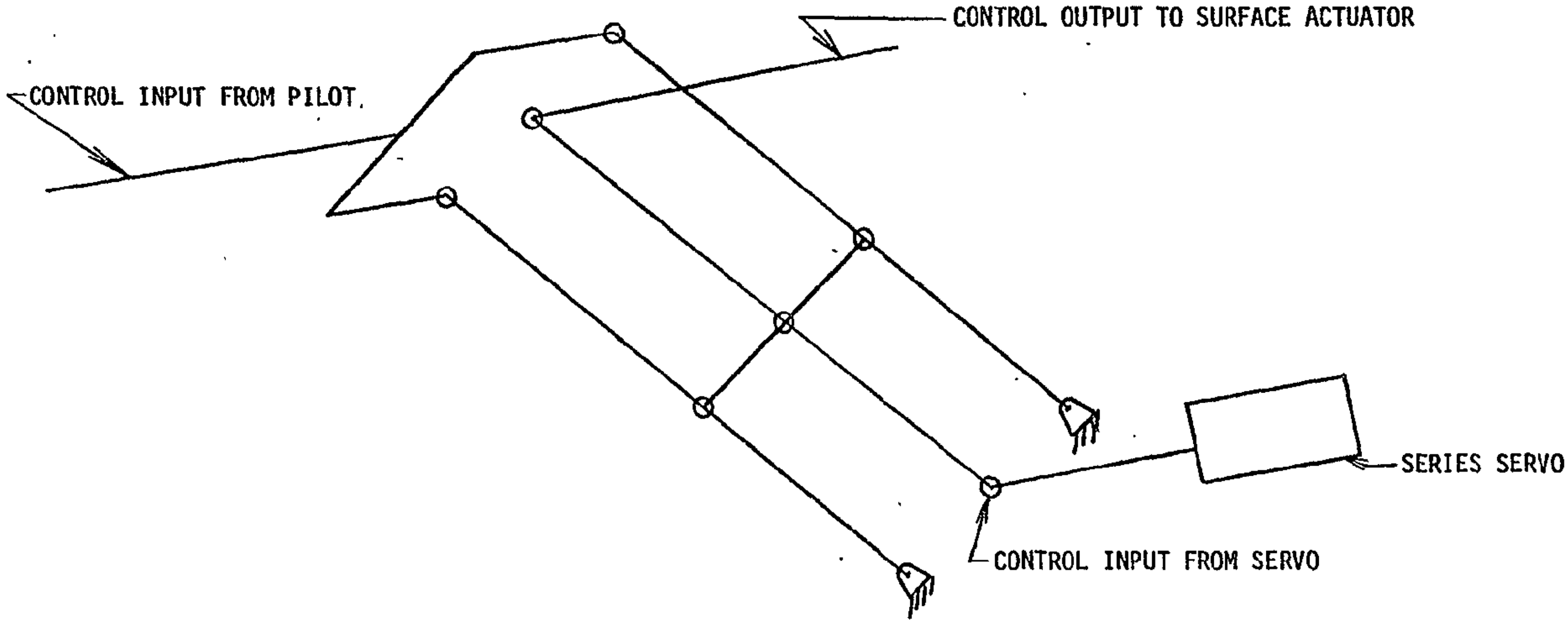

FIGURE 3-5 WALKING BEAM BELLCRANK SCHEMATIC

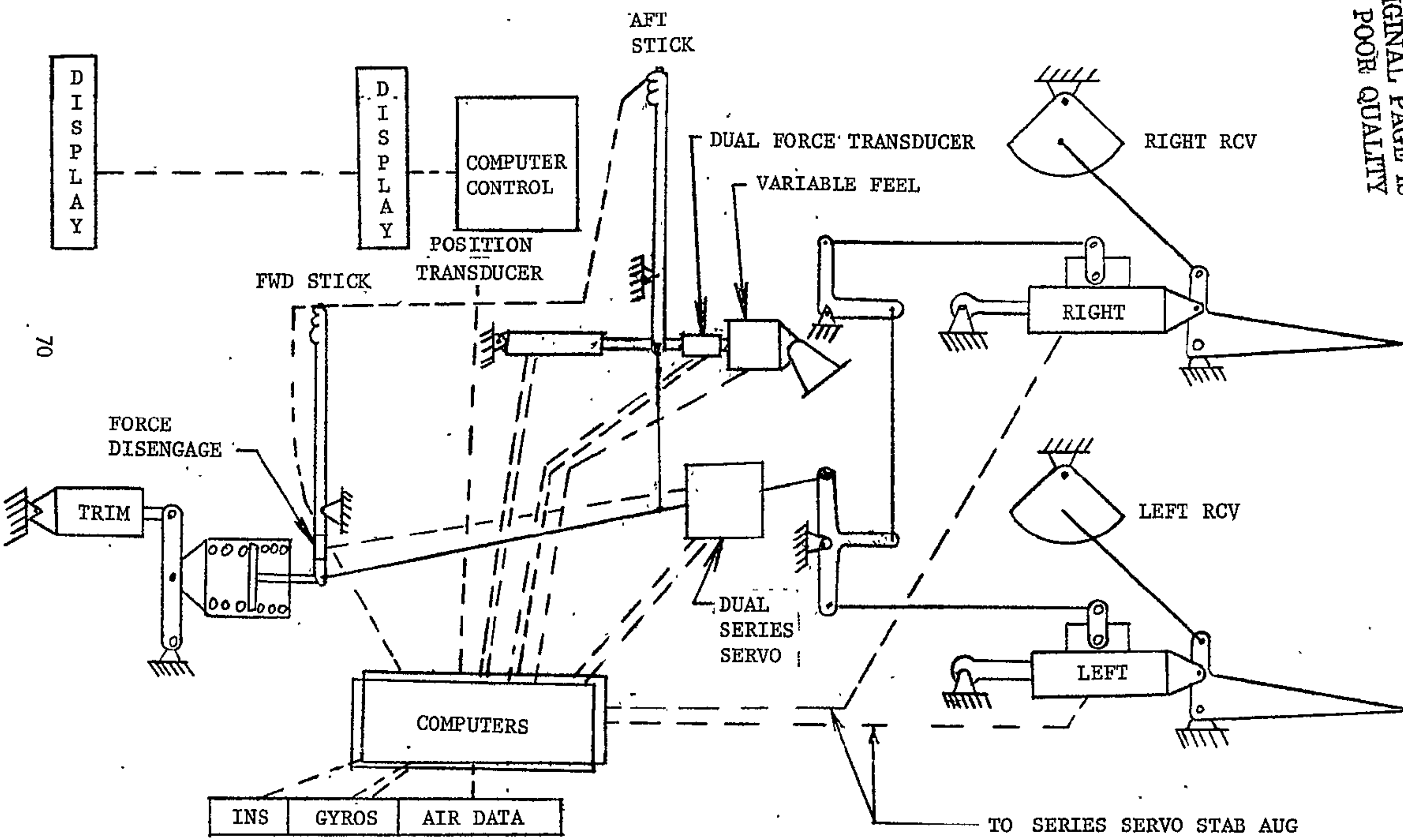

FIGURE 3-6 LATERAL CONTROL SYSTEM
DUAL SERIES

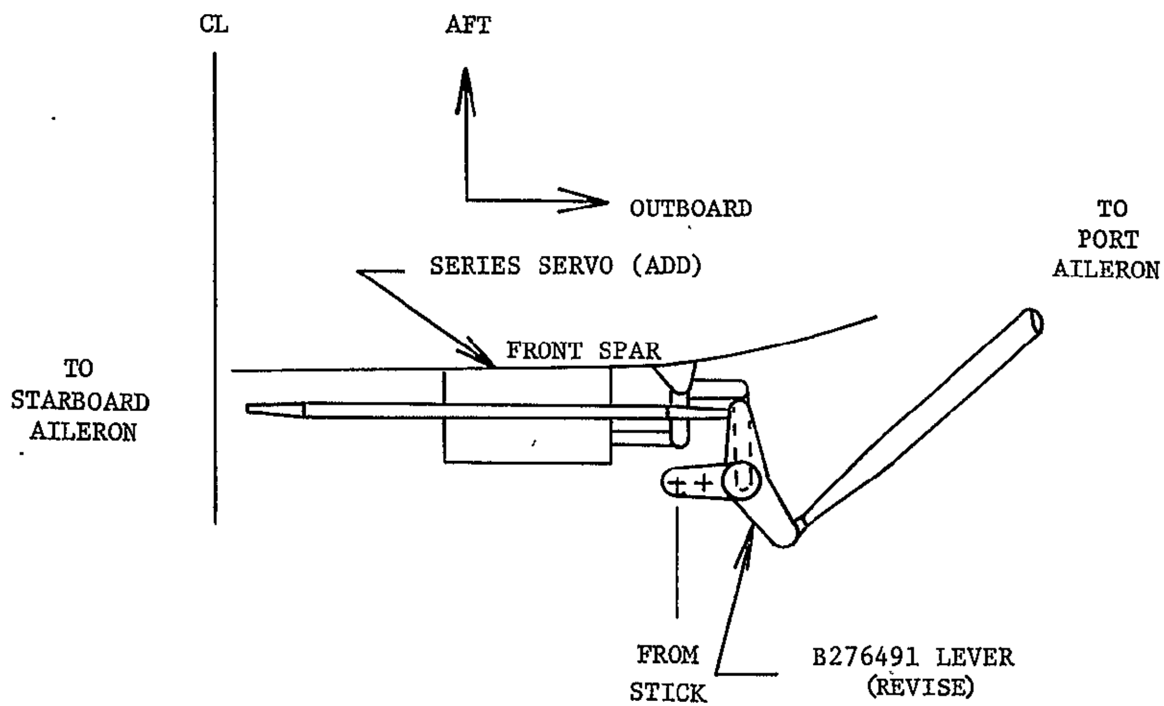

FIGURE 3-7 DUAL SERIES SERVO INSTALLATION
LATERAL SYSTEM

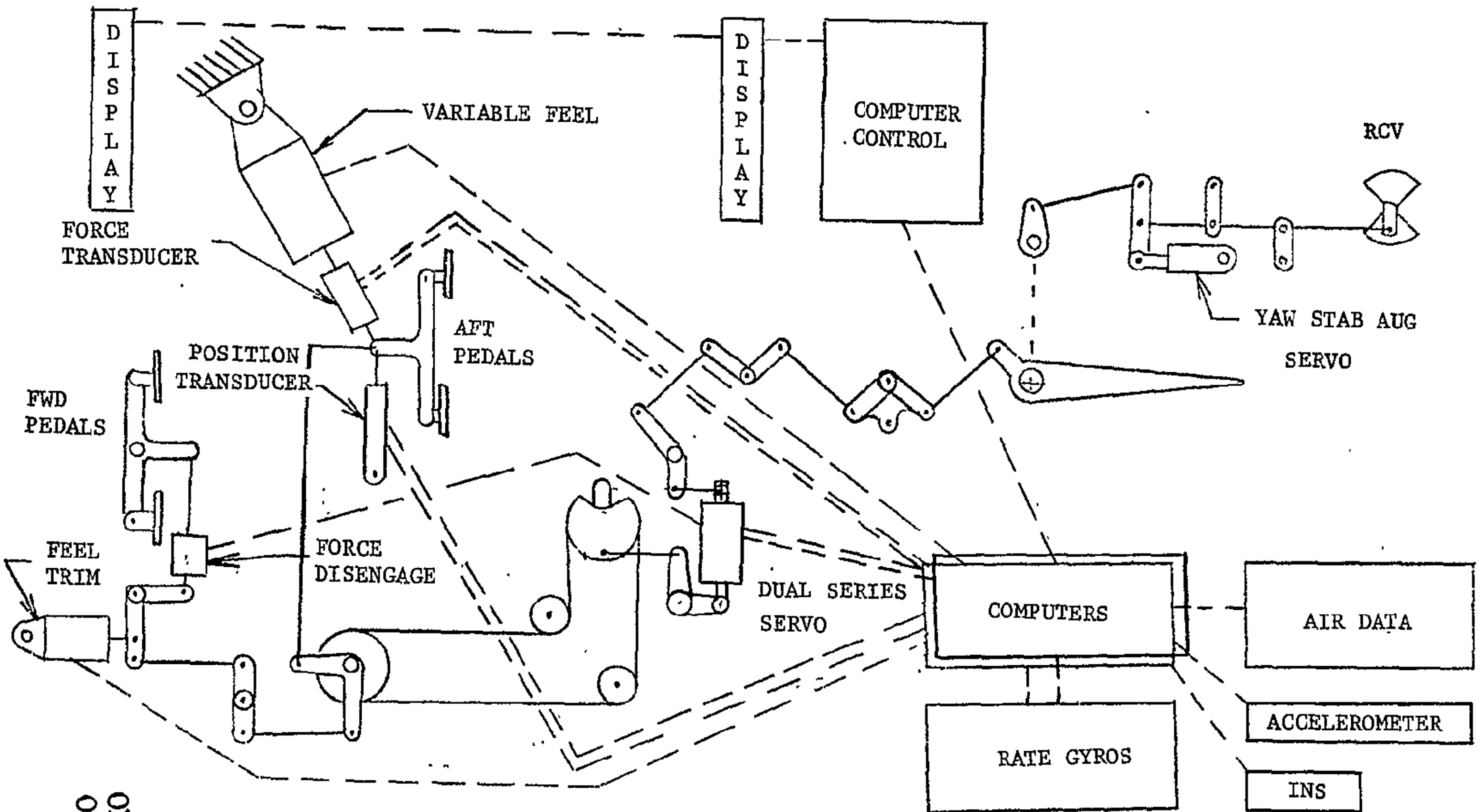

FIGURE 3-8 DIRECTIONAL CONTROL SYSTEM
DUAL SERIES

the dual series actuator. Trim is accomplished by adding a trim actuator to the forward cockpit area of the directional control systems.

Figure 3-9 shows the installation of the dual series servo actuator. This actuator consists of two F-15 rudder actuators combined in such a manner that a hydraulic actuator having a mechanical input and a two channel series electrical input is obtained. The mechanical input is to the actuator and is generated by the pilot through the rudder pedals. The electrical inputs are provided by the computer. When the digital fly-by-wire flight control system is disengaged the safety pilot will have a powered rudder rather than the manual rudder of the production Harrier.

The installation of the force and position transducer on the aft stick and rudder pedals is similar to that used for the simplex system discussed in Section 2.1.1. Also, the safety pilot can disengage the fly-by-wire system in the same way he can disengage the simplex system.

(a) Cockpit Information and Disengage Requirements - The cockpit installation of the duplex fly-by-wire system is nearly identical to that of the simplex system. The pilots make the same input to the system in either case. The dual system will have a series of additional warning lights to isolate a problem in a failure situation. See Section 2.1.2 for the description of the pilot functions since they are the same for both simplex and the duplex systems.

(b) Computer Hardware - The two onboard computers are the same as the single computer used in the simplex configuration and described in Section 2.1.3.

(c) Hydraulics - The duplex flight control system requires the addition of two channel full authority series actuators to each axis of flight control. Two F-4 lateral series servos (P/N 32-69054), mounted in parallel, will be connected in series in each of four control systems. The four control systems are the pitch, roll, throttle and nozzle.

Two F-15 rudder actuators, mounted in parallel, will be connected in series with the rudder control system. Aircraft Hydraulic System 1 will be utilized to supply one valve of each dual series servo. The other will be supplied by Hydraulic System 2. The 32-69054 series servo actuators will be modified to decrease centering time when the system is deenergized. Plumbing will consist of added lines connecting each servo valve into its respective system including a check valve and a last chance filter in each servo valve return line.

The flow/leakage requirements for the added series servo actuators will require new hydraulic pumps of greater capacity. YAV-8B pumps will therefore replace the existing hydraulic pumps and the engine gearbox will be modified to be compatible with the higher torque required. Plumbing from the pumps to the engine/airframe interface will be revised.

(d) Electrical - Same as Section 2.1.5.

(e) Equipment Installation - The areas where the new avionic duplex fly-by-wire system is located are shown in Figure 3-10. The additional computer and interface electronics unit are mounted on the lower access door between Frames 36 and 38 in the aft fuselage. See Figure 3-11.

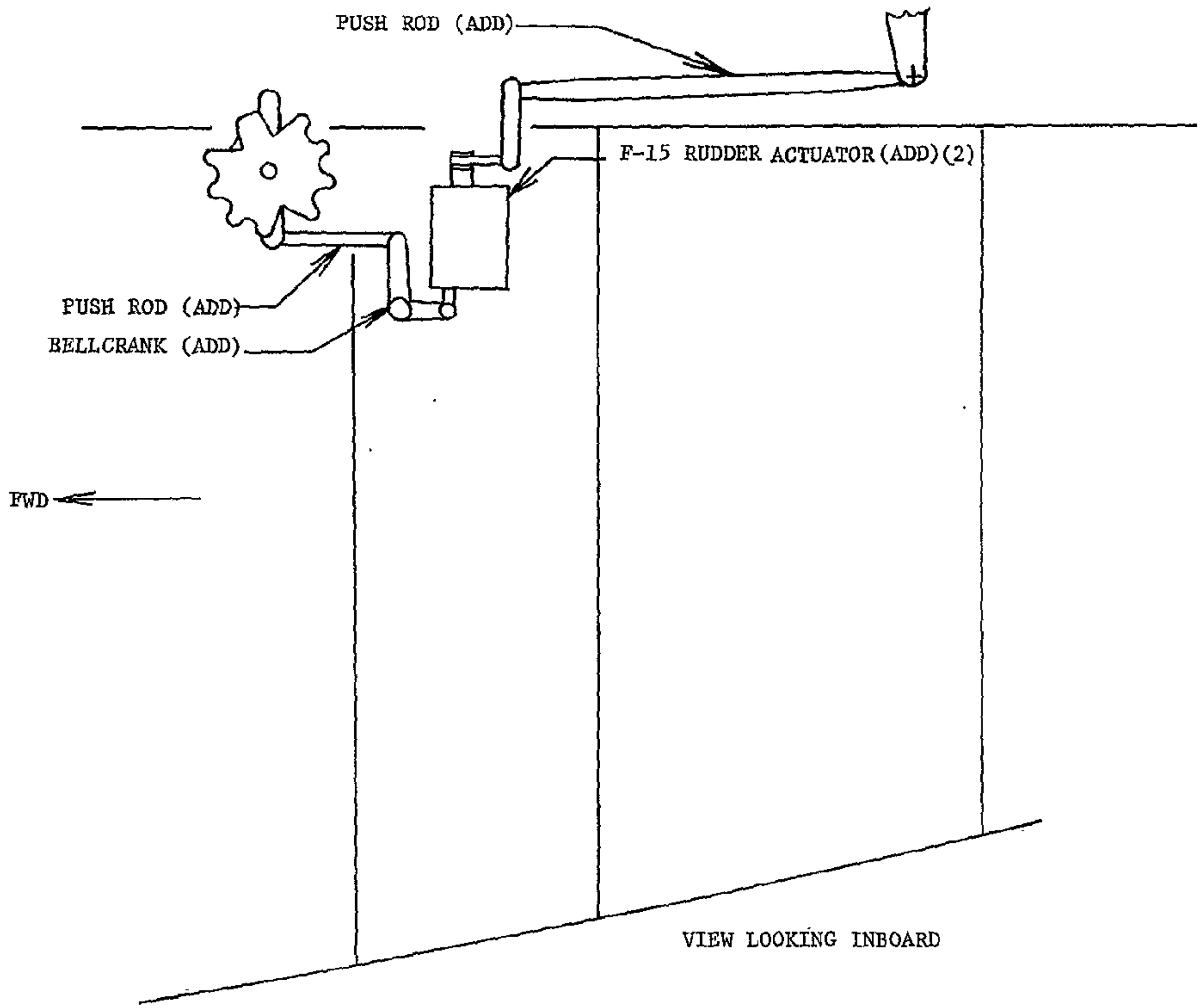

FIGURE 3-9 DUAL SERIES SERVO INSTALLATION
DIRECTIONAL SYSTEM

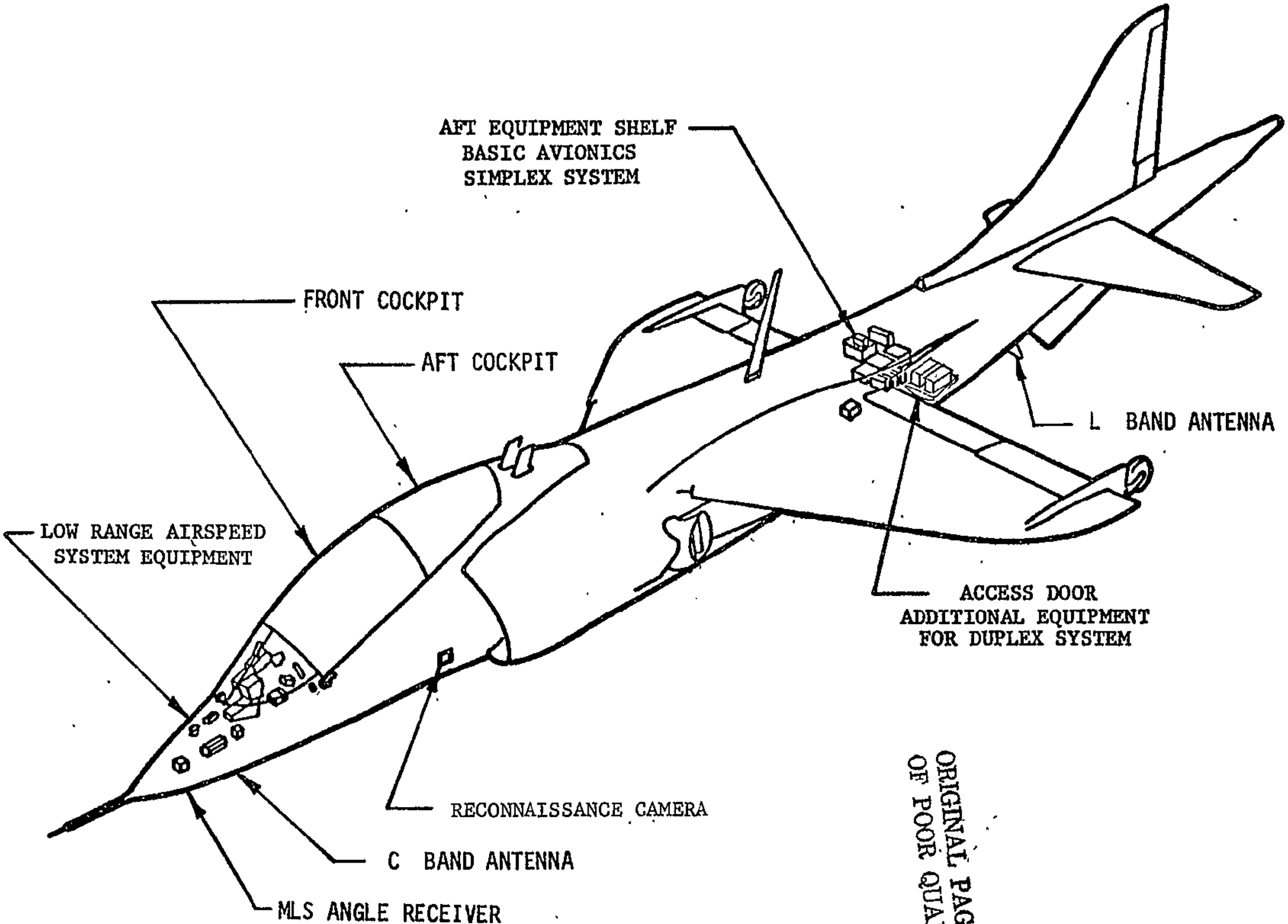

FIGURE 3-10 AVIONICS INSTALLATION
DUPLEX SYSTEM

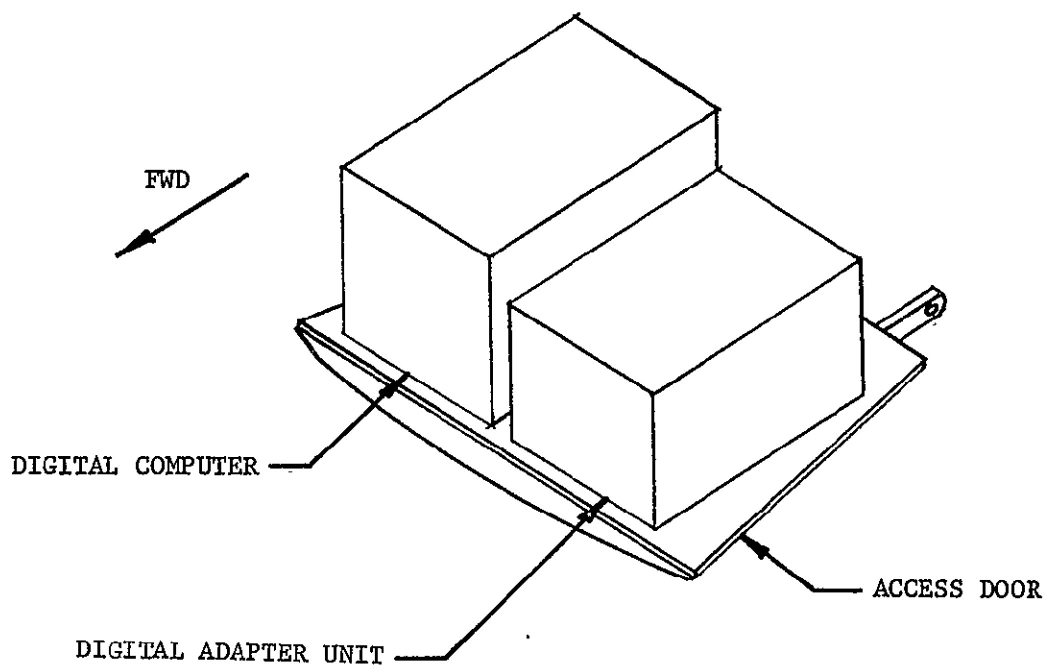

FIGURE 3-11 AVIONICS INSTALLATION
ADDITIONAL EQUIPMENT FOR DUPLEX INSTALLATION

3.1.2 DATA ACQUISITION SYSTEMS - Same as Section 2.2 except the definition of the measurands defined in the preliminary measurand list may be altered due to changes in the flight control system.

3.1.3 HEAD-UP DISPLAY - Same as Section 2.3.

3.1.4 LANDING GUIDANCE SYSTEM - Same as Section 2.4.

3.1.5 INERTIAL NAVIGATION SYSTEM - Same as Section 2.5.

3.1.6 SIMULATION - Same as Section 2.6.

3.1.7 SYSTEM SOFTWARE - The computer software requirements for the duplex configuration are very similar to those of the simplex configuration except for the replacement of the IFIM module with a Redundancy Management module. Computer memory estimates for the duplex configuration are shown in Figure 3-12. The memory estimates assume the use of HAL/S language in writing programs. The Redundancy Management module is less complex than for the smaller system and requires 2800 words less memory.

Computer memory estimates for each of the dual computers is shown in Figure 3-13. The flight control system part of the computer program is the same for each computer.

3.1.8 GROUND TESTS - Same as Section 2.8.

3.1.9 AIRWORTHINESS TESTS - Same as Section 2.9.

3.2 TRIPLEX FLIGHT CONTROL SYSTEM

The triplex flight control system design is based on the simplex system design discussed in Section 2.1.1. The computers and other critical components of the fly-by-wire system are implemented in a three channel fashion so that a component failure will not result in a potentially hazardous aircraft transient.

3.2.1 CONTROL SYSTEM MODIFICATIONS - A schematic diagram of the triplex longitudinal flight control system is given in Figure 3-14. This system is similar to the simplex longitudinal system except that three digital computers, three pitch rate gyros, triplex aft control stick force and position transducers, and a triplex parallel electromechanical servo are utilized.

A schematic diagram of the triplex lateral flight control system is presented in Figure 3-15. This system is similar to the simplex lateral system except that it has three digital computers, three roll gyros, triplex aft control stick force and rudder pedals, and triplex aft control stick force and position transducers, and a triplex parallel electromechanical servo are utilized.

A schematic diagram of the triplex directional flight control system is given in Figure 3-16. This system is similar to the simplex directional system except that it uses three digital computers, three yaw rate gyros, triplex aft control stick force and position transducers, and a triplex parallel electromechanical servo.

SOFTWARE MODULE	STORAGE (16 BIT WORDS)	
Control Laws		
- Solution of Control Law Equations & Algorithms	2000	
- Digital Filtering	300	
		2300
Redundancy Mgt		
- Signal Selection Algorithms	1200	
- Fault Recovery Routines	200	
- Synchronization Routine	100	
- In-line Monitoring of Rate Gyros, Accelerometers	100	
- In-line Monitoring of Secondary Actuators	300	
- In-line Monitoring of Single Sensors	400	
- Computer Self-Test	1000	
		3300
BIT		
- Power Supply Tests	100	
- Sensor Tests	600	
- Actuator Tests	400	
- Digital Interface Tests	700	
- Switch Tests	200	
- BIT Subexecutive & Display Routine	500	
		2500
Executive		
- Program Initialization	500	
- Interrupt Processing	700	
- Program Scheduling	200	
- Input/Output Processing	500	
- Mode Switching Logic	700	
		2400
Display		
- MLS/Flight Director		
TOTAL		13,100

FIGURE 3-12 COMPUTER MEMORY REQUIREMENTS - DUPLEX

	16 BIT MEMORY WORDS	
MODULE	COMPUTER 1	COMPUTER 2
CONTROL LAWS	2300	2300
REDUNDANCY MGT	3300	3300
EXECUTIVE	2500	2500
BIT	2600	2600
	10,700	10,700
MLS/FLIGHT DIRECTOR	2400	---
	13,100	10,700

FIGURE 3-13 OPERATIONAL FLIGHT PROGRAM MEMORY REQUIREMENTS

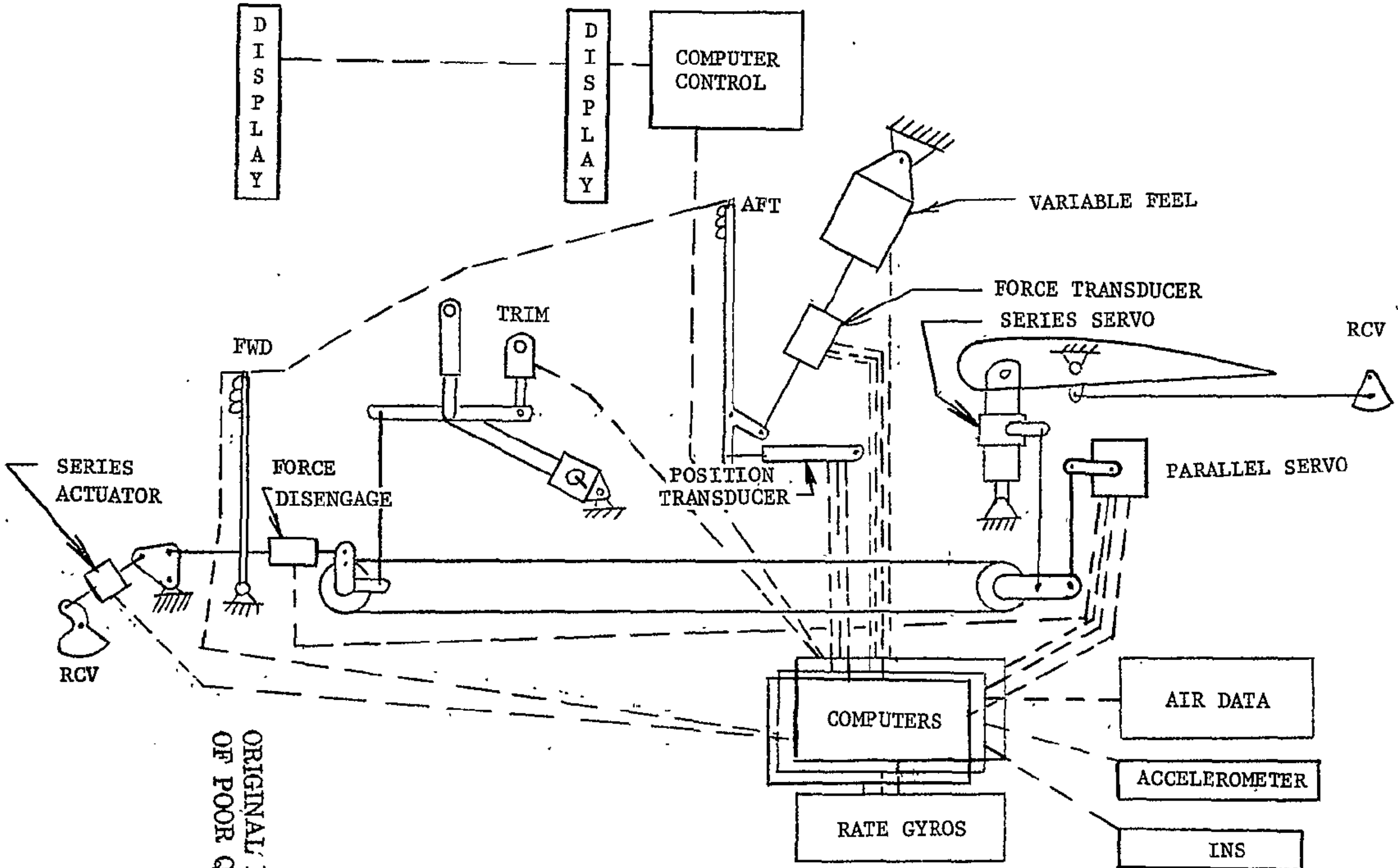

FIGURE 3-14 LONGITUDINAL CONTROL SYSTEM
TRIPLEX PARALLEL

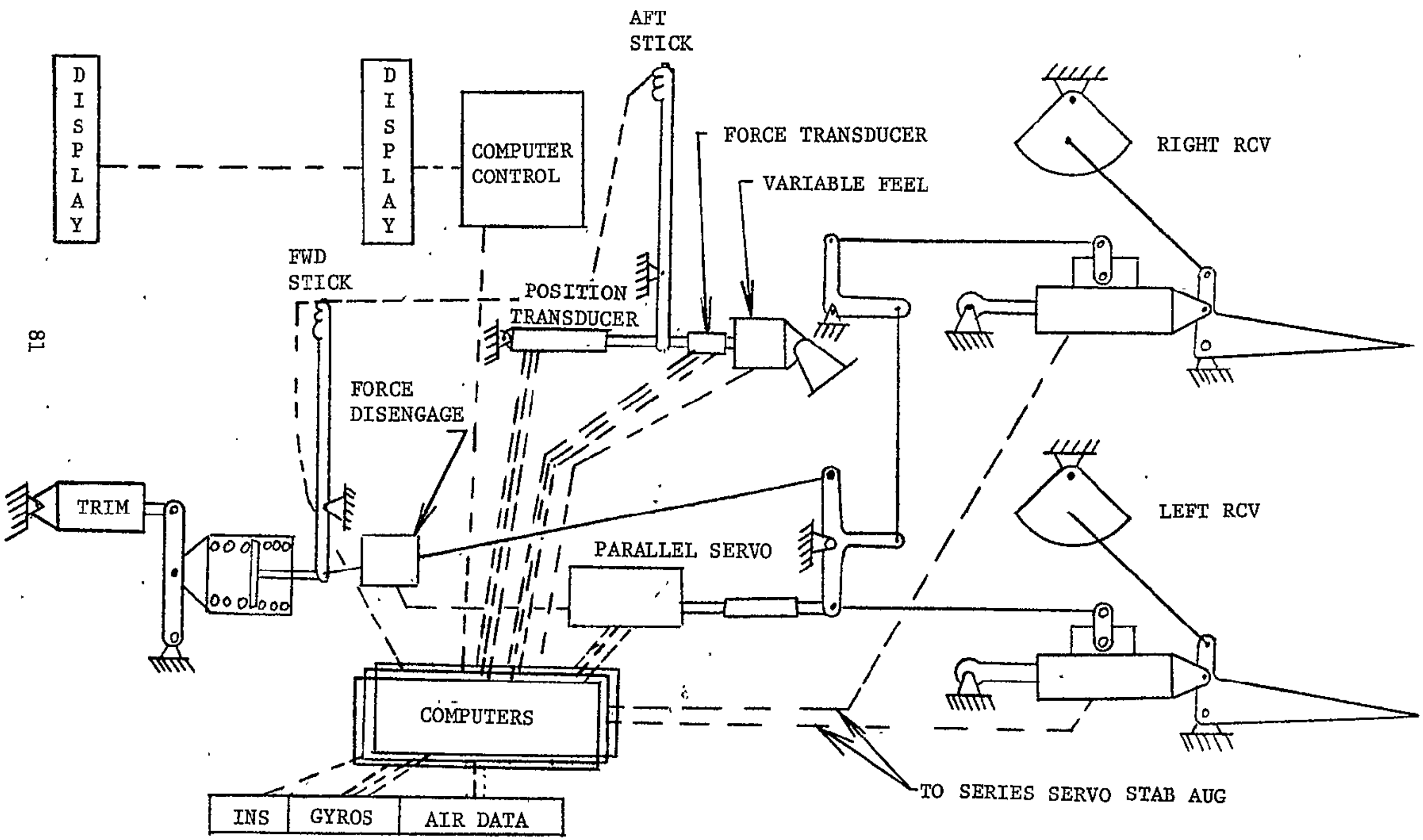

FIGURE 3-15 LATERAL CONTROL SYSTEM , TRIPLEX PARALLEL

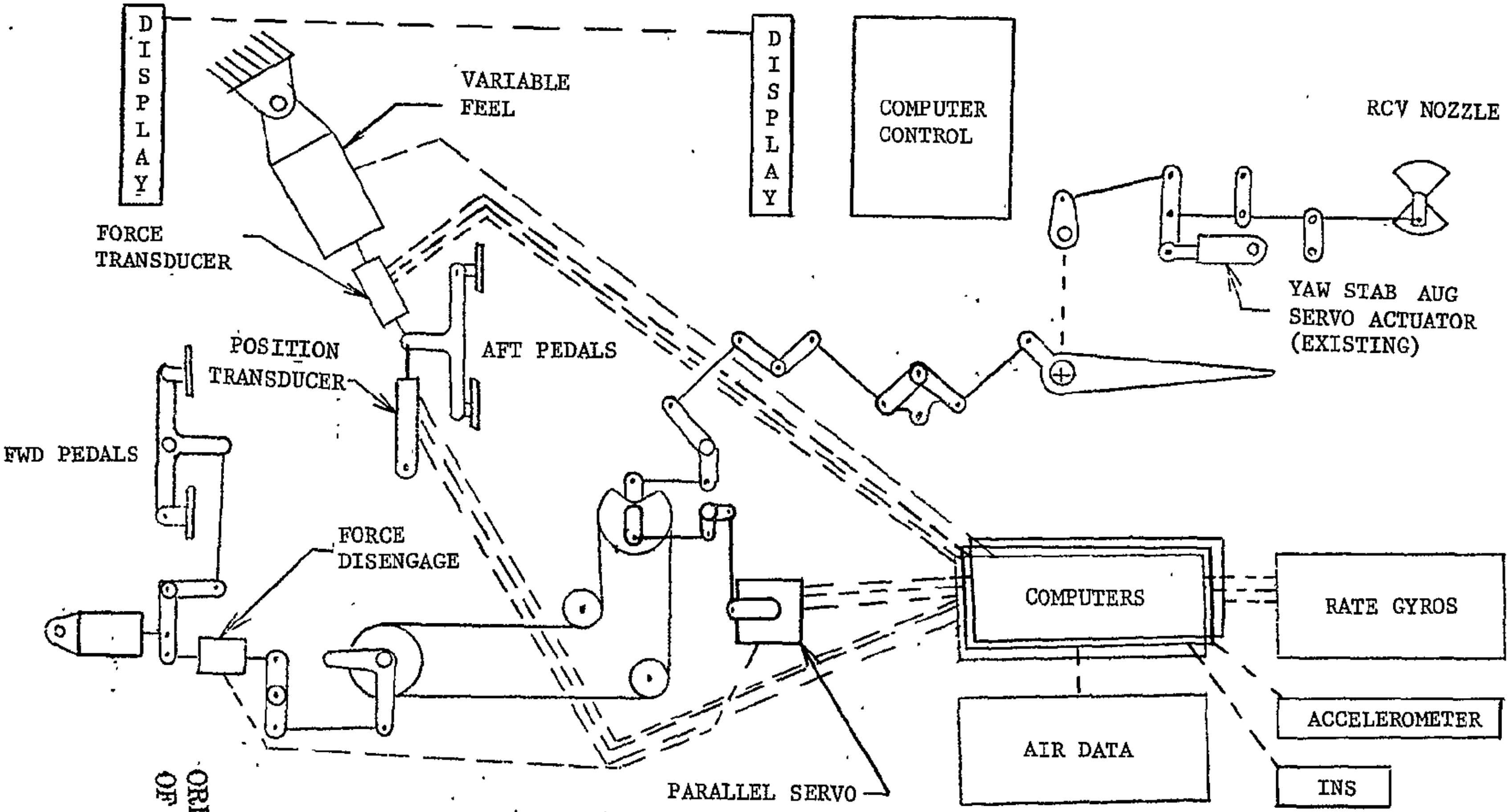

FIGURE 3-16 DIRECTIONAL CONTROL SYSTEM
TRIPLEX PARALLEL

The triplex parallel electromechanical servo actuators are constructed using three two-phase servo motors. A gear train is used to combine motor outputs as well as reduce speed and increase torque. RVDT's are used for position feedback and tachometers are used to provide rate feedback for stabilization. Brakes are not used so that when the servo actuator is electrically disengaged it rotates freely. When the pilot overpowers the servo actuator, the force generated by the pilot is proportional to the stall force of the servo motors.

The use of cross-element monitoring is planned for the triplex servo actuators by making a direct comparison to determine servo motor status. In this approach, the tachometer feedback in each servo motor is compared to the tachometer feedback in the remaining two servo motors. The logic circuitry defines a failure when all of the comparator limits associated with a particular servo motor are exceeded for a time duration determined by comparator delay time.

The installation of the parallel servos, position transducer, force transducer, and variable fuel system mechanization are the same as for the simplex flight control system. If one of the three channels of control fails, that channel is shut down and the system continues to operate using the other two channels.

(a) Cockpit Information and Disengage Requirements - The cockpit installation for a triplex fly-by-wire system is nearly identical to the simplex and the duplex installations. The pilots make the same inputs to the system in all three cases and will only have a small change to the warning lights in order to isolate a failure in a computer failure situation. See Section 2.1.2 for the description of the pilot functions.

(b) Computer Hardware - The three onboard computers are the same as the computer used in the simplex configuration and described in Section 2.1.3.

(c) Hydraulics - The triplex flight control system requires no modifications to the existing hydraulic system due to using electromechanical actuators for the fly-by-wire mode.

(d) Equipment Installation - The avionics installation for a triplex fly-by-wire system is shown in Figure 3-17. The additional computer and interface electronics unit needed are mounted in a gun pod shape under the center fuselage of the airplane. See Figure 3-18.

3.2.2 DATA ACQUISITION SYSTEM - Same as Section 2.2.

3.2.3 HEAD-UP DISPLAY - Same as Section 2.3.

3.2.4 LANDING GUIDANCE SYSTEM - Same as Section 2.4.

3.2.5 INERTIAL NAVIGATION SYSTEM - Same as Section 2.5.

3.2.6 SIMULATION - Same as Section 2.6.

3.2.7 SYSTEM SOFTWARE - The computer program is very similar to that for the duplex configuration. The memory estimates are shown in Figure 3-19. Operational flight program requirements for the three computers are shown in Figure 3-20. The flight control program for each of the computers is identical.

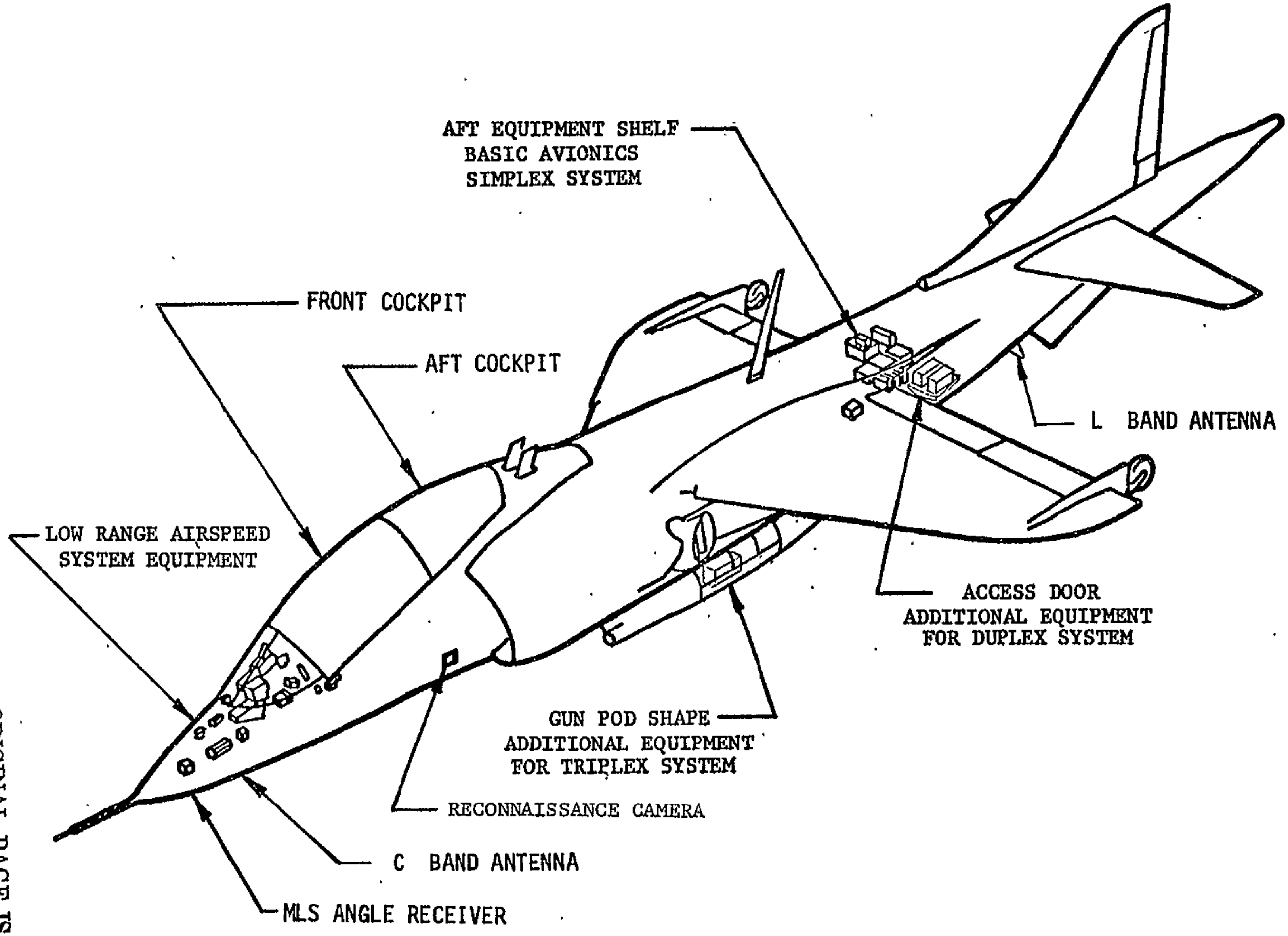

FIGURE 3-17 AVIONICS INSTALLATION
TRIPLEX SYSTEM

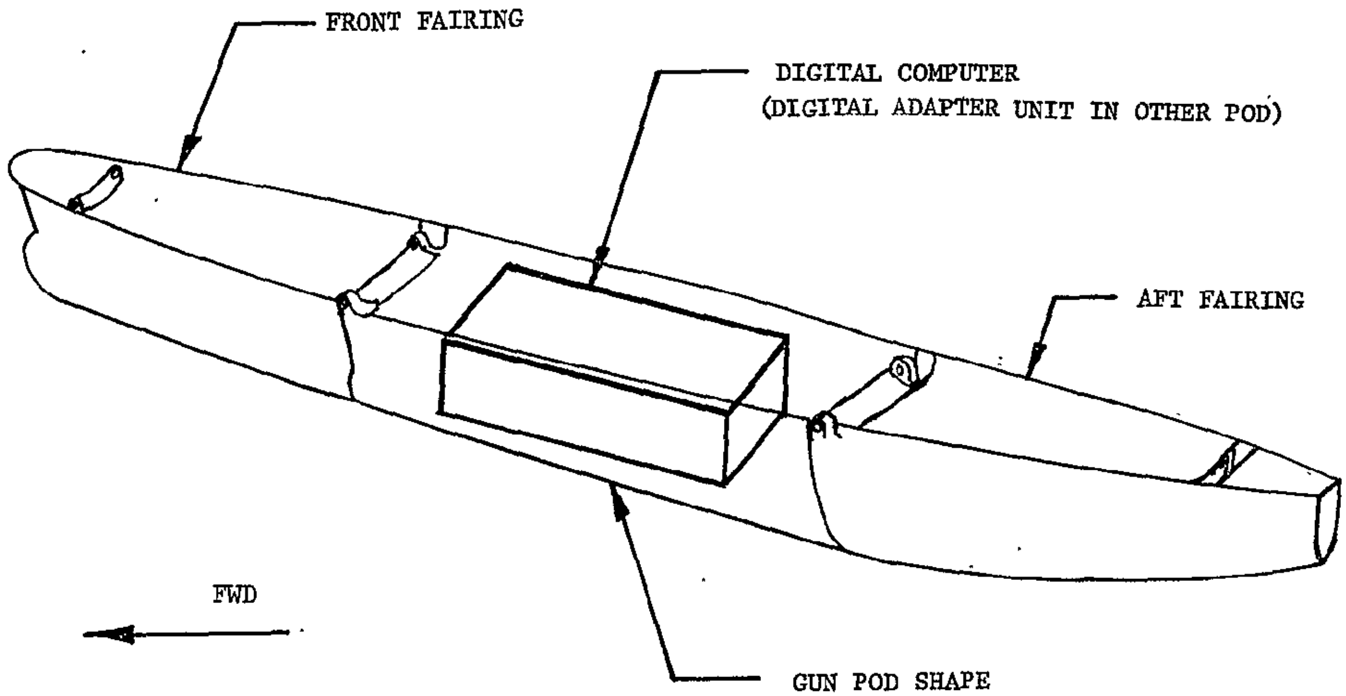

FIGURE 3-18 AVIONICS INSTALLATION,
ADDITIONAL EQUIPMENT FOR TRIPLEX SYSTEM

SOFTWARE MODULE	STORAGE (16 BIT WORDS)	
Control Laws		
- Solution of Control Law Equations & Algorithms	2000	
- Digital Filtering	300	
		2300
Redundancy Mgt		
- Signal Selection Algorithms	1700	
- Fault Recovery Routines	300	
- Synchronization Routine	100	
- In-line Monitoring of Rate Gyros & Accelerometers	100	
- In-line Monitoring of Secondary Actuators	300	
- In-line Monitoring of Single and Duplex Sensors	400	
- Computer Self-Test	1000	
		3900
BIT		
- Power Supply Tests	100	
- Sensor Tests	600	
- Actuator Tests	400	
- Digital Interface Tests	700	
- Switch Tests	200	
- BIT Subexecutive & Display Routine	500	
		2500
Executive		
- Program Initialization	500	
- Interrupt Processing	700	
- Program Scheduling	200	
- Input/Output Processing	500	
- Mode Switching Logic	700	
		2600
Display		
- MLS/Flight Director		2400
TOTAL		13,700

FIGURE 3-19 COMPUTER MEMORY REQUIREMENTS, TRIPLEX

MODULE	COMPUTER 1	COMPUTER 2	COMPUTER 3
CONTROL LAWS	2300	2300	2300
REDUNDANCY MGT	3900	3900	3900
EXECUTIVE	2500	2500	2500
BIT	2600	2600	2600
	11,300	11,300	11,300
MLS/FLIGHT DIRECTOR	2400	----	----
TOTAL	13,700	11,300	11,300

FIGURE 3-20 OPERATIONAL FLIGHT PROGRAM MEMORY REQUIREMENTS

TRIPLEX SYSTEM

3.2.8 GROUND TESTS - Same as Section 2.8.

3.2.9 AIRWORTHINESS TEST - Same as Section 2.9.

3.3 SIMPLEX THROTTLE AND NOZZLE CONTROL SYSTEMS

The simplex throttle and nozzle control system can be used in conjunction with any of the proposed flight control systems since only one digital computer is required. The parallel servo mechanization concept used for this option is similar to that used for the simplex parallel flight control system.

3.3.1 SIMPLEX THROTTLE CONTROL SYSTEM - The schematic diagram of the simplex throttle control system is given in Figure 3-21. The aft throttle lever is mechanically disconnected from the production Harrier throttle system linkages and equipped with a RVDT position transducer. A parallel electromechanical servo is connected to the outboard end of the fuel control torque tube. A force disengage switch (strain gage) is attached to the throttle mechanism.

When the fly-by-wire throttle system is engaged, the position transducer on the aft throttle provides throttle lever position information to the computer. The computer uses this information and, when necessary, other information such as nozzle position, gyro and accelerometer signals, air data, and inertial navigation system variables, to compute a command signal for the parallel servo. The parallel servo moves the entire throttle linkage including the forward cockpit throttle lever and the fuel control torque tube. The safety pilot can monitor the operation of the system by monitoring the motion of the forward cockpit throttle lever. If he detects a malfunction, he can disengage the entire fly-by-wire flight control system, including the throttle system, by use of the disengage button on his control stick, the disengage button on his throttle or the master switch on his control panel. He can also disengage the fly-by-wire throttle system alone by applying a force on the throttle lever of sufficient magnitude to activate the force disengage switch. If this switch fails, he can manually override the parallel servo.

The throttle system installation is shown in Figure 3-22. Detail A shows the electromechanical parallel servo installation. It can be seen that when the fly-by-wire system is disengaged the safety pilot has essentially the production Harrier throttle system.

The aft cockpit throttle is shown in Figure 3-23. The throttle lever is disconnected from the linkage connecting the forward cockpit throttle to the fuel control unit. An RVDT and linkage are added to provide the position measurements. A disengage switch is added so that the evaluating pilot can quickly disengage the system.

3.3.2 SIMPLEX NOZZLE CONTROL SYSTEM - The schematic diagram of the simplex parallel nozzle control system is given in Figure 3-24. The aft nozzle lever is mechanically disconnected from the production Harrier nozzle system linkages and equipped with a position transducer. A parallel electromechanical servo is connected to the Air Motor Servo Unit (AMSU). A force disengage switch, constructed using a strain gage, is attached to the nozzle mechanism.

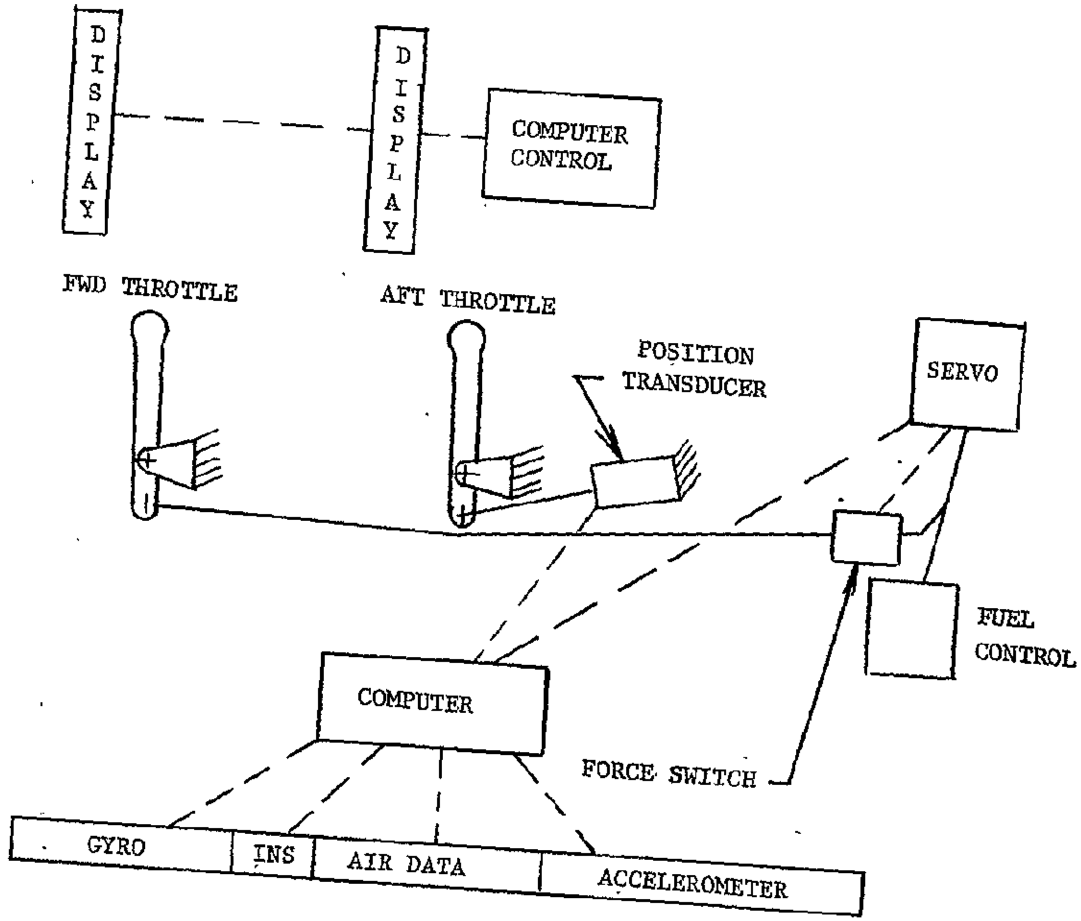

FIGURE 3-21 THROTTLE SYSTEM , SIMPLEX PARALLEL

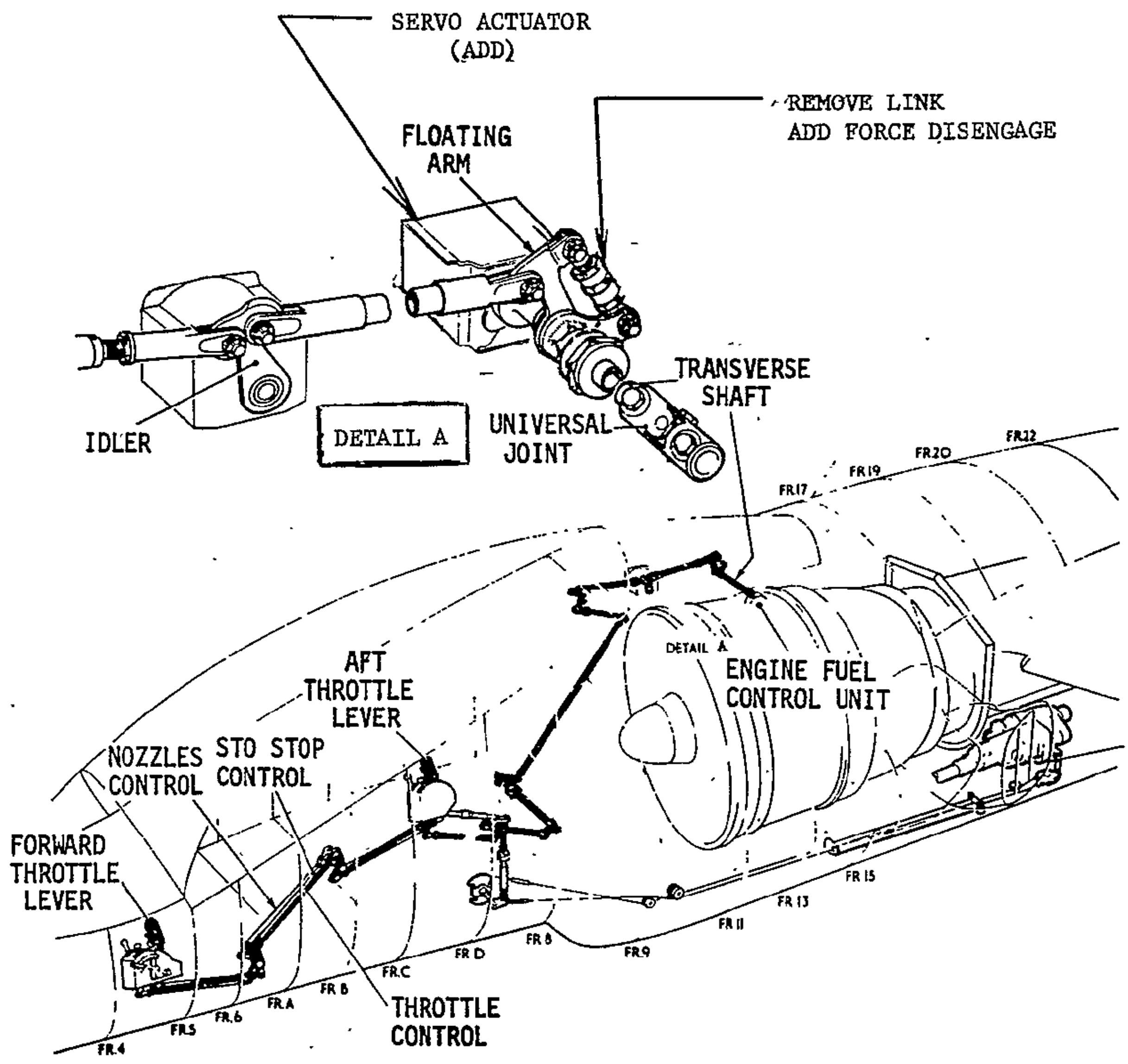

FIGURE 3-22 THROTTLE SYSTEM , PARALLEL SIMPLEX SERVO INSTALLATION

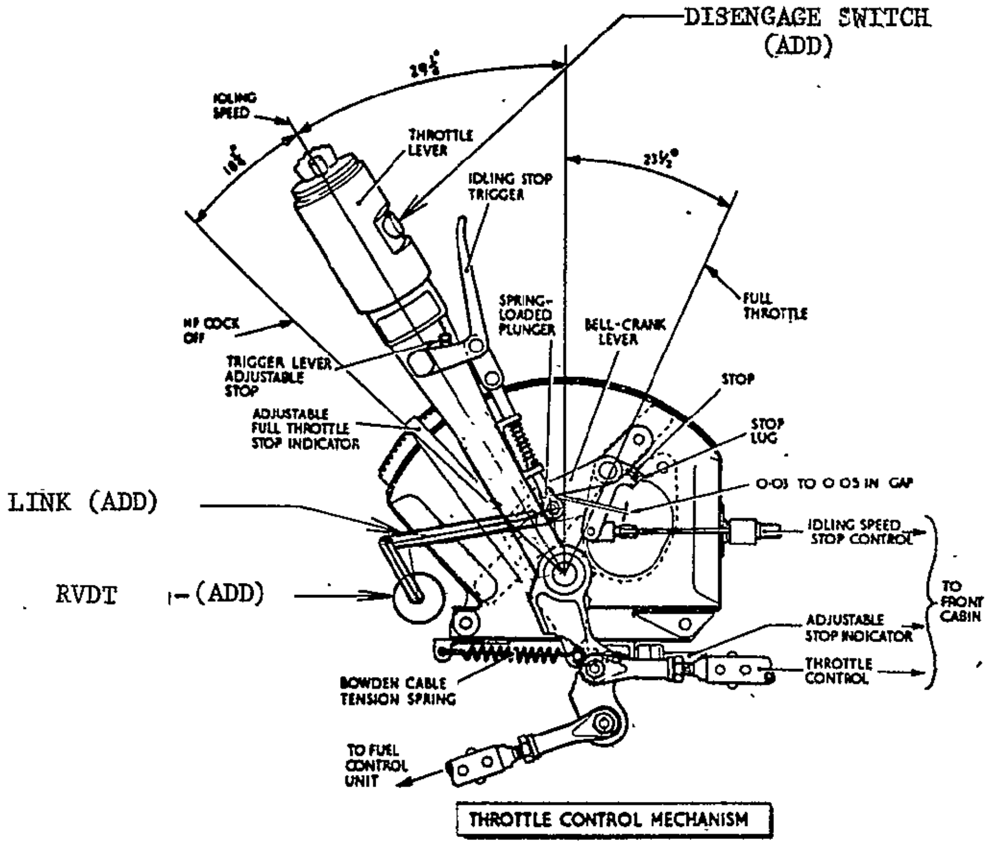

FIGURE 3-23 SIMPLEX·THROTTLE , CONTROL

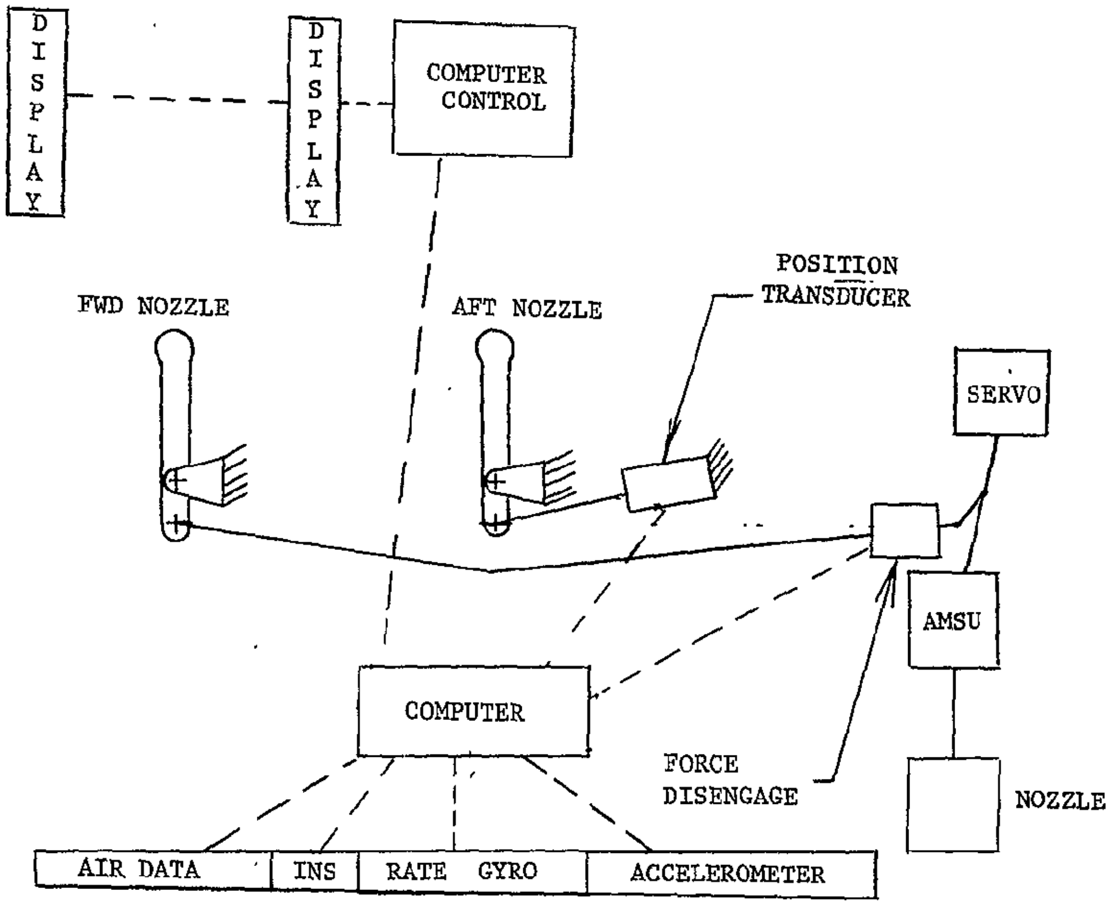

FIGURE 3-24 NOZZLE SYSTEM , SIMPLEX PARALLEL

When the fly-by-wire nozzle system is engaged, the position transducers on the aft nozzle provide nozzle lever position information to the computer. The computer uses the nozzle lever position, and possibly other information such as throttle position, gyro and accelerometer signals, air data, and inertial navigation system variables, to compute a command signal for the parallel servo. The parallel servo drives the mechanical input to the ASMU so that the nozzles are driven to the proper position. The parallel servo moves the entire nozzle linkage including the forward cockpit nozzle lever, so the safety pilot can monitor the operation of the fly-by-wire nozzle system by monitoring the motion of the forward cockpit nozzle lever. If he detects a malfunction, he can disengage the entire fly-by-wire flight control system, including the nozzle system, by means of the disengage button on his control stick, the disengage button on his throttle, or the master switch on his control panel. He can also disengage the fly-by-wire nozzle system by applying enough force on the nozzle control to activate the force disengage switch. If this switch fails, he can manually override the parallel servo.

The nozzle system installation is shown in Figure 3-25. Detail A shows the strain gage force link installation. Detail B shows the electromechanical parallel servo installation. It can be seen that when the fly-by-wire system is disengaged, the safety pilot has essentially the production Harrier throttle system.

The aft cockpit nozzle lever is shown in Figure 3-26. The nozzle lever is disconnected from the linkage connecting the forward cockpit throttle to the fuel control unit. An RVDT and linkage are added to provide the position measurements. A disengage switch is added so that the evaluation pilot can quickly disengage the system.

3.4 DUPLEX THROTTLE AND NOZZLE CONTROL SYSTEMS

The duplex throttle and nozzle control systems can be used in conjunction with either the duplex or triplex flight control system since two digital computers are required. The series servo mechanization concept used for this option is similar to that used for the duplex flight control system.

3.4.1 DUPLEX THROTTLE CONTROL - The schematic diagram of the duplex throttle control system is presented in Figure 3-27. In the duplex configuration, the aft cockpit throttle lever remains mechanically connected to the production Harrier throttle mechanism. It is equipped with a position transducer which is an RVDT. A dual electrohydraulic series servo is added in the throttle linkage.

The installation of the throttle system is shown in Figure 3-28. The dual series servo is an electrohydraulic servo fabricated using two F-4 lateral series servos. The orifices in the hydraulic recentering circuits of these actuators have been enlarged so that the servo recenters almost instantaneously when the servo is disengaged. A walking beam bellcrank connects the servo to the linkage from the throttle levers so that the servo output and the position of the throttle lever are added to obtain the mechanical motion which is applied to the fuel control shaft.

Figure 3-29 shows the throttle controller for the dual mechanization. The throttle lever remains mechanically connected to the production Harrier throttle mechanism. An RVDT and link are connected to the throttle lever to provide a position transducer. A disengage button is added to the throttle grip.

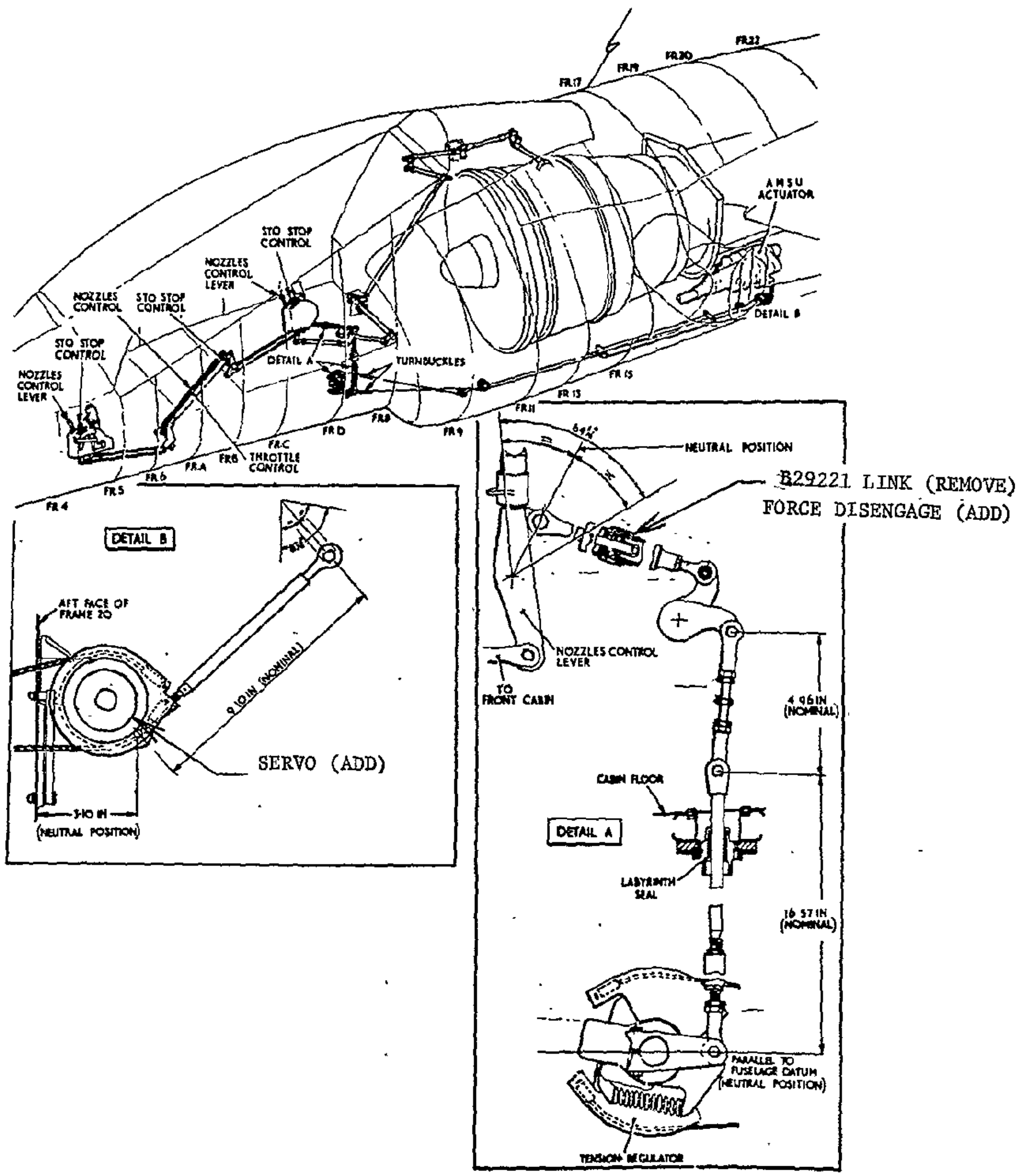

FIGURE 3-25 NOZZLE SYSTEM, SIMPLEX PARALLEL SERVO INSTALLATION

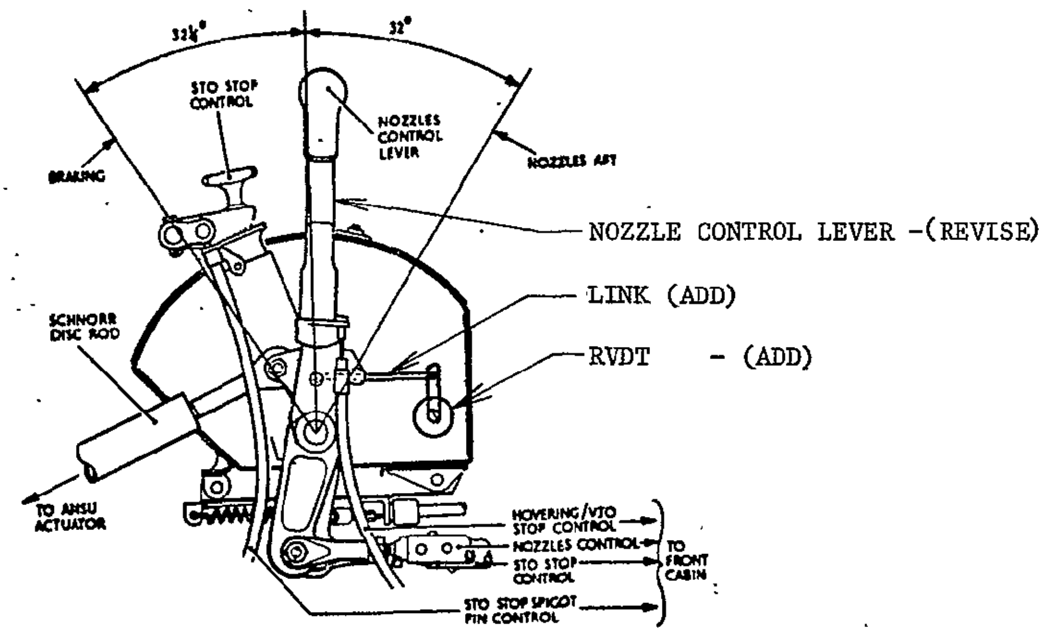

FIGURE 3-26 SIMPLEX NOZZLE CONTROL

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OF POOR QUALITY

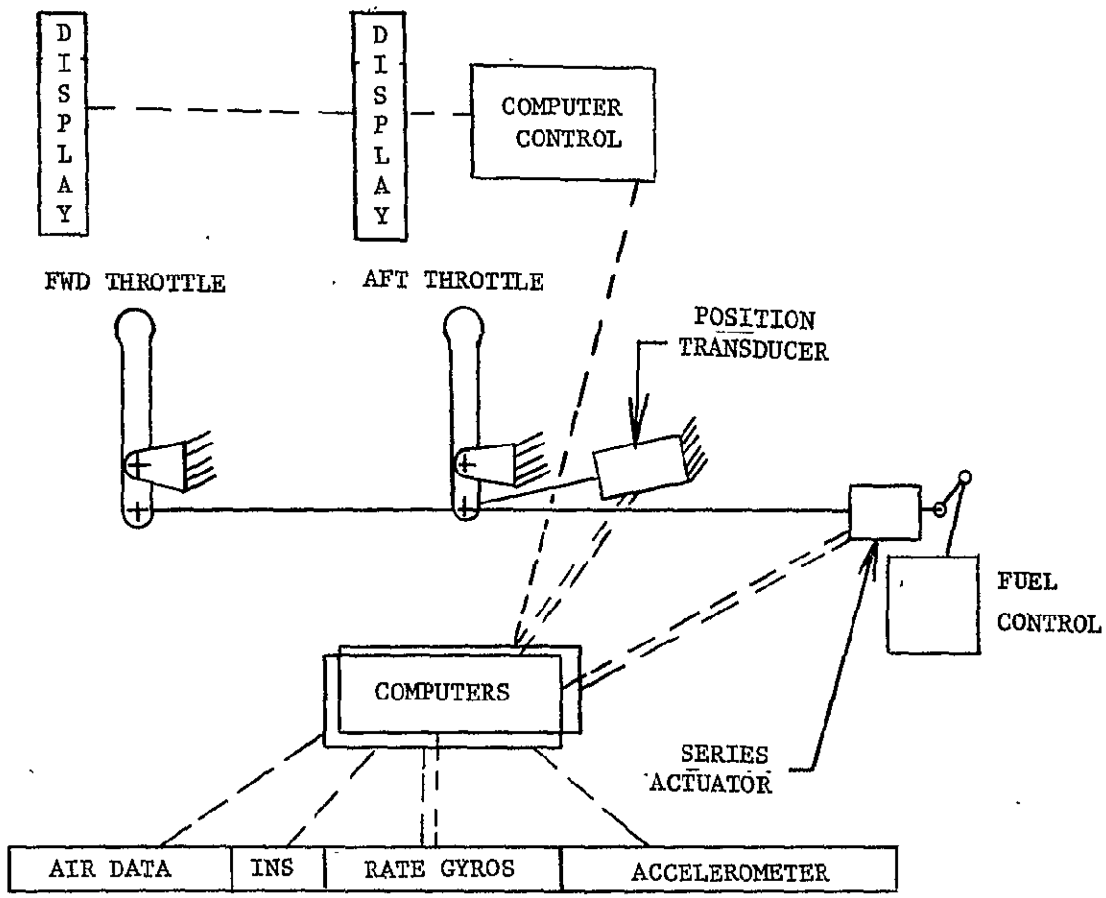


FIGURE 3-27 THROTTLE SYSTEM, DUAL SERIES

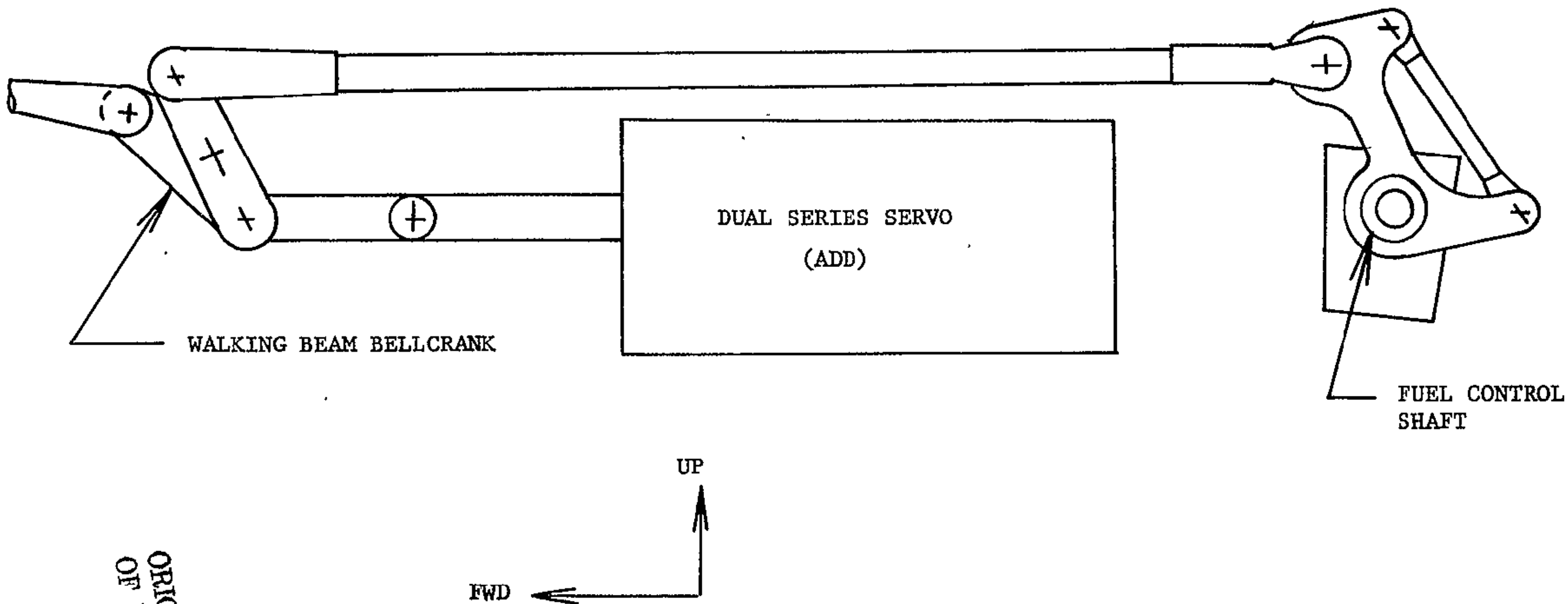


FIGURE 3-28 THROTTLE SYSTEM , DUAL SERIES SERVO INSTALLATION

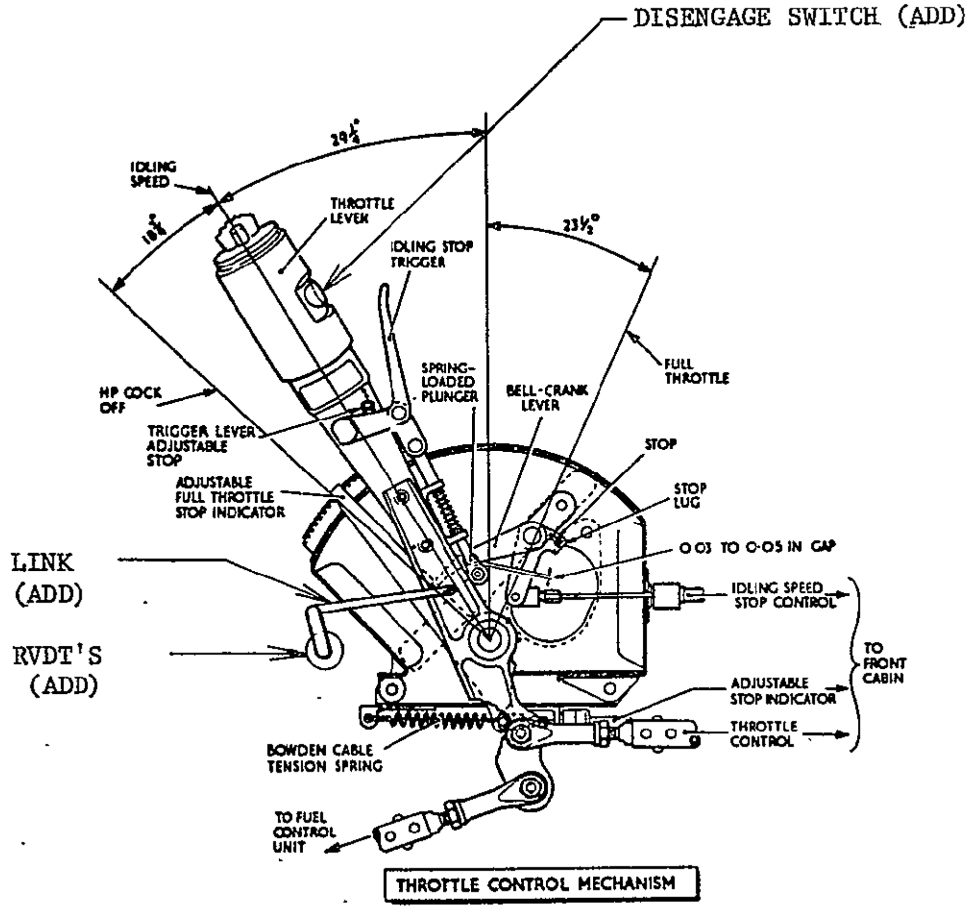


FIGURE 3-29 THROTTLE CONTROL , DUAL

In the dual mechanization, the forward cockpit and aft cockpit throttle levers move together so the safety pilot monitors the aft cockpit throttle lever position rather than the input to the engine fuel control unit as he does in the simplex mechanization. The disengage features of the simplex and duplex mechanizations are the same except that the duplex system does not have a force disengage switch since there is no convenient way to measure the force applied by the safety pilot.

3.4.2 DUPLEX NOZZLE CONTROL - The schematic diagram of the duplex nozzle control system is given in Figure 3-30. In the duplex configuration the aft cockpit nozzle level remains mechanically connected to the production Harrier nozzle mechanism. It is equipped with a position transducer which is an RVDT. A dual electrohydraulic series servo is added in the nozzle linkage.

The installation of the nozzle system is shown in Figure 3-31. The dual series servo is an electrohydraulic servo fabricated using two F-4 lateral series servos. The orifices in the hydraulic recentering circuits of these actuators have been enlarged so that the servo recenters almost instantaneously when the servo is disengaged. A walking beam bellcrank is used to connect the servo to the linkage from the nozzle levers so that the servo output and the position of the throttle lever are additive with respect to the mechanical motion applied to the ASMU.

Figure 3-32 shows the nozzle controller for the dual mechanizations. The nozzle lever remains mechanically connected to the production Harrier nozzle mechanisms. An RVDT and link are connected to the nozzle lever to provide a position transducer. A disengage button is added to the nozzle grip.

In the dual mechanization, the forward cockpit and aft cockpit nozzle levers move together. The safety pilot monitors the aft cockpit nozzle lever position rather than monitoring the input to the ASMU as he does in the simplex mechanization. The disengage features of the simplex and duplex mechanizations are the same except that the duplex system does not have a force disengage switch since there is no convenient way to measure the force applied by the safety pilot.

3.5 TRIPLEX THROTTLE AND NOZZLE CONTROL SYSTEMS - The triplex throttle and nozzle control systems can only be used in conjunction with the triplex flight control system since three digital computers are required. The parallel servo mechanization concept used for this option is similar to that used for the triplex flight control system.

3.5.1 TRIPLEX THROTTLE SYSTEM - The schematic diagram of the triplex throttle control system is given in Figure 3-33. It is essentially the same as the simplex throttle system described in Section 3.3.1 except that three RVDTs are attached to the throttle lever to obtain a triplex position transducer, a three channel electromechanical actuator is used instead of a single channel electromechanical actuator, and a triplex computer configuration is used to generate the servo command signal.

The installation and operational aspects of the simplex and triplex throttle systems are similar. The main difference is that in the triplex configuration, a component failure will not produce an appreciable throttle transient.

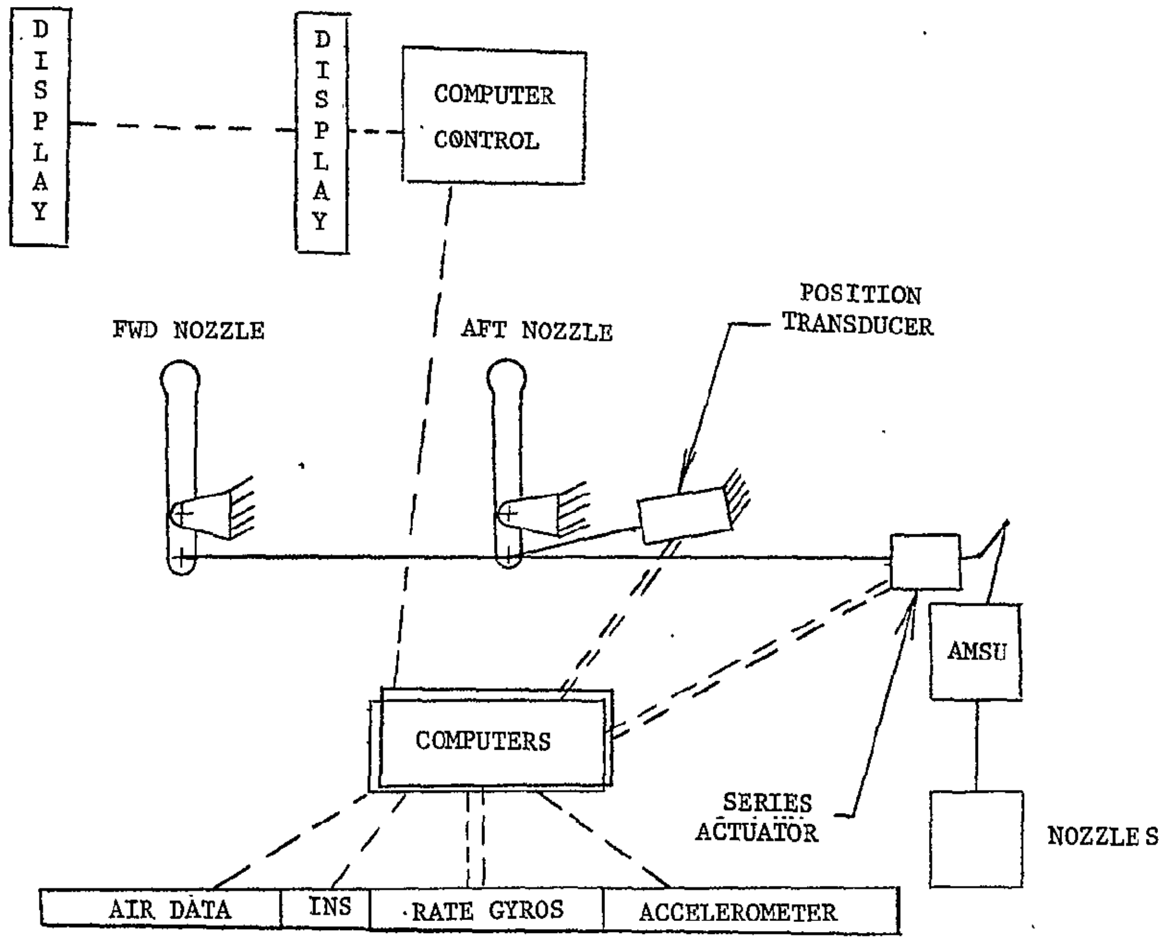


FIGURE 3-30 NOZZLE SYSTEM , DUAL SERIES

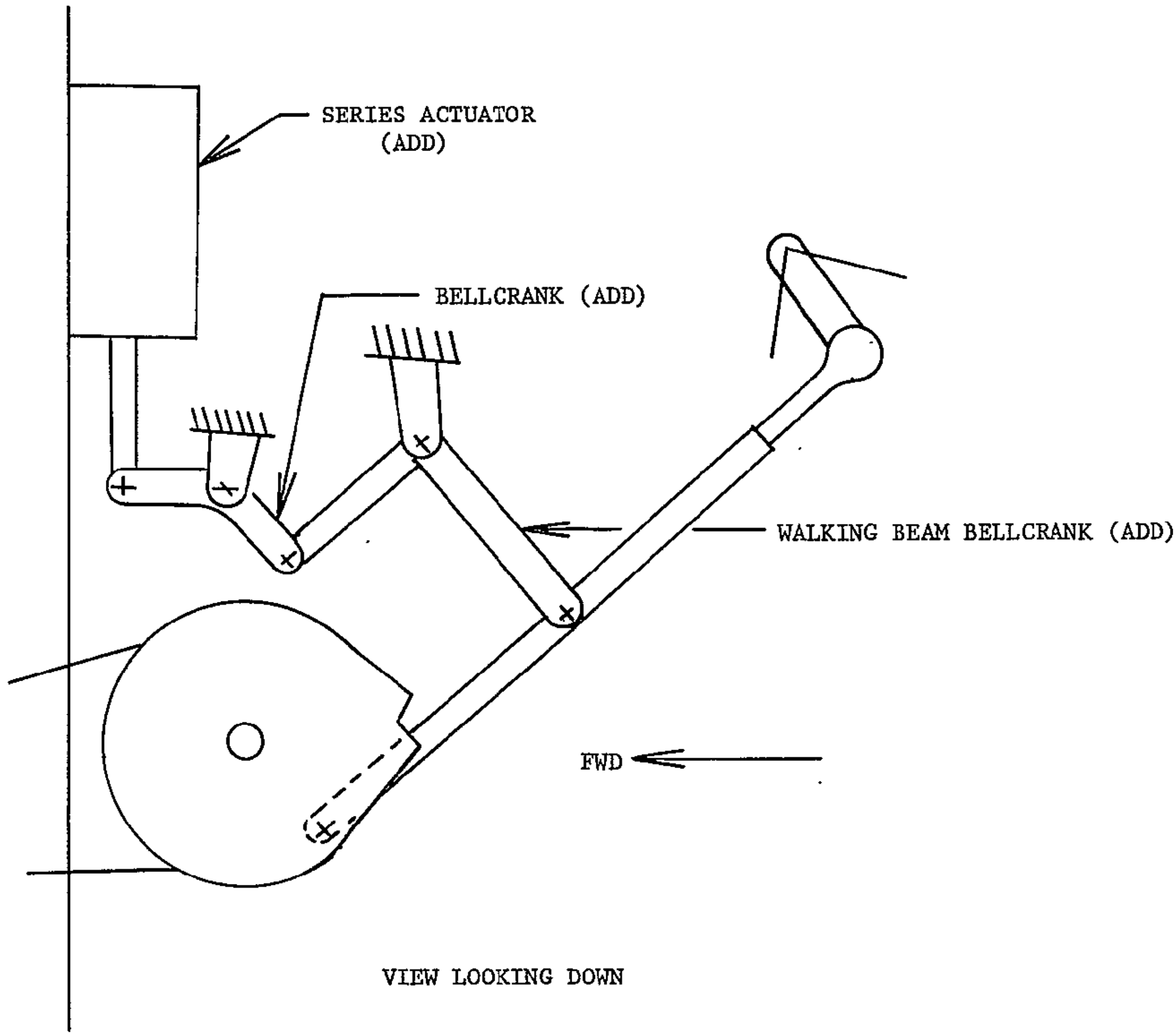


FIGURE 3-31 DUAL SERIES NOZZLE SYSTEM INSTALLATION

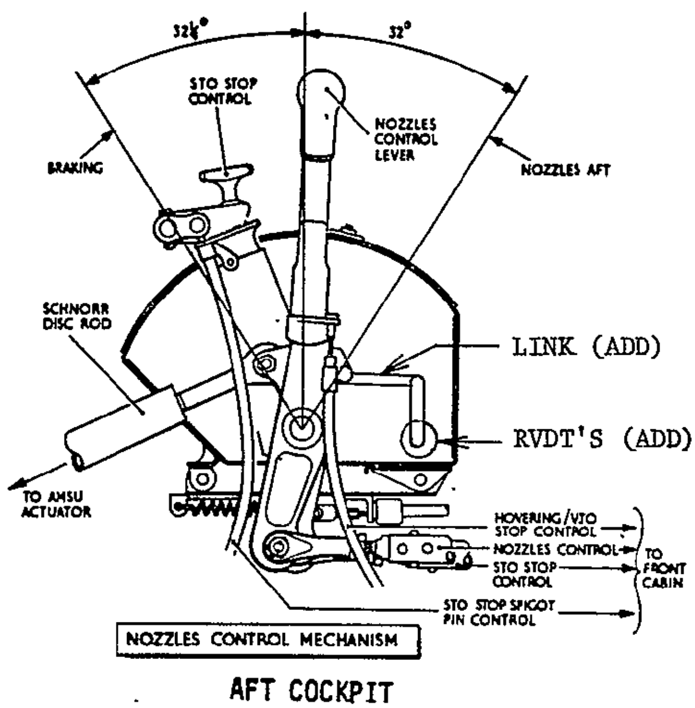


FIGURE 3-32 NOZZLE CONTROL - DUAL

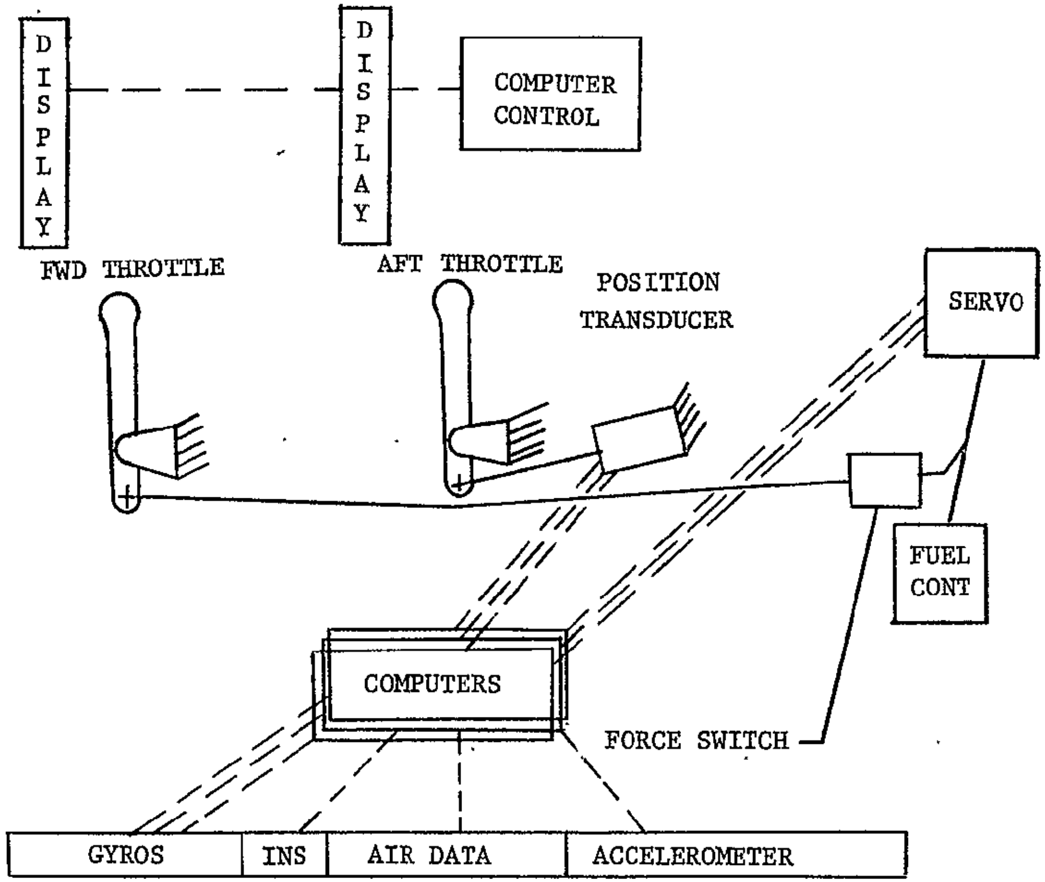


FIGURE 3-33 THROTTLE SYSTEM , TRIPLEX PARALLEL

3.5.2 TRIPLEX NOZZLE SYSTEM - The schematic diagram of the triplex nozzle control system is given in Figure 3-34. This system is essentially the same as the simplex nozzle system described in Section 3.3.2 except that three RVDTs are attached to the nozzle lever to obtain a triplex position transducer, a three channel electromechanical parallel servo is used instead of a single channel electromechanical parallel servo, and a triplex computer configuration is used to generate the servo command signal.

The installation and operational aspects of the simplex and triplex nozzle systems are similar. The main difference is that in the triplex configuration a component failure will not produce an appreciable nozzle transient.

3.6 LOW RANGE AIR DATA SYSTEM - The air data sensor which has been selected for the low range air data system option is the Applied Devices sensor which has also been selected for the YAV-8B. As shown in Figure 3-35, two sensors will be mounted on the aircraft. One will be mounted in the same location as it will be mounted on the YAV-8B so that the design, testing and calibration performed on the YAV-8B program will be directly applicable to this program. It will provide low speed air data measurements in the asymmetrical (X-Y) plane. The other will be mounted on the left side of the nose of the aircraft in a position which will be a mirror image of the location of the production angle of attack probe. This sensor will provide low speed air data measurements in the symmetrical (X-Z) plane. Installation design and calibration must be performed for this sensor since the YAV-8B will not have a similar sensor installation.

The air data probe is a small flat plate which rotates at right angles to the sensing plane. It is coupled to a piezoelectric generator which serves as both a restoring spring and signal generator. Its operation is such that when a steady state air flow impinges upon the rotating flat plate an oscillating torque is exerted on the crystal restraint. This gives a measure of total vector direction and amplitude of the air mass flow with respect to the sensor mount. The proposed two sensor system would therefore measure the magnitude and direction of the air flow in both the symmetrical and asymmetrical aircraft planes.

In order to determine the direction of the input air, it is necessary to have a reference signal to which the output signals can be compared to effect a coordinate conversion from the rotating frame to the fixed frame. This is accomplished by mounting a reference generator configured with two 90° phase shifted outputs on the spin axis with their null points aligned accurately to the alignment pin on the sensor outer case. Using these reference signals in conjunction with the probe output, it is then possible to determine the direction of the components in terms of body axis coordinates.

The sensor and its associated electronic unit are relatively small and light. Each sensor is 3 inches long, 1.25 inches in diameter and weighs 0.5 pounds. The electronic unit for each sensor is 3.5 inches by 2.5 inches and weights 1.2 pounds.

The two sensor units will be interfaced to the flight control computer and head up display through the digital adapter unit. The digital computer will be used to process the signals from the sensor electronic units in order to obtain the desired data format and improve data accuracy.

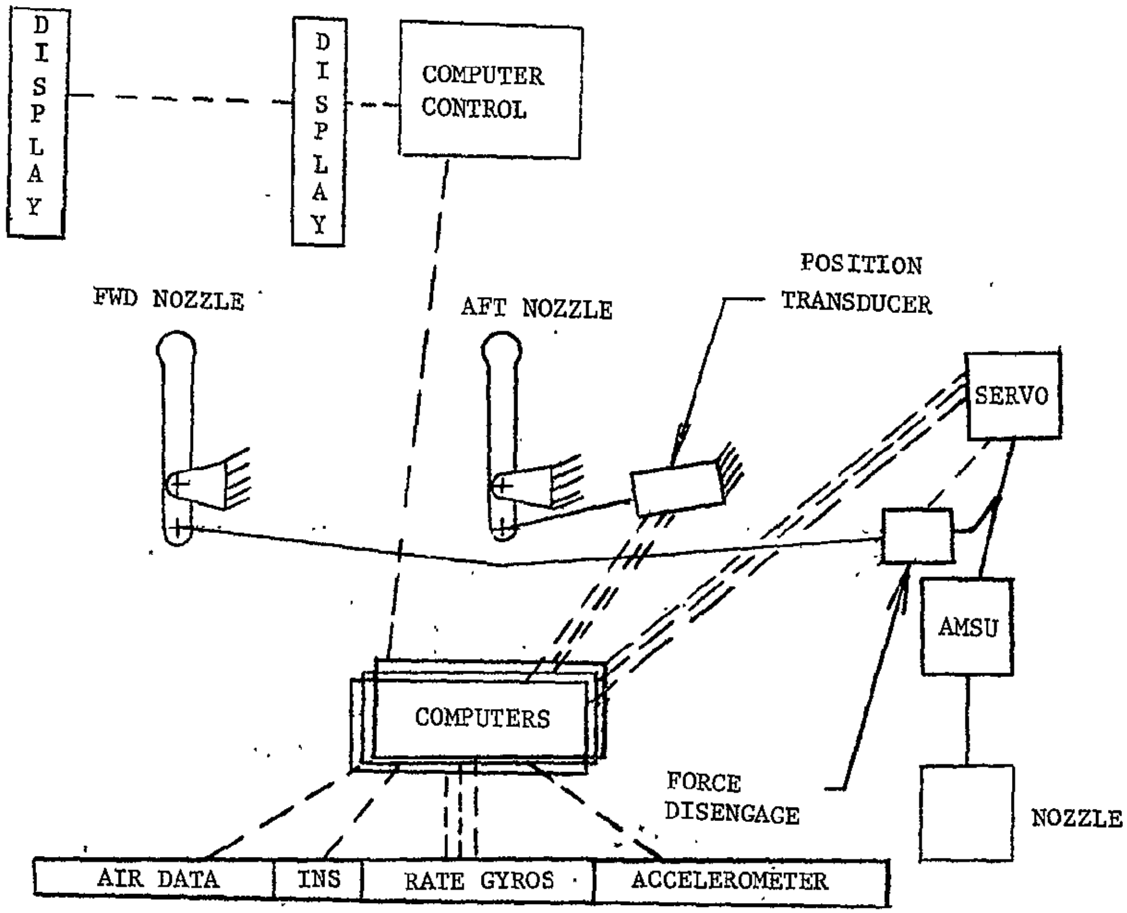


FIGURE 3-34 NOZZLE SYSTEM , TRIPLEX PARALLEL

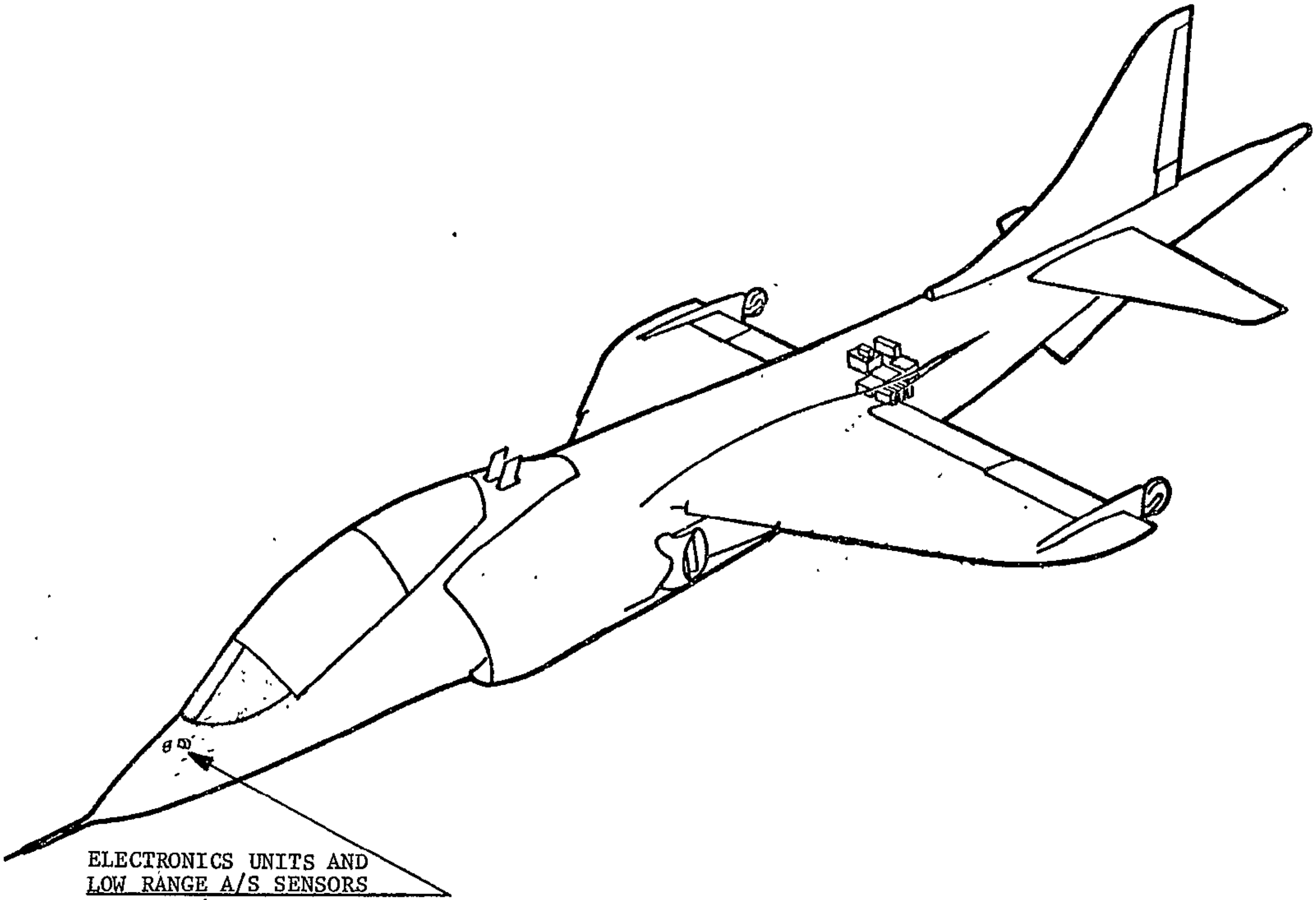


FIGURE 3-35 LOW RANGE AIRSPEED SYSTEM

3.7 SIDE ARM CONTROLLER - The controller selected for this option is the two axis base pivot unit which was developed in the F-15 High Acceleration Cockpit (HAC) program. The controller is equipped with three LVDTs in each axis and three separate wire bundle connectors so that the unit can be used with either the simplex, duplex or triplex flight control configuration. Feel force gradients in the pitch and roll axis are provided by two spring cartridges located inside the controller. If a feel force gradient change is desired the spring cartridges can be removed and replaced with cartridges which have the desired gradients. A procurement specification (P/N 68-031012-101) has been prepared but the unit has not been purchased or qualified.

The inclusion of the side arm controller in the aft cockpit requires structural preparation in the right console area. The oxygen indicators located just above the right console will be moved to the forward right hand subinstrument panel where they will replace the TACAN control panel which is not required for this test. The small oxygen indicator shelf is removed to give the side arm control stick enough clearance for full control travel. The top of the right console will be cleared of equipment control boxes and a structural support will be added to accept the unit. The support will include adjustment provisions to assure pilot comfort and smooth operation.

Figure 3-36 shows the location in the aft cockpit where the side arm controller will be mounted. Also shown are the new location of the oxygen indicators and the indicator shelf which will be removed.

Figures 3-37 and 3-38 give side and top views of the controller mounted in the cockpit. As shown in these figures the controller mounting is such that ample clearance is available for the designed controller displacements of $\pm 10^{\circ}$ in the pitch axis and $\pm 7.5^{\circ}$ in the roll axis.

FIGURE 3-36

AFT COCKPIT
SPECIAL CONTROLS
(SIDE ARM CONTROLLER)

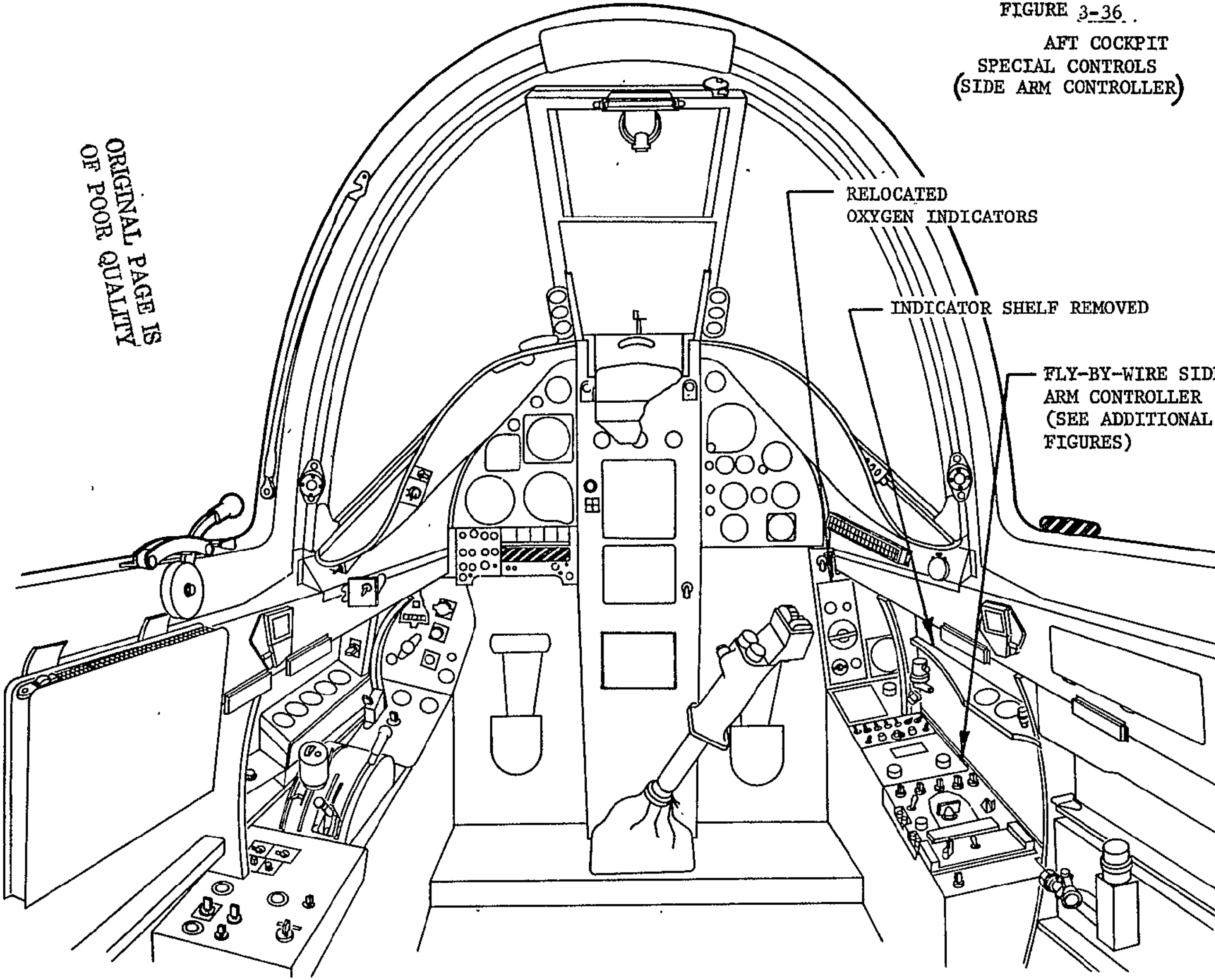


FIGURE 3-37 SIDE ARM CONTROLLER - SIDE VIEW

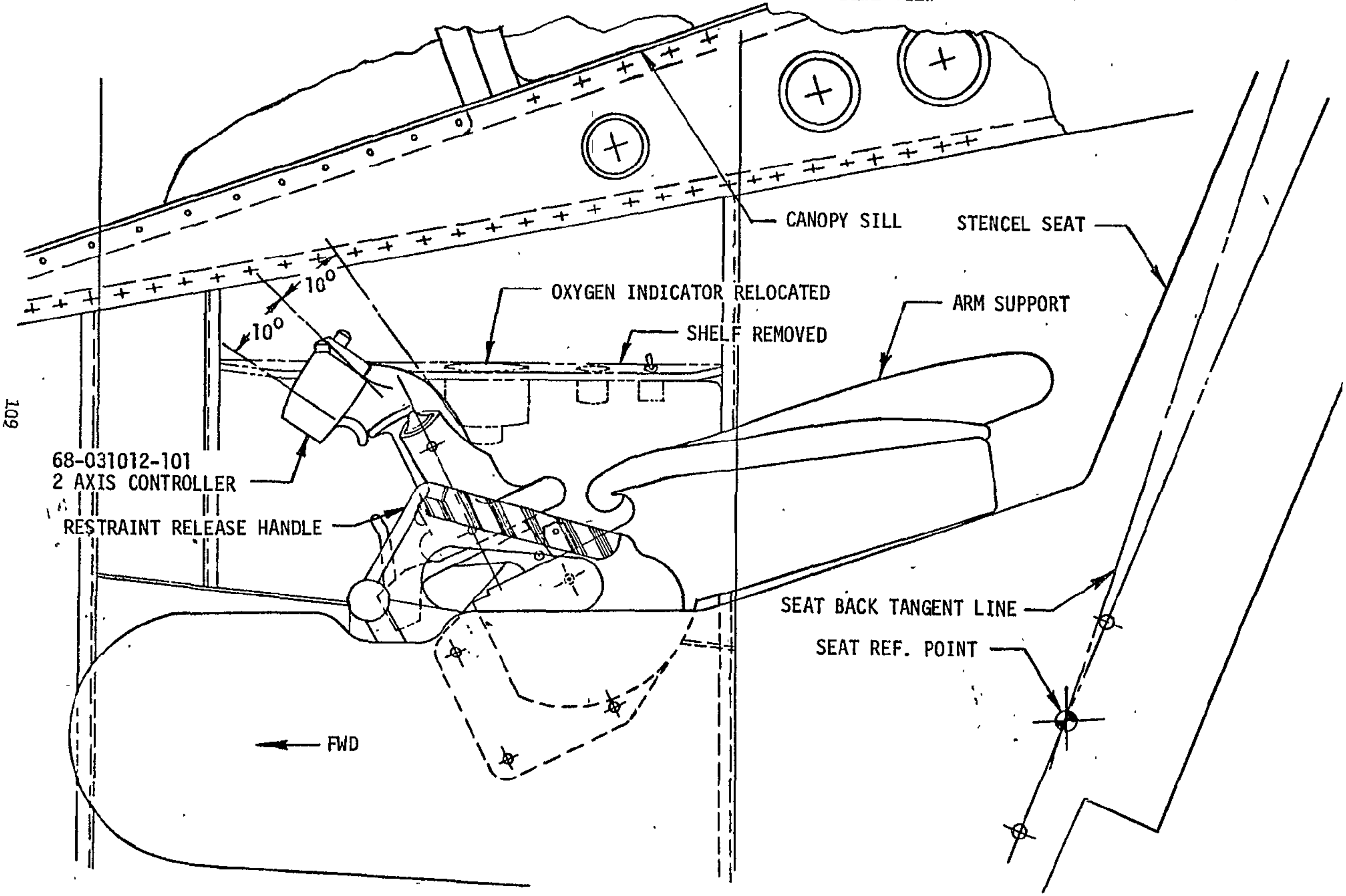
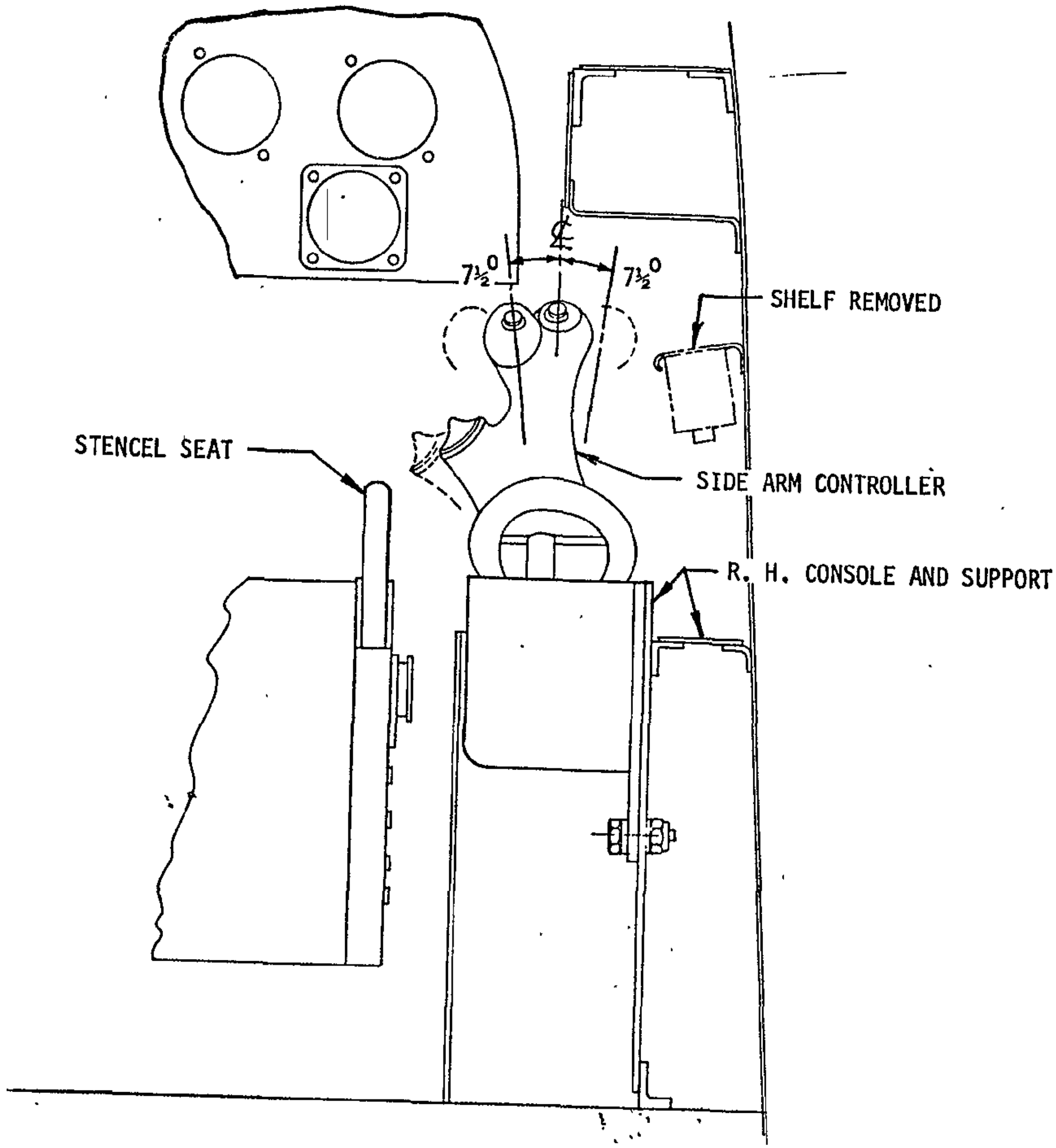


FIGURE 3-38 SIDE ARM CONTROLLER - LOOKING FORWARD



4. PLANS AND SCHEDULES

The modification of a two place Harrier to the NASA V/STOL Research Aircraft configuration will be controlled by a carefully prepared modification plan. Subsequent testing will also be carefully planned and scheduled. The preparation of these plans and schedules is described in this section.

4.1 MODIFICATION PLAN AND SCHEDULE

Once the options have been specified and Authority to Proceed has been issued, Manufacturing Planning will prepare a modification book. This book will be in the form of a Parts List Assembly Order and will contain instructions for implementation of the modifications required. It will describe the required operations in sequence to facilitate manloading and establish clear inspection points. Inspectors will stamp the pages as the work is completed and turn them over to Inspection Permanent Records. The pages can thus be used at any time in the future to verify that any portion of the work was done. It has been demonstrated on many previous programs that this system provides the most efficient, cost effective means of guaranteeing a quality product to the customer on schedule and within budgeted cost.

The modification plan has the following highlights:

- Coordinated Engineering Release Schedule - After receipt of Authority to Proceed, Manufacturing and Engineering will negotiate a mutually agreeable Item Release Schedule for the timely release of drawings to Manufacturing.

- Tools and Parts Fabrication - Manufacturing and Planning will order tools to fabricate the necessary parts.

- Aircraft Receipt and Inventory - Upon receipt of the aircraft, an inventory of installed equipment will be recorded to establish the configuration.

- Prepare for Modification - For safety reasons and accessibility it is necessary to remove the engine, defuel and purge the fuel system.

- Removals - Certain equipment will be removed to gain access to areas that require structural modifications for new and relocated equipment.

- Structural Modifications - The required structural modifications will be incorporated.

- Installations - New equipment, and equipment previously removed, will be installed.

- Operations and Checkout - A systems operation and checkout will be performed (flying controls, landing gear, flaps, etc.).

- Paint - The aircraft will be painted with the paint scheme agreed upon by the customer.

- Preflight - A complete preflight including engine run will be performed to ensure all systems are operational and the aircraft will be released for flight.

- First Flight - A MCAIR pilot will fly the aircraft. This flight will be limited to checking out the retained Harrier flight control system (front cockpit).

The Modification Schedule is given in Figure 4-1.

4.2 TEST PLAN AND SCHEDULE

The Test Plan for the NASA V/STOL Research Vehicle is presented in Figure 4-2. The primary elements in this test plan are:

- Check out of all aircraft systems while aircraft is on ramp status.

- Check out of installed fly-by-wire system by simulating signals into the computer and measuring the output signals.

- Ground tests on the aircraft (fly-by-wire system action):
  Rudder Ground Vibration Test (GVT).
  Free play and rigidity tests control surfaces.
  Frequency response of control surfaces.

- Engine ground run will measure vibration environment in the equipment shelf and the gun pod.

- The basic aircraft systems (excluding the fly-by-wire system) will be evaluated prior to delivery of the aircraft to NASA, these checks will require five flights out of the MCAIR facility at St. Louis, Missouri.

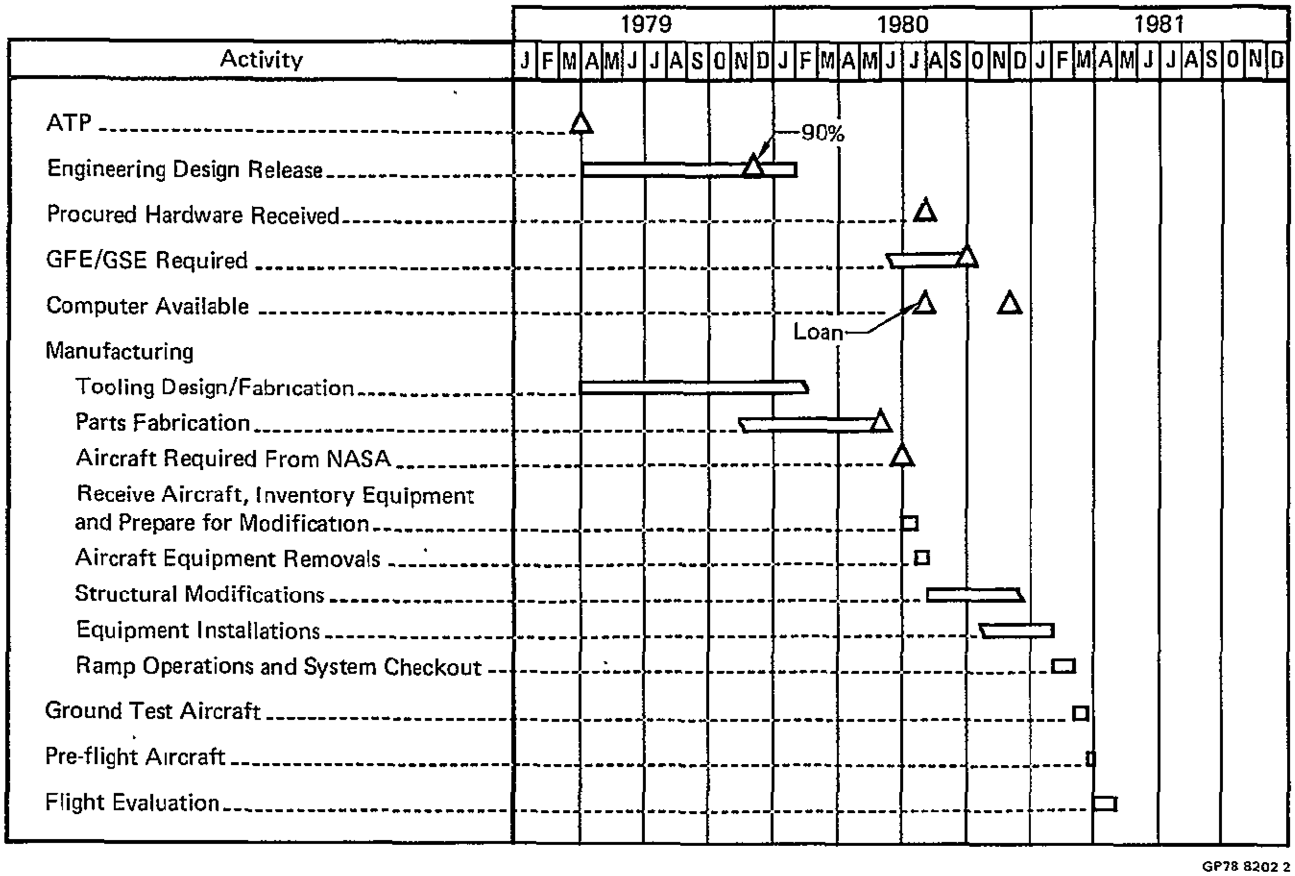


FIGURE 4-1 MODIFICATION PLAN FOR NASA V/STOL RESEARCH AIRCRAFT

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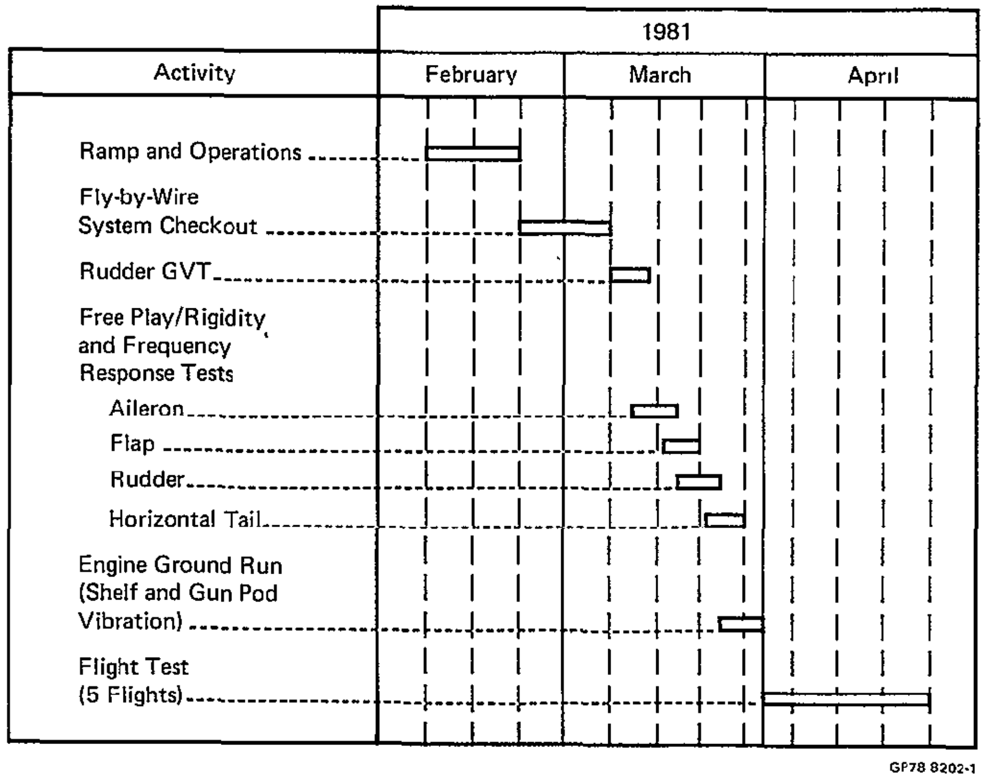


FIGURE 4-2 NASA V/STOL RESEARCH AIRCRAFT TEST PLAN

5. STUDIES AND ANALYSES

Investigations which were performed during this program included studies and analyses in the areas of reliability, performance, and weight and balance. Results of these studies are summarized in this section. The detailed failure mode and effects analyses which were developed during the reliability studies are contained in Appendix C.

5.1 RELIABILITY ANALYSIS

Failure Mode and Effects Analysis (FMEA) is a design evaluation procedure which documents potential failures and determines, by analysis, the effect of those failures on system operation. FMEAs are prepared early in the design formulation process through the coordinated efforts of reliability, maintainability, system safety, and design engineers. System deficiencies are thus detected early and guidance for corrective action provided.

5.1.1 FMEA GUIDELINES AND RESULTS - FMEA's are prepared using MIL-STD-2070(AS), "Procedures for Performing a Failure Mode, Effects and Criticality Analysis" as a guide. The criticality (hazard) classifications used are divided into four classes consistent with MIL-STD-882 hazard levels:

a. Class I - Negligible - A failure that does not degrade performance or operation but which requires corrective maintenance.

b. Class II - Marginal - A failure that can degrade performance or result in degraded operation. Special operating techniques or alternative modes of operation necessitated by the loss can be tolerated throughout the mission but should be corrected upon its completion.

c. Class III - Critical - A single failure that can result in the loss of mission or serious hazard or injury to personnel.

d. Class IV - Catastrophic - A single failure that can result in death or severe injury or loss of system (aircraft, missile, etc.).

FMEA's for the simplex, duplex and triplex flight control systems are presented in Appendix C. Their comparison reveals the effectiveness of progressive levels of redundancy in alleviating the effects of two of the more serious failures in electronic flight controls. Hardovers and disconnects are seen to be the most serious potential problem.

For safe operation with a simplex system in the aft cockpit, the safety pilot must be able to detect, respond to and recover from hardovers and disconnects at any point in the flight envelope. Going to a duplex system virturally eliminates hardovers but has at least twice the disconnects of the simplex system. Thus, with a duplex system the safety pilot would still have to recover from disconnects but would not have to contend with hardovers. A triplex system in the back seat would virtually eliminate both disconnects and hardovers on first failure, the only out of the ordinary indication to either pilot would be a channel-fail light. Control of the aircraft could then be passed from evaluation pilot to safety pilot at the convenience of the crew.

Because safe recovery from hardover and disconnect failures in the simplex and duplex systems is highly dependent on the safety pilot's reaction time, it is recommended that manned simulator studies for simplex or duplex configurations include uncommanded hardovers and disconnects. These should be introduced at random intervals and without prior notice to the pilots throughout the flight envelope. Analysis of the simulator data should help provide a basis for decisions as to whether a simplex or duplex system constitutes an acceptable risk.

5.1.2 POTENTIAL LOSS OF CONTROL PREDICTIONS - The preliminary FMEAs have identified uncommanded hardovers and back seat disengagement as potential causes for concern for the simplex and duplex electronic flight control systems. This concern led to math modeling and calculation of control loss probability from these two failure modes for each of the three levels of redundancy under consideration.

For the purpose of this evaluation, the existing mechanical flight control system of the TAV-8A aircraft was considered sufficiently similar to that of the single place AV-8A to permit the use of AV-8A failure frequency data. This is extremely desirable since most of the more than 70,000 flight hours accumulated by the Marine Corps have been logged on the AV-8A. The source of the baseline mechanical flight control system failure data for this evaluation is the Navy's 3M (Logistics and Maintenance) data system for the time period 1974-76 (39,500 Flight Hours). Certain "How Malfunctioned" codes (i.e. 070-Broken, 135 Bound Binding, Stuck, or Jammed and 780 Bent, Buckled, or Collapsed) were considered to represent real failures which could result in degraded flight control characteristics. Examination of these codes revealed that 140 of them had occurred in the lateral, longitudinal and directional flight control systems. In contrast, safety data reviewed for the same time period revealed only 1 accident in which the flight control system was even a contributing factor (and in that single case, pilot error was attributed to be the primary cause). Conservatively, it was therefore considered that 1% of the flight control system broken/bound/bent failures would result in a potential loss of control situation. Utilizing this approach, the distribution of TAV-8A baseline flight control failures which could result in a potential loss of control situation was estimated as follows:

Lateral Control = .0117 Failures/1000 Flight Hours
Directional Control = .0165 Failures/1000 Flight Hours
Longitudinal Control = .0074 Failures/1000 Flight Hours
Total = .0356 Failures/1000 Flight Hours

5.1.3 HARDWARE IMPLEMENTATION IMPACT - The impact of adding the hardware necessary to implement the rear cockpit digital flight control system capability was estimated by using failure rate data from Rome Air Development Center studies (AFFDL-TR-75-59) and component vendors. These data, in conjunction with estimates of the percent of component failures which could result in a potential loss of control situation and the Built In Test (BIT)/In Flight Integrity Management (IFIM) percentages considered to be reasonably achievable for different types of components, were used to arrive at an undetected-potential-loss-of-control-failure rate for the alternate system implementations. For purposes of simplification only added major flight control system components were considered.

Figure 5-1 illustrates the predicted effect of adding the single channel parallel digital flight control system to the basic TAV-8A aircraft. In effect, a failure every 225 flight hours is added to the flight control system by the simplex system. More importantly, the predicted potential-loss-of-control-failure rate for the flight control system is increased approximately ten-fold over the baseline mechanical flight control system from one per 30,000 flight hours to approximately one per 3000 flight hours. Assuming adequate and IFIM for the single channel, the system will experience disengagements at a rate of approximately 1 1/2 times the digital flight control system component failure rate, because of in line monitoring measurement techniques. Therefore, a disengagement can be expected every 150 flight hours.

Figure 5-2 illustrates the predicted effect of adding the dual channel series digital flight control system to the basic TAV-8A aircraft. In effect, a failure every 100 flight hours is added to the flight control system by the dual system. The predicted potential loss-of-control failure rate for the flight control system is increased approximately 10% over the baseline mechanical flight control system to approximately one per 25,000 flight hours. Assuming a simple cross-channel comparison BIT/IFIM system, a disengagement can be expected every 100 flight hours.

Figure 5-3 illustrates the predicted effect of adding the triplex parallel digital flight control system to the basic TAV-8. A failure every 75 flight hours is effectively added to the flight control system by the triplex system. The predicted potential loss-of-control failure rate for the flight control system is effectively unchanged from the baseline mechanical flight control system. Assuming a voting scheme (2 out of 3), a warning light indicating a failure in one of the channels can be expected every 75 flight hours. The evaluation pilot can then transfer aircraft control to the safety pilot at their mutual convenience without either a potential loss-of-control failure or an uncommanded disengagement. After the occurrence of the first failure in a channel, that channel is disconnected and the remaining two channels are compared for failure detection.

Each of these analyses assumes that the integrity of each channel of the added system will be validated (by BIT, visual checks, etc.) between flights. Each flight is assumed to average one hour.

5.1.4 EFFECT OF SELECTIVE REDUNDANCY - Figures 5-2 and 5-3 reflect the predicted reliability of redundant systems in which all the added elements have been duplicated or triplicated, as the case may be, which is most desirable from a reliability standpoint. In the event of a design decision not to duplicate (or triplicate) a given component type such as the accelerometer, the summary numbers are modified as follows:

FIGURE 5-1

SINGLE CHANNEL RELIABILITY MODEL (DIGITAL FLIGHT CONTROL SYSTEM)

LATERAL CONTROL - ELECTRICAL ACTUATOR

	RATE GYRO	ACCEL	A/D CONV	COMPUTER	D/A CONV	PWR AMPLS	ACTUATOR MOTOR	TOTAL
FAILURES/1000 FH	.4	.2	.15	1.6	.15	.03(EA)	.05	1.610
POTENTIAL LOSS OF CONTROL FR	.12	.06	.075	.8	.075	.015(EA)	.001	1.161
PERCENT DETECTION (BIT/IFIM)	75%	95%	85%	90%	85%	75%	10%	--
UNDETECTED POTENTIAL LOSS OF CONTROL FR	.03	.003	.011	.08	.011	.008(TOT)	.001	.144

LONGITUDINAL CONTROL - ELECTRICAL ACTUATOR

	RATE GYRO	ACCEL	A/D CONV	COMPUTER	D/A CONV	PWR AMPLS	ACTUATOR MOTOR	TOTAL
FAILURES/1000 FH	.4	.2	.15	COMMON	.15	.03(EA)	.05	1.010
POTENTIAL LOSS OF CONTROL FR	.12	.06	.075	COMMON	.075	.015(EA)	.001	.361
PERCENT DETECTION (BIT/IFIM)	75%	95%	85%	COMMON	85%	75%	10%	--
UNDETECTED POTENTIAL LOSS OF CONTROL FR	.03	.003	.011	COMMON	.011	.008(TOT)	.001	.064

DIRECTIONAL CONTROL - ELECTRICAL ACTUATOR

	RATE GYRO	ACCEL	A/D CONV	COMPUTER	D/A CONV	PWR AMPLS	ACTUATOR MOTOR	TOTAL
FAILURES/1000 FH	.4	COMMON	.15	COMMON	.15	.03(EA)	.05	.810
POTENTIAL LOSS OF CONTROL FR	.12	COMMON	.075	COMMON	.075	.015(EA)	.001	.301
PERCENT DETECTION (BIT/IFIM)	75%	COMMON	85%	COMMON	85%	75%	10%	--
UNDETECTED POTENTIAL LOSS OF CONTROL FR	.03	COMMON	.011	COMMON	.011	.008(TOT)	.001	.061

TOTAL SINGLE CHANNEL DFCS -	
FAILURES/1000 FH	4.430
POTENTIAL LOSS OF CONTROL FR	1.823
UNDETECTED POTENTIAL LOSS OF CONTROL FR	0.269

FH = FLIGHT HOURS
FR = FAILURE RATE

FIGURE 5-2

DUAL CHANNEL RELIABILITY MODEL (DIGITAL FLIGHT CONTROL SYSTEM)

LATERAL CONTROL - HYDRAULIC ACTUATOR

	RATE GYRO	ACCEL	A/D CONV	COMPUTER	D/A CONV	HYDRAULIC ACTUATOR	TOTAL
FAILURES/1000 FH	.8	.4	.3	3.2	.3	.6	5.600
POTENTIAL LOSS OF CONTROL FR	.24	.12	.15	1.6	.15	.06	2.320
PERCENT DETECTION (BIT/IFIM)	99.99%	99.99%	99.99%	99.99%	99.99%	99.99	--
UNDETECTED POTENTIAL LOSS OF CONTROL FR	--	--	--	--	--	--	0.003

LONGITUDINAL CONTROL - HYDRAULIC ACTUATOR

	RATE GYRO	ACCEL	A/D CONV	COMPUTER	D/A CONV	HYDRAULIC ACTUATOR	TOTAL
FAILURES/1000 FH	.8	.4	.3	COMMON	.3	.6	2.400
POTENTIAL LOSS OF CONTROL FR	.24	.12	.15	COMMON	.15	.06	0.720
PERCENT DETECTION (BIT/IFIM)	99.99%	99.99%	99.99%	COMMON	99.99%	99.99	--
UNDETECTED POTENTIAL LOSS OF CONTROL FR	--	--	--	--	--	--	0.0006

DIRECTIONAL CONTROL - HYDRAULIC ACTUATOR

	RATE GYRO	ACCEL	A/D CONV	COMPUTER	D/A CONV	HYDRAULIC ACTUATOR	TOTAL
FAILURES/1000 FH	.8	COMMON	.3	COMMON	.3	.6	2.000
POTENTIAL LOSS OF CONTROL FR	.24	COMMON	.15	COMMON	.15	.06	0.600
PERCENT DETECTION (BIT/IFIM)	99.99%	COMMON	99.99%	COMMON	99.99	99.99	--
UNDETECTED POTENTIAL LOSS OF CONTROL FR	--	COMMON	--	--	--	--	0.0006

TOTAL DUAL CHANNEL DFCS -	
FAILURES/1000 FH	10.000
POTENTIAL LOSS OF CONTROL FR	3.640
UNDETECTED POTENTIAL LOSS OF CONTROL FR	0.0042

FH = FLIGHT HOURS
FR = FAILURE RATE

FIGURE 5-3

TRIPLEX RELIABILITY MODEL (DIGITAL FLIGHT CONTROL SYSTEM)

LATERAL CONTROL - ELECTRICAL ACTUATOR

	RATE GYRO	ACCEL	A/D CONV	COMPUTER	D/A CONV	PWR AMPLS	ACTUATOR MOTOR	TOTAL
FAILURES/1000 FH	1.2	.6	.45	4.8	.45	.18	.15	7.830
POTENTIAL LOSS OF CONTROL FR	.36	.18	.125	2.4	.125	.045	.003	3.238
PERCENT DETECTION (BIT/IFIM)	99.999%	99.999%	99.999%	99.999%	99.999%	99.999%	99.999%	--
UNDETECTED POTENTIAL LOSS OF CONTROL FR	--	--	--	--	--	--	--	--

LONGITUDINAL CONTROL - ELECTRICAL ACTUATOR

	RATE GYRO	ACCEL	A/D CONV	COMPUTER	D/A CONV	PWR AMPLS	ACTUATOR MOTOR	TOTAL
FAILURES/1000 FH	1.2	.6	.45	COMMON	.45	.18	.15	3.030
POTENTIAL LOSS OF CONTROL FR	.36	.18	.125	COMMON	.125	.045	.003	0.838
PERCENT DETECTION (BIT/IFIM)	99.999%	99.999%	99.999%	COMMON	99.999%	99.999%	99.999%	--
UNDETECTED POTENTIAL LOSS OF CONTROL FR	--	--	--	--	--	--	--	--

DIRECTIONAL CONTROL - ELECTRICAL ACTUATOR

	RATE GYRO	ACCEL	A/D CONV	COMPUTER	D/A CONV	PWR AMPLS	ACTUATOR MOTOR	TOTAL
FAILURES/1000 FH	1.2	COMMON	.45	COMMON	.45	.18	.15	2.430
POTENTIAL LOSS OF CONTROL FR	.36	COMMON	.125	COMMON	.125	.045	.003	0.658
PERCENT DETECTION (BIT/IFIM)	99.999%	COMMON	99.999%	COMMON	99.999%	99.999%	99.999%	--
UNDETECTED POTENTIAL LOSS OF CONTROL FR	--	--	--	--	--	--	--	--

TOTAL TRIPLEX DFCS -	
FAILURES/1000 FH	13.290
POTENTIAL LOSS OF CONTROL FR	4.734
UNDETECTED POTENTIAL LOSS OF CONTROL FR	HIGHLY IMPROBABLE

FH = FLIGHT HOURS
FR = FAILURE RATE

Dual System (except accelerometer)	
Total Failures/1000 Flight Hours (FH)	9.600
Potential Loss of Control Failure Rate (FR)	3.520
Undetected Potential Loss of Control FR	0.0102
Triplex System (except accelerometer)	
Total Failures/1000 FH	12.490
Potential Loss of Control FR	4.494
Undetected Potential Loss of Control FR	0.006

The study results indicate that a significant numerical increase in loss of control probability is associated with the use of a simplex system. Adding a second channel to make it a duplex system reduces the control loss probability to slightly move than that of the mechanical system. A third channel makes the electronic system approximately equal to the mechanical system.

The selective redundancy study indicated that the system failure rate becomes essentially that of the nonredundant components.

5.2 AIRCRAFT PERFORMANCE

This section summarizes the performance data for the NASA V/STOL research aircraft, and provides typical variations of data for V/STOL performance with ambient temperature and pressure altitude. Performance data is included for conventional wing-borne flight, partially jet-borne flight and jet-borne flight. Performance comparisons are presented for the simplex, duplex and triplex flight control system.

5.2.1 LEVEL FLIGHT - The NASA V/STOL research aircraft is a high performance vectored thrust aircraft. The level flight envelope for the clean configuration with a gross weight of 7585 kg (16707 lb) is shown in Figure 5-4. Level flight at speeds lower than $M = 0.4$ requires use of vectored thrust while flight at speeds lower than $M = 0.25$ requires that the throttle be moved into the lift rating range (RPM $\geq$ 100%).

5.2.2 V/STOL - Harrier V/STOL performance capability is a function of actual (as distinguished from nominal) engine performance. The major variables are:

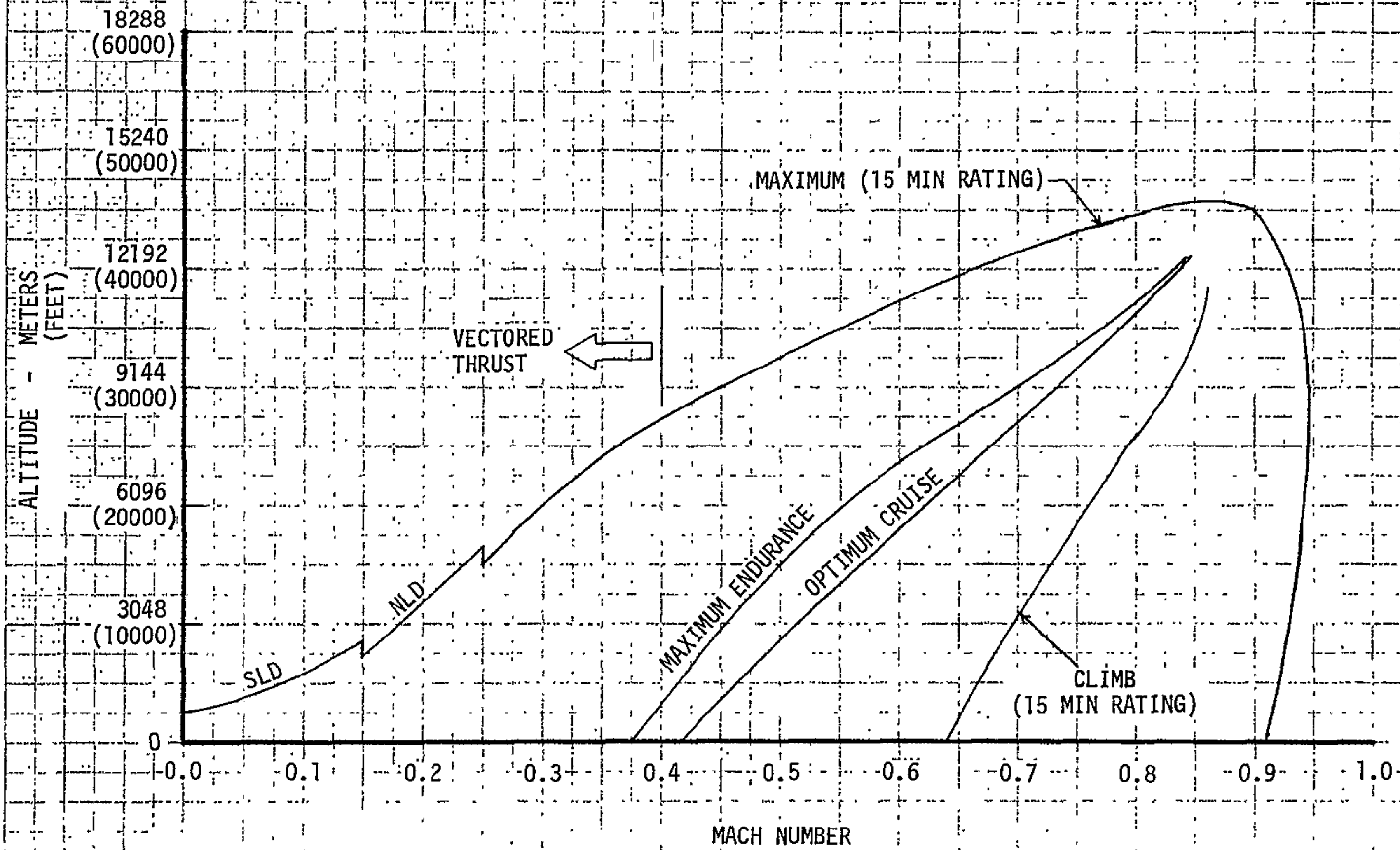
- o Relative hover performance
- o Relative JPT
- o JPT limiter settings
- o Maximum RPM settings

These variables will be discussed in the following sections as their effects on V/STOL performance are shown.

5.2.3 ENGINE OPERATING LIMITATIONS - An operational limitation imposed on the engine restricts continuous engine bleed to a 5 minute period. This is basicly an aircraft limitation on the use of the reaction control system (RCS). The hot bleed air flowing through the RCS ducts heats the aircraft structure and, when in hover, heats the landing gear, brakes and tires.

Each period of operating in the lift ratings must be separated by at least 5 minutes in or below the maximum continuous thrust rating.

FIGURE 5-4
NASA V/STOL RESEARCH AIRCRAFT
LEVEL FLIGHT ENVELOPE AND VELOCITY PERFORMANCE
GROSS WEIGHT = 7584 KG
(16707 LBS)



JPT is limited, on the nominal engine, to 710°C for the SLD rating, and to 740°C for the SLW rating.

5.2.4 VTO - Maximum VTO performance is shown in Figure 5-5. This figure shows the maximum gross weights for VTOs as a function of the outside air temperature. These curves represent standard VTO performance, i.e. VTOs which attain a height of 6 meters (20 ft) in 5 to 10 seconds. Also shown are the VL curves the the Short Lift Dry (SLD) and Short Lift Wet (SLW) throttle settings. Figure 5-6 shows the effects of altitude and type of day on VTO performance for the same conditions. It should be noted that water injection is used only when the outside temperature is 5°C or higher.

5.2.5 HOVER - The effects of specific engine performance is addressed in this section in terms of relative hover performance. Figures 5-7 and 5-8, taken from the AV-8A/TAV-8A NATOPS Manual, illustrate these effects. Figure 5-7 shows the engine RPM required as a function of hover gross weight. This figure shows that a ±3% difference in relative hover performance can mean a difference of up to 4% in RPM required to hover. Figure 5-8 shows the effect of JPT in hover. A ±30° difference in relative JPT can result in a 5.5% RPM difference in RPM required to hover.

The relative hover performance of a specific aircraft is easily determined by carefully establishing a steady hover and recording the RPM, JPT and aircraft gross weight. The ambient temperature and pressure at the hover altitude are also noted. This data point is then compared to the nominal engine by using the plots shown in Figures 5-7 and 5-8 to obtain the "Relative Hover" performance.

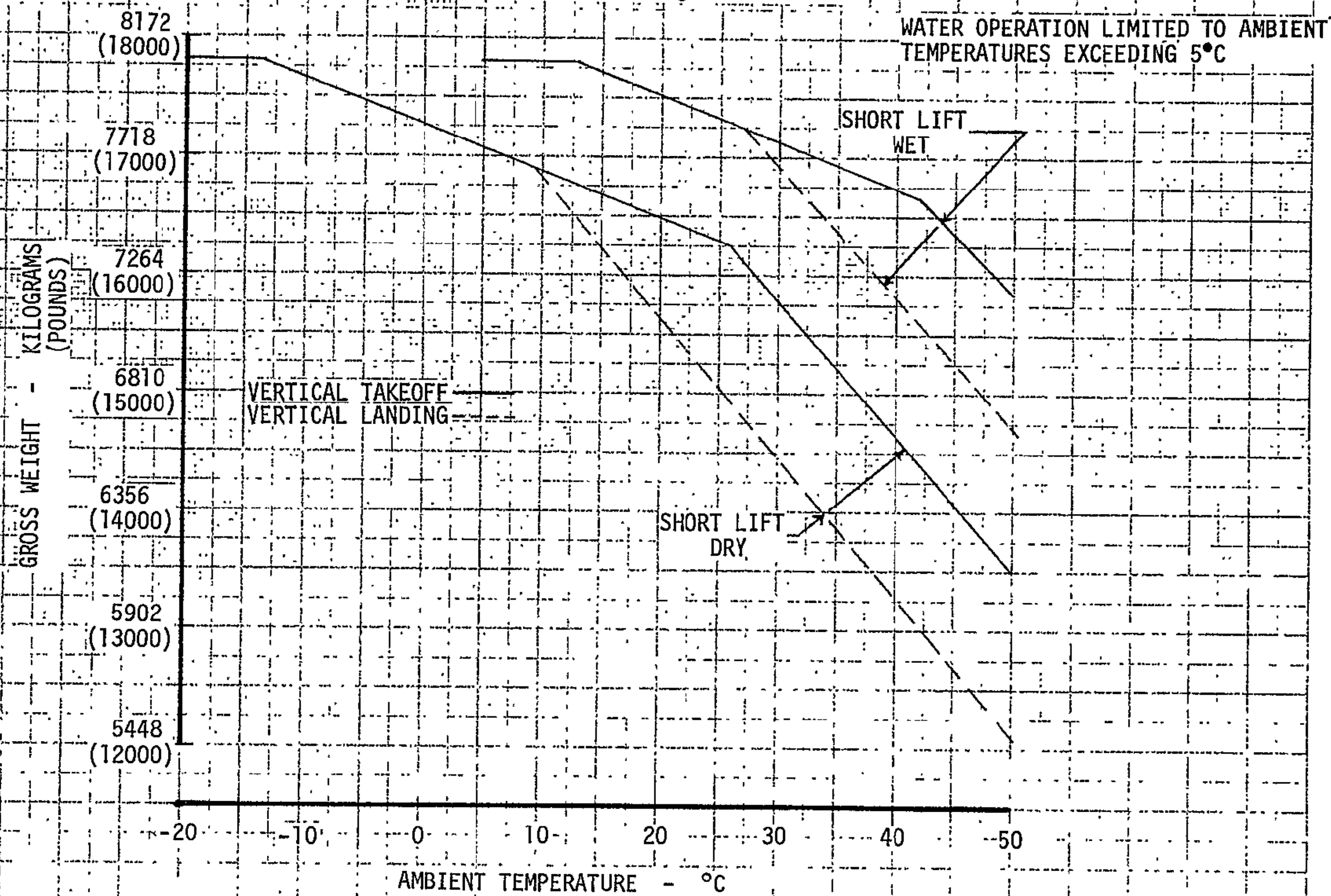
5.2.6 SHORT TAKEOFF - A large improvement in takeoff gross weight and mission range results from using the short takeoff instead of the vertical takeoff. Figure 5-9 shows the target rotation speed and nozzle angle for the corrected hover weight. Figure 5-10 shows the distances to takeoff and to 15 meters (50 ft) as a function of the rotation speed. Both of these figures are from the AV-8A/TAV-8A NATOPS Manual. Figures 5-11 and 5-12 show the effects of gross weight on both ground roll and to a 15 M (50 ft) obstacle. Figure 5-11 is for the short lift wet (SLW) rating and Figure 5-12 is for short lift dry (SLD).

5.2.7 CLIMB - Time, distance and fuel required to climb are shown in Figure 5-13 for the maximum thrust (15 min. rating) engine setting. This is for a drag index of 10 and shows the effects of 3 gross weights.

5.2.8 CEILING - The ceiling, as a function of gross weight, is shown in Figure 5-14. This is the 91 MPM (300 FPM) Rate Of Climb Ceiling.

5.2.9 CRUISE - Cruise performance is shown in Figures 5-15 through 5-16. Figure 5-15 shows the optimum and fast cruise as a function of gross weight. Figure 5-16 shows the sea level cruise performance for various drag indexes as functions of gross weight. Figure 5-17 shows the optimum altitude cruise performance for various drag indexes.

FIGURE 5-5
NASA V/STOL RESEARCH AIRCRAFT
MAXIMUM VERTICAL TAKEOFF AND LANDING CAPABILITY
NOMINAL ENGINE, ZERO BLEED
SEA LEVEL STATIC, θ_j = 81°



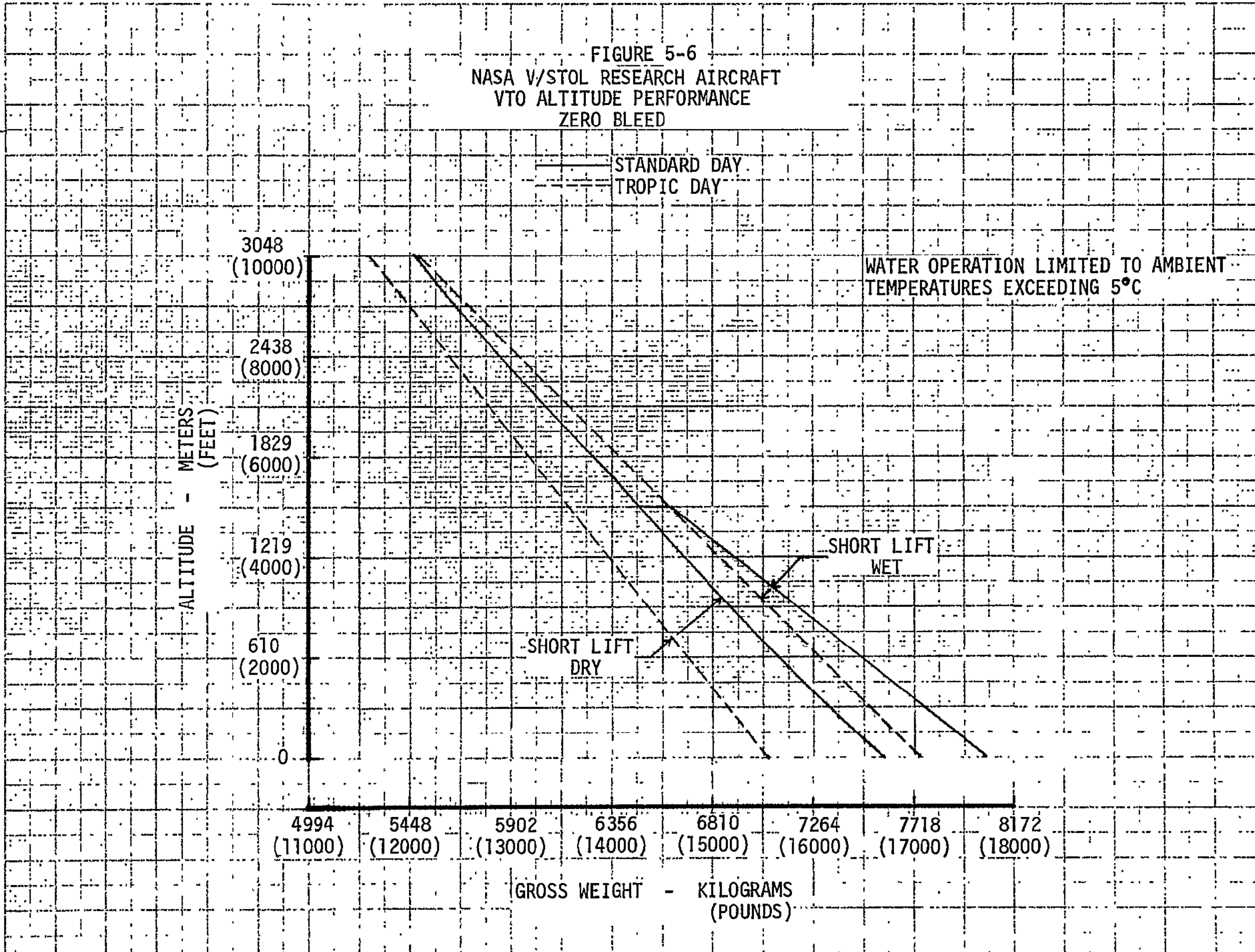


FIGURE 5-6
NASA V/STOL RESEARCH AIRCRAFT
VTO ALTITUDE PERFORMANCE
ZERO BLEED

FIGURE 5-7

ENGINE RPM REQUIRED TO HOVER
NOZZLES IN HOVER STOP WET OR DRY ENGINE OPERATION

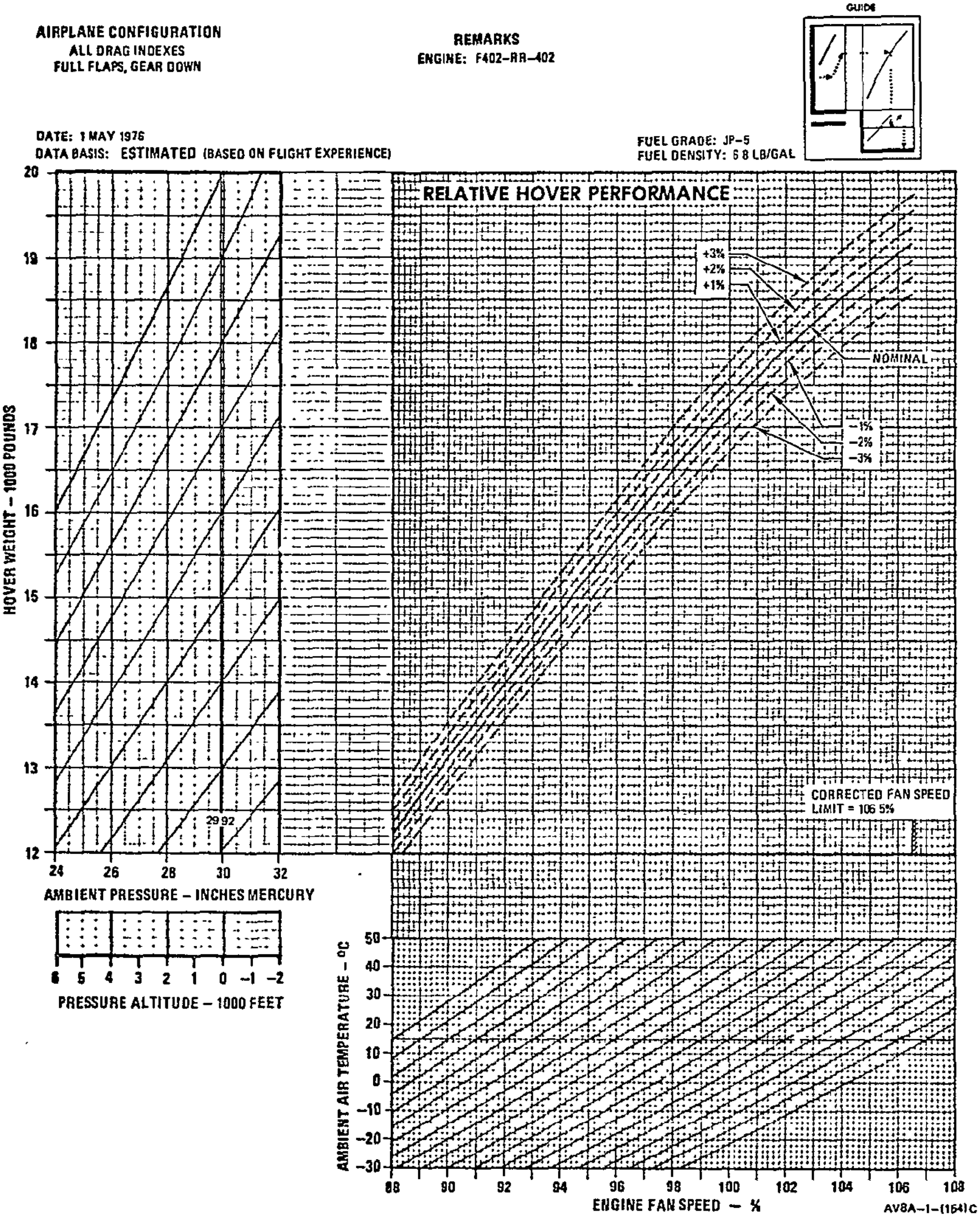


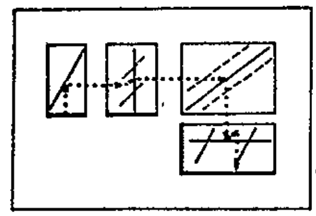
FIGURE 5-8

JPT IN HOVER

AIRPLANE CONFIGURATION
ALL DRAG INDEXES
FULL FLAPS, GEAR DOWN

REMARKS
ENGINE: F402-RR-402

GUIDE



NOTE

JPT SHOWN IS FOR DRY OPERATION WITH WATER FLOWING CORRECTED JPT IS REDUCED APPROXIMATELY 35°K.

JPTL OPERATION
SHORT LIFT WET – 740°C ±10°C
SHORT LIFT DRY – 710°C ±10°C
MAXIMUM THRUST – 600°C ±10°C

DATE: 1 MAY 1976
DATA BASIS: ESTIMATED (BASED ON FLIGHT EXPERIENCE)

FUEL GRADE: JP-5
FUEL DENSITY: 6.5 LB/GAL

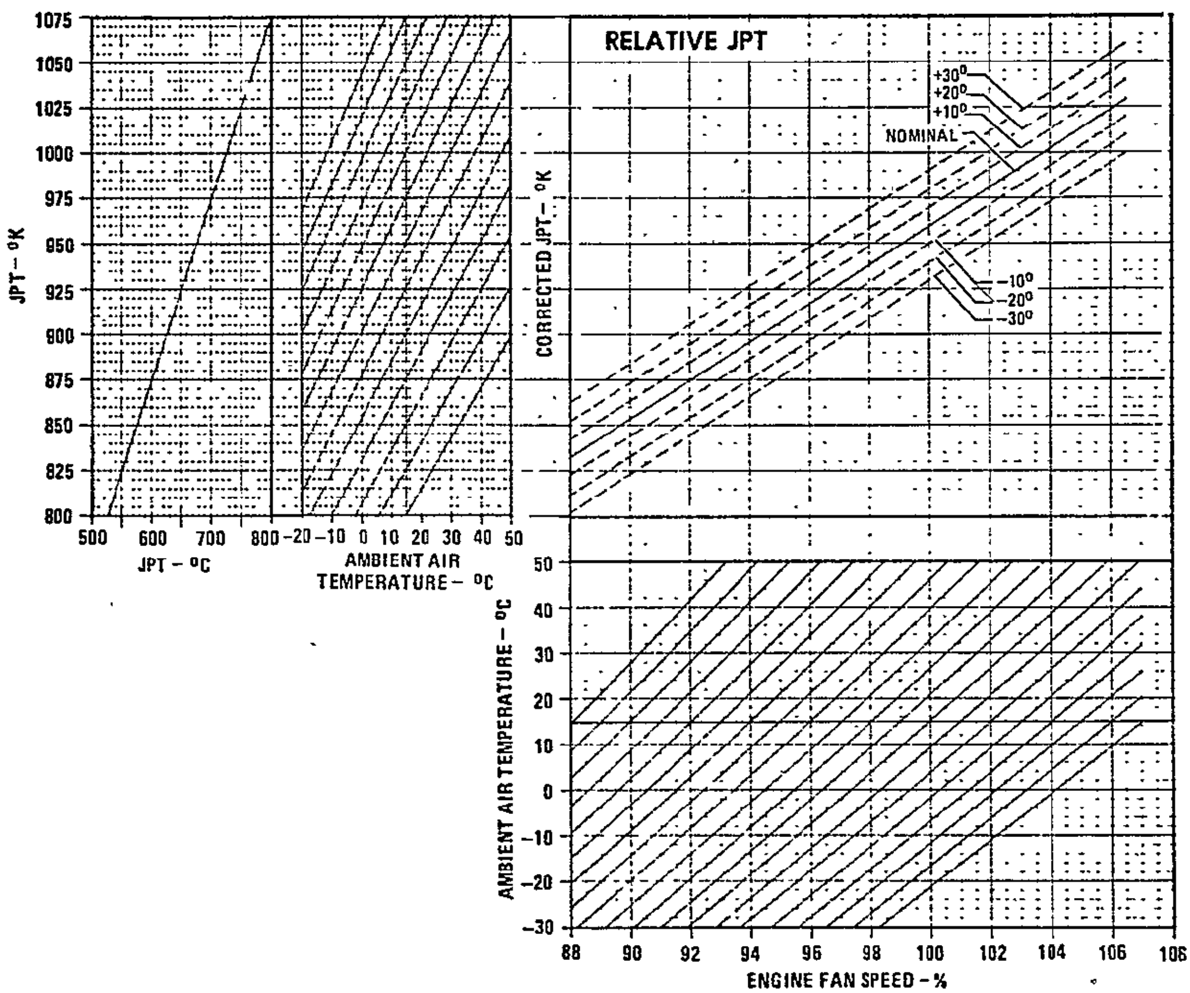


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FIGURE 5-9

SHORT TAKEOFF
ROTATION SPEED AND NOZZLE ANGLES

AIRPLANE CONFIGURATION
ALL DRAG INDEXES
FULL FLAPS, GEAR DOWN

REMARKS
ENGINE: F402-RR-402
SHORT LIFT RATING
NOZZLES: 10° IN GROUND RUN

GUIDE

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL

NOTE

- FOR MAXIMUM PERFORMANCE STO WITH CG AFT OF 13.1% MAC, THE NOZZLE ANGLE IS 50°. THE NOZZLE ROTATION SPEED IS DETERMINED IN THE SAME MANNER AS FOR A NORMAL STO WITH NOZZLE ANGLE GREATER THAN 50°.
- FOR STO WITH 300 GALLON EXTERNAL FUEL TANKS, ADD 5 KNOTS TO CALCULATED NOZZLE ROTATION SPEED.
- SUBTRACT THE APPLICABLE CORRECTION FROM THE CORRECTED HOVER WEIGHT WHEN MAXIMUM BLEED DEMANDS LIKELY TO BE ENCOUNTERED DUE TO GUSTING OR VEERING WIND OR OPERATING NEAR CROSSWIND LIMIT AND:
  - RPM LIMITED (BEFORE PPC 38) – UP TO 700 POUNDS
  - RPM LIMITED (AFTER PPC 38) – UP TO 350 POUNDS
  - JPT LIMITED USING MAXIMUM PERFORMANCE TECHNIQUE FROM OBSTACLE CRITICAL SITE – 600 POUNDS

DATE: 1 MAY 1976
DATE BASIS ESTIMATED
(BASED ON FLIGHT EXPERIENCE)

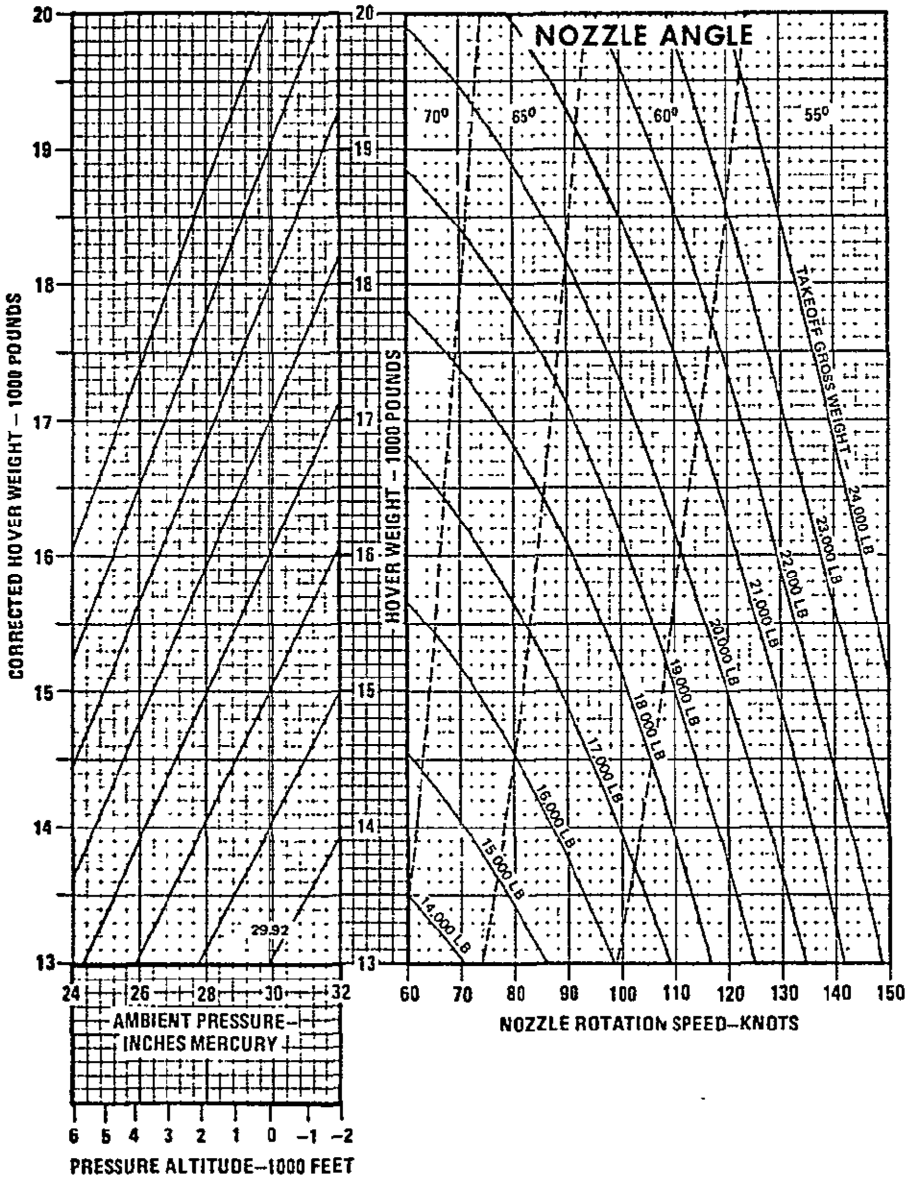

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FIGURE 5-10

SHORT TAKEOFF DISTANCE

SHORT LIFT RATING
10° NOZZLES IN GROUND ROLL

AIRPLANE CONFIGURATION
ALL DRAG INDEXES
FULL FLAPS, GEAR DOWN

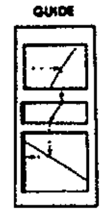
REMARKS
ENGINE: F402-RR-402

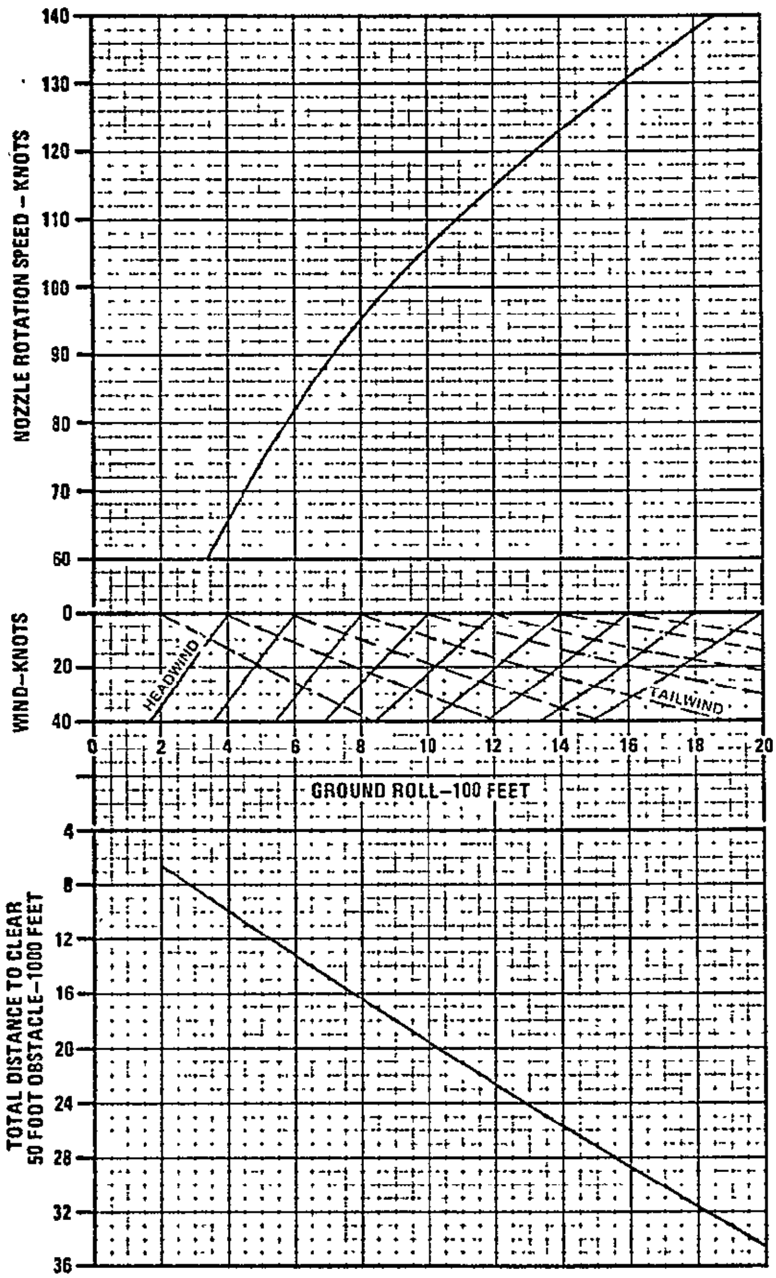
DATE: 1 MAY 1976
DATA BASIS: ESTIMATED
(BASED ON FLIGHT EXPERIENCE)

FUEL GRADE: JP-5
FUEL DENSITY: 6.8 LB/GAL

NOTE

- THE SHORT TAKEOFF DISTANCE OBTAINED FROM THIS FIGURE OR THE V/STOL COMPUTER IS BASED ON A 12 UNIT AOA DURING CLIMB OUT
- CARRIAGE OF LARGE STORES ON INBOARD PYLONS MAY CAUSE SIGNIFICANT INCREASES IN TAKEOFF DISTANCES.





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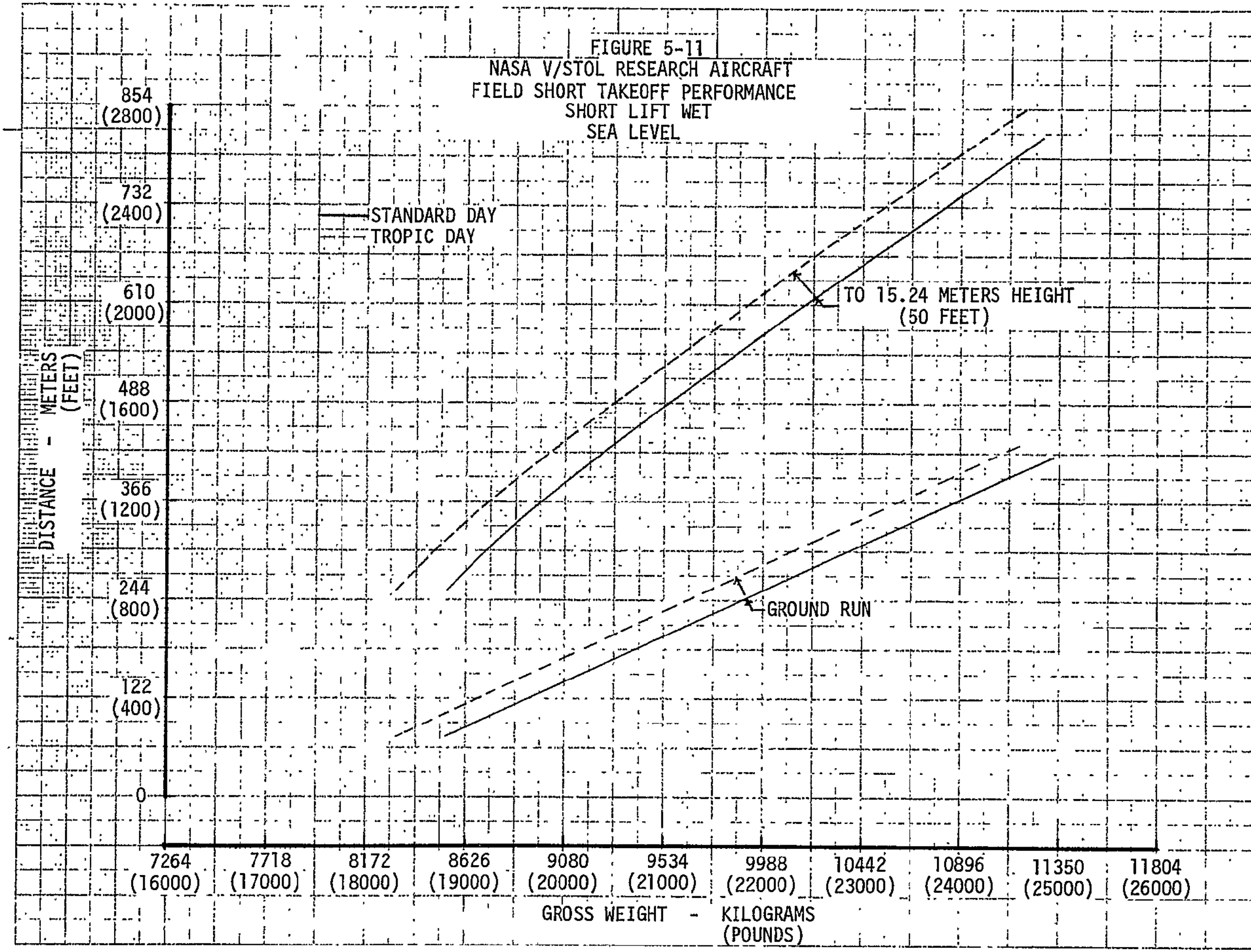


FIGURE 5-11
NASA V/STOL RESEARCH AIRCRAFT
FIELD SHORT TAKEOFF PERFORMANCE
SHORT LIFT WET
SEA LEVEL

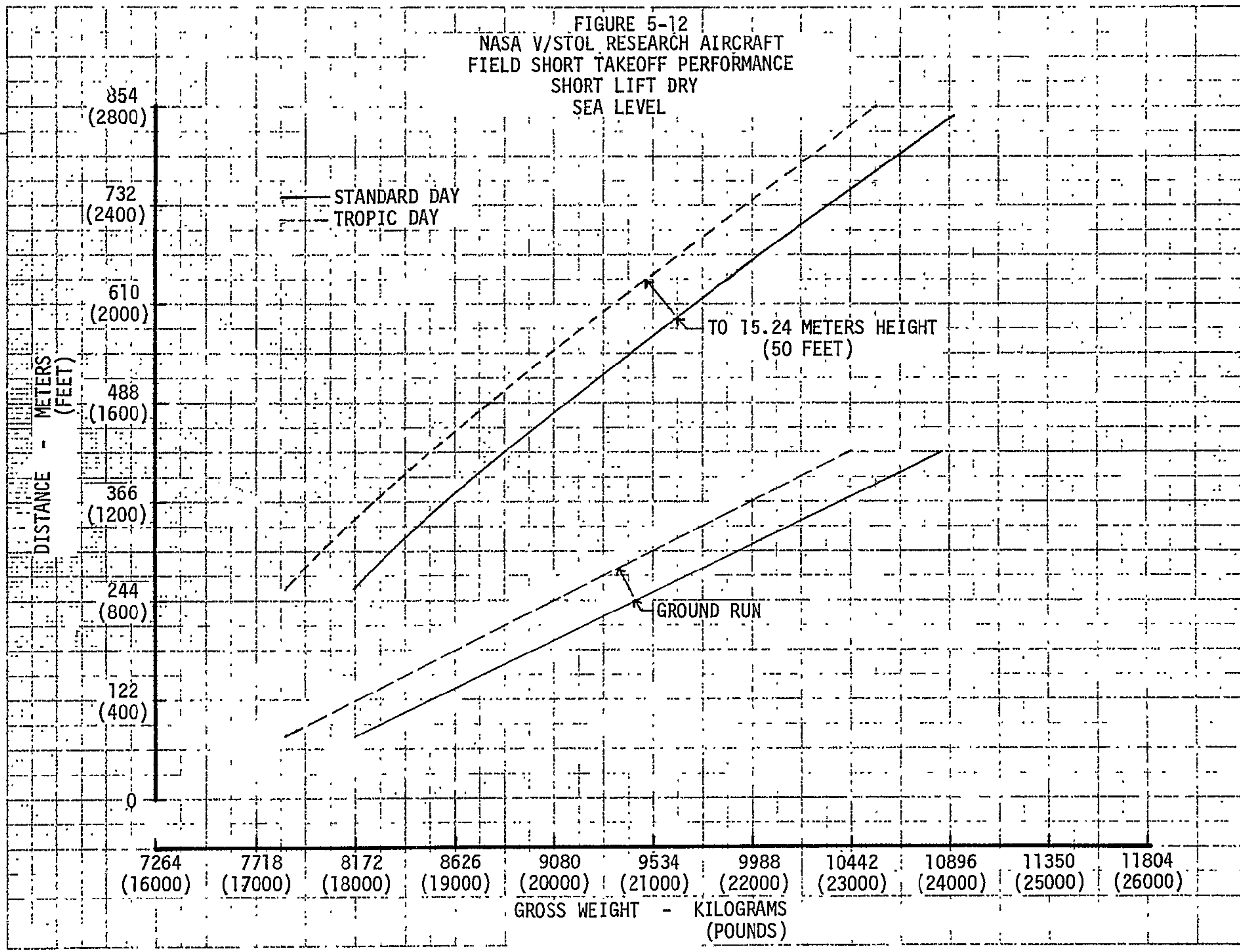


FIGURE 5-12
NASA V/STOL RESEARCH AIRCRAFT
FIELD SHORT TAKEOFF PERFORMANCE
SHORT LIFT DRY
SEA LEVEL
STANDARD DAY
TROPIC DAY
TO 15.24 METERS HEIGHT
(50 FEET)
GROUND RUN
DISTANCE - METERS
(FEET)
854
(2800)
732
(2400)
610
(2000)
488
(1600)
366
(1200)
244
(800)
122
(400)
0
7264
(16000)
7718
(17000)
8172
(18000)
8626
(19000)
9080
(20000)
9534
(21000)
9988
(22000)
10442
(23000)
10896
(24000)
11350
(25000)
11804
(26000)
GROSS WEIGHT - KILOGRAMS
(POUNDS)

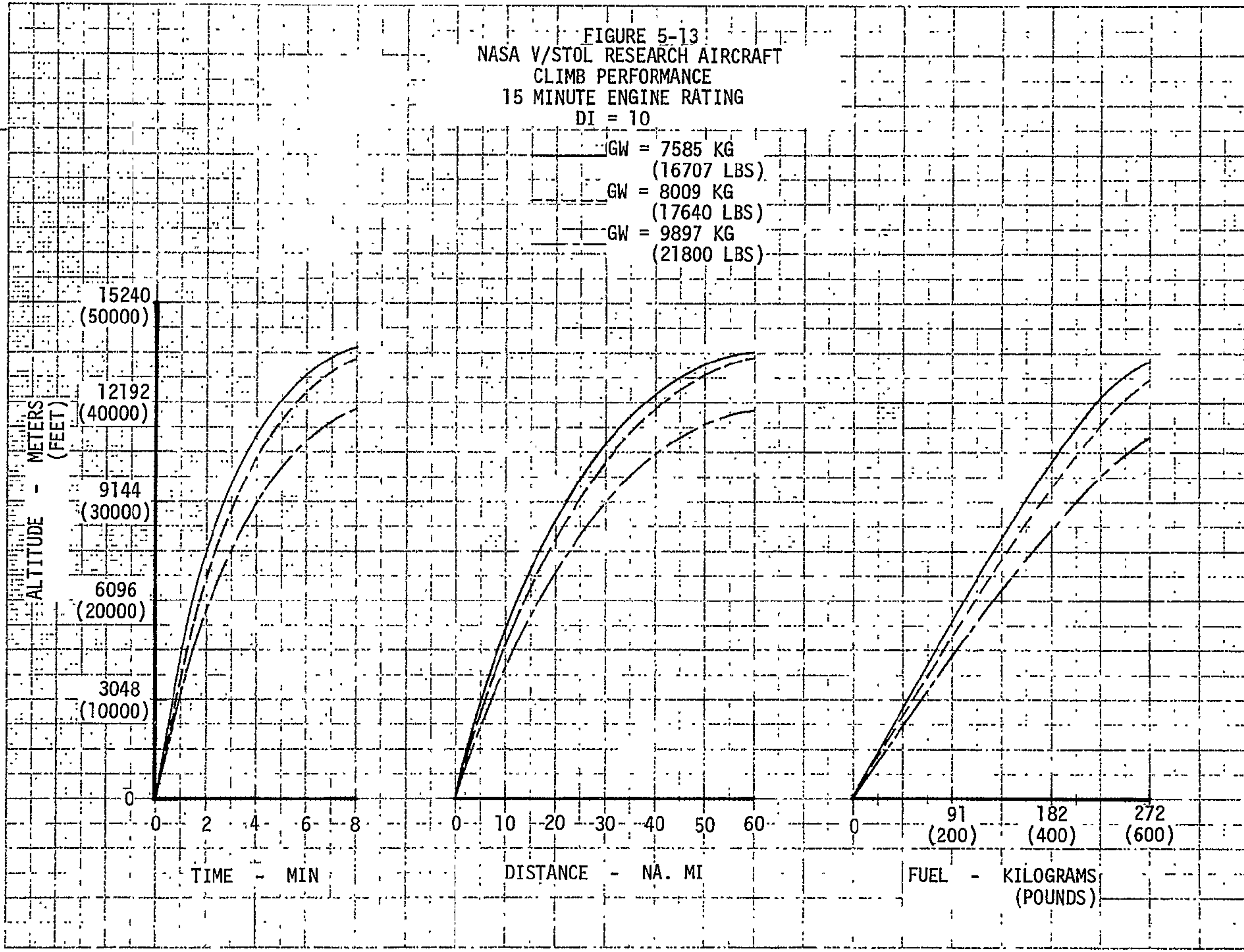
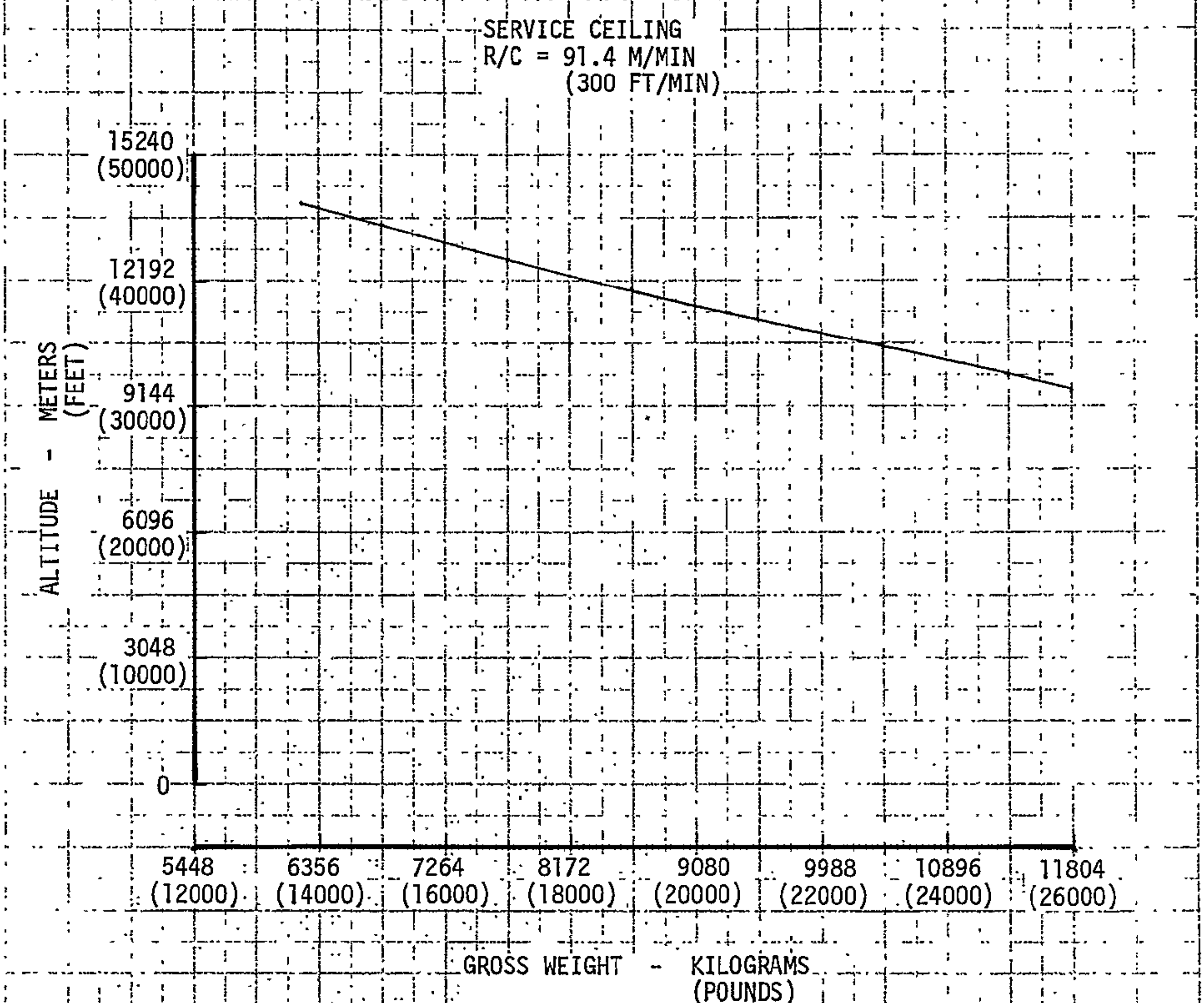


FIGURE 5-13
NASA V/STOL RESEARCH AIRCRAFT
CLIMB PERFORMANCE
15 MINUTE ENGINE RATING
DI = 10
GW = 7585 KG (16707 LBS)
GW = 8009 KG (17640 LBS)
GW = 9897 KG (21800 LBS)
ALTITUDE - METERS (FEET)
15240 (50000)
12192 (40000)
9144 (30000)
6096 (20000)
3048 (10000)
0
0 2 4 6 8
TIME - MIN
0 10 20 30 40 50 60
DISTANCE - NA. MI
0 91 (200) 182 (400) 272 (600)
FUEL - KILOGRAMS (POUNDS)

FIGURE 5-14
NASA V/STOL RESEARCH AIRCRAFT
CEILING PERFORMANCE
MAXIMUM CONTINUOUS RATING



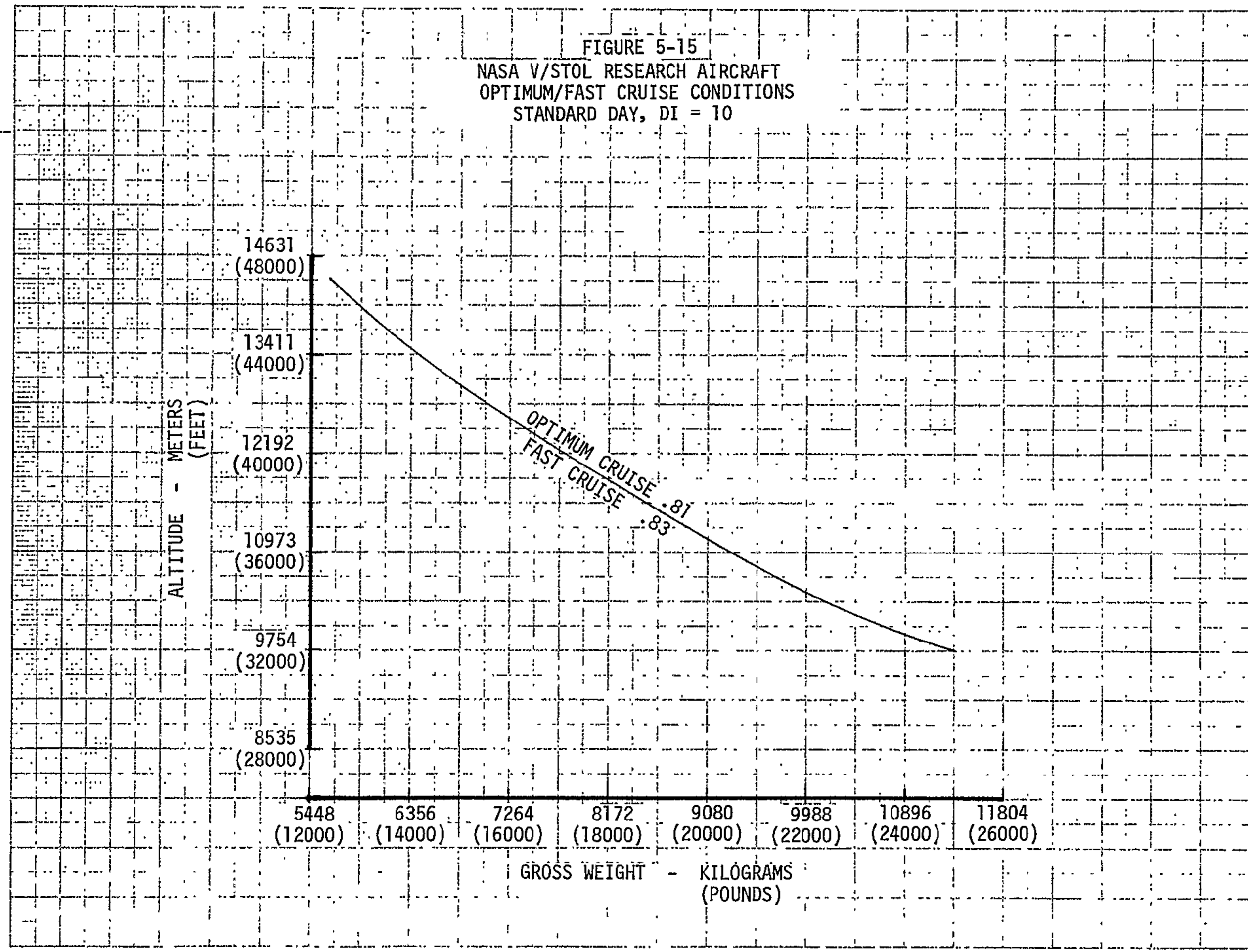


FIGURE 5-15
NASA V/STOL RESEARCH AIRCRAFT
OPTIMUM/FAST CRUISE CONDITIONS
STANDARD DAY, DI = 10
OPTIMUM CRUISE .81
FAST CRUISE .83
14631 (48000)
13411 (44000)
12192 (40000)
10973 (36000)
9754 (32000)
8535 (28000)
ALTITUDE - METERS (FEET)
5448 (12000)
6356 (14000)
7264 (16000)
8172 (18000)
9080 (20000)
9988 (22000)
10896 (24000)
11804 (26000)
GROSS WEIGHT - KILOGRAMS (POUNDS)

FIGURE 5-16
NASA V/STOL RESEARCH AIRCRAFT
CRUISE PERFORMANCE
SEA LEVEL
(FUNCTION OF GROSS WEIGHT)

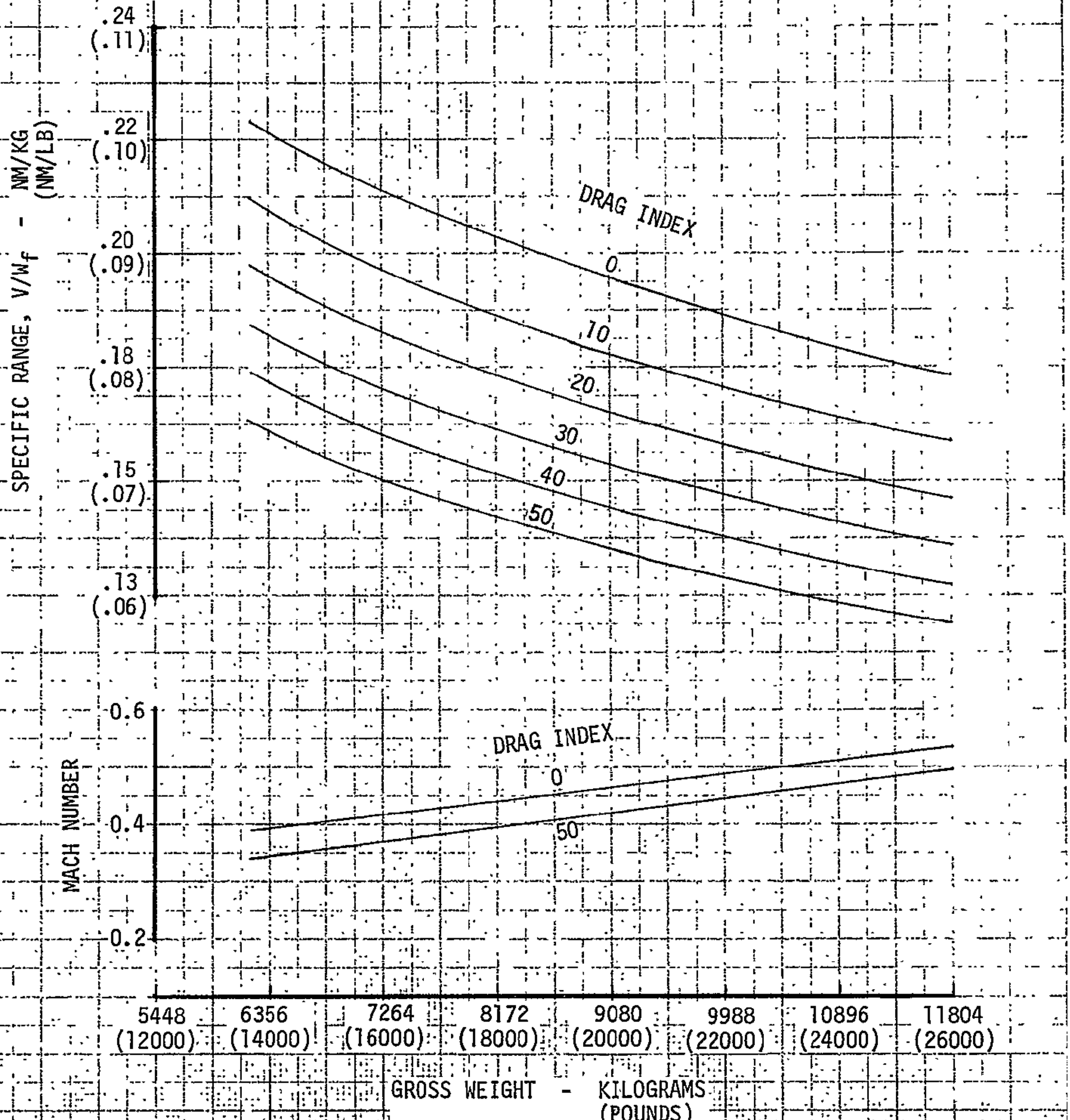
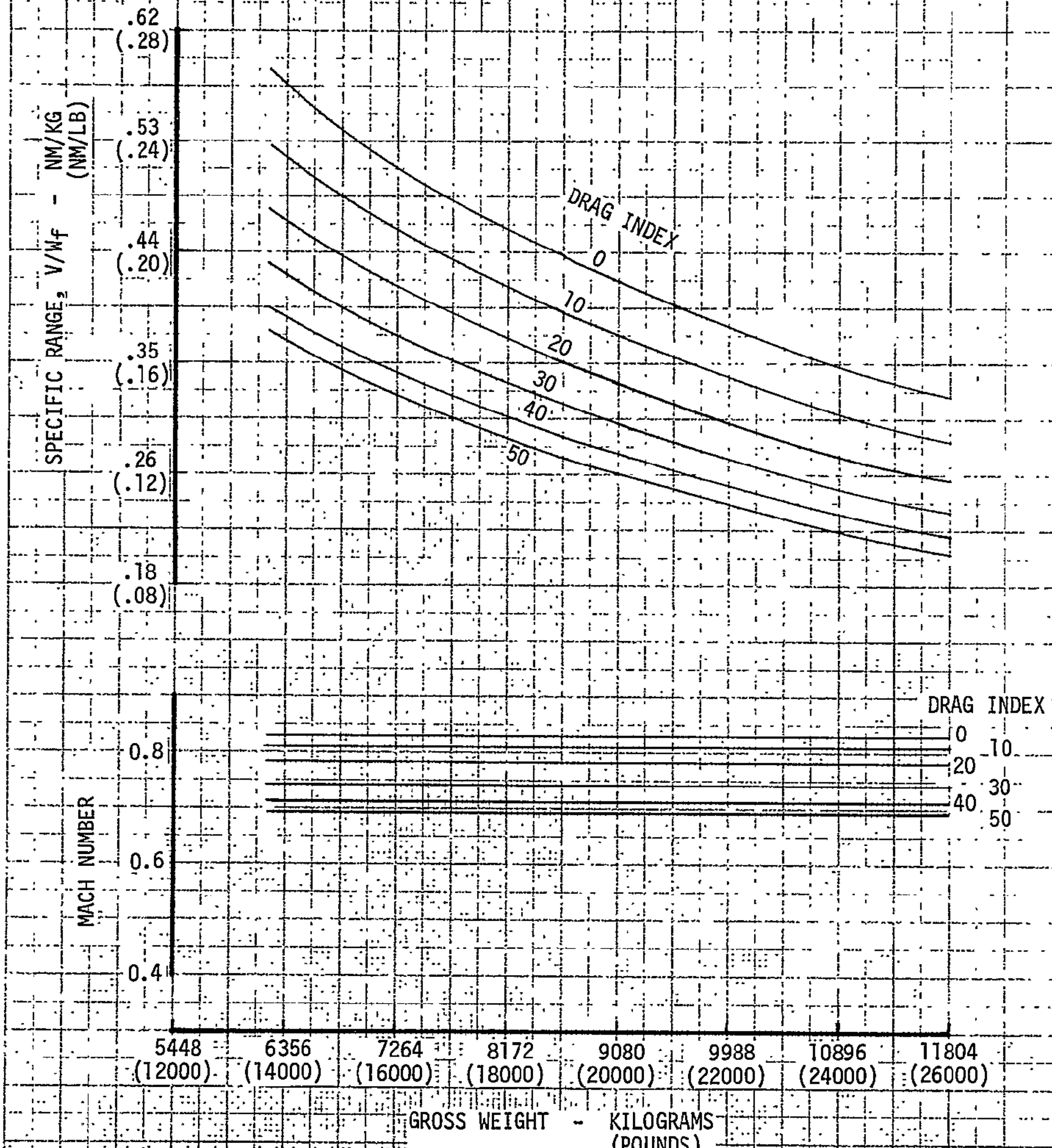


FIGURE 5-17
NASA V/STOL RESEARCH AIRCRAFT
CRUISE PERFORMANCE
OPTIMUM CRUISE ALTITUDE
(FUNCTION OF GROSS WEIGHT)



5.2.10 LOAD FACTOR - Figure 5-18 shows the load factor available as a function of Mach number and altitude. This is for buffet onset and is therefore not the maximum available.

5.2.11 TURN PERFORMANCE - Figure 5-19 shows the turn performance as a function of Mach number and altitude.

5.2.12 DESCENT - Maximum range descents at 230 KIAS, idle thrust, flaps up, and speedbrake retracted, are shown in Figure 5-20. For practical purposes these curves can be used for all aircraft gross weights since the variation with gross weight is rather small; e.g. 3% for a 25% weight increase. The aircraft gross weight used for the descent calculations is 15,000 lb.

5.2.13 VERTICAL LANDINGS - Vertical landing performance is shown in Figures 5-21 and 5-22. Figure 5-21 shows the gross weight effects. Figure 5-22 shows the altitude and type of day effects of maximum vertical landing weight.

The vertical landing performance at the higher ambients is based upon a 50°C JPT margin below the JPT limiter setting. In the SLD rating, a JPT above 660° at 90 KIAS in level transition will result in an excessive JPT or insufficient thrust at lower airspeeds. If the JPT margin is in excess of 50°C, the deceleration is continued to the hover. If the margin is less than 50°C the aircraft is accelerated to wing-borne flight and flown in this mode until a significant amount of fuel has been expended before the VL is attempted.

5.2.14 STALL SPEED - Figure 5-23 presents the stall speed as a function of gross weight for zero and 50° flap settings for the power off condition; nozzle angle is 0°.

5.2.15 CONVENTIONAL LANDING - Conventional landing performance is shown as a function of gross weight in Figure 5-24. This figure shows approach speed, touchdown speed, ground roll distance, and distance from a 15 M (50 ft) height for landings with and without nozzle braking. Nozzle braking reduces landing ground roll significantly. It is noted that nozzle braking is terminated when the airspeed decreases to 65 KIAS in order to preclude thermal reingestion in the inlets.

5.2.16 V/STOL RESEARCH MISSION ANALYSIS - The typical mission flight test pattern is shown in Figure 5-25. The takeoff, either VTO or STO, is performed, followed by a climb to 305 M (1000 ft) altitude. A race track pattern is then flown at 180 KIAS with gear and flaps down and nozzles aft. This pattern allows the 5 minutes operation with nozzles aft as required after each period of nozzles-deflected flight. A decelerating transition is then initiated followed by either a standard NATOPS deceleration to a VL or a slow approach to a SL. These are illustrated in Figure 5-26 and 5-27.

Figure 5-28 shows a detail of the VTOL research mission profile which includes fuel usage, distance, time and altitude. Two circuits are possible in this mission since the 150 kg (330 lbs) of fuel for starting are used only once and the 48 kg (105 lbs) for taxi and reposition are used only once. This leaves the aircraft with 272 kg (600 lbs) of fuel at the end of the second complete circuit.

FIGURE 5-18
NASA V/STOL RESEARCH AIRCRAFT
INSTANTANEOUS LOAD FACTOR
BUFFET ONSET C_L
$\theta_j = 0°$; DI = 0.; GW = 7584 KG (16707 LBS)

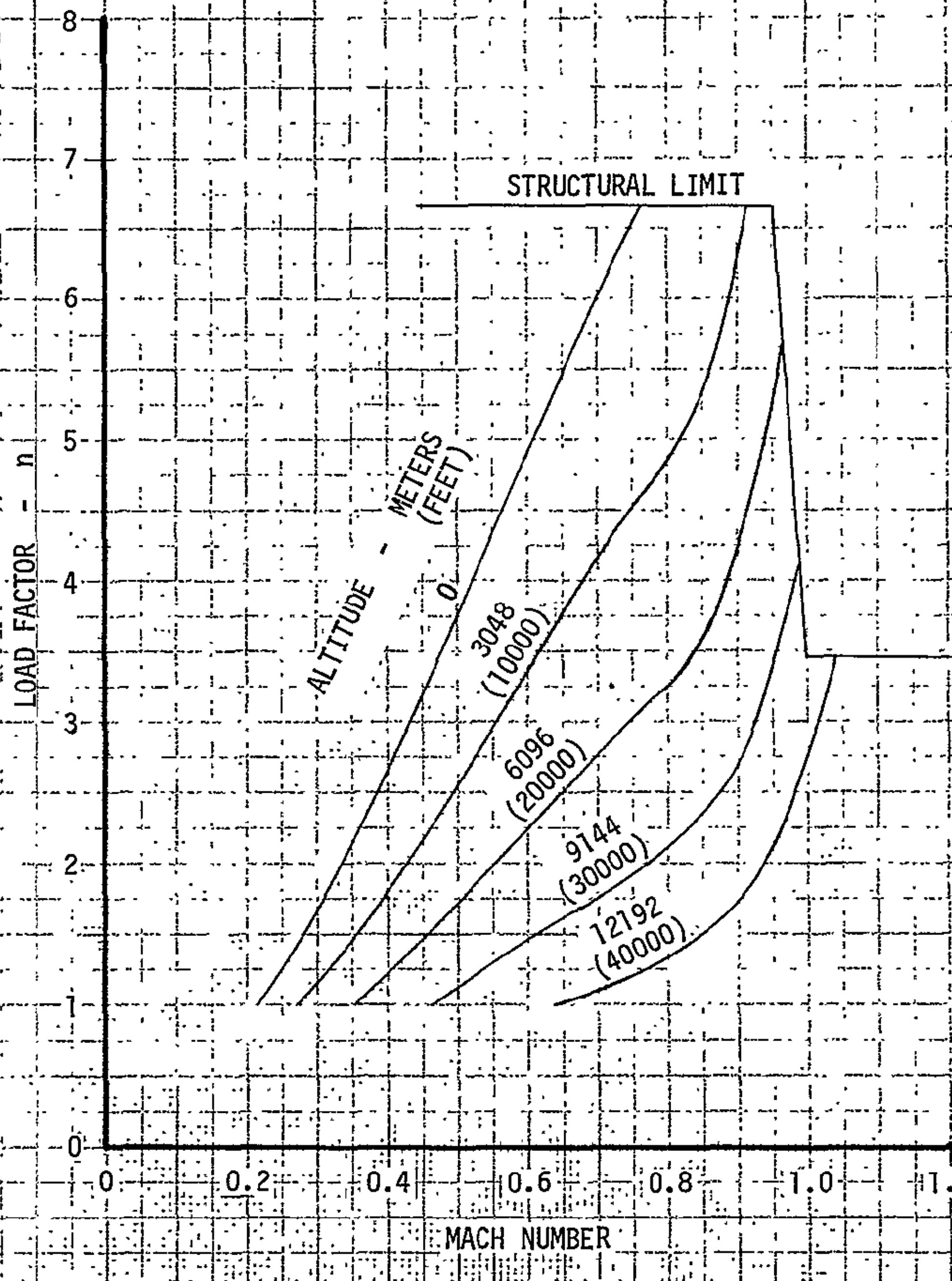
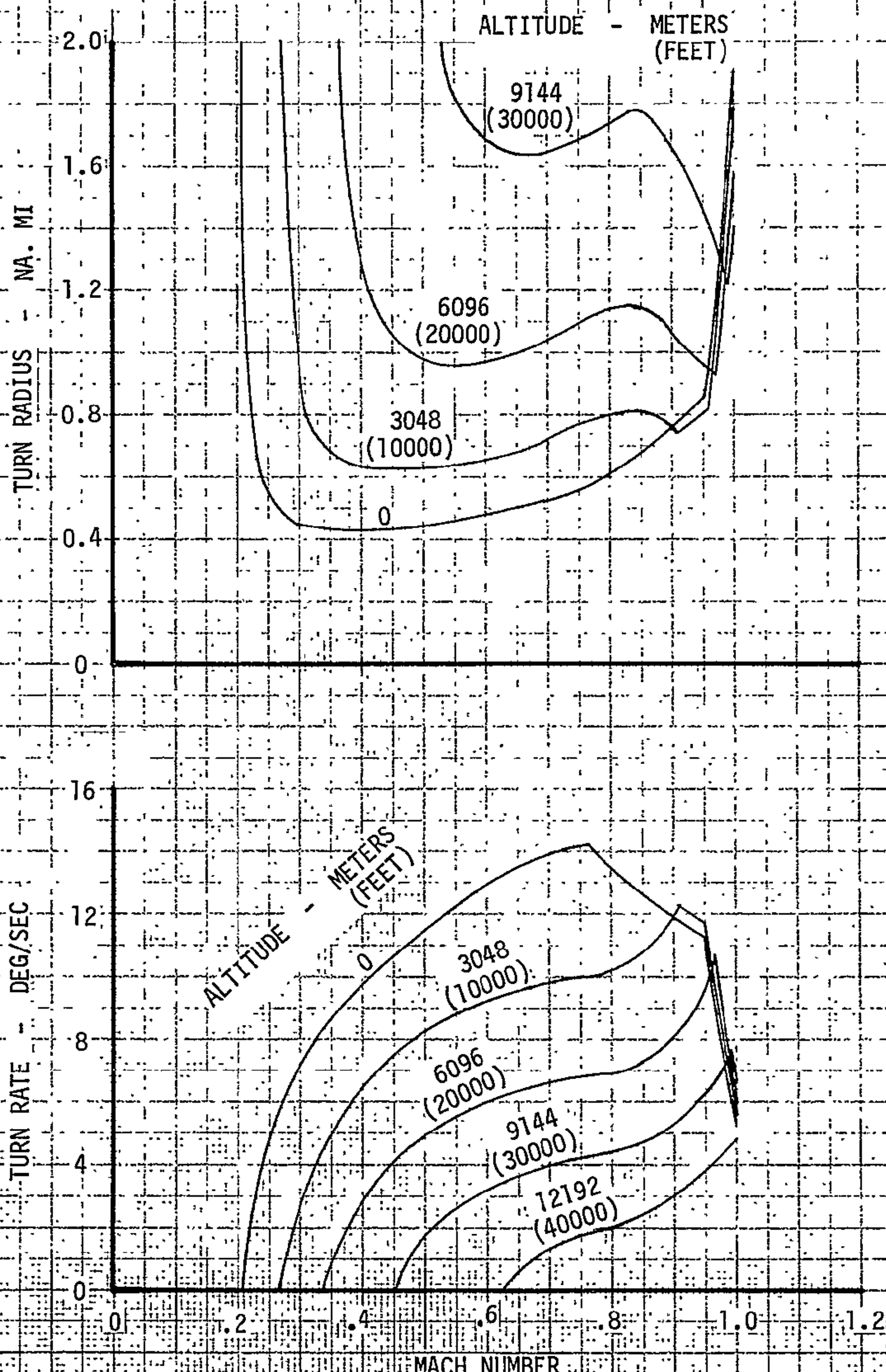


FIGURE 5-19
NASA V/STOL RESEARCH AIRCRAFT
BUFFET ONSET TURN PERFORMANCE
DI = 0.; θ_j = 0°., GW = 7584 KG
(16707 LBS)



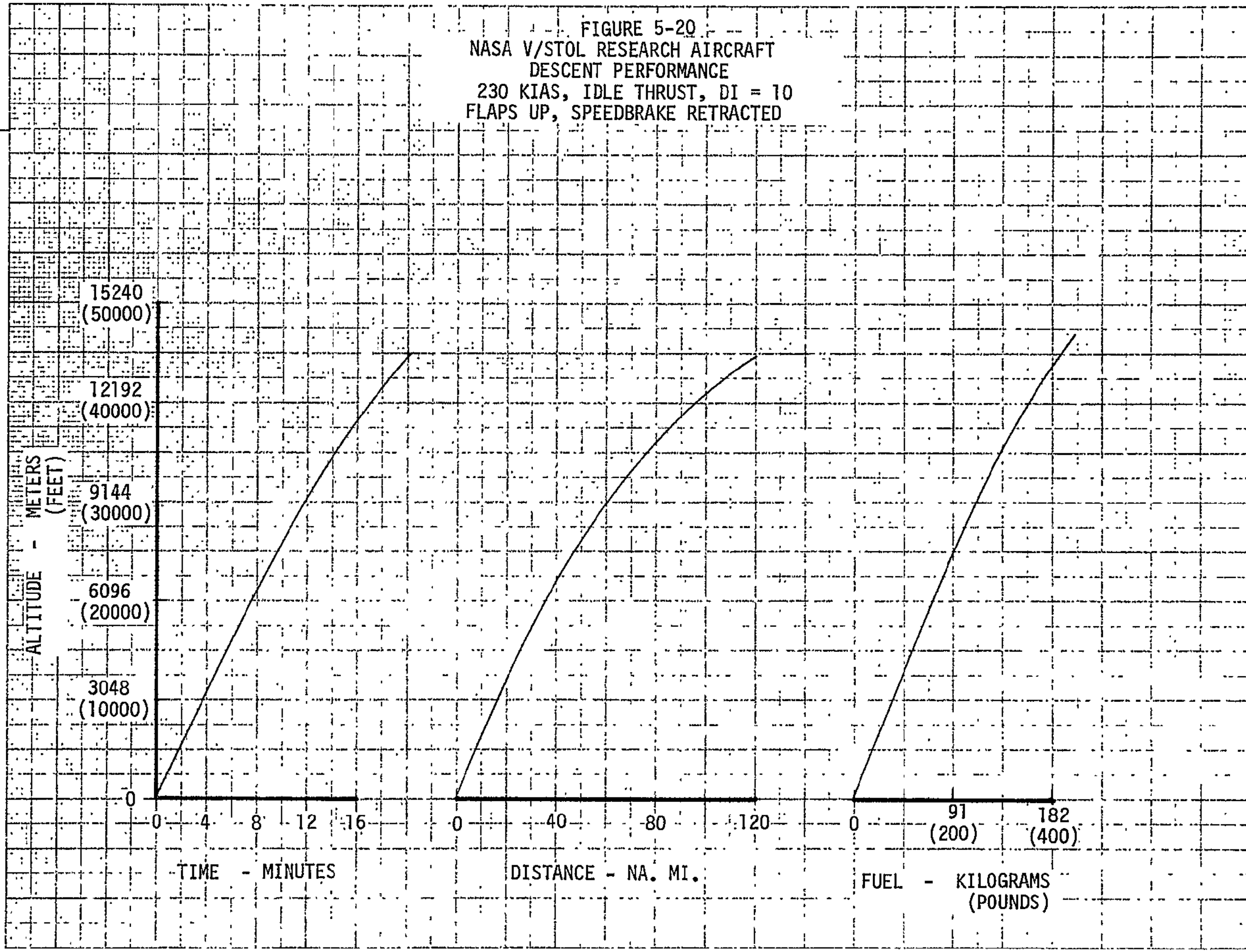


FIGURE 5-20
NASA V/STOL RESEARCH AIRCRAFT
DESCENT PERFORMANCE
230 KIAS, IDLE THRUST, DI = 10
FLAPS UP, SPEEDBRAKE RETRACTED

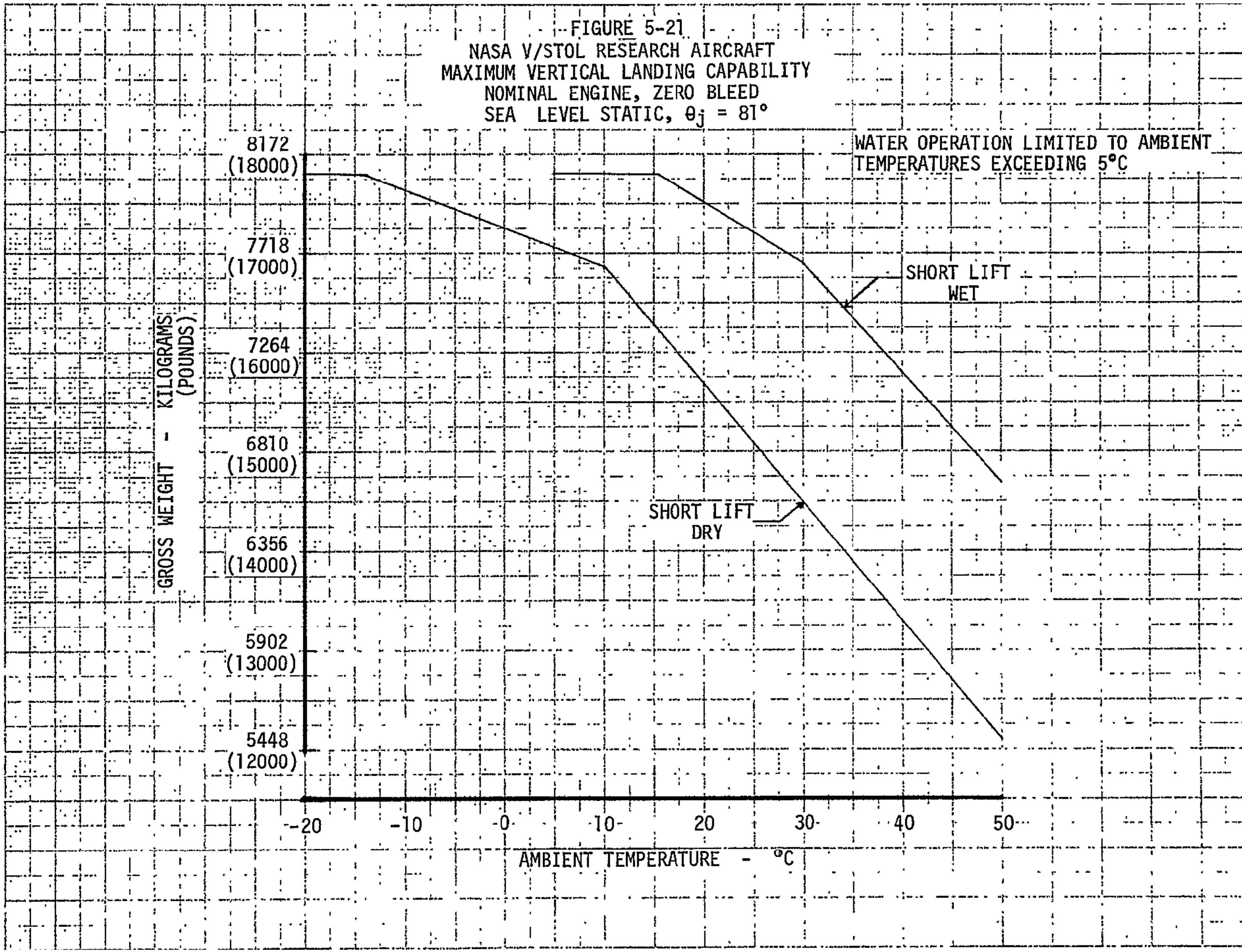
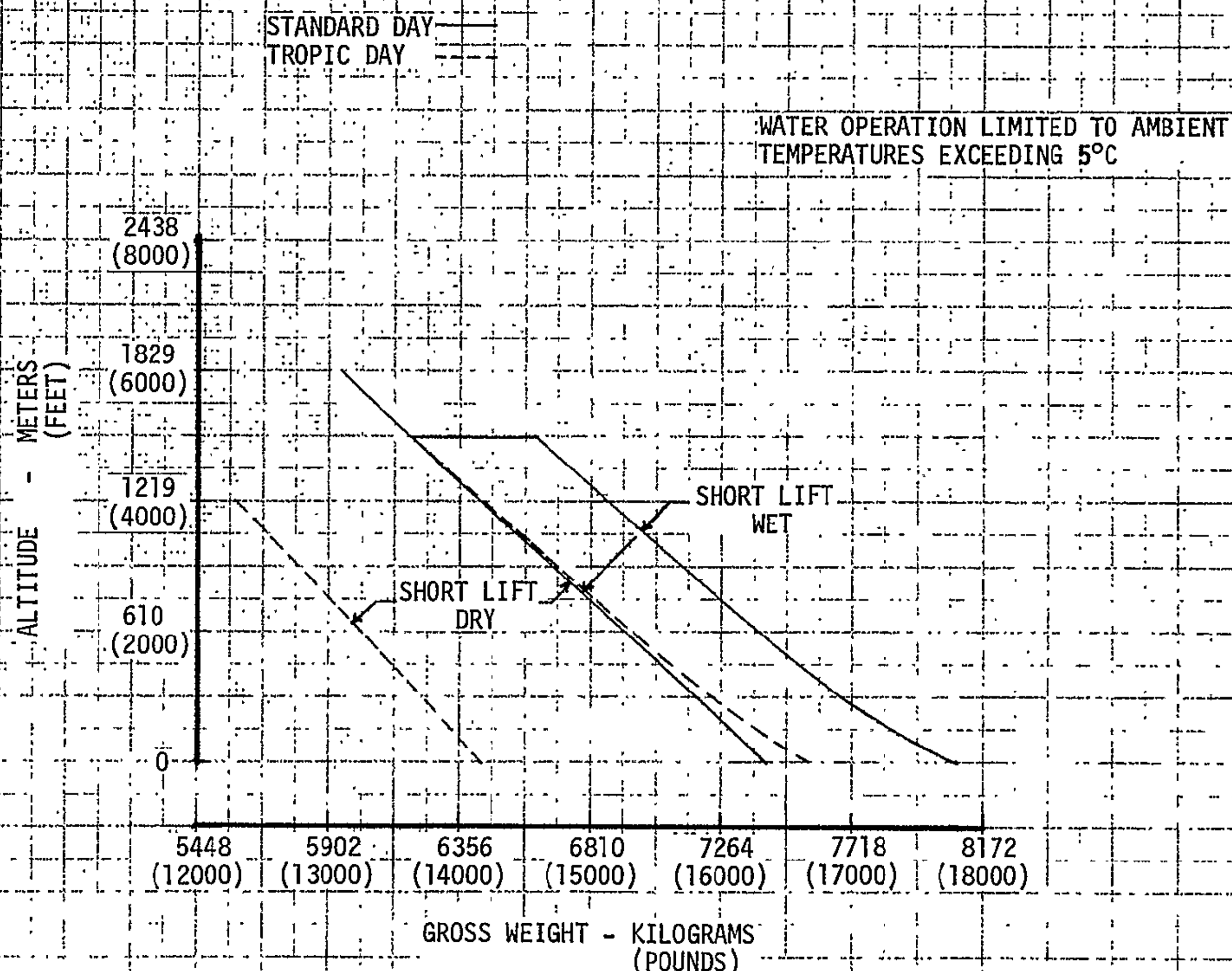


FIGURE 5-21
NASA V/STOL RESEARCH AIRCRAFT
MAXIMUM VERTICAL LANDING CAPABILITY
NOMINAL ENGINE, ZERO BLEED
SEA LEVEL STATIC, θj = 81°
WATER OPERATION LIMITED TO AMBIENT TEMPERATURES EXCEEDING 5°C
SHORT LIFT WET
SHORT LIFT DRY
GROSS WEIGHT - KILOGRAMS (POUNDS)
8172 (18000)
7718 (17000)
7264 (16000)
6810 (15000)
6356 (14000)
5902 (13000)
5448 (12000)
-20
-10
0
10
20
30
40
50
AMBIENT TEMPERATURE - °C

FIGURE 5-22
NASA V/STOL RESEARCH AIRCRAFT
VERTICAL LANDING ALTITUDE PERFORMANCE
ZERO BLEED



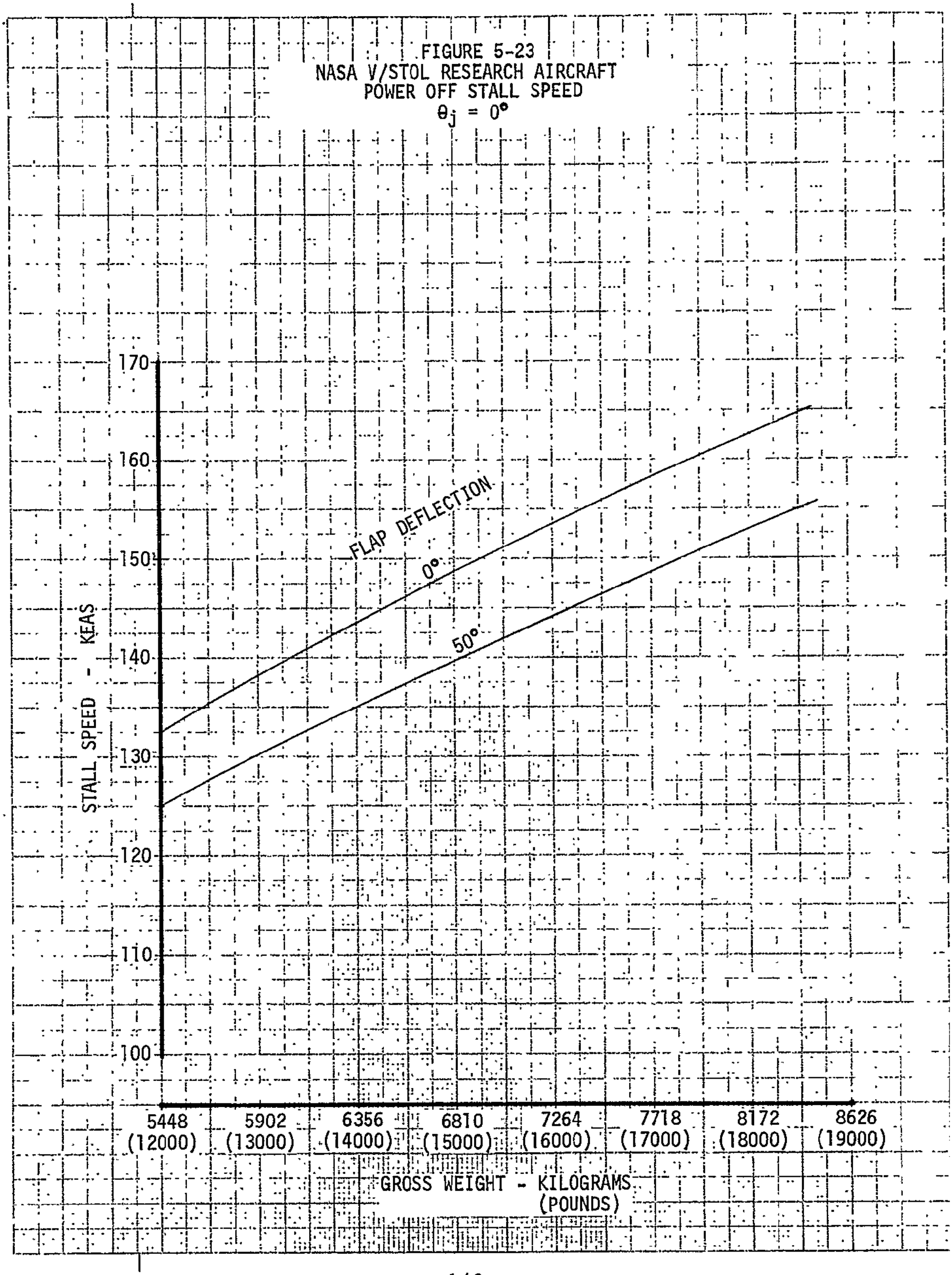


FIGURE 5-23
NASA V/STOL RESEARCH AIRCRAFT
POWER OFF STALL SPEED
$\theta_j = 0°$
STALL SPEED - KEAS
170
160
150
140
130
120
110
100
FLAP DEFLECTION
0°
50°
5448 (12000)
5902 (13000)
6356 (14000)
6810 (15000)
7264 (16000)
7718 (17000)
8172 (18000)
8626 (19000)
GROSS WEIGHT - KILOGRAMS
(POUNDS)

FIGURE 5-24
NASA V/STOL RESEARCH AIRCRAFT
CONVENTIONAL LANDING PERFORMANCE
SEA LEVEL STANDARD DAY

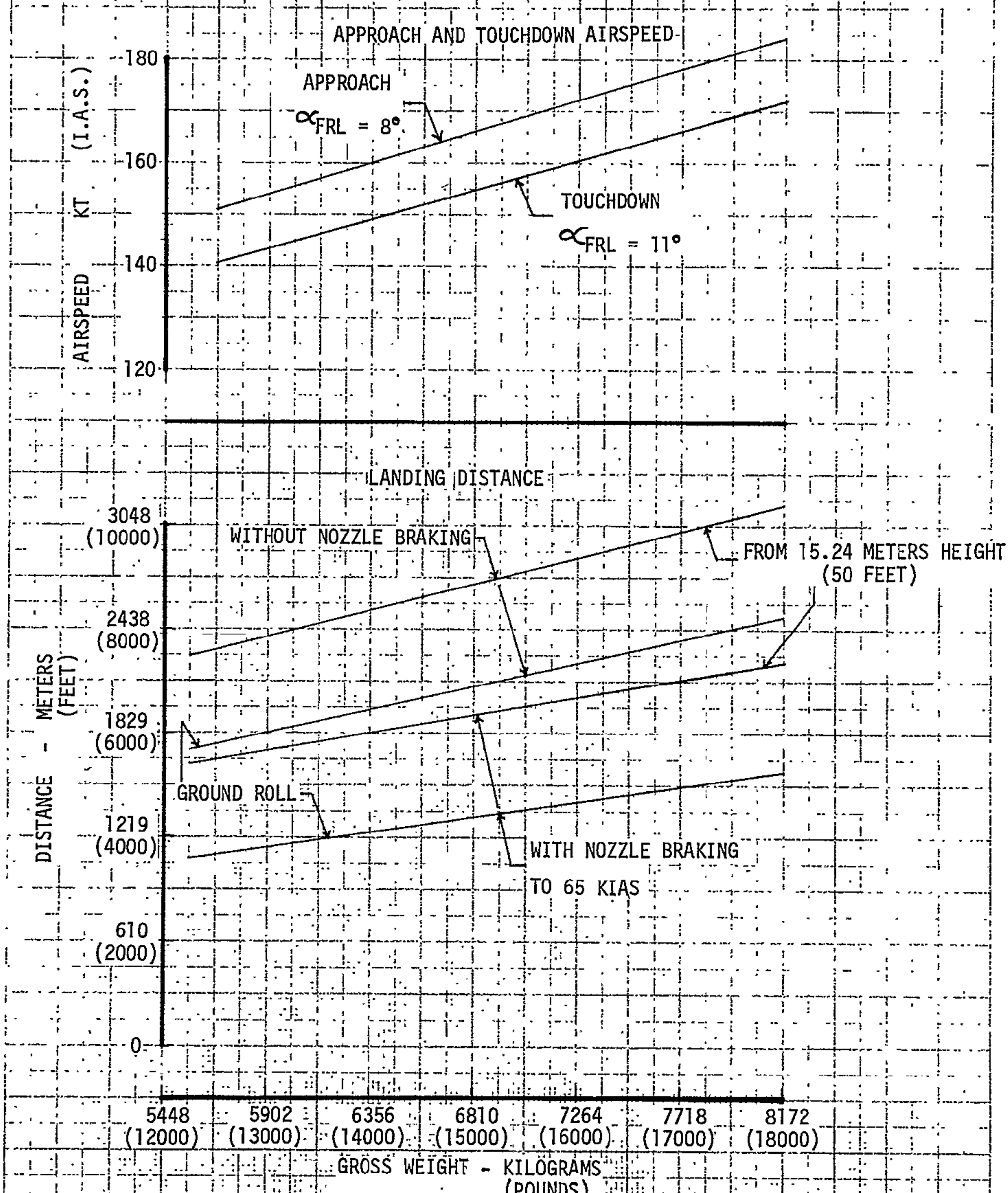


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FIGURE 5-25
NASA V/STOL RESEARCH AIRCRAFT
TYPICAL VTOL AND STOL FLIGHT TEST PATTEPNS

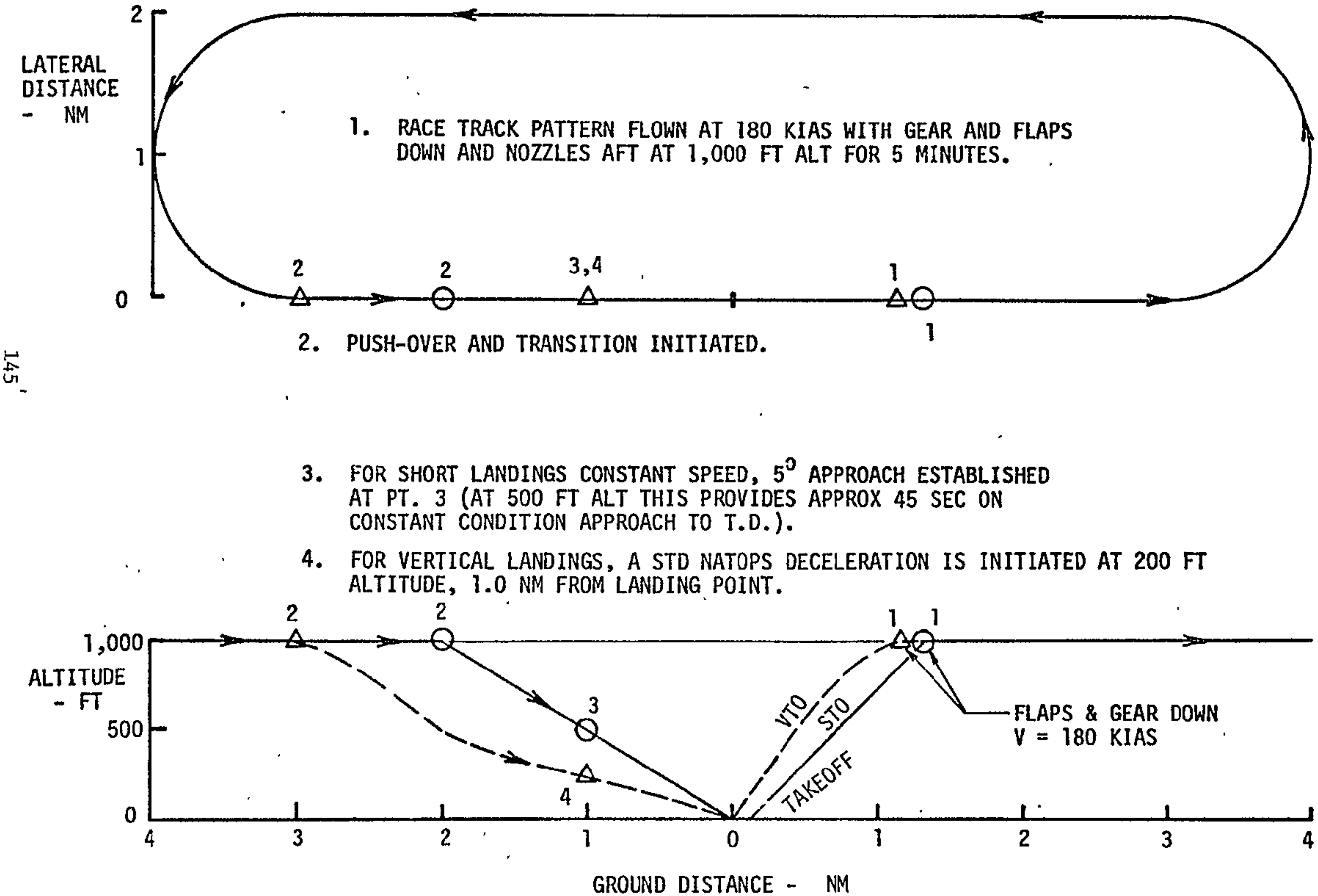

FIGURE 5-26

VERTICAL LANDING

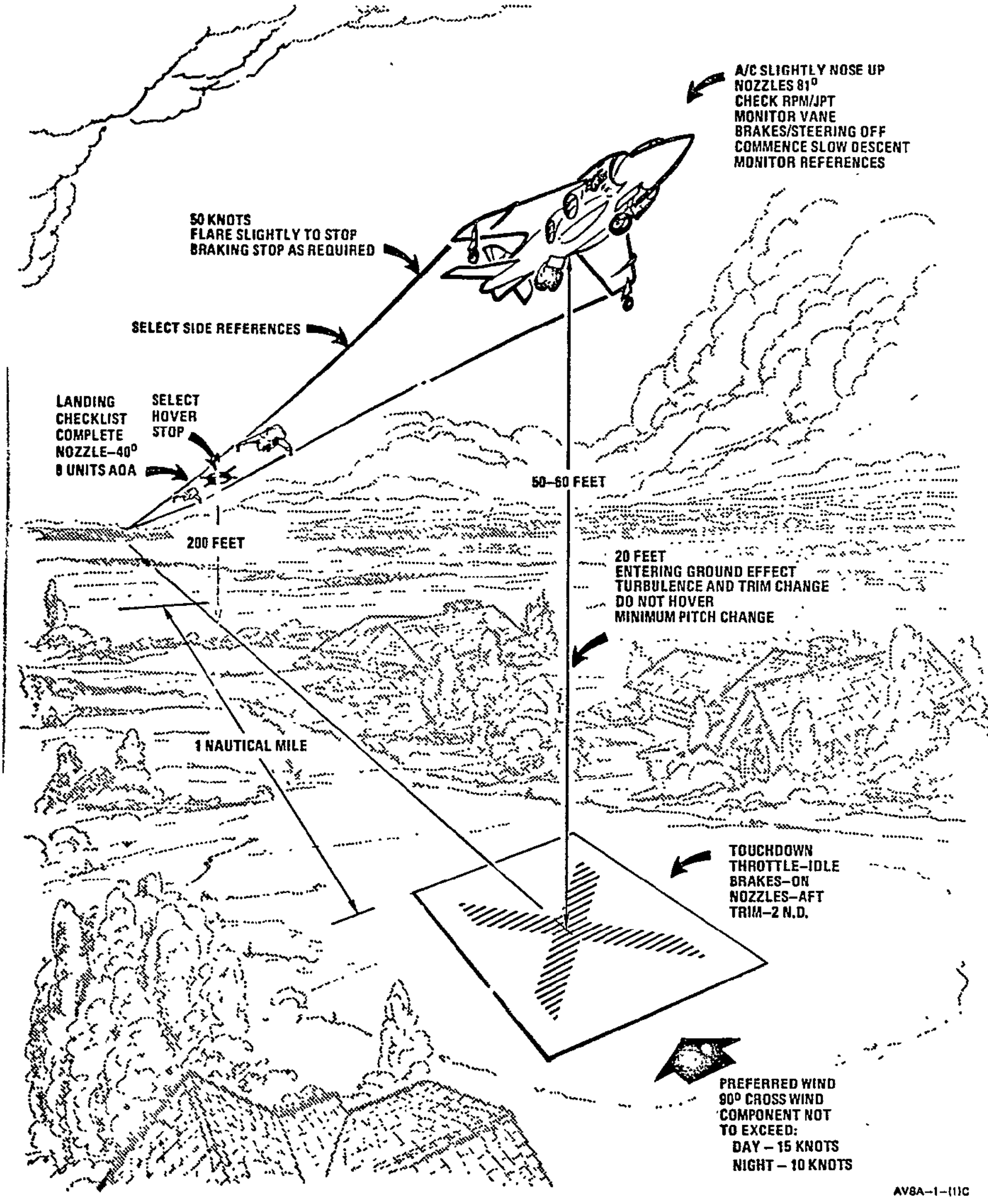

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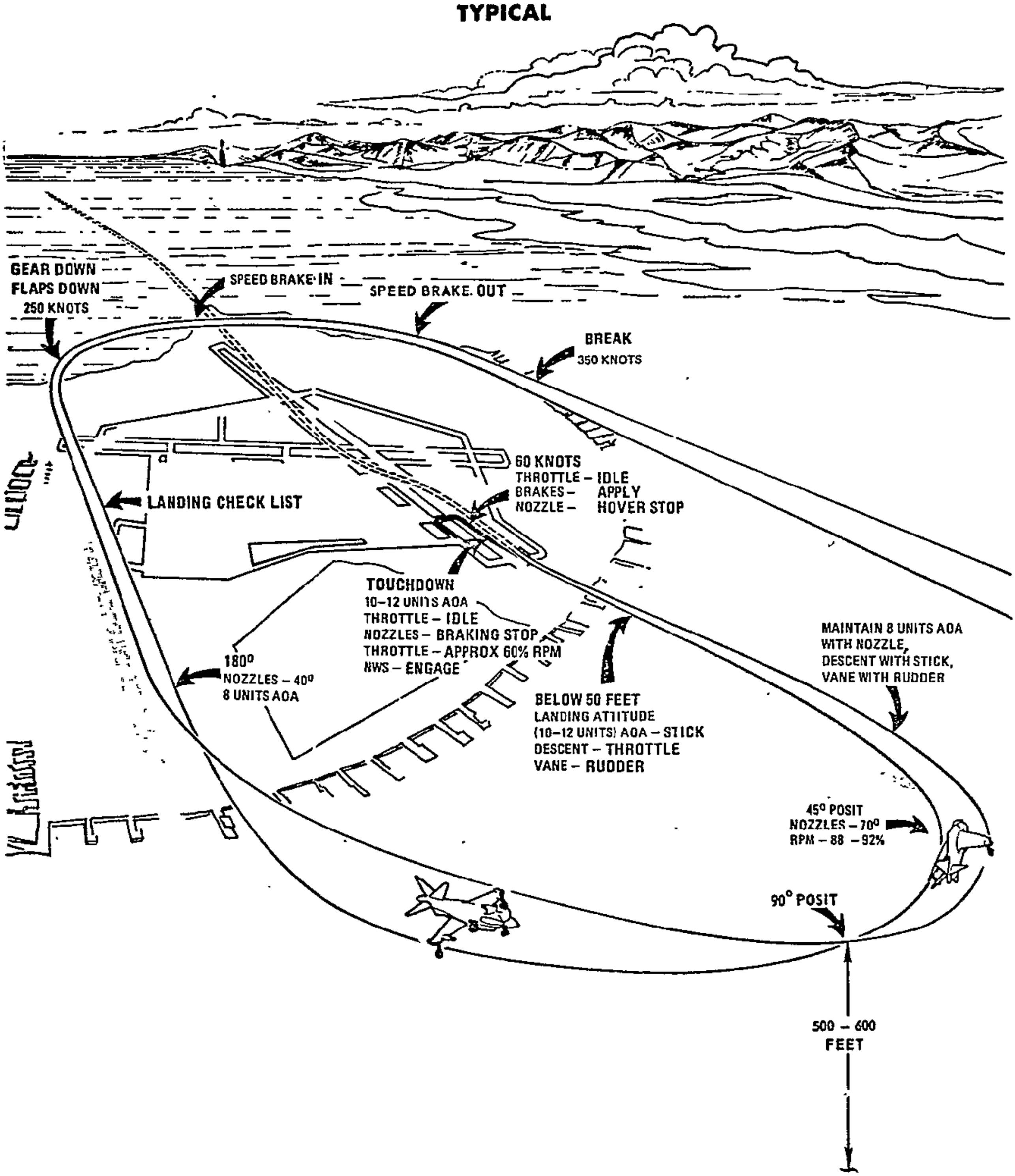
FIGURE 5-27

SLOW LANDING

TYPICAL



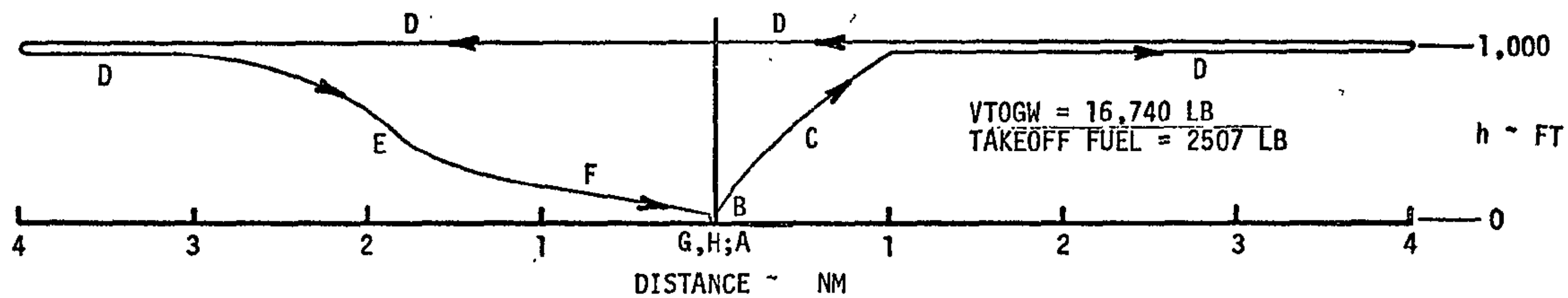
AV8A–1–(4)D

FIGURE 5-28

NASA V/STOL RESEARCH AIRCRAFT

TYPICAL RESEARCH MISSION PROFILE

VTOL



OPERATION	FUEL LB. Δ	CUM	Δ DIST. N.M.	TIME MIN Δ	CUM	ALTITUDE FT
A. START, CHECK ENGINES & TAX	330	330	-	5.0	5.0	0
B. VTO @ SLD ENGINE RATING	110	440	-	0.5	5.5	0→50
C. CLIMB & ACCELERATE TO 1,000' @ 180KCAS	110	550	1	0.5	6.0	50→1,000
D. FLY PATTERN AT 1,000' @ 180 KEAS; $\theta_J = 0°$	372	922	15	5.0	11.0	1,000
E. REDUCED PWR. PUSHOVER AND DESCENT TO INITIAL POINT AT 200' ALT, 1.0 N.M. FROM VL POINT	75	998	2	1.0	12.0	1,000
F. DECELERATING TRANSITION TO VL MODE	75	1073	1	0.50	12.5	200→60
G. VERTICAL LANDING FROM 60 FT.	50	1123	-	0.25	12.75	60→0
H. TAXI & REPOSITION FOR NEXT CIRCUIT	105	1228	-	5.0	17.75	0

Figure 5-29 shows a detail of the STOL research mission profile with the same conditions as in the VTOL mission profile. Using STOL allows 5 circuits. The fuel allowances listed represent the average gross weight associated with the 5 circuit capability.

A performance summary is shown in Figure 5-30 for each of the missions with each of the proposed control systems.

These circuit capabilities are for the SLD throttle setting on a standard day at sea level, and consequently represent a nominal capability. Due to the large effects of ambient temperature, field altitude and use of water, the V/STOL performance charts must be checked for specific mission planning.

Figure 5-31 shows the effect of fuel loading at takeoff on the number of cycles for the two different takeoff and landing operations.

Figure 5-32 shows the effect of fuel loading at takeoff for the total time including takeoff, circuit, landing, and repositioning the aircraft.

5.2.17 FERRY MISSIONS - Figures 5-33 and 5-34 are examples of the ferry mission capability. Figure 5-33 shows a typical ferry from Moffet Field to the Crows Landing auxiliary test site. It includes a weight, distance, time and altitude breakdown for this 44 nautical miles mission. Figure 5-34 shows the capability in the ferry mission with this aircraft.

5.3 WEIGHT AND BALANCE

Weight and balance data were computed for the basic and optional aircraft modifications. Data are given in units of pounds and inches to be consistent with performance data contained in NATOPS Flight Manual NAVAIR 01-AV8A-1. The computations were based on the assumption that the Harrier G-VTOL aircraft is brought up to TAV-8A standards before delivery to MCAIR. The update modification will decrease the present G-VTOL weight by 391 pounds.

Table 5-1 presents the weight and balance data for the equipment added and deleted to implement the basic modifications. Tables 5-2 and 5-3 give the weight and balance changes to the data of Table 5-1 if the duplex or triplex flight control system is implemented instead of the simplex system. Figures 5-4 through 5-8 give weight and balance data for the other options.

Table 5-9 gives a total aircraft weight summary for the simplex, duplex and triplex flight control systems installed in the Harrier G-VTOL and a production TAV-8A. Figure 5-35 shows the center of gravity envelope for the aircraft equipped with the simplex system and the aircraft equipped with a triplex system. The envelope for the aircraft equipped with the duplex system falls between the simplex and triplex envelopes. Also shown is the point on the production TAV-8A envelope which is for a crew of two and full internal fuel.

5.4 AIRCRAFT SUPPORT

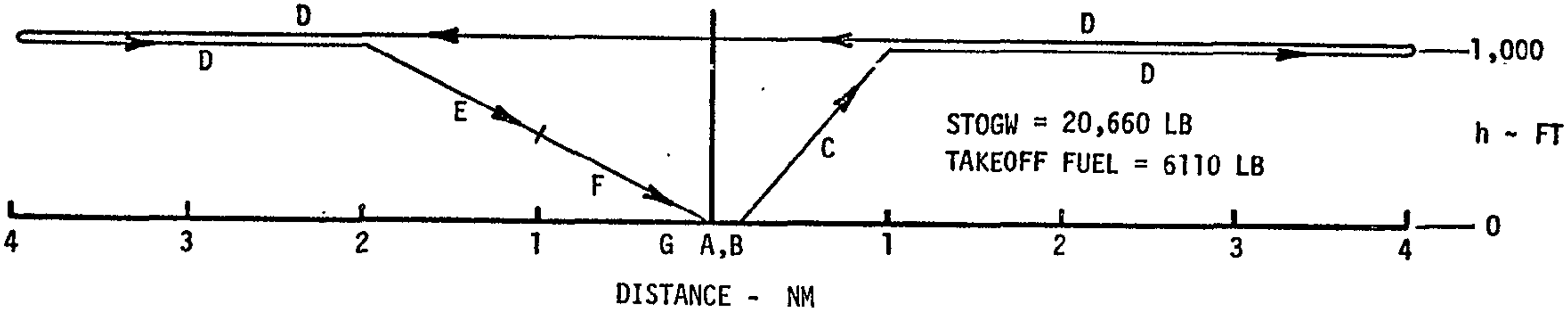
MCAIR recommends limited support of the NASA Research Airplane modification and flight test program.

FIGURE 5-29

NASA V/STOL RESEARCH AIRCRAFT

TYPICAL RESEARCH MISSION PROFILE

STOL



OPERATION	FUEL LB.		Δ DIST.	TIME MIN		ALTITUDE
	Δ	CUM	N.M.	Δ	CUM	FT
A. START, CHECK ENGINES & TAXI	330	330	-	5.0	5.0	0
B. STO AT SLD ENGINE RATING	110	440	-	0.5	5.5	0
C. CLIMB & ACCELERATE TO 1,000 FT. @ 180 KCAS	110	550	1	0.5	6.0	0→1,000
D. FLY PATTERN AT 1,000' @ 180 KEAS: $\theta_J = 0°$	440	990	15	5.0	11.0	1,000
E. PUSHOVER, ESTABLISH APPROACH NOZZLE SETTING AND DECELERATE TO APPROACH SPEED	150	1140	1	0.75	11.75	1,000→500
F. CONSTANT SPEED APPROACH TO SHORT LANDING V ≈ 110 KIAS	150	1290	1	0.75	12.5	500→0
G. LDG. RUNOUT, TAXI & REPOSITION FOR NEXT CIRCUIT	105	1395	-	5.0	17.5	0

FIGURE 5-30

NASA V/STOL RESEARCH AIRCRAFT

PERFORMANCE SUMMARY

		CONTROL SYSTEM		
		SIMPLEX	DUPLEX	TRIPLEX
A. CONFIGURATION DATA				
1. OWE INCLUDING FLIGHT TEST INSTRUMENTATION	LB	14,233	14,443	14,491
2. GUN POD CARRIAGE		NO	NO	YES
B. VTOL CAPABILITY				
3. VTO G.W. (STD DAY)*	LB	16,740/17,640	16,740/17,640	16,740/17,640
4. TAKEOFF FUEL (INTERNAL)	LB	2,507/2,907	2,297/2,697	2,249/2,649
5. WATER		0/500	0/500	0/500
6. NUMBER OF TEST CIRCUITS AVAILABLE		2/2	2/2	2/2
7. TOTAL CIRCUIT TIME AVAILABLE	MIN	22/28	19/24	19/24
C. STOL CAPABILITY				
8. STO G.W. (STD DAY, 1000' G.R.)*	LB	20,660/21,590**	20,660/21,800**	20,660/21,848**
9. TAKEOFF FUEL	LB	6063/6793	5853/6793	5805/6793
10. WATER		0/200	0/200	0/200
11. NO. OF TEST CIRCUITS AVAILABLE		5/6	5/6	5/6
12. TOTAL CIRCUIT TIME AVAILABLE	MIN	61/70	59/70	58/70

*NOTE: SHORT LIFT DRY/SHORT LIFT WET OPERATION ASSUMES NOMINAL ENGINE PERFORMANCE.

**NOTE: FUEL LIMITED.

FIGURE 5-31

NASA V/STOL RESEARCH MISSION

NOMINAL ENGINE PERFORMANCE

(5) MINUTE CYCLES @ 180 KIAS

LANDING WITH 600 LB FUEL REMAINING

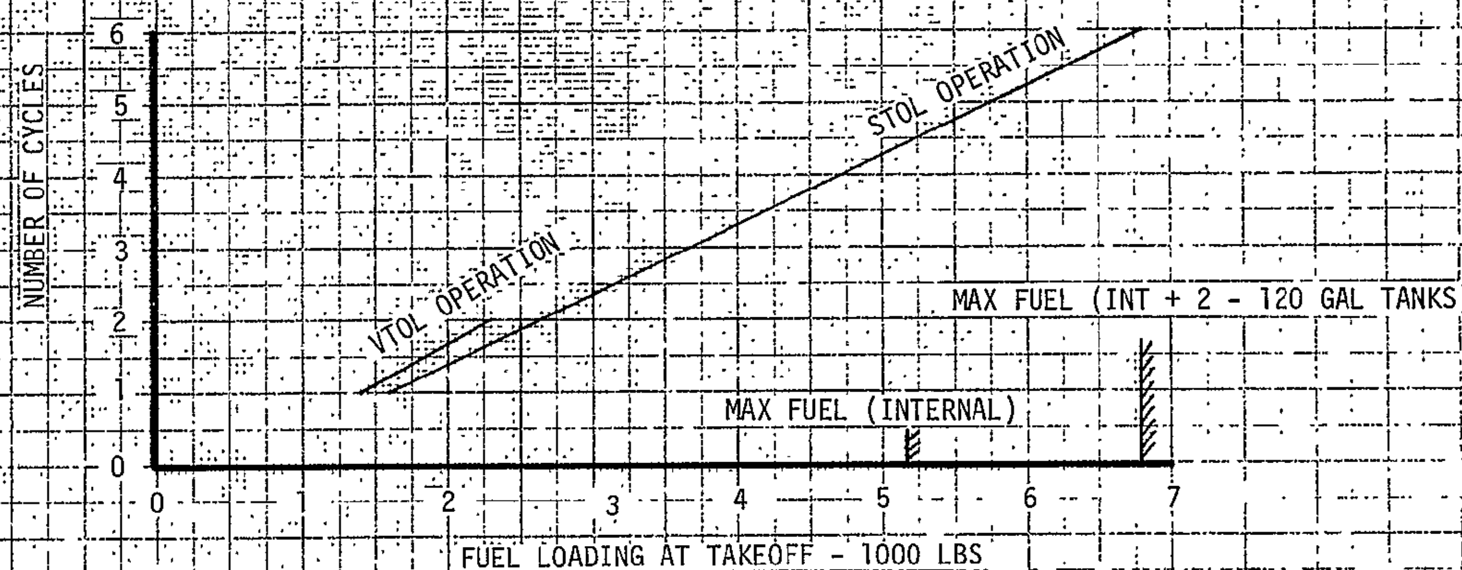


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FIGURE 5-32
NASA V/STOL RESEARCH AIRCRAFT

RESEARCH MISSION TOTAL TIME

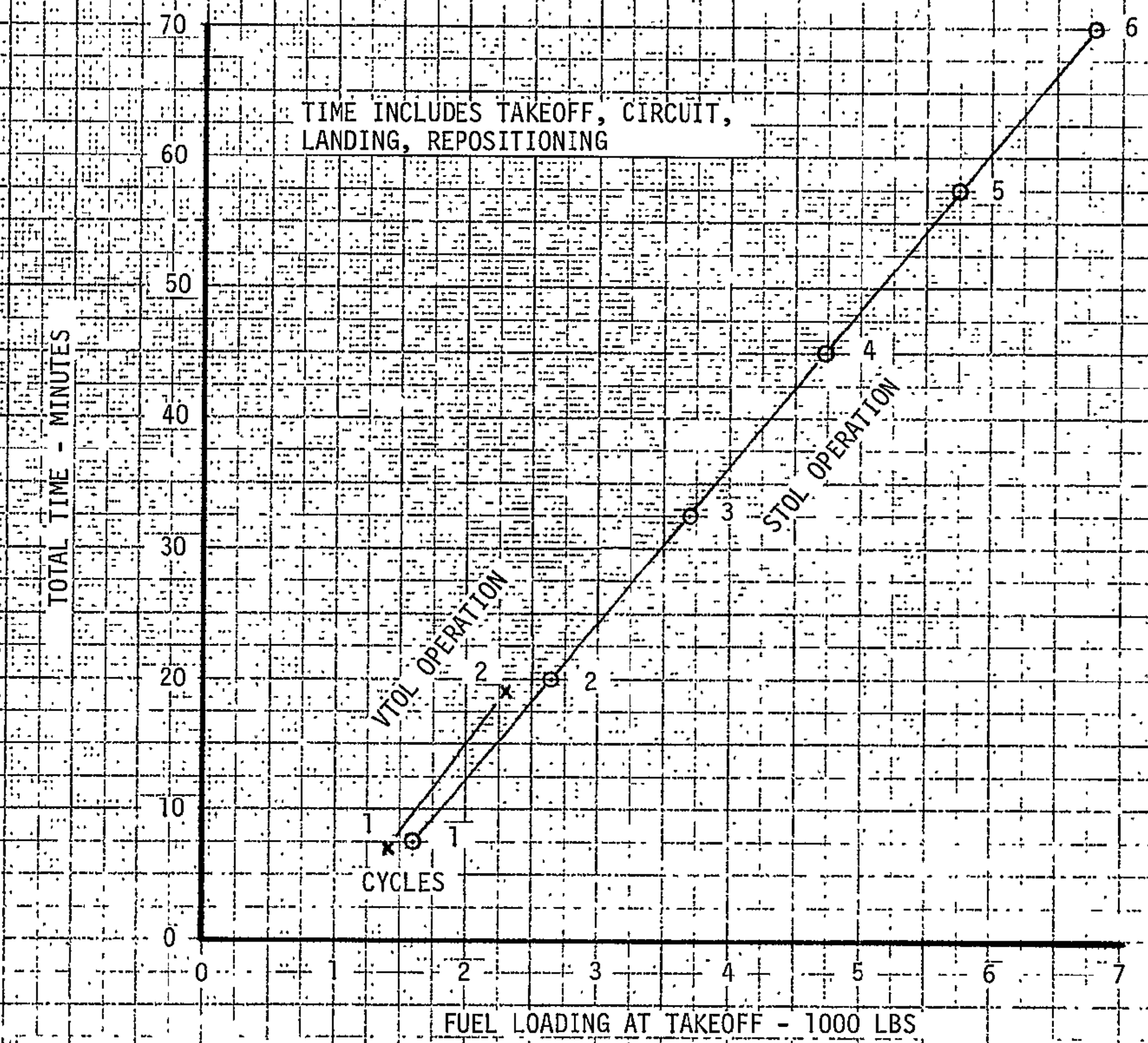
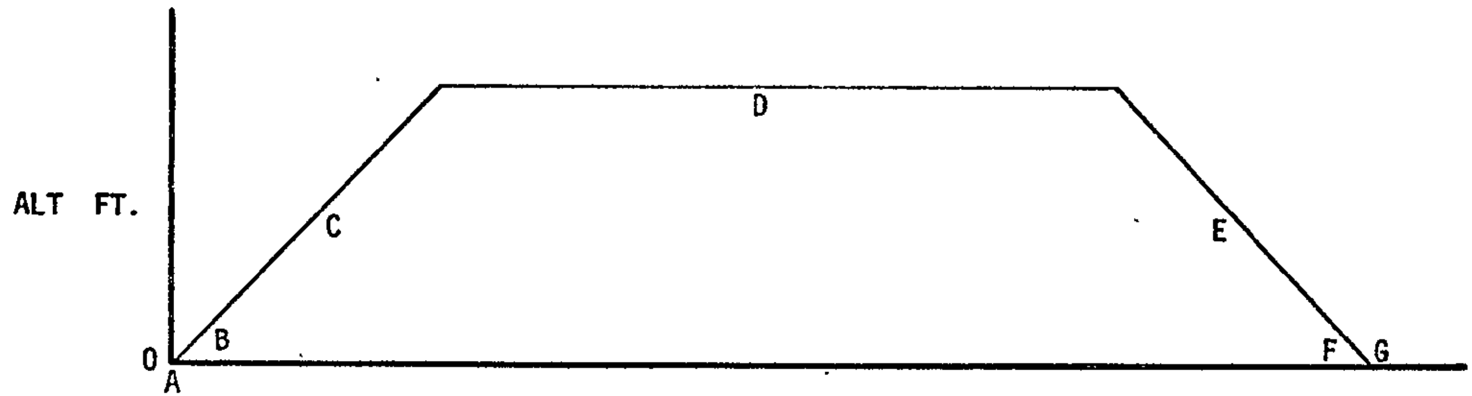


FIGURE 5-33

NASA V/STOL RESEARCH AIRCRAFT

TYPICAL RESEARCH FERRY MISSION

(MOFFET FIELD TO CROWS LANDING AUXILIARY LANDING FIELD)



DIST. = 44 NM

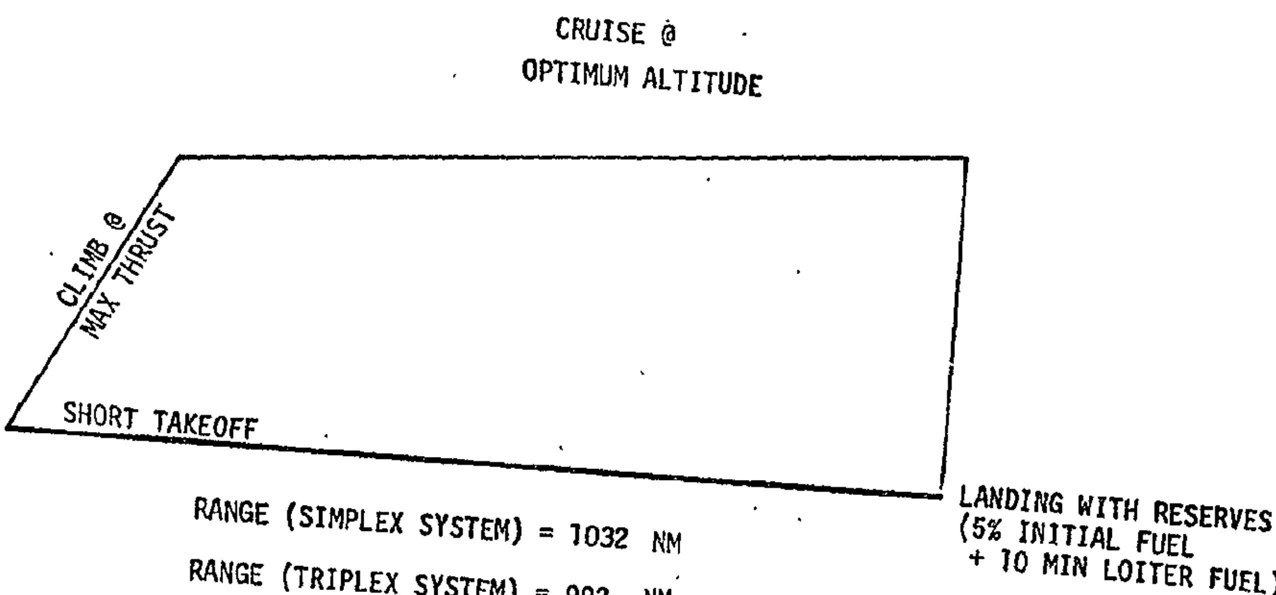
OPERATION	FUEL LB Δ	FUEL LB CUM	Δ DIST N.M.	TIME MIN Δ	TIME MIN CUM	ALTITUDE FT
A. START, ENGINE CHECK & TAXI	330	330	-	5.0	5.0	0
B. STO @ SLD ENGINE RATING	110	440	-	0.5	3.0	0
C. MAX. PWR. CLIMB TO 10,000 FT. CRUISE ALT.	160	660	6	0.9	3.9	0 → 10,000
D. CRUISE OUT AT 10,000 FT. ALTITUDE, M≈.55	225	825	23	3.9	7.8	10,000
E. IDLE PWR. DESCENT TO SEA LEVEL	70	895	15	3.5	11.3	10,000 → 0
F. DECELERATE & SL @ TEST SITE	300	1195	-	1.5	12.8	0
G. TAXI FOR 5 MINUTES AT TEST SITE	105	1300	-	5.0	17.8	0
			44 NM			

FIGURE 5-34

NASA V/STOL RESEARCH AIRCRAFT

<u>FERRY MISSION CAPABILITY</u>

INTERNAL FUEL +
(2) 120 GALLON TANKS
(TANKS RETAINED)



RANGE (SIMPLEX SYSTEM) = 1032 NM

RANGE (TRIPLEX SYSTEM) = 992 NM

TABLE 5-1

NASA RESEARCH AIRPLANE

WEIGHT STATEMENT

BASIC MODIFICATIONS

	WEIGHT (LB)	ARM (IN)	MOMENT (IN-LB)
REMOVE:	(-248.0)	(346.4)	(-85908)
Stand By Sight	-1.0	180.0	-180
Weapon Control Panel	-10.0	125.2	-1252
Sidewinder Control Panel	-1.8	127.0	-229
Throttle (2)	-1.4	162.1	-227
Control Column Grip (2)	-2.8	162.5	-455
Ballistics Panel	-2.0	128.0	-256
Waveform Generator, HUD	-18.3	465.0	-8510
Displays, HUD (2)	-33.2	161.7	-5368
VHF Radio, ARC-114 (2)	-13.2	166.7	-2200
VHF Antenna	-8.0	390.0	-3120
VHF Coupler	-0.9	232.0	-209
IFF Test Set	-3.0	476.0	-1428
IFF Test Set, Mt.	-1.5	476.0	-714
IFF Secure Voice	-14.5	465.9	-6756
Vertical Speed Xducer	-1.5	468.5	-703
SEAM Box, ASA-83	-5.0	444.0	-2220
Rate Gyros	-2.5	204.0	-510
Camera, F-95	-23.1	198.0	-4574
Camera Control	-1.5	127.0	-191

TABLE 5-1 (Continued)

	WEIGHT (LB)	ARM (IN)	MOMENT (IN-LB)
REMOVE:			
IWAC Computer	-15.1	456.8	-6898
Equipment Rack	-66.7	463.2	-30895
AHRS ASN-116	-21.0	429.2	-9013
ADD:	(483.7)	(332.2)	(160708)
Emerg. Spring Cartridges (3)	4.5	340.0	1530
Feel Sys. Servoactuator - Yaw	5.6	88.0	493
Feel Sys. Servoactuator - Pitch	6.8	180.0	1224
Feel Sys. Servoactuator - Roll	2.7	180.0	486
System LVDT's (3)	3.0	267.7	803
Strain Gages	2.5	340.0	850
Series Servoactuator	2.8	95.0	266
Parallel Servoactuator - Roll	2.6	310.0	806
Parallel Servoactuator - Yaw	6.9	530.0	3657
Parallel Servoactuator - Pitch	5.4	540.0	2916
Control Rods, Bellcranks & Instl.	12.8	304.5	3897
Power Amplifiers (6)	1.5	560.0	840
DME Indicator	1.1	127.0	140
Sidewinder Panel	1.8	127.0	229
Throttle (2)	1.5	162.0	243
Control Column Grip (2)	2.9	162.4	471
INS Mode Select Panel	0.8	127.0	102
DME Frequency Selector	2.2	1127.0	279
MLS Angle Receiver	9.5	100.0	950

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TABLE 5-1 (Continued)

	WEIGHT (LB)	ARM (IN)	MOMENT (IN-LB)
MLS Angle Receiver Mt.	1.8	100.0	180
MLS Control	1.8	127.0	229
C-Band Antenna	0.3	88.0	26
DME Interrogater	10.5	465.0	4883
L-Band Antenna	0.4	525.0	210
Digital Computer AP-101	46.0	465.0	21390
Inertial Nav. Unit	61.1	465.0	28412
Control Display Unit	5.1	127.0	648
Programable Graphics Gen.	15.0	465.0	6975
Rate Gyro Assy.	5.4	468.0	2527
Lateral Accelerometer	2.0	468.0	936
Vertical Accelerometer	2.0	130.0	260
Digital Adapter	24.0	465.0	11160
Engaging Control	6.0	127.0	762
Displays, HUD (2)	33.0	91.5	3020
Compass C-2J	11.0	408.0	4488
Installation - Mtg. Prov.	28.4	362.2	10286
Installation - Wire	59.0	362.2	21370
Nose Boom	52.0	80.0	4160
Equipment Rack	42.0	463.2	19454
TOTAL CHANGE	(235.7)	(317.4)	(74800)
C_L Pod (Data Acquisition)	290.0	322.2	93438
C_L Pylon	52.0	326.7	16988
Flt Test Equipment	30.0	341.1	10233
	(372.0)	(324.4)	(120659)

TABLE 5-2

NASA RESEARCH AIRPLANE

WEIGHT STATEMENT

DUPLEX FLIGHT CONTROL SYSTEM

CHANGES TO BASIC SIMPLEX SYSTEM

	WEIGHT (LB)	ARM (IN)	MOMENT (IN-LB)
REMOVE:	(-21.5)	(356.0)	(-7654)
Hydraulic Pumps (2)	-15.3	294.2	-4502
Power Amplifiers (6)	-1.5	560.0	-840
Access Door 266408	-4.7	492.0	-2312
ADD:	(231.7)	(429.4)	(99483)
Hydraulic Pumps (2)	17.3	294.2	5090
Cooling Sys.	14.0	294.0	4116
System RDVT's (3)	3.0	267.7	803
Dual Series Servoactuator - Yaw	24.0	530	12720
Dual Series Servoactuator - Pitch (2)	12.4	540	6696
Dual Series Servoactuator - Roll (2)	12.4	310	3844
Power Amplifiers (3)	0.8	560	448
Trim Motor - Yaw	6.5	180	1170
Control Rods, Bellcranks & Instl	17.7	442	7825
Access Door	8.0	402.0	3936
Digital Computer	46.0	492.0	22632
Digital Adapter	24.0	492.0	11808
Rate Gyro Assembly	5.4	468.0	2527
Installation - Mtg. Prov.	10.2	490.4	5002
Installation - Wire	30.0	362.2	10866
TOTAL CHANGE TO SIMPLEX SYSTEM	210.2	436.9	91829

TABLE 5-3

NASA RESEARCH AIRPLANE

WEIGHT STATEMENT

TRIPLEX FLIGHT CONTROL SYSTEM

CHANGES TO BASIC SIMPLEX SYSTEM

	WEIGHT (LB)	ARM (IN)	MOMENT (IN-LB)
REMOVE:	(-4.7)	(492.0)	(-2312)
Access Door 266408	-4.7	492.0	-2312
ADD:	(256.9)	(420.9)	(108136)
Parallel Servoactuator - Yaw	10.0	530	5300
Parallel Servoactuator - Pitch	10.0	540	5300
Parallel Servoactuator - Roll	10.0	310	3100
Control Rods, Bellcranks & Instl.	9.0	460	4140
Access Door	8.0	492.0	3936
Digital Computers (2)	92.0	418.5	38502
Digital Adapters (2)	48.0	418.5	20088
Rate Gyro Assy (2)	10.8	468.0	5054
Installation - Mtg Rov.	19.1	425.6	8128
Installation - Wire	40.0	362.2	14488
TOTAL CHANGE TO SIMPLEX SYSTEM	(252.2)	419.6	(105824)
Remove C_L Pod	-290.0	322.2	-93438
Remove C_L Pod	-52.0	326.7	-16988
Data Acquisition (2 Gun Pods)	378.0	334.7	126517
TOTAL CHANGE TO DATA ACQUISITION SYSTEM	(36.0)	(447.0)	(16091)

TABLE 5-4

NASA RESEARCH AIRPLANE

WEIGHT STATEMENT

LOW RANGE AIR DATA SYSTEM

	WEIGHT (LBS)
Sensor	1.0
Electronic Unit	2.4
Installation	1.0
TOTAL CHANGE	4.4

TABLE 5-5

NASA RESEARCH AIRPLANE

WEIGHT STATEMENT

SIDE-ARM CONTROLLER

REMOVE:	
ARC-114 Radio	-7.0
ARC-150 Radio	-10.0
Voice Recorder	-3.0
Oxygen Indicator	-.2
Shelf	-.6
TOTAL WEIGHT REMOVED	(-20.8)
ADD:	
Side-Arm Controller	4.8
D403 Radio	5.5
Oxygen Indicator	.2
Shelf	.5
Installation	3.3
TOTAL WEIGHT ADDED	(14.3)
TOTAL CHANGE	-6.5

TABLE 5-6

NASA RESEARCH AIRPLANE

WEIGHT STATEMENT

SIMPLEX THROTTLE AND NOZZLE SYSTEM

	WEIGHT
REMOVE:	
Link Assy	-.5
Connecting Rod	-.9
Quadrant Assy	-.5
TOTAL WEIGHT REMOVED	(-1.9)
ADD:	
Force Link	.5
Servoactuator	2.8
RVDT	.1
Quadrant Assy	.8
Parallel Servoactuator	2.8
Force Link	.9
RVDT	.1
Link	.3
Installation	1.7
TOTAL WEIGHT ADDED	(10.0)
TOTAL CHANGE	8.1

TABLE 5-7

NASA RESEARCH AIRPLANE

WEIGHT STATEMENT

DUPLEX THROTTLE AND NOZZLE SYSTEM

	WEIGHT
REMOVE:	
Idler	-.3
Control Rod	-.2
TOTAL WEIGHT REMOVED	(-.5)
ADD:	
Walking Beam Bellcrank	1.0
Dual Series Servo	12.4
RVDT (2)	.2
Link	.3
Dual Series Servoactuator	12.4
Bellcrank	.3
Walking Beam Bellcrank	1.0
RVDT (2)	.2
Link	.3
Installation	7.4
TOTAL WEIGHT ADDED	(35.5)
TOTAL CHANGE	35.0

TABLE 5-8

NASA RESEARCH AIRPLANE

WEIGHT STATEMENT

TRIPLEX THROTTLE AND NOZZLE SYSTEM

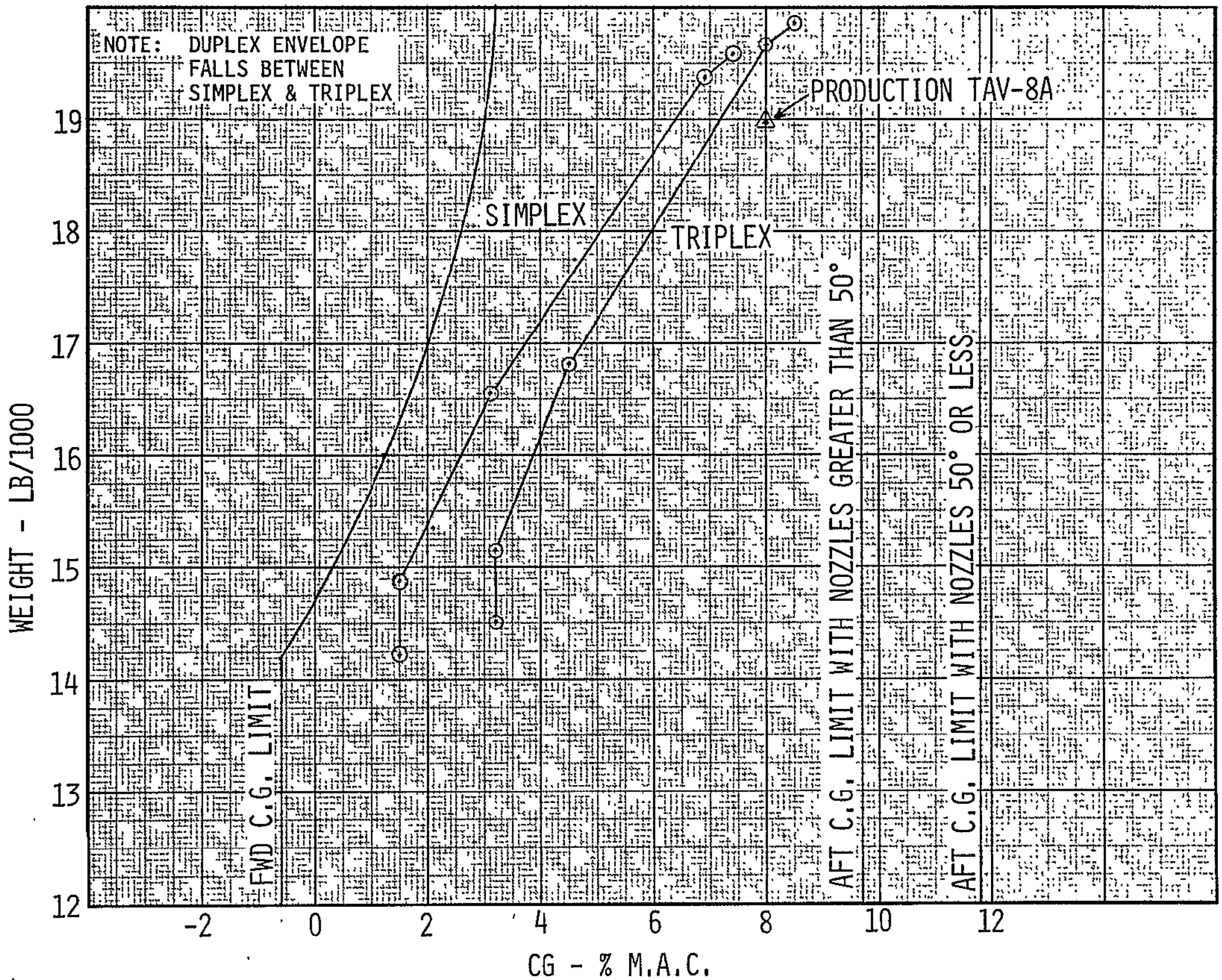
	WEIGHT
REMOVE:	
Link Assy	-.5
Connecting Rod	-.9
Quadrant Assy	-.5
TOTAL WEIGHT REMOVED	(-1.9)
ADD:	
Force Link	.5
Servoactuator	4.0
RVDT	.6
Quadrant Assy	.8
Servoactuator	4.0
Force Link	.9
RVDT	.6
Link	.3
Installation	2.4
	(14.1)
TOTAL CHANGE	12.2

TABLE 5-9
WEIGHT SUMMARY
NASA RESEARCH AIRPLANE

	G/VTOL	TAV-8A
OPERATING WEIGHT EMPTY (INCLUDES UPDATE MODS)	13540	13625
SIMPLEX SYSTEM	236	236
FLIGHT TEST EQUIPMENT	30	30
DATA ACQUISITION (C_L POD + C_L PYLON)	342	342
OWE - SIMPLEX SYSTEM	14148	14233
DELTA CHANGE TO SIMPLEX SYSTEM	210	210
OWE - DUPLEX SYSTEM	14358	14443
DELTA CHANGE TO SIMPLEX SYSTEM	252	252
REMOVE C_L POD & C_L PYLON	-342	-342
ADD GUN PODS - DATA ACQUISITION EQUIPMENT	378	378
OWE - TRIPLEX SYSTEM	14436	14521
OPTIONS:		
o LOW RANGE AIR DATA SYSTEM	4	4
o SIDE-ARM CONTROLLER	-7	-7
o THROTTLE AND NOZZLE SYSTEM		
- DELTA CHANGE TO SIMPLEX SYSTEM	8	8
- DELTA CHANGE TO DUPLEX SYSTEM	35	35
- DELTA CHANGE TO TRIPLEX SYSTEM	12	12

FIGURE 5-35

CENTER OF GRAVITY ENVELOPE - NASA RESEARCH AIRPLANE



We have a staff of Integrated Logistics Support (ILS) personnel assigned to the AV-8 Program. These personnel will be available to NASA to provide support information. Normally, requests for support assistance would be made through the MCAIR AV-8 ILS manager.

5.4.1 VENDOR DATA - MCAIR will obtain vendor data for hardware selected for the NASA peculiar modifications. We will also identify spares requirements and prepare Technical Manuals.

5.4.2 SPARES, REPAIR PARTS AND GROUND SUPPORT EQUIPMENT - Appendix A itemizes MCAIR recommendations for spares, repair parts and ground support equipment required for support of the aircraft over a four year period. These recommendations are based on an aircraft utilization rate of 10 hours per month. Appendix A will be revised, as required, to reflect aircraft configuration changes and additional requirements resulting from USMC usage. NASA, at its option, may contract with NAVAIR for material support on an as needed basis, or may procure outfitting requirements through MCAIR. In the event NASA elects to procure support materials from MCAIR, it is recommended that their purchase be made concurrent with that of the aircraft due to limited quantity and lead times.

5.4.3 GROUND SUPPORT EQUIPMENT (GSE) - Existing AV-8A/TAV-8A Organizational Level GSE items required to support the program are identified in the GSE section of Appendix A. Additional test equipment and integration benches to support peculiar equipment will be developed by MCAIR as "laboratory type" equipment. These additional items will be made available for NASA use upon delivery of the reconfigured aircraft.

5.4.4 TECHNICAL MANUALS - MCAIR will utilize engineering drawings and vendor source data to provide Technical Manuals for the NASA flight test program. MCAIR will provide NASA with one set of AV-8A/TAV-8A manuals of the latest configuration and one set of supplements identifying the modifications made by MCAIR for the NASA flight test program.

5.4.5 ENGINEERING CHANGE PROPOSAL (ECP) SUPPORT - MCAIR will provide NASA with a copy of all ECPs approved by NAVAIR for the TAV-8A aircraft. If NASA desires to have any of these ECPs installed in the aircraft they can negotiate directly with NAVAIR for kit procurement.

5.4.6 REPAIR OF REPAIRABLES - Spares, repair parts and GSE identified in Appendix A provides material support for organizational level maintenance (on-aircraft maintenance). It is recommended that NASA contract with NAVAIR for repair of components that are common to USMC TAV-8A configurations and with MCAIR for repair of components peculiar to the NASA Research Aircraft. Common components are identified in Appendix A by assignment of National Stock Numbers. Items not assigned National Stock Numbers are peculiar to this airplane.

APPENDIX A

STATEMENT OF WORK

A-1 INTRODUCTION

MCAIR Report MDC A4949 "Proposal for a Conceptual Design Study of Modifications to Harrier G-VTOL" was submitted to the National Aeronautics and Space Administration on 29 August 1977. It was submitted in response to Request for Proposal 2-26931(HK), "Conceptual Design Study of Modifications to Harrier G-VTOL," dated 5 August 1977. On 3 October 1977 MCAIR began work on the program under NASA Contract NAS2-9748. The Scope of Work defined by this contract was the same as proposed by MCAIR except that MCAIR was to provide, to the extent possible, the cost estimates for the individual items in the basic modifications; as a minimum the costs for the control system modifications, data acquisition systems, simulation, ground tests, and airworthiness tests. Amendment One to the contract added the conceptual design of a triplex parallel flight control system and triplex parallel throttle and nozzle controls.

This appendix contains those portions of the documents which define the statement of work for this program. Section A-2 contains the Study Approach and Deliverable Items sections from the MCAIR proposal. Sections A-3 and A-4 contain the Scope of Work sections from the contract and contract amendment.

A-2 MCAIR STATEMENT OF WORK

The statement of work proposed by MCAIR is contained in Section 2 and Section 3 of MDC A4949. These sections are quoted below:

"2.0 STUDY APPROACH

"MCAIR will provide the materials and services necessary to perform a conceptual design study which will define modification approaches and estimate costs associated with converting the Harrier G-VTOL to a VTOL control, display, and guidance research aircraft. This study will include, but not necessarily be limited to, defining modifications required to the control system and cockpit, exploring various methods of mechanization, defining research equipment and software requirements, exploring methods for aircraft weight reduction and defining the weight of added equipment, selecting major hardware components, defining provisions for safety, and determining the weight and performance capabilities of the modified aircraft. The design modifications will be established in such a way that the ability to operate from a ship will be retained or easily restored. The conceptual design study will identify the basic modifications required in the overall modification program as well as optional modifications which can be added initially or at a later date in order to enhance the research capability of the aircraft. Where necessary certain simplifying assumptions will be made."

"2.1 BASIC MODIFICATIONS

"MCAIR will develop the conceptual designs and the budgetary costing data for the following basic aircraft modifications:

"2.1.1 CONTROL SYSTEM MODIFICATIONS - The flight control system design modifications will be established on the basis that the rear cockpit will be the evaluation pilot station and the front cockpit will be the safety pilot and solo pilot station. The aft cockpit's control stick and rudder pedals will be mechanically disconnected from the Harrier flight control system and equipped with electrical position and force transducers as well as actuators for providing variable feel characteristics."

"The design approach for implementing the research aircraft control system capability will be to use an onboard digital computer to compute pitch, roll and yaw servo command signals for the Harrier control surface/reaction control system from the aft pilot's control stick and rudder pedal command signals, aircraft motion sensor signals, and air data computer measurements. The digital computer will also provide signals for the aft control stick and rudder pedal variable feel systems."

"Sufficient information will be provided in the front cockpit to enable the safety pilot to monitor the activity of the evaluation pilot and of the digital control system and to disengage these systems should the necessity arise. The front seat pilot will then fly the aircraft using an essentially unmodified Harrier flight control system. The front cockpit will also contain a digital computer console which will permit the front seat pilot to change the aircraft dynamic response and feel system characteristics provided to the evaluation pilot."

"This modification comprises the installation of a digital flight control system and other systems, system modifications, and studies including:

a. The stick and rudder pedals shall be disconnected from the flight control system and a force-feel system installed.

b. Installation of a powered rudder system or an acceptable alternate.

c. Installation of a digital flight control system and other systems consisting of:

(1) A single thread system using full authority parallel along with limited authority series servos utilizing as much existing hardware as possible. In each channel of control (pitch, roll, and yaw) the parallel servo will move the existing Harrier mechanical control linkages, including the safety pilot's controller, in response to commands computed by the digital computer. Since the frequency response capability of the parallel servo system will be limited due to the mass of the control linkages, a limited authority series servo, which would not cause the safety pilot's controller to move, will be included in each channel to provide the frequency response characteristics required for closed loop control purposes. A high pass ("washout") network will be included in each series servo so that control surface/reaction control will be primarily due to the parallel servos. Therefore, the safety pilot can effectively monitor control system activities by monitoring the motion of his stick and rudder pedals. If required, he can manually override and electrically disconnect the evaluation pilot's control system.

(2) A digital computer having sufficient memory capability and computational speed to perform the required control, display, and guidance computations. A computer sizing analysis will be performed to an appropriate depth for estimating computer requirements. Candidates considered in the computer selection process will include, but not be limited to, the IBM AP101, the ROLM 1600, and the ROLM Ruggedized Eclipse. Included in the computer selection criteria will be the

availability of a higher order language compiler/code generator which can be resident on either the IBM 360 or CDC 7600 Ames system. The computer system selected will include all necessary peripheral equipment.

(3) An inertial platform similar to Litton LTN 51 to replace existing Navigation Display Unit and Control (FE-541) to provide position, rate, and acceleration information to the computational system.

(4) Transducers to provide input data to the computer and data acquisition system.

d. Hydraulic and electrical subsystem modifications as required.

e. The investigation of methods for improving control power. Included will be methods which would not be desirable for production aircraft, such as boom mounted roll reaction controls.

f. An abbreviated failure mode and effects analysis of the complete system."

"2.1.2 DATA ACQUISITION SYSTEM - MCAIR will prepare a conceptual design of a data acquisition system to the depth required for cost estimation. A system with 150 measurands will be assumed. A Teledyne Controls AIFTDS-4000 "Airborne Integrated Flight Test Data System" shall be considered GFE. GFE will not include data sensors, signal conditioning interface wiring, and necessary component fixtures and bracketry. Sensors which are components of existing aircraft systems will be utilized wherever possible. Use of the outputs from the inertial platform will be used for both the control system and the data system."

"2.1.3 HEAD UP DISPLAY - MCAIR will determine the modifications required to allow flexibility in presentation format of the Head Up Display (HUD). The displays in both the front and rear cockpits will be identical. A programmable graphic generator will replace the existing waveform generator. The graphic generator will be easily reprogrammable in order to provide flexibility for change and be interfaced with the system computer. MCAIR will consider using the digital computer to generate the HUD symbology instead of a new graphic generator."

"2.1.4 LANDING GUIDANCE SYSTEM - MCAIR will determine the modifications required to incorporate a landing guidance system. The landing guidance system will consist of MLS glide slope and DME receivers (GFE), MLS antennas (GFE), and the appropriate electronics to interface with the digital computer. The MLS angle receiver interface will conform to ARINC 582 and the MLS DME receiver interface will conform to ARINC 568. In addition, the interface electronics unit will have room for adding the electronics to interface an additional landing guidance sensor system."

"2.1.5 SIMULATION - MCAIR will estimate the planning price of modifying the existing AV-8A math model to be representative of the G-VTOL aircraft and compatible with the Ames FSAA Simulator. This simulation will model aircraft aerodynamics, propulsion, control, and landing gear characteristics. The support requirements for a four week simulation by the Ames Research Center on the FSAA during the design phase shall be determined. This simulation will evaluate the control system modifications and the flight safety aspects of the design."

"2.1.6 SYSTEM SOFTWARE - MCAIR will prepare a preliminary plan for developing the required system software. The software addressed in this software development plan will consist of:

a. All software required for functionally operating the system (e.g., for accessing all sensor information, sending servo commands, driving displays and setting flags, accomplishing a system preflight checkout, accomplishing a computer self-check, and a utility program not necessarily resident in the computer).

b. A software control system program that will functionally provide the same control system performance as the basic vehicle control system (i.e., manual control system plus SAS).

c. A HUD software program that provides the same functional capability as the standard AV-8A HUD system."

"2.1.7 GROUND TESTS - MCAIR will define the requirements for a minimum ground test of the modified control system and aircraft to assure the safe operation of the aircraft, digital computer, inertial platform, sensors, and software."

"2.1.8 AIRWORTHINESS TESTS - MCAIR will define the requirements for a minimum airworthiness flight test following the modification of the aircraft. This flight test will address only the modified basic control system from the front cockpit."

"2.2 OPTIONAL MODIFICATIONS

MCAIR will conceptually design and develop the budgetary costing data for the following aircraft modifications:

"2.2.1 LOW SPEED AIR DATA SYSTEM - MCAIR will install a low speed air data system in the aircraft. To accomplish this objective, MCAIR will review available low speed air data sensors and review aircraft locations suitable for their installation. Candidate sensors and their locations on the aircraft will be determined. Selection criteria will include the consideration that inputs from a low speed air data system may eventually be required to provide inputs to the aircraft control system. If possible, the system installed will have had prior flight experience so development or flight certification of a new system will not be required."

"2.2.2 THRUST AND NOZZLE SERVO SYSTEM - A conceptual design of a thrust and nozzle servo system for the rear cockpit will be made. In the design, the rear cockpit's throttle lever and nozzle lever will be mechanically disconnected from the Harrier engine and nozzle controls and equipped with electrical position transducers which would provide lever position information to the digital computer. Single channel servo systems utilizing full authority parallel servos and limited authority series servos will be installed to permit the engine fuel control unit and the air motor servo unit (which positions the nozzles) to be controlled by signals from the digital computer. The parallel servos will be used to move the existing Harrier throttle and nozzle control mechanisms, including the throttle lever and nozzle lever in the front cockpit. Since the frequency response capability of the parallel servo system will be limited due to the mass of the linkages, limited authority series servos will be included at the engine fuel control unit and air motor servo unit to provide the frequency response characteristics required for closed loop control purposes. A high pass ("washout") network will be included in each series servo so that the mechanical inputs to the engine fuel control unit and air motor servo unit would be due primarily to the parallel servos. Therefore, the safety pilot can effectively monitor throttle and nozzle activities by monitoring the motion of his throttle and nozzle levers. This thrust and nozzle servo system will permit integrated power management control concepts to be investigated. Provisions will be made to readily replace the aft cockpit's throttle/nozzle control box which contains the throttle and nozzle levers with a new integrated power management control unit which will supply pilot command signals to the digital computer."

"2.2.3 SIDE-ARM CONTROLLER - A conceptual design for a side-stick controller for pitch, roll, and yaw control incorporated with functions that parallel the electrical inputs from the stick and rudder pedals will be developed. A design for the cockpit installation will be prepared."

"2.2.4 REDUNDANT CONTROL SYSTEMS - MCAIR will consider dual full authority series servo mechanizations for the rear cockpit pitch, roll, yaw, throttle and nozzle control systems in place of the single thread parallel/series servo systems. An abbreviated failure mode and effects analysis will be performed."

"2.2.5 REDUNDANT THROTTLE AND NOZZLE CONTROL - MCAIR will prepare a conceptual design of a dual, full authority series servo system for a throttle and nozzle control system to operate in conjunction with single thread pitch, roll and yaw control systems."

"3.0 DELIVERABLE ITEMS

As a minimum, MCAIR will deliver at the end of the study, a report including the following:

a. Conceptual designs for the Harrier modification including the designation of recommended purchased equipment. The basic modification package will consist of items 2.1.1 through 2.1.8. Items 2.2.1 through 2.2.5 will be considered as options and the effect of including each of these options individually and sequentially into the program will be documented.

b. A modification plan, a test plan, and a schedule for the basic modifications of the Harrier G-VTOL aircraft and each of the options.

c. A cost estimate for the basic modification package (Paragraph 2.1) and for each option item (2.2.1, 2.2.2, 2.2.3, 2.2.4, 2.2.5) will be provided. Each of these cost estimates will contain information for the following categories: Engineering, Manufacturing, Procurement, and Other Direct Costs.

d. The estimated performance of the modified aircraft will be documented along with a weight breakdown statement including weight of parts removed and parts added."

A-3 CONTRACT SCOPE OF WORK

The Scope of Work section from NASA Contract NAS2-9748 is the following:

"A. Scope of Work

1. The contractor shall furnish the necessary services and materials to provide a Conceptual Design Study of Modifications to Harrier G-VTOL as set forth in McDonnell Douglas Corporation, McDonnell Aircraft Company Report Number MDC A4949 (Volume 1), dated August 29, 1977 to RFP 2-26931, dated August 5, 1977. The McDonnell Douglas Corporation, McDonnell Aircraft Company Report

Number MDC A4949 and NASA Statement of Work 2-26931, dated August 5, 1977 are hereby made a part of this contract by reference."

2. The following addition is hereby made to Paragraph 3.0(c) of the McDonnell Aircraft Company Report No. MDC A4949:

   "In addition to the above cost estimates the contractor shall provide, to the extent possible, the total cost estimates for the individual items in Section 2.1; as a minimum the costs shall be provided for 2.1.1, 2.1.2, 2.1.5, 2.1.7 and 2.1.8."

A-4 CONTRACT AMENDMENT SCOPE OF WORK

Amendment One to NASA Contract NASA-9748 modified the contract scope of work as follows:

"1. Article II, Paragraph A (Scope of Work) is modified to include the following additional work:

   Alternate Control System Modifications for Harrier G-VTOL

   (a) Redundant Parallel/Series Control Systems

   The contractor shall conceptually design a triplex parallel-full authority servo plus a single limited authority series servo mechanization for the rear cockpit pitch, roll, yaw, throttle, and nozzle control systems in place of the single thread parallel/series servo systems. Should the study indicate sufficient response in throttle and nozzle operation can be obtained from the parallel servos, the single thread series servo will not be included in these systems.

   (b) Redundant Parallel/Series Throttle and Nozzle Control

   The contractor shall prepare a conceptual design of a triplex parallel-full authority servo plus a single limited authority series servo (if required) system for a throttle and nozzle control system to operate in conjunction with single thread pitch, roll, and yaw control systems.

   The results of the conceptual design and estimated cost of each of these two option studies will be included under the deliverable items under this contract."

APPENDIX B

TASK DESCRIPTIONS

B-1 INTRODUCTION

Technical descriptions of the aircraft modifications which will convert the British civil registry Harrier G-VTOL to a V/STOL control, display, and guidance research aircraft are contained in the main body of this report. Brief descriptions of the major tasks required to implement these aircraft modifications are contained in this appendix. These task descriptions were prepared for use in deriving planning prices for NASA. It was assumed in preparing these task descriptions that the aircraft would be modified to the applicable and appropriate TAV-8A standard configuration before delivery to MCAIR.

The modifications studied for NASA fall into two categories: basic modifications and optional modifications. The basic modifications include a simplex parallel digital fly-by-wire flight control system, a data acquisition system, a head-up display (HUD), a landing guidance system (MLS), and an inertial navigation system (INS). This part of the program includes simulation, system software, ground tests, and airworthiness tests. The optional modifications include a duplex series digital fly-by-wire flight control system, a triplex parallel digital fly-by-wire flight control system, a low speed air data system, a side-arm controller, and a thrust and nozzle servo system (simplex parallel, duplex series, and triplex parallel). The task descriptions for the basic modifications are given in Section B-2. The task descriptions for the duplex and triplex flight control systems are given in Sections B-3 and B-4. The technical and task descriptions for the low speed air data system, side-arm controller, and thrust and nozzle servo systems are given in Sections B-5, B-6, and B-7.

B-2 BASIC MODIFICATIONS

Task descriptions for implementing the basic modifications to the two place Harrier are given in this section. The control system modifications (which include component hardware and the inertial navigation system) are given in Section B-2.1. The data acquisition system, head up display, and landing guidance system are discussed in Sections B-2.2 through B-2.4. System software and simulation are discussed in Sections B-2.5 and B-2.6. Ground tests and airworthiness tests are discussed in Sections B-2.7 and B-2.8.

B-2.1 FLIGHT CONTROL SYSTEM MODIFICATIONS - The flight control system modifications include aircraft modifications and equipment requirements.

B-2.1.1 AIRCRAFT MODIFICATIONS - The following aircraft modifications are required:

(a) Nose - Longitudinal Reaction Control System: The C292616 nozzle lever is removed. A new electromechanical series servo actuator with associated linkage is added and the existing nose boom is replaced with a flight test nose boom.

(b) Forward Cockpit - A fly-by-wire control panel with a DME Indicator is designed to be located in the space left by the removal of the weapon control panel from the left hand main instrument panel. This panel has an engage enable switch and indicators lights to display the functions of the control system which are selected, which are operational, and any failures that occur. (The aft cockpit will have the selection capability and will decide when the system will be activated.)

The Sidewinder control panel on the left hand subinstrument panel is revised to give a jettison capability for stores such as fuel tanks. (This stores jettison capability was lost when the weapon control panel was removed above.)

An INS Mode Select Panel is mounted on the right hand main instrument panel where formerly the Ferranti INAS Control Panel was mounted.

The pilot's control stick is modified to remove the safety cover from the bomb button. This switch is wired for system disengage so that the safety pilot can quickly disengage the fly-by-wire system. The throttle grip on the left hand console is also revised to include a system disengage switch.

The ballistic panel is removed from the pedestal panel between the forward pilot's legs and a DME frequency selector installed. On the right hand console the ARC-114 radio is replaced with the INS control panel.

Rate gyros (405 RGU/1 and 307 RGU/1), lockon/reject (9900-06), and camera controls (B314828) are removed.

The standby sight on the left hand side of the Head-Up Display is removed to save weight.

(c) Aft Cockpit - The weapon monitor panel on the left side of the main instrument panel is replaced with a fly-by-wire control panel which shows the system status with function selection and failure indications repeated in the front cockpit. The panel includes variable stability controls, computer displays, and master engage switch.

A microwave landing system control panel is added to the center pedestal where the aft cockpit pilot will control the selection of glideslope, sensitivity, and other variables.

The following changes will be made to implement the feel systems:

- Lateral Feel System - The C292279 aileron lever is revised so that the B267697 rotatable control rod can be disconnected from the lateral control system. A new bellcrank, a new RVDT, a new torque motor, and a new force link are added.

- Longitudinal Feel System - The B290034 tailplane control rod is removed. A new force link, RVDT, and torque motor are added.

- Directional Feel System - The B290103 rudder control rod and C287292 lever assembly are removed. A new force link, RVDT, and torque motor are added.

The VHF Voice Radio (ARC-114) will be removed and the gyro for the attitude heading reference set (under floor) will be removed.

(d) Wing - A new lateral parallel servo is added to the front spar of the wing and the B276941 aileron control lever assembly is revised to accept the motion output of the lateral servo. A new spring cartridge is added.

(e) Aft Equipment Bay - The aft equipment bay shelf, P/N 317739, between Frame 33 and Frame 36 is replaced with the new equipment shelf using the same structural pickup points. The following equipment is removed:

- IFF Test Set (TS-1843/APX)
- IFF Test Set Mount (MT-3513A/APX-72)
- Secure IFF provisions
- Vertical speed transducer
- SEAM (ASA-83)
- Waveform Generator (202SUE/3)
- Attitude heading reference set components

The following equipment is added:

- Digital Computer (IBM AP-101)
- Rate gyros (3)
- Lateral accelerometer
- Vertical accelerometer
- Digital adapter
- Litton LTN-51 Inertial Navigation Unit

(f) Aft Fuselage - The new longitudinal servo actuator is installed in the aft fuselage. The B136980 tailplane compensator is revised to accept the output of the longitudinal servo actuator. A new spring cartridge is installed between the longitudinal servo and the B136980 tailplane compensator. A new directional servo actuator is fitted on Frame 41. A new bellcrank, new spring cartridge, and new push rod are added. The 168864 rudder compensator is revised to accept the directional servo actuator input. Install six (6) power amplifiers (1 for each feel motor and 1 for each parallel servo motor) on Frame 42.

(g) Top Center Fuselage - The VHF antenna and coupler are removed.

(h) Hydraulics - The simplex system requires no hydraulic system modifications.

(i) Electrical - The wiring for the basic avionics changes, the electromechanical parallel actuators, and the feel systems actuators are as follows:

- 31 wires from the nose to the forward cockpit.
- 12 wires from the nose to the equipment bay shelf.
- 119 wires from the forward cockpit to the equipment bay shelf.
- 115 wires from the aft cockpit to the equipment bay shelf.
- 12 wires from the wing to the equipment bay shelf.
- 24 wires from the engine compartment to the equipment bay shelf.
- 1 coax from the equipment bay shelf to an added antenna located on the bottom of the aircraft just forward of the ventral fin.

- o Approximately 200 wires are added or reterminated within the equipment bay/aft fuselage area.
- o Existing wiring not used will be left in the aircraft but capped off. The removal of equipment will delete 10 fuses in the distribution panel. The addition of equipment will add back 9 fuses.

B-2.1.2 EQUIPMENT REQUIREMENTS - The following tasks are involved in obtaining the digital flight control and avionics equipment associated with the control systems modification:

(a) Flight Control Components - Flight control system components which are to be purchased or manufactured by MCAIR are listed in Figure B-1. This list does not include purchased avionics which are discussed later. Development effort will be minimized by selecting components which can be qualified by similarity whenever possible with only acceptance tests required of the supplier. Supplier tests are expected to be analog. Where qualified components cannot be obtained, a reduced qualification test program will be utilized which is informally agreed on by the supplier and MCAIR. A brief description of the components follows:

- o Feel System Servoactuators - To provide artificial feel forces at the stick and rudder pedals in the aft cockpit. Single channel electromechanical servoactuators are planned. Servoactuator ratings differ for each axis.

- o RVDTs and Strain Gages - Position and force transducers which will be attached to aft cockpit stick and rudder pedals to provide command signals to the flight control and feel system servoactuators.

- o Force Switch Strain Gages - Transducers attached to the forward cockpit stick and rudder pedals to disengage the fly-by-wire controls when the forward cockpit pilot generates a sufficient override force.

- o Series Servoactuator and Solenoid - Limited authority center-locked electromechanical servoactuator and solenoid to provide series control of the pitch forward reaction control valve presently installed in YAV-8B aircraft.

- o Emergency Spring Cartridge - Required to protect against a jammed motor.

- o Parallel Servoactuators - The flight control servoactuators used to drive the mechanical controls in yaw, pitch, and roll will be electromechanical. Supplier tests are expected to be analog. Ratings for the servoactuators differ for each axis.

FIGURE B-1
SIMPLEX PARALLEL FLIGHT CONTROL EQUIPMENT

NAME	STATUS	TYPE	MANUFACTURER	PART NUMBER	CHANNELS PER UNIT	NUMBER OF UNITS A/C	NUMBER OF UNITS SPARE	WEIGHT
Feel System Servoactuator-Yaw	New	E/M E/M	MPC Products Plessey Dynamics	proposal #15974 DDRWA-2	1	1	1	5.6
Feel System Servoactuator-Pitch	New	E/M E/M	MPC Products Plessey Dynamics	#15973 DDRWA-1	1	1	1	6.8
Feel System Servoactuator-Roll	New	E/M E/M	MPC Products Plessey Dynamics	#15972 DDRWA-3	1	1	1	2.7
System RVDT's	New			TBD	1	3	2	.5
System Strain Gages	Existing		MDC	TBD	1	3	2	
Force Switch Strain Gages	Existing		MDC	TBD	1	3	2	
Series Servoactuator	Existing	E/M	MPC Products	11650-3-L59	1	1	1	2.8
Series Servoactuator Solenoid	Existing		LEDEX	184-945-001	1	1	1	
Emergency Spring Cartridge	New		MDC		1	3		4.5
Parallel Servoactuator-Yaw	New	E/M E/M	MPC Products Plessey Dynamics	#15977	1	1	1	6.9
Parallel Servoactuator-Pitch	New	E/M E/M	MPC Products Plessey Dynamics	#15979-1	1	1	1	5.4
Parallel Servoactuator-Roll	New	E/M E/M	MPC Products Plessey Dynamics	#15978-1	1	1	1	2.6

(b) Avionics - Avionic system components are the following:

- o Digital Flight Control System (CFE)

  - Computer (IBM AP-101)
  - Rate Gyros
  - Lateral Accelerometer
  - Vertical Accelerometer
  - Digital Adapter

- o Inertial Navigation System (GFE)

  - Litton LTN-51 Inertial Navigation Unit
  - INS Control Display Unit
  - INS Mode Select Unit

- o Compass System (GFE)

The MCAIR and supplier tasks required to acquire, integrate and install the avionics are:

- o Digital Flight Control System and Digital Adapters (CFE) - MCAIR will prepare the specifications and do the procurement. Two units are to be procured (1 installed, 1 spare). Acceptance tests will be done (included in ground tests), likewise a brief integration test. MCAIR is also responsible for installation design and actual installation, as checkout tests will be performed for reference.

- o Inertial Navigation and Compass System (GFE) - MCAIR will perform acceptance tests, likewise integration testing. These tasks are included in the ground tests. Two units are to be obtained (1 installed, 1 spare).

(c) Failure Mode and Effects Analysis - The failure mode and effects analysis (FMEA) will comprise three tasks:

- o Determine Built-in-Test (BIT) percentage necessary and define means to achieve these levels.

- o Determine Inflight-Integrity Monitoring required and perform detailed design to achieve the necessary coverage.

- o Determine failure modes and their probability of occurrence. Using these failure modes and those of the components, conduct and FMEA of:

  - Lateral Flight Control System
  - Longitudinal Flight Control System
  - Directional Flight Control System

- o Provide results of above efforts to simulator personnel for use in simulating failures.

(d) Logistic Support - The publications, spares and GSE required to support the aircraft over a 10 year period will be determined.

B-2.2 DATA ACQUISITION SYSTEM - The data acquisition system for the aircraft will be built around a Teledyne Remote Multiplexer/Demultiplexer Unit (RMDU) Model AIFTDS-4000 and a Bell and Howell Airborne Tape Recorder, model MARS 1414 LT30. Both of these units, along with a suitable telemetry system and a time code generator, will be provided to MCAIR as GFE equipment by NASA Ames. The tasks described herein comprise the definition of the design, fabrication, installation, checkout and documentation of a data acquisition system incorporating the GFE components.

B-2.2.1 DESIGN - The data acquisition system will be developed to support the NASA flight development program. In order to ensure that all NASA requirements have been satisfied, a series of coordination interface meetings will be held at Ames with NASA project and instrumentation personnel to firm up measurand requirements and obtain familiarization with the Teledyne data acquisition system. A preliminary list of 90 measurands has been defined and coordinated with project and NASA personnel.

Procurement specifications will be generated to the extent that Instrumentation Equipment Drawings (IED) will be prepared for each deliverable piece of CFE that is to be delivered to MCAIR. Bid request and vendor coordination will be initiated. Vendor proposals will be evaluated. Delivery dates and quantities will be negotiated. Purchase orders will be prepared, vendors selected, and long lead purchases initiated. Unique signal conditioning components necessary to interface vendor supplied items with the Teledyne RMDU will be identified and designed during this phase of activity. These signal conditioners will be new units requiring electrical design and packaging support.

Instrumentation installation drawings to support the data acquisition requirements will be developed and coordinated with the cognizant Project Design Groups prior to being released to manufacturing. Installation drawings consist of mechanical and electrical interface definitions. New electrical specification drawings will be required for all 90 measurands. New mechanical drawings will be required for approximately 75 of the measurands. Engineering Orders (EOs) and Design Change Notices (DCNs) will be required for approximately 15 measurands.

An external instrumentation pod will be required to carry the GFE equipment. The design of the pod will be initiated using available conventional pod silhouettes. If available silhouettes are not satisfactory, a new configuration will be designed and detail drawings developed for delivery to NASA for manufacture. These design drawings will consist of new mechanical and electrical specifications. Internal shelf installation and equipment layouts will be designated. Electrical interfaces will be designed and appropriate interface connectors will be specified. It is anticipated that suitable qualification tests will be required of a new pod prior to release for carry on a Harrier aircraft. The centerline instrumentation pod will require provisions for cooling air during specific flight conditions. A new mechanical interface will be required utilizing bleed air from the Environmental Control System. An electrical-pneumatic control system will be designed to provide this capability. A flight test noseboom will be designed using data developed by MCAIR to support other Harrier related programs.

B-2.2.2 FABRICATION - All components necessary to support the data acquisition system will be fabricated using current MCAIR technology which has been proven on earlier programs. Fabrication of new electrical signal conditioning components will include development of engineering models for verification and proof fit. The instrumentation centerline pod will be fabricated by NASA Ames and will be delivered to MCAIR as GFE.

B-2.2.3 INSTALLATION - MCAIR will install all applicable instrumentation and interface wiring with the GFE system components. All system wiring will be checked out as an operating system.

B-2.2.4 CHECKOUT - The checkout of the data acquisition system will consist of acceptance tests of contractor and government furnished components, complete verification of system wiring and end to end confidence checks including simulated sensor inputs through suitable readout equipment. Verification of the total system will include end to end operation from the sensor to the tape recorder playback. All ground support equipment required to interrogate and decommutate the RMDU digital data will be provided as GFE from NASA Ames. Ground simulation testing will provide initial total system checkout. The aircraft evaluation measurands will be verified during the flight airworthiness testing conducted in St. Louis.

B-2.2.5 DOCUMENTATION - MCAIR will furnish two sets of system block diagrams, wire lists, mechanical installation drawings and test and operating procedures for the PCM system.

B-2.3 PROGRAMMABLE HEAD-UP DISPLAY (HUD) - The aircraft modifications, equipment modifications, and procurement required to incorporate the HUD are described below.

B-2.3.1 AIRCRAFT MODIFICATIONS - The HUD Display Waveform Generator (DWG) and Interface/Weapon Aiming Computer (I/WAC) are deleted and a Programmable Display Processor (PDP) installed. The wiring between the DWG and I/WAC is deleted and approximately 10 new wires are added between the PDP and various aircraft sensors. Digital interface wiring (2 shielded pairs) is added from the aircraft digital computer to the PDP and the existing I/WAC wiring is reterminated as required to be compatible with the PDP.

B-2.3.2 EQUIPMENT - The equipment to be procured comprises two PDPs (1 installed and 1 spare) one programming adapter, a PDP test set and twelve spare PROM elements. Because of their similarity with YAV-8B equipment, these items will be procured sole source from Smiths Industries, Incorporated because of their similarity with YAV-8B equipment. The PDP will be built and tested to MCAIR specification.

B-2.3.3 PROCUREMENT - The MCAIR procurement activity will include design specification generation, definition of the support requirements and data requirements list, and necessary RFP and PO activity.

B-2.3.4 SUPPORT - The PDP supplier will provide engineering support for the duration of the MCAIR modification effort. This support shall include:

- o Spare PDP piece parts
- o Maintenance
- o Programming assistance

B-2.3.5 TESTS - See Paragraph B-2.7.

B-2.3.6 SYMBOLOGY - The initial symbology programmed into the PDP will be based on YAV-8B symbology.

B-2.4 LANDING GUIDANCE SYSTEM - The following tasks described below will be performed. The equipment listed will be required.

B-2.4.1 AIRCRAFT MODIFICATIONS - The aircraft modifications consist of adding the angle receiver in the nose, a C-band antenna to the lower fuselage in the nose area, a DME indicator and MLS control in the cockpit, the DME interrogator to the equipment bay, and an L-band antenna to the lower aft fuselage. This stand alone system has no interface with the flight control system. It does interface via the digital adapter unit to the computer for the flight director computation and display on the ADI and head up display.

B-2.4.2 EQUIPMENT - The following WRAs (GFE) are required:

- MLS angle receiver
- Receiver mount
- MLS Control Panel
- C-Band antenna
- DME Interrogator
- DME Indicator
- L-Band antenna

B-2.4.3 DEVELOPMENT AND TEST REQUIREMENTS - There is no development requirement. Acceptance tests of WRAs prior to installation using delivered GSE and operational tests of installed equipment are described in Paragraph B-2.7.

B-2.5 COMPUTER SOFTWARE - Development of the computer software is dependent on NASA's furnishing MCAIR with the latest version of the HAL/S compiler. Required from the computer vendor will be the support software needed in program development such as the linkage editor, functional simulator, and utilities as well as preflight and inflight computer self-test program modules.

The Flight Control System software will be developed during the System Design Phase. The System Design Phase will be devoted to problem analysis, planning and establishing standards for subsequent software activities. Tasks to be accomplished during this phase are:

- Generate Integration Block Diagrams (IBDs).
- Extract major computation tasks from IBD's.
- Establish timing and program operation.
- Determine modes and mode switching requirements.
- Define logic for selection of secondary modes.
- Establish variables, iteration rates range scaling, format and engineering units for computer interface signals.
- Organize software standards.
- Generate test specifications and software design verification.

B-2.5.1 DEVELOPMENT OF PROGRAM MODULES - After the system design has been completed and the software requirements defined, the actual programming phases of the software development will begin. Top-level flow diagrams will be generated from the IBD's, the equations will be formulated then put into a form suitable for a digital computer. Detailed math flows will be converted. The flight program will be organized into functional modules and will be programmed in the HAL/S language. The program will be compiled and the program modules debugged.

The individual software module will be tested in an isolated environment before being combined with other modules. The objective is to determine that the module does its job as a black box, i.e., provides the proper outputs for a given set of inputs. The program module will be tested on an IBM 360/370 with a test driver which is a special test routine that provides the proper test environment by simulating inputs and outputs to the module.

The individual modules will then be tested on a host computer (IBM 360/370) using a functional simulator. This simulator is a computer program that provides a bit-by-bit simulation of the airborne computer instruction set while executing on the host computer. The functional simulator will be used in conjunction with a User's Control Program, a Pathfinder program and Dynamic Statistics Program, all of which are available from the F-15 program. These programs will be modified for use on the NASA V/STOL research aircraft. The software modules will then be modified based on the results of the instruction level simulator tests and MCAIR manned flight simulation.

B-2.5.2 MODULE - MODULE INTEGRATION - As groups of modules are completed and tested, they will be integrated. The module integration will be completed when the individual module source decks or tapes have been combined by the linkage editor into a single object deck or tape. This tape will include the inflight diagnostic support software which is furnished by the computer vendor.

The integrated modules will then be tested much as individual modules were tested, i.e., by using test drivers, and performing tests with the functional simulator and associated programs. The integration and testing will be continued at ever-higher program levels until the complete operational program is debugged and verified. An OFP tape will then be prepared in object computer (IBM-AP101) language for insertion into the AP-101 computer for on line testing.

B-2.5.3 ON-LINE SOFTWARE TEST - The flight computer program which was validated off-line will next be tested on-line in a Software Test Facility. The Software Test Facility will contain:

- A flight computer loaded with the flight program.
- An auxiliary computer loaded with the software test facility operating system program.
- Tapes containing software utility routines and an assembler for the auxiliary computer.
- Peripheral equipment which interfaces with the auxiliary computer.

B-2.5.4 HARDWARE/SOFTWARE BENCH INTEGRATION - The integrated bench test is described in Paragraph B-2.7.

B-2.5.5 SUPPORT SOFTWARE MODIFICATION - The support software which is used in flight program generation and verification and which must be modified is shown in Figure B-2.

B-2.6 SIMULATION - The simulation program will be conducted on the NASA Ames Flight Simulator for Advanced Aircraft (FSAA). It will evaluate control system modifications, associated flight safety aspects, and failure effects. This three phase simulation program will be conducted by NASA Ames with MCAIR technical assistance.

B-2.6.1 PHASE I - MCAIR will provide the software simulation model of the two place Harrier aircraft, including aerodynamics, propulsion, flight control and landing gear characteristics. MCAIR will provide software for the HUD and the Landing Guidance System and will flight test the aircraft model in MCAIR simulation facilities. NASA pilots will familiarize themselves with the handling qualities and flight characteristics of the aircraft on the MCAIR simulator. (A three-day flight simulation familiarization program for two Harrier-qualified NASA pilots is suggested.) Familiarization will include VTOs, hover, transition, STOs, SLs and high speed maneuvers. Ground effects, wind and turbulence will be simulated. Flights will be conducted with and without stability augmentation. The MCAIR model will be programmed on the CDC Cyber 175 computer. This phase will be of approximately 9 weeks' duration.

B-2.6.2 PHASE II - MCAIR will provide the basic simulation to NASA Ames in sufficient detail to permit conversion to the Honeywell Information Systems (HIS) Sigma 8 digital computer. Sufficient check cases will be furnished to permit verification of the model following conversion. MCAIR will provide technical assistance to NASA Ames for:

- Two place Harrier model conversion.
- Control system failure mode conversion.
- Software/hardware checkout.
- Determination of data requirements, formats and units.

NASA Ames will provide the FSAA simulation facility with two cockpits, one to simulate the Two Place Harrier front cockpit with standard Harrier flight controls and the other, the rear cockpit with the experimental flight controls. All hardware for the front cockpit such as the Harrier stick grip and throttle/nozzle control quadrant and the HUD will be provided by NASA Ames. Developmental hardware such as a thrust and nozzle servo control and sidestick controller, and HUD for the simulated rear cockpit will likewise be provided by NASA Ames. Conversion of the software package to the Sigma 8 digital computer will be the basic responsibility of NASA Ames, with MCAIR technical assistance. Check cases provided by MCAIR will be used to verify the software conversion including the control system failure effects. Use of a spare flight control research computer for onboard computations within the simulation is suggested. This phase will be about 6 weeks in duration.

FIGURE B-2

SUPPORT SOFTWARE MODIFICATION

	Function	Program Size	Per Cent Rework
STF Operating System	Provides real-time inputs to Airborne Computer from modeled aircraft peripheral systems previously recorded on mag tapes in time history format. Debugging facility for airborne computer program.	8000 words (24 BIT)	30
User Control Program (UCP)	Airborne Computer Functional Simulator Interface with Environment Program.	56K Bytes	50
Pathfind	Preprocessor to Compiler to source cards to floating point coding to enable path analysis	16.5K Bytes	50
Dynamic Statistics	Analysis of Trace Lines Output from UCP and Functional Simulator to give dynamic statistics.	29K Bytes	50
Data Base Catalog	Bookkeeping program for maintenance and sorting of OFP Data Base.	29K Bytes	20
Electrical Interface Program	Checks Airborne Computer I/O Interface with all peripherals.	119K Bytes	80

B-2.6.3 PHASE III - MCAIR will provide a simulation test plan for NASA use. The test plan will be balanced insofar as is possible for the effects of pilot learning, pilot fatigue, time of day, day of test and candidate order of appearance. This will permit a sufficient number of flight operating points to be gathered to ensure that sufficient data of quality will be generated to permit the determination of the statistical significance of results. NASA Ames will furnish Harrier qualified pilots, operating personnel and technicians. NASA Ames will generate and analyze the data. MCAIR will provide technical assistance for the simulation, reduction and analysis of data, and report preparation.

Phase III may be conducted as a continuous 8 week simulation or two 4 week simulations with sufficient time to analyze data and modify the control system.

B-2.7 GROUND TESTS - The ground tests which will be required include both airframe tests and experiment tests.

B-2.7.1 AIRFRAME - Ground vibration tests of the airframe will identify the structural modes of significance to control system analysis:

- o Fuselage first vertical bending mode.
- o Fuselage first lateral bending mode.
- o Wing first bending mode.
- o Vertical tail bending mode.
- o Stabilator bending mode.
- o Rudder rotation mode.
- o Stabilator rotation mode.

In conjunction with the ground vibration tests, tests will be performed to determine free play and rigidity and frequency response of the aileron, stabilator, and rudder. The weight and center of gravity of the aircraft will also be determined.

B-2.7.2 EQUIPMENT - The tasks required to perform the equipment tests include equipment acceptance tests, software development, integrated bench test, aircraft installation and checkout, and EMI tests (EMI survey only). The equipment tests will require MCAIR to design, develop, and fabricate three major test equipment items: a digital adapter test bench, a control system test bench and a flight line analyzer. These test benches will be delivered to NASA with the modified aircraft.

(a) Equipment Acceptance Tests - Except for the Digital Adapter (DA), all equipment acceptance tests will be performed using delivered GSE and test sets. Thus, only the DA and its cockpit controls will require a test bench. The equipment to be acceptance tested is that addressed above.

(b) Software Development - Software development is described in Section B-2.5.

(c) Integrated Bench Test - An integrated bench test will be performed on the avionics system to validate the flight control system on the bench prior to aircraft installation and flight. The integrated bench will have the following capabilities:

- o Interface capability with the MCAIR F-18 software test facility for the duration that the bench is at MCAIR.

- o The ability to interface with and perform integrated bench test on the following flight hardware:

  - Aircraft Digital Computer (AP-101)
  - Digital Adapter and Control
  - Control Surface Electromechanical Actuators

- o The ability to simulate:

  - Air Data Computer outputs
  - Inertial Navigation System outputs
  - Angle of attack
  - Rate gyros
  - Accelerometers
  - Control stick
  - Actuator dummy loads (if actuators not installed)

- o The ability to accept rate and acceleration inputs from rate gyros/accelerometers mounted on an adjacent rate table (the rate table is not deliverable as part of the bench).

- o The wiring and test points necessary to verify the microwave landing system (MLS) and programmable display processor (PDP) interface with the digital computer. The MLS and PDP equipment will be exercised using delivered GSE and test equipment.

The integrated bench test will be performed by integrating the bench with the MCAIR software test facility (STF). The STF will simulate the airborne system on the bench and simulate the equations of motion. It will also provide (displays and data) output for assessing system operation. The software required by the STF for the integrated bench test is shown in Figure B-2.

B-2.7.3 AIRCRAFT INSTALLATION AND CHECKOUT - A flight line analyzer (FLA) will be designed and fabricated to permit rapid flight control system checkout on the aircraft. It will provide open loop stimuli to the flight control system to the extent necessary to assess the integrity of the system for flight. System software checks performed by the FLA will be simplified tests such as memory scan checks. This equipment will be delivered to NASA with the aircraft.

B-2.7.4 EMI SURVEY - An EMI test shall be performed on the aircraft prior to flight. This test shall be the minimum necessary to assure that no EMI effects will compromise the performance of the flight control system.

B-2.8 AIRWORTHINESS TESTS - The functional test flights of the aircraft will be conducted at the MCAIR, St. Louis facility. Five flights will be performed over a one month period to accomplish the checkout of the basic aircraft control system and the airborne data acquisition system. The digital fly-by-wire flight control system will not be turned on during the MCAIR flight tests. The instrumentation ground support equipment, to be supplied as GFE, will be required at St. Louis during the flight test period and also during the preceding ground test of the fly-by-wire control system.

The functional checkout of the basic airplane will be accomplished in accordance with NAVAIR 01-AV8A-1F, NATOPS Functional Checkflight Checklist, modified as required to accommodate the aircraft configuration. Additional flights, over and above those normally required to complete the NATOPS checklist, will be required to evaluate the frictional effects of the fly-by-wire system servos attached to the basic flight control system. Maneuvers will be performed throughout the conventional flight envelope and in the V/STOL mode.

B-3 DUPLEX FLIGHT CONTROL SYSTEM

Task descriptions for implementing the duplex flight control system are given in this section. The duplex series digital fly-by-wire flight control system, data acquisition system, head up display, landing guidance system (MLS), simulation system software, ground tests and airworthiness tests are included.

B-3.1 CONTROL SYSTEMS MODIFICATIONS - The flight control system modifications include aircraft modifications and equipment requirements.

B-3.1.1 AIRCRAFT MODIFICATIONS - The following aircraft modifications will be required:

(a) Nose - Same as Paragraph B-2.1.1(a).

(b) Forward Cockpit - Same as Paragraph B-2.1.1(b).

(c) Aft Cockpit - Same as Paragraph B-2.1.1(c) except as follows:

- o Longitudinal Feel System - Same as the longitudinal feel system in Paragraph B-2.1.1(c) except the RVDT is dual and the stick is still connected to the control column.
- o Lateral Feel System - Same as the lateral feel system in Paragraph B-2.1.1(c) except the RVDT is dual and the stick is still connected to the control column.
- o Directional Feel System - Same as the directional feel system in Paragraph B-2.1.1(c) except the RVDT is dual and the pedals are still connected to the control column.

(d) Wing -

- o Revise the B276491 Aileron Control Lever to accept the output of the dual series servo unit.

- Add the dual series servo to the front spar.

(e) Equipment Bay - Same as Paragraph B-2.1.1(e).

(f) Aft Fuselage - Same as Paragraph B-2.1.1(f) except as follows:

- Longitudinal System

  - Remove the B278853 Control Pod
  - Add dual series servo actuator
  - Add walking beam bellcrank
  - Add two new control rods

- Directional System

  - Add two F-15A rudder actuators
  - Add new bellcrank
  - Add new pushrod
  - Power Amplifiers - Same as Paragraph B-2.1.1(f) except only 3 three power amplifiers are required.
  - Equipment Door - The second digital computer makes it necessary to remove the present airplane door 266408 and build a new door similar to the AV-8C flare and chaff door (75A338202).

(g) Top Center Fuselage - Same as Paragraph B-2.1.1(g).

(h) Hydraulics - Change engine gear box. Install Abex Pump 53227. Revise plumbing from pump to engine/airframe interface. Add hydrulic lines to connect hydraulic systems 1 and 2 to the pitch, roll, and yaw dual series servo actuators.

(i) Electrical System - The wiring which will be required for the basic avionics changes, the electrohydraulic series actuator, and the feel system actuators is as follows:

- 31 wires from the nose to the forward cockpit.
- 18 wires from the nose to the equipment bay area.
- 151 wires from the forward cockpit to the equipment bay area.
- 166 wires from the aft cockpit to the equipment bay area.
- 16 wires from the wing to the equipment bay area.
- 32 wires from the engine compartment to the equipment bay area.
- 1 coax from the equipment bay shelf to an added antenna located on the bottom of the aircraft just forward of the ventral fin.
- Approximately 270 wires will be added or reterminated within the equipment bay/aft fuselage area.
- Wiring not used will remain in the aircraft and be capped off.

The removal of equipment will delete 10 fuses from the distribution panel. The addition of equipment will add back 13 fuses.

B-3.1.2 EQUIPMENT REQUIREMENTS - The following tasks are required to acquire the digital flight control and electronic equipment for implementing the duplex flight control system:

(a) Flight Controls - Flight control system components which are to be purchased or manufactured by MCAIR are listed in Figure B-3. This list does not include purchased avionics which are discussed later. Development effort will be minimized by selecting components which can be qualified by similarity whenever possible, only acceptance tests being required of the supplier. Supplier tests are expected to be analog. Where qualified components cannot be obtained, a reduced qualification test program will be utilized which is informally agreed on by the supplier and MCAIR. A brief description of the components follows:

- Feel System Servoactuators - Same as Paragraph B-2.1.2(a).
- RVDTs and Strain Gages - Position and force transducers attached to the aft cockpit stick and rudder pedals will be dual channel.
- Force Switch Strain Gages - Same as Paragraph B-2.1.2(a).
- Series Servoactuator and Solenoid - The forward reaction control valve servoactuator and solenoid will be the same as for the simplex parallel configuration discussed in Paragraph B-2.1.2(a) except that it will be dual and will have full authority.
- Modified Pumps and Gear Box - It is anticipated that two modified pumps will be required, one for each system, plus a modified engine gearbox.
- Yaw Dual Series Servoactuator - The yaw axis dual series servoactuator is expected to consist of two F-15 rudder actuators. Control from the forward cockpit will be fully powered through the actuator.
- Pitch and Roll Dual Series Servoactuators - It is planned that the dual series servoactuators for pitch and roll will each use two F-4 lateral series servos tied together and modified for high response. Supplier tests are expected to be analog.
- Yaw Axis Trim Motor - A yaw axis trim motor will be required for the dual series configuration. Use of the F-15 rudder trim is planned.

(b) Avionics - The avionic equipment requirements for the dual fly-by-wire system are the same described in Paragraph B-2.1.2(b) except that additional avionics will be required:

- One airborne computer
- One digital adapter
- One rate gyro assembly (pitch, roll, yaw)

FIGURE B-3

DUAL SERIES FLIGHT CONTROL EQUIPMENT

NAME	STATUS	TYPE	MANUFACTURER	PART NUMBER	CHANNELS PER UNIT	NUMBER OF UNITS NUMB A/C	SPARE	WEIGHT
Feel System Servoactuator-Yaw	New	E/M E/M	MPC Products Plessey Dynamics	proposal #15974 TBD	1	1	1	5.6
Feel System Servoactuator-Pitch	New	E/M E/M	MPC Products Plessey Dynamics	#15974 TBD	1	1	1	6.8
Feel System Servoactuator-Roll	New	E/M	Plessey Dynamics	TBD				
System RVDTs	Existing			TBD	2	3	2	1.0
System Strain Gages	Existing		MDC	TBD	2	3	2	
Force Switch Strain Gages	Existing		MDC	TBD	1	3	2	
YAV-8B Series Servoactuator	Existing	E/M	MPC Products	1165-3-L59	2	1	1	2.8
YAV-8B Series Servoactuator Solenoid	Existing		LEDEX	184945-001	1	1	1	
Hydraulic Pump-Modified, PC-1					1	1		
Engine Gear Box-Modified		TBD			1	1		
Dual Series Servoactuator-Yaw	Existing	E/H	(F-15 Rudder Actuator) Ronson	3U3151-8	2	1	1	24.0
Dual Series Servoactuator-Pitch	Existing	E/H	(F-4 Lateral Series Servo) MOOG	PS 32-69054	1	2	1	12.4
Dual Series Servoactuator-Roll	Existing	E/H	MOOG	PS 32-69054	1	2	1	12.4
Trim Motor-Yaw	Existing	E/M	MPC Plessey DYN	Similar to 11650-3-L59 R15023M1-3 (F-15 Rudder Trim)	1	1	1	6.5

(a) Two units will be combined to form a dual unit

(c) Failure Mode and Effects Analysis - Same as Paragraph 2.1.2(c).

(d) Logistic Support - Same as Paragraph 2.1.2(d).

B-3.2 DATA ACQUISITION SYSTEM - Same as Paragraph B-2.2.

B-3.3 PROGRAMMABLE HEAD-UP DISPLAY - Same as Paragraph B-2.3.

B-3.4 LANDING GUIDANCE SYSTEMS - Same as Paragraph B-2.4.

B-3.5 COMPUTER SOFTWARE - The tasks required to generate the software for the dual configuration are identical to those for the simplex system, Paragraph B-2.5, except that the inflight integrity management module will be replaced with the redundancy management module. Also, there is a slight increase in BIT programming. The memory requirements of the Operating Flight Program are given in Figure B-4.

The support software generation task will be the same as that for the simplex configuration.

B-3.6 SIMULATION - Same as Paragraph B-2.6.

B-3.7 GROUND TESTS - The ground tests which will be required include both airframe and equipment tests.

B-3.7.1 AIRFRAME - Same as Paragraph B-2.7.1.

B-3.7.2 EQUIPMENT - This section defines only those items affected by duplex flight control implementation. Items not addressed in this section are unchanged from the simplex implementation addressed in Paragraph B-2.7.2 herein.

(a) Equipment Acceptance Tests - This item is the same as the equivalent item in Paragraph B-2.7.2 except that the equipment to be tested is that addressed in Paragraph B-3.1.2.

(b) Integrated Bench Test - The changes to this item from Paragraph B-2.7.2 for duplex implementation include making provisions for the following items:

- o Dual digital adapters
- o Dual AP-101 digital computers
- o Dual rate gyros
- o Dual electrohydraulic actuators

The test bench will be configured to perform integrated bench test on this duplex system in a manner similar to that described for the simplex system. Provision will be made to have a hydraulic power source adjacent to the integrated bench. This power source will not be a deliverable item.

B-3.8 AIRWORTHINESS TESTS - Same as Paragraph B-2.7.

FIGURE B-4

OPERATIONAL FLIGHT PROGRAM MEMORY REQUIREMENTS

- DUAL SYSTEM (HAL/S PROGRAMMING)

MEMORY WORDS - 16 BITS

MODULE	COMPUTER 1	COMPUTER 2
Control Laws	7200	7200
Redundancy Mgt.	6100	6100
Executive	4300	4300
BIT	1800	1800
I/O	500	500
	19900	19900
MLS/Flt. Director	2400	—
TOTAL	22300	19900

B-4 TRIPLEX FLIGHT CONTROL SYSTEM

Task descriptions for implementing the triplex flight control systems are given in this section. The triplex parallel digital fly-by-wire flight control system, data acquisition system, head up display, landing guidance system (MLS), system software, simulation, ground tests, and airworthiness tests are presented.

B-4.1 CONTROL SYSTEM MODIFICATIONS - The flight control system modifications include aircraft modifications and equipment requirements.

B-4.1.1 AIRCRAFT MODIFICATIONS - The following aircraft modifications will be required:

(a) Nose - Same as Paragraph B-2.1.1(a).

(b) Forward Cockpit - Same as Paragraph B-2.1.1(b).

(c) Aft Cockpit - Same as Paragraph B-2.1.1(c) except as follows:

- o Lateral Feel System - Same as Paragraph B-2.1.1(c) except that the RVDTs and force link are triplex.
- o Longitudinal Feel System - Same as Paragraph B-2.1.1(c) except that the RVDTs and force link are triplex.
- o Directional Feel System - Same as Paragraph B-2.1.1(c) except that the RVDTs and force link are triplex.

(d) Wing - Same as Paragraph B-2.1.1(d) except that lateral servo is triplex.

(e) Equipment Bay - Same as Paragraph B-2.1.1(e).

(f) Aft Fuselage - Same as Paragraph B-2.1.1(f) except that the longitudinal and lateral servos are triplex, there are 12 power amplifiers mounted on Frame 42, and the equipment door described in Paragraph B-2.1.1(f) is installed.

(g) Top Center Fuselage - Same as Paragraph B-2.1.1(g).

(h) Hydraulics - No hydraulic system modifications are required.

(i) Electrical System - Changes involving the avionics, the triplex electromechanical parallel actuators, and the feel system actuators are:

- o 31 wires from the nose to the forward cockpit.
- o 18 wires from the nose to the equipment bay area.
- o 6 wires from the nose to the gun pod containing the digital adapter.
- o 151 wires from the forward cockpit to the equipment bay area.
- o 18 wires from the forward cockpit to the gun pod containing the digital adapter.

- o 166 wires from the aft cockpit to the equipment bay area.
- o 56 wires from the aft cockpit to the gun pod containing the digital adapter.
- o 24 wires from the wing to the equipment bay area.
- o 12 wires from the wing to the gun pod containing the ditial the digital adapter.
- o 48 wires from the engine compartment to the equipment bay area.
- o 24 wires from the engine compartment to the gun pod containing the digital adapter unit.
- o 1 coax from the equipment bay shelf to the added antenna located on the bottom of the aircraft just forward of the ventral fin.
- o 58 wires between the left and right gun pods.

Approximately 300 wires will be added or reterminated within the equipment bay/aft fuselage area. Wiring not used will be capped and left in the aircraft.

The removal of equipment will delete 10 fuses in the distribution panel. The addition of equipment will add back 17 fuses.

(j) Gun Pod - The third computer requires the installation of a redesigned and rebuilt gun pod. To incorporate an equipment shelf within the gun pod for the required unit, a new mid section to replace the gun mounting cradle and a new mid fairing to replace the present cradle fairing will be necessary. The front and aft gun pod fairings will be adapted to the new mid section using similar attach points. The equipment mount will attach to the airplane at the same points as the gun mounting cradle did and the equipment will be installed inverted to simplify the mount structure.

B-4.1.2 EQUIPMENT - The tasks required to acquire the digital flight control and electronic equipment for implementing the triplex flight control system are described below.

(a) Flight Controls - Flight control system components which are to be purchased or manufactured by MCAIR are listed in Figure B-5. This list does not include purchased avionics which are discussed later. Development effort will be minimized by selecting components which can be qualified by similarity whenever possible, with only acceptance tests required of the supplier. Supplier tests are expected to be analog. Where qualified components cannot be obtained, a reduced qualification test program will be utilized which is informally agreed on by the supplier and MCAIR. A brief description of the components follows:

- o Feel System Servoactuators - Same as Paragraph B-2.1.2(a).

- o RVDTs and Strain Gages - Position and force transducers attached to the aft cockpit stick and rudder pedals are triplex.

- o Force Switch Strain Gages - Same as Paragraph B-2.1.2(a).

FIGURE B-5

TRIPLEX PARALLEL FLIGHT CONTROL EQUIPMENT

NAME	STATUS	TYPE	MANUFACTURER	PART NUMBER	CHANNELS PER UNIT	NUMBER OF UNITS A/C	NUMBER OF UNITS SPARE	WEIGHT
Feel System Servoactuator-Yaw	New	E/M	MPC Products	Proposal #15974	1	1	1	5.6
		E/M	Plessey Dynamics	TBD				
Feel System Servoactuator-Pitch	New	E/M	MPC Products	#15974	1	1	1	6.8
		E/M	Plessey Dynamics	TBD				
Feel System Servoactuator-Roll	New	E/M	MPC Products	#15974	1	1	1	2.7
		E/M	Plessey Dynamics	TBD				
System RVDTs	Existing			TBD	3	3	2	1.5
System Strain Gages	Existing		MDC	TBD	3	3	2	
Force Switch Strain Gages	Existing		MDC	TBD	1	3	2	
YAV-8B Series Servoactuator	Existing	E/M	MPC Products	11650-3-L59	1	1	1	2.8
YAV-8B Series Servoactuator Solenoid	Existing	LEDEX		184 945-001	1	1	1	
Emergency Spring Cartridge	New	MDC			1	3		4.5
Parallel Servoactuator-Yaw	New	E/M	MPC Products	TBD	3	1	1	10.0
		E/M	Plessey Dynamics					
Parallel Servoactuator-Pitch	New	E/M	MPC Products	Proposal No. #15978	3	1	1	10.0
		E/M	Plessey Dynamics					
Parallel Servoactuator-Roll	New	E/M	MPC Products	Proposal No. #15978	3	1	1	10.0
		E/M	Plessey Dynamics					

- Series Servoactuator and Solenoid - The forward reaction control valve servoactuator and solenoid will be the same as for the simplex parallel configuration discussed in Paragraph B-2.1.2(a).

- Emergency Spring Cartridges - Same as Paragraph B-2.1.2(a).

- Pitch, Roll and Yaw Servoactuators - The pitch, roll and yaw parallel electromechanical servoactuators are triplex.

(b) Avionics - The avionic equipment requirements for the triplex fly-by-wire system are as described in Paragraph B-2.1.2(b) except that additional avionics will be required:

- Two airborne computers.
- Two digital adapters.
- Two rate gyro assemblies (pitch, rate, and yaw).

(c) Failure Mode and Effects Analysis - Same as Paragraph 2.1.2(c).

(d) Logistics Support - Same as Paragraph 2.1.2(d).

B-4.2 DATA ACQUISITION SYSTEM - Same as Paragraph B-2.2.

B-4.3 PROGRAMMABLE HEAD-UP DISPLAY - Same as Paragraph B-2.3.

B-4.4 LANDING GUIDANCE SYSTEM - Same as Paragraph B-2.4.

B-4.5 SYSTEM SOFTWARE - The task complexity for providing software for the triplex configuration is the same as described for the dual configuration in Paragraph B-3.5. Support software requirements are the same as described for the simplex configuration in Paragraph B-2.5. The memory requirements of the Operating Flight Program are given in Figure B-6.

B-4.6 SIMULATION - Same as Paragraph B-2.6.

B-4.7 GROUND TESTS - The ground tests which will be required include both airframe tests and equipment tests.

B-4.7.1 AIRFRAME - Same as Paragraph B-2.3.1.

B-4.7.2 EQUIPMENT - This paragraph defines only those items affected by the triplex flight control system implementation. Items not addressed in this paragraph are unchanged from the simplex implementation addressed in Paragraph B-2.7.1 herein.

(a) Equipment Acceptance Tests - This item is the same as the equivalent item in Paragraph B-2.7.2 except that the equipment to be tested is that addressed in Paragraph B-4.1.2.

FIGURE B-6

OPERATIONAL FLIGHT PROGRAM MEMORY REQUIREMENTS

-TRIPLEX SYSTEM (HAL/S PROGRAMMING)

MODULE	MEMORY WORDS (16 bits)		
	COMPUTER 1	COMPUTER 2	COMPUTER 3
CONTROL LAWS	7200	7200	7200
REDUNDANCY MGT.	4100	4100	4100
EXECUTIVE	4300	4300	4300
BIT	2500	2500	2500
I/O	500	500	500
	18,600	18,600	18,600
MLS/FLT DIRECTOR	2400	-	-
	21,000	18,600	18,600

(b) Integrated Bench Test - The changes to this item for triplex implementation includes triple equipment as follows:

- o Three channel digital adapters
- o Three AP-101 digital computers
- o Three rate gyros
- o Three electromechanical actuators

The test bench will be configured to perform integrated bench test on this triplex system in a manner similar to that described for the simplex system.

B-4.8 AIRWORTHINESS TESTS - Same as Paragraph B-2.7.

B-5 LOW SPEED AIR DATA SYSTEM

B-5.1 AIRCRAFT MODIFICATION - One sensor will be mounted on the upper surface of the nose immediately forward of the forward canopy and the associated electronic unit will be mounted in the nose area. YAV-8B derived structural modifications for the containment of the sensor in the nose upper fuselage are directly applicable to the two place Harrier. The other sensor will be mounted on the left side of the nose in a position which is a mirror image of the location of the production angle of attack probe. Structural modifications for mounting this probe must be designed. Since both the YAV-8B and V/STOL research aircraft will have a flight test nose boom, the modifications related to the installation of the electronics units in the nose area of the YAV-8B will also be applicable to the V/STOL research aircraft.

B-5.2 EQUIPMENT REQUIRED - The equipment required is Applied Devices low range air data sensors and two electronics units. The sensor is three inches long, 1.25 inches in diameter and weighs 0.5 pound. The associated electronic unit is 3.5 inches by 2.5 inches by 10 inches and weighs 1.2 pound. The output of this system interfaces with the airborne computer. The system is CFE.

B-5.3 DEVELOPMENT AND TEST REQUIREMENTS - The sensor units and electronics will have been developed for the YAV-8B. However, unlike the YAV-8B the units must be interfaced to the flight control computer and head up display through the digital adapter unit. Thus, two hardware identical channels will have to be developed for the digital adapter unit. Precision calibration of the units will be required in the mini-model lab and tunnel. Analysis of local field effects such as the nose boom and sideslip effect on the sensor mounted on the port side of the aircraft must be determined.

B-5.4 SOFTWARE - Software for processing the raw sensor data must be developed.

B-5.5 SPARES - Two spare systems, consisting of one sensor and electronic unit each, will be required.

B-5.6 GSE - Available standard test equipment (multimeter, oscilloscope) will be used for checkout so no special GSE is required.

B-6 SIDE-ARM CONTROLLER

The tasks required to implement the side-arm controller are described below.

B-6.1 AIRCRAFT MODIFICATIONS - The side-arm controller will be installed in the aft cockpit in the right hand console area. The right hand console will be reworked to prepare for the side-arm controller by removing the ARC-114 radio, the ARC-150 radio and the voice recorder. (The ARC-150 is replaced by the D403 radio outside of the cockpit.) A cover plate is to be used until a side-arm controller is required. Some adjustment will be required to determine the optimum angles for pilot comfort.

B-6.2 EQUIPMENT REQUIREMENTS - The controller selected is the two axis, base pivot unit which was developed in the F-15 High Acceleration Cockpit (HAC) program. It has built-in electrical outputs through three separate wire bundle connectors. P/N 68-031012-101 is the F-15 HAC procurement specification prepared for such a unit but none has been purchased or qualified.

B-6.3 DEVELOPMENT AND TESTS - Qualification tests will be performed by the supplier.

B-6.4 SOFTWARE - The software for the sidestick should be essentially the same as the software for the center stick.

B-7 SIMPLEX THROTTLE AND NOZZLE SYSTEMS

The tasks required to implement the simplex parallel throttle and nozzle systems are described below.

B-7.1 AIRCRAFT MODIFICATIONS - The tasks required for the installation of the throttle control system are:

- o Remove link assembly.
- o Add force link.
- o Add servo to end of fuel control shaft.

The tasks required to revise the throttle control lever are:

- o Revise throttle lever to allow motion carry-through from forward cockpit.
- o Add RVDT.
- o Add disengage switch.

The tasks required for the installation of the nozzle control system are:

- o Revise quadrant assembly to accept parallel servo.
- o Add parallel servo actuator.
- o Remove B292227 connecting rod.
- o Add force link.

The tasks required to revise the nozzle control lever are:

- o Revise nozzle lever to allow motion carry-through from forward cockpit.
- o Add RVDT.
- o Add link.

B-7.2 EQUIPMENT REQUIRED - The equipment required for the nozzle and throttle systems is listed in Figure B-7.

B-7.3 DEVELOPMENT AND TESTS - The development and tests will be similar to those described for the simplex flight control system in Sections B-2.1.2(a), B-2.7.2(a), B-2.7.2(c), and B-2.7.3.

B-7.4 SOFTWARE - Position transducers on the throttle and nozzle levers will provide throttle and nozzle lever position information to the onboard digital computer. The computer will then calculate the command signals for the parallel servos attached to the fuel control unit and the air motor servo unit. Since MCAIR will provide the same throttle and nozzle system characteristics as the production Harrier software development will be minimal, essentially gains. Software for implementing more complex systems, such as independent altitude and velocity control, will be developed and tested by NASA after MCAIR delivers the aircraft.

B-8 DUPLEX THROTTLE AND NOZZLE SYSTEMS

The tasks required to implement the duplex series throttle and nozzle systems are described below.

B-8.1 AIRCRAFT MODIFICATIONS - The tasks required to make the installation of the dual throttle and control system are:

- o Remove B301672 idler lever.
- o Add walking beam bellcrank.
- o Add dual series servo.

The tasks required to revise the throttle lever are:

- o Revise throttle lever.
- o Add RVDTs.
- o Add link.
- o Add disengage switch.

The tasks required for the installation of the dual nozzle system are:

- o Remove A263618 control rod.
- o Add series actuator.
- o Add bellcrank.
- o Add walking beam bellcrank.

The tasks required to revise the throttle lever are:

- o Revise nozzle lever.
- o Add RVDTs.
- o Add link.

B-8.2 EQUIPMENT REQUIRED - The equipment required to implement the dual series throttle and nozzle systems is listed in Figure B-8.

B-8.3 DEVELOPMENT AND TESTS - The development and tests required will be similar to the development and tests required for the duplex fly-by-wire flight control system.

FIGURE B-7

SIMPLEX PARALLEL THROTTLE AND NOZZLE SYSTEMS EQUIPMENT

NAME	STATUS	TYPE	MANUFACTURER	PART NUMBER	CHANNELS PER UNIT	NUMBER OF UNITS A/C	SPARE	WEIGHT
Parallel Throttle Servoactuator	New	E/M E/M	MPC Products Plessey Dynamics-	Proposal #15975	1	1	1	2.8
Parallel Nozzle Servoactuator	New	E/M	MPC Products	#15976	1	1	1	2.8
Throttle RVDT	Existing				1	1	1	.2
Nozzle RVDT	Existing				1	1	1	.2

FIGURE B-8

DUAL SERIES THROTTLE AND NOZZLE SYSTEMS EQUIPMENT

NAME	STATUS	TYPE	MANUFACTURER	PART NUMBER	CHANNELS PER UNIT	NUMBER OF UNITS A/C	SPARE	WEIGHT
Dual Series Throttle Servoactuator	New	E/H	MOOG	32-69054	1	2	1	12.4
Dual Series Nozzle Servoactuator	New	E/H	MOOG	32-69054	1a	2	1	12.4
Existing					2	1	1	0.4
Existing					2	1	1	0.4

(a) two units will be combined to form a dual unit

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B-8.4 SOFTWARE - Same as Section B-7.4.

B-9 TRIPLEX THROTTLE AND NOZZLE SYSTEMS

The tasks involved in mechanizing the triplex parallel throttle and nozzle systems are described below.

B-9.1 AIRCRAFT MODIFICATIONS - The aircraft modifications are the same as described in Paragraph B-7.1, except that triplex actuators and RVDTs are used.

B-9.2 EQUIPMENT REQUIRED - The equipment required for the triplex parallel throttle and nozzle systems is given in Figure B-9.

B-9.3 DEVELOPMENT AND TESTS - The development and tests required will be similar to the development and tests required for the triplex fly-by-wire flight control systems.

B-9.4 SOFTWARE - Same as Section B-7.4

FIGURE B-9

TRIPLEX PARALLEL THROTTLE AND NOZZLE SYSTEMS EQUIPMENT

NAME	STATUS	TYPE	MANUFACTURER	PART NUMBER	CHANNELS PER UNIT	NUMBER OF UNITS A/C	NUMBER OF UNITS SPARE	WEIGHT
Parallel Throttle Servoactuator	New	E/M E/M	MPC Products Plessey Dynamics	Proposal #15975	3	1	1	4.0
Parallel Nozzle Servoactuator	New	E/M E/M	MPC Products Plessey Dynamics	Proposal #15976	3	1	1	4.0
Throttle RVDT	Existing				3	1	1	0.6
Nozzle RVDT	Existing				3	1	1	0.6

APPENDIX C

FAILURE MODE AND EFFECTS ANALYSIS

This appendix presents the Failure Mode and Effects Analysis (FMEA) performed for the NASA Two Place V/STOL Research Aircraft. Table C-1 presents the FMEA of the simplex system, Table C-2 presents the FMEA for the duplex system and Table C-3 presents that for the triplex system.

System FLT CONTROLS LATERAL/LONG/DIRECTIONAL
Subsystem Single Chan-Parallel Flight ____
Equipment ____ GSE ____
Module ____

FAILURE MODE AND EFFECTS ANALYSIS
NASA TWO PLACE V/STOL RESEARCH AIRCRAFT

TABLE C-1 FMEA, SIMPLEX SYSTEM

Page 1 of 26 Pages
Date ____
By ____
Approved ____

Item Identification			Reliability Logic Diagram Number	Function	FAILURE MODE	Operation Phase	Failure Effect On			Failure Detection Method	Corrective Action Time Available/Time Required	Design Provisions To Reduce Criticality	Hazard Classification/ Remarks
Name	Ident Number	Drawing Reference Designation					Component/ Functional Assembly	Next Higher Subsystem	Uppermost System				
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)
Computer (1)			S40.01	Accept Position and Force Transducer Inputs From Aft Stick/ Pedals Force Link Discretes from Fwd Stick/ Pedals Data Analogs from Motion Sensors and Air Data Computation. Provide Command Outputs for Parallel Servo and Aft Stick/ Pedals Variable Feel; Formatted Data for Displays. Provide Storage, Execute, and Decision Capability for BIT and IFIM	Incorrect Output (Program Fault or Hardware Malfunction) to Parallel Servo		Incorrect Input to Parallel Servo	Possible Erratic Operation of Control Surfaces	Sudden Unexpected Hardover Could Be Difficult to Recover	BIT Prior to Flt, Inflt Monitor and Pilot Observation When Airborne		Computer Self Test Capability Prior to and During Flight. Safety Pilot Can Take Control	III (At Altitude) IV (Low Level)
					Incorrect Output (Program Fault or Hardware Malfunction) to Variable Feel Device		Incorrect Input to Variable Feel Device	Incorrect Feel Forces on Aft Stick/ Pedals	Back Seat Pilot Would Have Tendency to Over-control or Under-control	Stick/Pedal Feel Unacceptable to Rear Pilot		Safety Pilot Can Take Over Control	I
					Incorrect Output (Program Fault or Hardware Malfunction) to BIT		Incorrect Data for BIT	Wrong Steps or Trip Levels	Most Likely Results is False BIT Failures, Some Possibility that a Bonifide Failure Would Go Undetected	Computer Self-Test		Diagnostic Routines for Computer Self-Check	I
					Incorrect Output to Displays		Incorrect Input to Displays	Wrong Data Analog Levels to Displays	Pilots Do Not Have Accurate Flt Control Status Info	Pilot Comparison with Other Instrumentation and Assessment of Aircraft's Response to Controls		Safety Pilot Can Fly Aircraft with Conventional Controls and Displays	I

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FAILURE MODE AND EFFECTS ANALYSIS
NASA TWO PLACE W/STOL RESEARCH AIRCRAFT

System FLT CONTROLS - LATERAL/LONG/DIRECTIONAL
Subsystem Single Chan-Parallel Flight
Equipment ____ GSE ____
Module ____

Date ____
By ____
Approved ____

TABLE C-1 CONT.

Item Identification: Name (1)	Ident Number (2)	Drawing Reference Designation (3)	Reliability Logic Diagram Number (4)	Function (5)	FAILURE MODE (6)	Operation Phase (7)	Failure Effect On: Component/Functional Assembly (8)	Next Higher Subsystem (9)	Uppermost System (10)	Failure Detection Method (11)	Corrective Action Time Available/Time Required (12)	Design Provisions To Reduce Criticality (13)	Hazard Classification/Remarks (14)
Motion Sensors			S40.02	Provide rotational and translational motion data to computer	Incorrect inputs to computer resulting in incorrect outputs to parallel servo and display		Incorrect Input to Parallel Servo	Possible Erratic Operation of Control Surfaces	Sudden Unexpected Hardover Could be Difficult to Recover	BIT Prior to Flt, Inflt Monitor and Pilot Observation When Airborne		BIT Test Capability Prior to flt. Safety Pilot Can Take Control	III (At Altitude) IV (Low Level)
							Incorrect Input to Displays	Wrong Data Analog Levels to Display	Pilots Do Not have accurate flt control Status info	Pilot Comparison with other Instrumentation and Assessment of A/C's Response to Controls		Safety Pilot Can Fly Aircraft with conventional Controls and Displays	II

FAILURE MODE AND EFFECTS ANALYSIS
NASA TWO PLACE W/STOL RESEARCH AIRCRAFT

TABLE C-1 CONT.

System FLT CONTROLS - LATERAL/LONGITUDINAL
Subsystem Single Chan-Parallel Flight ______
Equipment ______ GSE ______
Module ______

Date ______
By ______
Approved ______

Item Identification: Name (1)	Ident Number (2)	Drawing Reference Designation (3)	Reliability Logic Diagram Number (4)	Function (5)	FAILURE MODE (6)	Operation Phase (7)	Failure Effect On: Component/Functional Assembly (8)	Next Higher Subsystem (9)	Uppermost System (10)	Failure Detection Method (11)	Corrective Action Time Available/Time Required (12)	Design Provisions To Reduce Criticality (13)	Hazard Classification/Remarks (14)
Rear Stick			S40.03	Evaluation Pilot Input	Jammed at Pivot		Rear Stick Won't Move	No Input to Computer Roll Axis from Rear Seat.	No Lateral /Long. Control From Rear Seat.	None Needed - Readily Apparent to Pilot		Front Seat Can Fly Aircraft	III
					Broken Off at Pivot		Rear Stick Useless, Feel Sys Should Center Transducer	No Rear Seat Input to Computer	No Lateral /Long. Control from Rear Seat	None Needed Readily Apparent to Pilot		Safety Pilot Can Electrically Disengage Back Seat	III
					Loose at Pivot		No Solid Reaction Point for Rear Seat Input	Erratic Outputs from Rear Stick Transducer	Erratic Lateral/ Long. Control from Rear Seat	May Feel Loose to Evaluation Pilot - Both Pilots Should Note Erratic Movement of Aircraft		Safety Pilot Can Electrically Disengage Back Seat	III

System FLT CONTROLS - LATERAL/LONGITUDINAL
Sybsystem Single Chan-Parallel Flight
Equipment ______ GSE ______
Module ______

FAILURE MODE AND EFFECTS ANALYSIS

NASA TWO PLACE W/STOL RESEARCH AIRCRAFT

TABLE C-1 CONT.

Date ______
By ______
Approved ______

Item Identification			Reliability Logic Diagram Number	Function	FAILURE MODE	Operation Phase	Failure Effect On			Failure Detection Method	Corrective Action Time Available/Time Required	Design Provisions To Reduce Criticality	Hazard Classification/ Remarks
Name	Ident Number	Drawing Reference Designation					Component/ Functional Assembly	Next Higher Subsystem	Uppermost System				
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)
Fwd Stick			S40.04	Safety Pilot Override Input	Jammed at Pivot		Stick Won't Move	No Input to Computer or to Aileron and Stabilator Control Valves	No Long/ Lateral System Control with Either Stick	None Needed Readily Apparent to Pilots		Pilot Has Considerable Mechanical Advantage to Push through Some Potential Jams	IV
					Broken Off at Pivot		Fwd Stick Useless	No Front Seat Input to Computer or to Aileron/ Stab Control Valves	No Lat/ Long Sys Control from Front Seat	None Needed- Readily Apparent to Front Seat Pilot		Back Seat Can Fly Aircraft by Use of "Emergency" SW	III
					Loose at Pivot		No Solid Reaction Point for Fwd Pilot Input	Erratic or Sloppy Inputs to Long/ Lateral System	Erratic Lateral/ Long Control from Front Seat	None Needed, Readily Apparent to Pilot		Back Seat Can Fly Aircraft by Use of "Emergency" SW	III
Fwd Stick Paddle Switch			S40.041	Emergency Electrical Disengage of Back Seat Controls	Failed Closed		Can't Break Circuit		Loss of Stick-Mounted Emergency Disengage Capability	Pre-Flt Checks		Emergency Disengage Available Fwd Stick Force Link	II
					Failed Open		Circuit Won't MAke		Constant Disengage of Back Seat Controls	BIT, Pre-Flt Checks		Safety Pilot Can Fly Aircraft	II

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FAILURE MODE AND EFFECTS ANALYSIS
NASA TWO PLACE W/STOL RESEARCH AIRCRAFT

TABLE C-1 CONT.

System FLT CONTROLS – LATERAL/LONGITUDINAL
Subsystem Single Chan-Parallel Flight
Equipment ______ GSE ______
Module ______

Page 5 of 26 Pages
Date ______
By ______
Approved ______

Item Identification			Reliability Logic Diagram Number	Function	FAILURE MODE	Operation Phase	Failure Effect On			Failure Detection Method	Corrective Action Time Available/Time Required	Design Provisions To Reduce Criticality	Hazard Classification/ Remarks
Name	Ident Number	Drawing Reference Designation					Component/ Functional Assembly	Next Higher Subsystem	Uppermost System				
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)
Fwd Stick Force Link			S40.042	Provide Automatic Takeover by Safety Pilot	Jammed or Breakout Too High		Won't Switch at Specified Force	Automatic Override Inoperative	Safety Pilot Does Not Have Automatic Override Function	None Needed - Readily Apparent to Safety Pilot		Safety Pilot Can Use Paddle Switch to Disengage Rear Seat	II
					Breakout Too Low		Switches Below Spec Force	Undesired Disengagement of Evaluation Pilot's Controls	Nuisance Disengagements Would Invalidate Some Test Results	None Needed - Readily Apparent to Both Pilots		Ground Test and Adjustment or Replacement After Pilot Squawk	I

System FLT CONTROLS - LATERAL
Subsystem Single Chan-Parallel Flight ______
Equipment ______ GSE ______
Module ______

FAILURE MODE AND EFFECTS ANALYSIS
NASA TWO PLACE W/STOL RESEARCH AIRCRAFT

TABLE C-1 CONT.

Date ______
By ______
Approved ______

Item Identification			Reliability Logic Diagram Number	Function	FAILURE MODE	Operation Phase	Failure Effect On			Failure Detection Method	Corrective Action Time Available/Time Required	Design Provisions To Reduce Criticality	Hazard Classification/ Remarks
Name	Ident Number	Drawing Reference Designation					Component/ Functional Assembly	Next Higher Subsystem	Uppermost System				
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)
Aft Stick Position Transducer			S40.11	Provide Aft Stick Position Data to Computer	Open or Shorted		Distorted or No Output	Incorrect or No Aft Stick Position Data to Computer	Possible Reduction in Ability to Fly Aircraft from Back Seat	BIT, Inflt Monitor		Inflt Monitoring Backed Up by Force Transducer. Safety Pilot Can Disengage Back Seat and Fly Aircraft	II
Aft Stick Force Transducer			S40.12	Provide Aft Stick Force Data to Computer	Open or Shorted		Distorted or No Output	Incorrect or No Aft Stick Force Data to Computer	Possible Reduction in Ability to Fly Aircraft from Back Seat	BIT, Inflt Monitor		Inflight Monitoring, Backed Up by Position Transducer. Safety Pilot can Disengage Back Seat and Fly Aircraft	II
Aft Stick Variable Feel (Torque Motor)			S40.13	Provide Q-programmed Variable Feel to Rear Seat Stick	Jammed		Rear Stick Immobilized	No Stick Position Transducer Output	Possible Reduction in Ability to Fly Aircraft from Back Seat	BIT, Inflt Monitor		BIT, Inflt Monitor Force Transducer Still Operational. Safety Pilot can Disengage Back Seat and Fly Aircraft	II

(Continued Next Page)

System FLT CONTROLS - LATERAL
Subsystem Single Chan-Parallel Flight ____________
Equipment ____________ GSE ____________
Module ____________

FAILURE MODE AND EFFECTS ANALYSIS

NASA TWO PLACE W/STOL RESEARCH AIRCRAFT

TABLE C-1 CONT.

Date ____________
By ____________
Approved ____________

Item Identification			Reliability Logic Diagram Number	Function	FAILURE MODE	Operation Phase	Failure Effect On			Failure Detection Method	Corrective Action Time Available/Time Required	Design Provisions To Reduce Criticality	Hazard Classification/ Remarks
Name	Ident Number	Drawing Reference Designation					Component/ Functional Assembly	Next Higher Subsystem	Uppermost System				
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)
(Continued) Aft Stick Variable Feel (Torque Motor)			S40.13 (Continued)	(Continued) Provide Q-programmed Variable Feel to Rear Seat Stick	Shorted or Open		Little or No Resistance to Stick Movement	Little or No Force Transducer Output	Little or No Tactile Cues for Stick Positioning, Tendency to Over-control Via Position Sensor	None Needed-Readily Apparent to Back Seat Pilot		BIT, Inflt Monitor Position Transducer Still Operational Safety Pilot Can Disengage Back Seat and Fly Aircraft	II
Parallel Servo (Electrical)			S40.14E	Move Control Rod Linkage as Commanded by Computer in Response to Aft Stick Transducer Outputs	Jammed		Won't Move	No Response to Back Seat Pilot Commands	Back Seat Controls Useless, Front Seat Controls Stiffer than Usual, May Trip Circuit Breaker	Readily Apparent to Back Seat Pilot, BIT would Detect if Existing Prior to Flight		BIT, Safety Pilot Can Electrically Disengage Rear Controls and Fly Aircraft by Overpowering Parallel Servo Output Safety Spring Cartridge, Circuit Breaker	III

(Continued Next Page)

System FLT CONTROLS - LATERAL
Subsystem Single Chan-Parallel Flight__________
Equipment__________ GSE__________
Module__________

FAILURE MODE AND EFFECTS ANALYSIS

NASA TWO PLACE W/STOL RESEARCH AIRCRAFT

TABLE C-1 CONT.

Date__________
By__________
Approved__________

Item Identification			Reliability Logic Diagram Number	Function	FAILURE MODE	Operation Phase	Failure Effect On			Failure Detection Method	Corrective Action Time Available/Time Required	Design Provisions To Reduce Criticality	Hazard Classification/ Remarks
Name	Ident Number	Drawing Reference Designation					Component/ Functional Assembly	Next Higher Subsystem	Uppermost System				
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)
(Continued) Parallel Servo (Electrical)			S40.014E		(Continued) Burned Out or Shorted		Little or No Movement	Little or No Response to Back Seat Pilot Commands	Back Seat Controls Virtually Useless, Front Seat Controls May Be Stiffer than Usual If Shorted Will Probably Trip Circuit Breaker	Readily Apparent to Back Seat Pilot, Fwd Pilot Would Probably Notice Change in Feel of Stick		BIT Prior to Flt, Ckt Breaker to Prevent Overheat, Safety Pilot Can Electrically Disengage Rear Controls and Fly Aircraft by Overpowering Parallel Servo Spring Cartridge or Back-Driving the Parallel Servo	III
					Broken Rod End or Head End Attachment		No Reaction Point for Command Forces	No Output	Rear Seat Controls Useless	Problem Readily Apparent to Rear Pilot		Normal Operation from Front Seat with Rear Seat Controls Electrically Disengaged	III

System FLIGHT CONTROLS- LATERAL

Subsystem Single Chan-Parallel Flight____________

Equipment____________ GSE____________

Module____________

FAILURE MODE AND EFFECTS ANALYSIS

NASA TWO PLACE W/STOL RESEARCH AIRCRAFT

TABLE C-1 CONT.

Date____________

By____________

Approved____________

Item Identification			Reliability Logic Diagram Number	Function	FAILURE MODE	Operation Phase	Failure Effect On			Failure Detection Method	Corrective Action Time Available/Time Required	Design Provisions To Reduce Criticality	Hazard Classification/ Remarks
Name	Ident Number	Drawing Reference Designation					Component/ Functional Assembly	Next Higher Subsystem	Uppermost System				
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)
Parallel Servo Safety Spring Cartridge			S40.15	Couple Parallel Servo to Mech Control System and Provide Capabilities for Mech Sys to Override Electrical System	Jammed		Cartridge Won't Collapse or Extend	Direct Link Between Servo and Mech System	Loss of Direct Mechanical Override Protection No Effect Unless Coupled with an Electrical System Failure			Single Jam in Cartridge not Significant a Second Failure in Back Seat Controls Required to Cause Problems.	II
					Broken Spring		No Resistance in One Direction	Servo Could Not Move Linkage in One Direction	Problems Would Be Noted by Pilot as Erratic Response to Back Seat Commands	BIT Prior to Flight Pilot Observation in Flight		Safety Pilot Can Electrically Disengage Back Seat Controls and Fly Aircraft From Front Seat	II
					Broken Attachments		No Resistance in Either Direction	Servo Could Not Move Linkage in Either Direction	No Response to Back Seat Commands	Readily Apparent to Back Seat Pilot		Safety Pilot Can Electrically Disengage Back Seat Controls and Fly Aircraft from Front Seat	III

FAILURE MODE AND EFFECTS ANALYSIS

NASA TWO PLACE W/STOL RESEARCH AIRCRAFT

TABLE C-1 CONT.

System FLIGHT CONTROLS - LATERAL
Sybsystem Single Chan-Parallel Flight ______
Equipment ______ GSE ______
Module ______

Date ______
By ______
Approved ______

Item Identification			Reliability Logic Diagram Number	Function	FAILURE MODE	Operation Phase	Failure Effect On			Failure Detection Method	Corrective Action Time Available/Time Required	Design Provisions To Reduce Criticality	Hazard Classification/ Remarks
Name	Ident Number	Drawing Reference Designation					Component/ Functional Assembly	Next Higher Subsystem	Uppermost System				
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)
Front Seat (Mechanical) Trim			S40.16	Reacts to Pilot Beep Switch Command to Trim Ailerons by Changing Stick Neutral Point	Open		Winding Burnout	Trim Motor Won't Run	No Mechanical Trim Capability, Trim Fixed at One Position			Pilot Can Hold in Trim with Stick, Rear Seat Pilot Can Put in Trim Bias Electrically	II
					Short		Shorted Winding	Sluggish Operation, May Trip Circuit Breaker	Poor or No Mechanical Trip Operation			Circuit Breaker to Prevent Overheat, Rear Seat Pilot Can Put in Trim Bias Electrically	II
					Jammed, or Stripped Gearing		Jammed or Stripped	No Trim Output	Loss of Mech Trim Capability, Trim Fixed at One Position			Pilot Can Hold in Trim with Stick, Rear Seat Pilot Can Put in Trim Bias Electrically	II
					Loss of Structural Ground (Attachments Broken)		No Reaction Point for Trim Motor Torque	No Trim Motion	Loss of Mechanical Trim Capability			Pilot Can Hold in Trim with Stick, Rear Seat Pilot Can Put in Trim Bias Electrically	II

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FAILURE MODE AND EFFECTS ANALYSIS

NASA TWO PLACE W/STOL RESEARCH AIRCRAFT

TABLE C-1 CONT.

System FLT. CONTROLS - LATERAL
Subsystem Single Chan-Parallel Flight ______
Equipment ______ GSE ______
Module ______

Date ______
By ______
Approved ______

Item Identification			Reliability Logic Diagram Number	Function	FAILURE MODE	Operation Phase	Failure Effect On			Failure Detection Method	Corrective Action Time Available/Time Required	Design Provisions To Reduce Criticality	Hazard Classification/ Remarks
Name	Ident Number	Drawing Reference Designation					Component/ Functional Assembly	Next Higher Subsystem	Uppermost System				
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)
Front Stick Feel Spring			S40.17	Provides Artificial Feel for Fwd Pilot	Jammed		Stick Won't Move	No Input to Computer or to Aileron Control Valves	No Lateral System Control with Either Stick	None Needed-Readily Apparent to Pilots		Pilot Has Considerable Mechanical Advantage to Push through Some Potential Jams	IV
					Broken Spring		Absence of Feel and Trim in One Direction	Unbalanced Feel Forces, Erratic Trim	Difficult to Fly Aircraft with Front Stick	None Needed-Readily Apparent to Pilot		With Hands Off in Front Seat, Back Seat Pilot Can Fly Aircraft Electrically. "Emergency" SW Can be Used to Prevent DFCS Disengage.	III
					Broken Head End or Rod End Attachments		Absence of Feel Forces and Trim Position in Both Directions	Loss of Tactile Cues for Proper Stick Positioning	Difficult to Fly Aircraft with Front Stick	None Needed-Readily Apparent to Pilot		With Hands Off in Front Seat, Back Seat Pilot Can Fly Aircraft Electrically. "Emergency" SW Can Be Used to Prevent DFCS Disengage.	III

System FLT CONTROLS - LONGITUDINAL
Subsystem Single Chan-Parallel Flight
Equipment ____ GSE ____
Module ____

FAILURE MODE AND EFFECTS ANALYSIS
NASA TWO PLACE W/STOL RESEARCH AIRCRAFT

Date ____
By ____
Approved ____

TABLE C-1 CONT.

Item Identification			Reliability Logic Diagram Number	Function	FAILURE MODE	Operation Phase	Failure Effect On			Failure Detection Method	Corrective Action Time Available/Time Required	Design Provisions To Reduce Criticality	Hazard Classification/ Remarks
Name	Ident Number	Drawing Reference Designation					Component/ Functional Assembly	Next Higher Subsystem	Uppermost System				
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)
Aft Stick Position Transducer			S40.21	Provide Aft Stick Position Data to Computer	Open or Shorted		Distorted or No Output	Incorrect or No Aft Stick Position Data to Computer	Possible Reduction in Ability to Fly Aircraft from Back Seat	BIT, Inflt Monitor		Inflt Monitoring Backed Up by Force Transducer. Safety Pilot Can Disengage Back Seat and Fly Aircraft	II
Aft Stick Force Transducer			S40.22	Provide Aft Stick Force Data to Computer	Open or Shorted		Distorted or No Output	Incorrect or No Aft Stick Force Data to Computer	Possible Reduction in Ability to Fly Aircraft from Back Seat	BIT, Inflt Monitor		Inflight Monitoring, Backed Up by Position Transducer. Safety Pilot can Disengage Back Seat and Fly Aircraft	II
Aft Stick Variable Feel (Torque Motor)			S40.23	Provide Q-programmed Variable Feel to Rear Seat Stick	Jammed		Rear Stick Immobilized	No Stick Position Transducer Output	Possible Reduction in Ability to Fly Aircraft from Back Seat	BIT, Inflt Monitor		BIT, Inflt Monitor Force Transducer Still Operational. Safety Pilot can Disengage Back Seat and Fly Aircraft	II

(Continued Next Page)

FAILURE MODE AND EFFECTS ANALYSIS

NASA TWO PLACE W/STOL RESEARCH AIRCRAFT

TABLE C-1 CONT.

System FLT CONTROLS - LONGITUDINAL
Subsystem Single Chan-Parallel Flight
Equipment GSE
Module

Date
By
Approved

Item Identification			Reliability Logic Diagram Number	Function	FAILURE MODE	Operation Phase	Failure Effect On			Failure Detection Method	Corrective Action Time Available/Time Required	Design Provisions To Reduce Criticality	Hazard Classification/ Remarks
Name	Ident Number	Drawing Reference Designation					Component/ Functional Assembly	Next Higher Subsystem	Uppermost System				
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)
(Continued) Aft Stick Variable Feel (Torque Motor)			S40.23 (Continued)	(Continued) Provide Q-programmed Variable Feel to Rear Seat Stick	Shorted or Open		Little or No Resistance to Stick Movement	Little or No Force Transducer Output	Little or No Tactile Cues for Stick Positioning, Tendency to Over-control Via Position Sensor	None Needed-Readily Apparent to Back Seat Pilot		BIT, Inflt Monitor Position Transducer Still Operational Safety Pilot Can Disengage Back Seat and Fly Aircraft	II
Parallel Servo (Electrical)			S40.24E	Move Control Rod Linkage as Commanded by Computer in Response to Aft Stick Transducer Outputs	Jammed		Won't Move	No Response to Back Seat Pilot Commands	Back Seat Controls Useless, Front Seat Controls Stiffer than Usual, May Trip Circuit Breaker	Readily Apparent to Back Seat Pilot, BIT would Detect if Existing Prior to Flight		BIT, Safety Pilot Can Electrically Disengage Rear Controls and Fly Aircraft by Overpowering Parallel Servo Output Safety Spring Cartridge, Circuit Breaker	III

(Continued Next Page)

FAILURE MODE AND EFFECTS ANALYSIS

NASA TWO PLACE W/STOL RESEARCH AIRCRAFT

TABLE C-1 CONT.

System FLT CONTROLS - LONGITUDINAL

Sybsystem Single Chan-Parallel Flight ______

Equipment ______ GSE ______

Module ______

Date ______

By ______

Approved ______

Item Identification			Reliability Logic Diagram Number	Function	FAILURE MODE	Operation Phase	Failure Effect On			Failure Detection Method	Corrective Action Time Available/Time Required	Design Provisions To Reduce Criticality	Hazard Classification/ Remarks
Name	Ident Number	Drawing Reference Designation					Component/ Functional Assembly	Next Higher Subsystem	Uppermost System				
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)
(Continued) Parallel Servo (Electrical)			S40.14E		(Continued) Burned Out or Shorted		Little or No Movement	Little or No Response to Back Seat Pilot Commands	Back Seat Controls Virtually Useless, Front Seat Controls May Be Stiffer than Usual If Shorted Will Probably Trip Circuit Breaker	Readily Apparent to Back Seat Pilot, Fwd Pilot Would Probably Notice Change in Feel of Stick		BIT Prior to Flt, Ckt Breaker to Prevent Overheat, Safety Pilot Can Electrically Disengage Rear Controls and Fly Aircraft by Overpowering Parallel Servo Spring Cartridge or Back-Driving the Parallel Servo	III
					Broken Rod End or Head End Attachment		No Reaction Point for Command Forces	No Output	Rear Seat Controls Useless	Problem Readily Apparent to Rear Pilot		Normal Operation from Front Seat with Rear Seat Controls Electrically Disengaged	III

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221

System FLIGHT CONTROLS - LONGITUDINAL
Subsystem Single Chan-Parallel Flight
Equipment ______ GSE ______
Module ______

FAILURE MODE AND EFFECTS ANALYSIS
NASA TWO PLACE W/STOL RESEARCH AIRCRAFT

TABLE C-1 CONT.

Date ______
By ______
Approved ______

Item Identification			Reliability Logic Diagram Number	Function	FAILURE MODE	Operation Phase	Failure Effect On			Failure Detection Method	Corrective Action Time Available/Time Required	Design Provisions To Reduce Criticality	Hazard Classification/ Remarks
Name	Ident Number	Drawing Reference Designation					Component/ Functional Assembly	Next Higher Subsystem	Uppermost System				
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)
Parallel Servo Safety Spring Cartridge			S40.25	Couple Parallel Servo to Mech Control System and Provide Capabilities for Mech Sys to Override Electrical System	Jammed		Cartridge Won't Collapse or Extend	Direct Link Between Servo and Mech System	Loss of Direct Mechanical Override Protection No Effect Unless Coupled with an Electrical System Failure			Single Jam in Cartridge not Significant a Second Failure in Back Seat Controls Required to Cause Problems.	II
					Broken Spring		No Resistance in One Direction	Servo Could Not Move Linkage in One Direction	Problems Would Be Noted by Pilot as Erratic Response to Back Seat Commands	BIT Prior to Flight Pilot Observation in Flight		Safety Pilot Can Electrically Disengage Back Seat Controls and Fly Aircraft From Front Seat	II
					Broken Attachments		No Resistance in Either Direction	Servo Could Not Move Linkage in Either Direction	No Response to Back Seat Commands	Readily Apparent to Back Seat Pilot		Safety Pilot Can Electrically Disengage Back Seat Controls and Fly Aircraft from Front Seat	III

FAILURE MODE AND EFFECTS ANALYSIS

NASA TWO PLACE W/STOL RESEARCH AIRCRAFT

TABLE C-1 CONT.

System FLIGHT CONTROLS - LONGITUDINAL
Subsystem Single Chan-Parallel Flight
Equipment
GSE
Module

Date
By
Approved

Item Identification: Name (1)	Ident Number (2)	Drawing Reference Designation (3)	Reliability Logic Diagram Number (4)	Function (5)	FAILURE MODE (6)	Operation Phase (7)	Failure Effect On: Component/Functional Assembly (8)	Next Higher Subsystem (9)	Uppermost System (10)	Failure Detection Method (11)	Corrective Action Time Available/Time Required (12)	Design Provisions To Reduce Criticality (13)	Hazard Classification/Remarks (14)
Front Seat (Mechanical) Trim			S40.26	Reacts to Pilot Beep Switch Command to Trim Stabilator by Changing Stick Neutral Point	Open		Winding Burnout	Trim Motor Won't Run	No Mechanical Trim Capability, Trim Fixed at One Position			Pilot Can Hold in Trim with Stick, Rear Seat Pilot Can Put in Trim Bias Elect-ically	II
					Short		Shorted Winding	Sluggish Operation, May Trip Circuit Breaker	Poor or No Mechanical Trip Operation			Circuit Breaker to Prevent Overheat, Rear Seat Pilot Can Put in Trim Bias Electrically	II
					Jammed or Stripped Gearing		Jammed or Stripped	No Trim Output	Loss of Mech Trim Capability Trim Fixed at One Position			Pilot Can Hold in Trim with Stick, Rear Seat Pilot Can Put in Trim Bias Electrically	II
					Loss of Structural Ground (Attachments Broken)		No Reaction Point for Trim Motor Torque	No Trim Motion	Loss of Mechanical Trim Capability			Pilot Can Hold in Trim with Stick, Rear Seat Pilot Can Put in Trim Bias Electrically	II

FAILURE MODE AND EFFECTS ANALYSIS

NASA TWO PLACE W/STOL RESEARCH AIRCRAFT

TABLE C-1 CONT.

System FLT. CONTROLS - LONGITUDINAL

Subsystem Single Chan-Parallel Flight

Equipment ______ GSE ______

Module ______

Date ______

By ______

Approved ______

Item Identification			Reliability Logic Diagram Number	Function	FAILURE MODE	Operation Phase	Failure Effect On			Failure Detection Method	Corrective Action Time Available/Time Required	Design Provisions To Reduce Criticality	Hazard Classification/ Remarks
Name	Ident Number	Drawing Reference Designation					Component/ Functional Assembly	Next Higher Subsystem	Uppermost System				
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)
Front Stick Feel Spring			S40.27	Provides Artificial Feel for Fwd Pilot	Jammed		Stick Won't Move	No Input to Computer or to Stabilator Control Valves	No Pitch System Control with Either Stick	None Needed-Readily Apparent to Pilots		Pilot Has Considerable Mechanical Advantage to Push through Some Potential Jams	IV
					Broken Spring		Absence of Feel and Trim in One Direction	Unbalanced Feel Forces, Erratic Trim	Difficult to Fly Aircraft with Front Stick	None Needed-Readily Apparent to Pilot		With Hands Off in Front Seat, Back Seat Pilot Can Fly Aircraft Electrically. "Emergency" SW Can be Used to Prevent DFCS Disengage.	III
					Broken Head End or Rod End Attachments		Absence of Feel Forces and Trim Position in Both Directions	Loss of Tactile Cues for Proper Stick Positioning	Difficult to Fly Aircraft with Front Stick	None Needed-Readily Apparent to Pilot		With Hands Off in Front Seat, Back Seat Pilot Can Fly Aircraft Electrically. "Emergency" SW Can Be Used to Prevent DFCS Disengage.	III

FAILURE MODE AND EFFECTS ANALYSIS

NASA TWO PLACE W/STOL RESEARCH AIRCRAFT

TABLE C-1 CONT.

System FLT CONTROLS – LONGITUDINAL

Subsystem Single-Chan Parallel Flight ______

Equipment Pitch Axis GSE ______

Module ______

Date ______

By ______

Approved ______

Item Identification: Name (1)	Ident Number (2)	Drawing Reference Designation (3)	Reliability Logic Diagram Number (4)	Function (5)	FAILURE MODE (6)	Operation Phase (7)	Failure Effect On: Component/Functional Assembly (8)	Next Higher Subsystem (9)	Uppermost System (10)	Failure Detection Method (11)	Corrective Action Time Available/Time Required (12)	Design Provisions To Reduce Criticality (13)	Hazard Classification/Remarks (14)
Forward RCS Valve Series Actuator			S40.28	Provide SAS Capability on front RCS Valve	Jammed to bellcrank		Won't move relative to bellcrank	Solid link Stick to RCS Valve	Loss of front RCS SAS. Valve still operational	BIT, in-flt monitor		A/C still completely flyable with increase pilot work-load. Some pitch RCS SAS still available via rear RCS Valve. No interferences with conventional landing.	II
					broken, burned out or shorted Which Allows Free Movement of Actuator		Servo inop	No transmittal of pilot command	loss of use of front RCS Valve. VTOL operations difficult with fwd C.G. Aircraft	BIT, in-flight monitor		BIT or in-flt monitor warns pilot not to attempt vertical operations with failed actuator. No interference with conventional landing.	II

FAILURE MODE AND EFFECTS ANALYSIS

NASA TWO PLACE W/STOL RESEARCH AIRCRAFT

TABLE C-1 CONT.

System FLT CONTROLS- DIRECTIONAL
Subsystem Single Chan-Parallel Flight
Equipment ______ GSE ______
Module ______

Date ______
By ______
Approved ______

Item Identification			Reliability Logic Diagram Number	Function	FAILURE MODE	Operation Phase	Failure Effect On			Failure Detection Method	Corrective Action Time Available/Time Required	Design Provisions To Reduce Criticality	Hazard Classification/ Remarks
Name	Ident Number	Drawing Reference Designation					Component/ Functional Assembly	Next Higher Subsystem	Uppermost System				
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)
Fwd Pedals			S40.38	Safety Pilot Override Input	Jammed at Pivot		Pedals Won't Move	No Input to Computer or to Rudder	No Directional System Control	None Needed Readily Apparent to Pilots		Pilot Has Considerable Mechanical Advantage to Push through Some Potential Jams	III (At Altitude) IV (Low Level)
					Broken Off at Pivot		Fwd Pedals Useless	No Front Seat Input to Computer or to Rudder	No Directional Sys Control from Front Seat	None Needed-Readily Apparent to Front Seat Pilot		Back Seat Can Fly Aircraft by Use of "Emergency" SW	III
					Loose at Pivot		No Solid Reaction Point for Fwd Pilot Input	Erratic or Sloppy Inputs to Directional System	Erratic Control from Front Seat	None Needed, Readily Apparent to Pilot		Back Seat Can Fly. Aircraft by Use of "Emergency" SW	III

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System FLT CONTROLS-DIRECTIONAL
Subsystem Single Chan-Parallel   Flight ________
Equipment ________   GSE ________
Module ________

FAILURE MODE AND EFFECTS ANALYSIS
NASA TWO PLACE W/STOL RESEARCH AIRCRAFT

TABLE C-1 CONT.

Date ________
By ________
Approved ________

Item Identification: Name	Item Identification: Ident Number	Item Identification: Drawing Reference Designation	Reliability Logic Diagram Number	Function	FAILURE MODE	Operation Phase	Failure Effect On: Component/Functional Assembly	Failure Effect On: Next Higher Subsystem	Failure Effect On: Uppermost System	Failure Detection Method	Corrective Action Time Available/Time Required	Design Provisions To Reduce Criticality	Hazard Classification/ Remarks
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)
Fwd Pedal Force Link			S40.39	Provide Automatic Takeover by Safety Pilot	Jammed or Breakout Too High		Won't Switch at Specified Force	Automatic Override Inoperative	Safety Pilot Does Not Have Automatic Override Function	None Needed - Readily Apparent to Safety Pilot		Safety Pilot Can Use Paddle Switch to Disengage Rear Seat	II
					Breakout Too Low		Switches Below Spec Force	Undesired Disengagement of Evaluation Pilot's Controls	Nuisance Disengagements Would Invalidate Some Test Results	None Needed - Readily Apparent to Both Pilots		Ground Test and Adjustment or Replacement After Pilot Squawk	I

System FLT. CONTROLS-DIRECTIONAL
Sybsystem Single Chan-Parallel Flight
Equipment
Module
CSE

FAILURE MODE AND EFFECTS ANALYSIS
NASA TWO PLACE W/STOL RESEARCH AIRCRAFT

TABLE C-1 CONT.

Date
By
Approved

Item Identification: Name	Ident Number	Drawing Reference Designation	Reliability Logic Diagram Number	Function	FAILURE MODE	Operation Phase	Failure Effect On: Component/Functional Assembly	Next Higher Subsystem	Uppermost System	Failure Detection Method	Corrective Action Time Available/Time Required	Design Provisions To Reduce Criticality	Hazard Classification/Remarks
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)
Front Pedal Feel Spring			S40.37	Provides Artificial Feel for Fwd Pilot	Jammed		Pedals Won't Move	No Input to Computer or to Rudder Control Valves	No Directional System Control with Fwd or Aft Pedals	None Needed-Readily Apparent to Pilots		Pilot Has Considerable Mechanical Advantage to Push through Some Potential Jams	III (At Altitude) IV (Low Level)
					Broken Spring		Absence of Feel in One Direction	Unbalanced Feel Forces.	Difficult to Fly Aircraft with Front Pedals	None Needed-Readily Apparent to Pilot		With Hands Off in Front Seat, Back Seat Pilot Can Fly Aircraft Electrically. "Emergency" SW Can be Used to Prevent DFCS Disengage.	III
					Broken Head End or Rod End Attachments		Absence of Feel Forces in Both Directions	Loss of Tactile Cues for Proper Pedal Positioning	Difficult to Fly Aircraft with Front Pedals	None Needed-Readily Apparent to Pilot		With Hands Off in Front Seat, Back Seat Pilot Can Fly Aircraft Electrically. "Emergency" SW Can Be Used to Prevent DFCS Disengage.	III

FAILURE MODE AND EFFECTS ANALYSIS

NASA TWO PLACE W/STOL RESEARCH AIRCRAFT

TABLE C-1 CONT.

System FLT CONTROLS - DIRECTIONAL
Subsystem Single Chan-Parallel Flight ______
Equipment ______ GSE ______
Module ______

Date ______
By ______
Approved ______

Item Identification: Name (1)	Ident Number (2)	Drawing Reference Designation (3)	Reliability Logic Diagram Number (4)	Function (5)	FAILURE MODE (6)	Operation Phase (7)	Failure Effect On: Component/Functional Assembly (8)	Next Higher Subsystem (9)	Uppermost System (10)	Failure Detection Method (11)	Corrective Action Time Available/Time Required (12)	Design Provisions To Reduce Criticality (13)	Hazard Classification/Remarks (14)
Rear Pedals			S40.38	Evaluation Pilot Input	Jammed at Pivot		Rear Pedals Won't Move	No Input to Computer Yaw Axis from Rear Seat.	No Directional Control from Rear Seat.	None Needed - Readily Apparent to Pilot		Front Seat Can Fly Aircraft	III
					Broken Off at Pivot		Rear Pedals Useless, Feel Sys Should Center Transducer	No Rear Seat Input to Computer	No Directional Control from Rear Seat.	None Needed Readily Apparent to Pilot		Safety Pilot Can Electrically Disengage Back Seat	III
					Loose at Pivot		No Solid Reaction Point for Rear Seat Input	Erratic Outputs from Rear Pedal Transducer	Erratic Directional Control from Rear Seat	May Feel Loose to Evaluation Pilot - Both Pilots Should Note Erratic Movement of Aircraft		Safety Pilot Can Electrically Disengage Back Seat	III

System FLT CONTROLS - DIRECTIONAL

Subsystem Single Chan-Parallel Flight

Equipment ________ GSE ________

Module ________

FAILURE MODE AND EFFECTS ANALYSIS

NASA TWO PLACE W/STOL RESEARCH AIRCRAFT

TABLE C-1 CONT.

Date ________

By ________

Approved ________

Item Identification			Reliability Logic Diagram Number	Function	FAILURE MODE	Operation Phase	Failure Effect On			Failure Detection Method	Corrective Action Time Available/Time Required	Design Provisions To Reduce Criticality	Hazard Classification/ Remarks
Name	Ident Number	Drawing Reference Designation					Component/ Functional Assembly	Next Higher Subsystem	Uppermost System				
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)
Aft Pedal Position Transducer			S40.31	Provide Aft Pedal Position Data to Computer	Open or Shorted		Distorted or No Output	Incorrect or No Aft Pedal Position Data to Computer	Possible Reduction in Ability to Fly Aircraft from Back Seat	BIT, Inflt Monitor		Inflt Monitoring Backed Up by Force Transducer. Safety Pilot Can Disengage Back Seat and Fly Aircraft	II
Aft Pedal Force Transducer			S40.32	Provide Aft Pedal Force Data to Computer	Open or Shorted		Distorted or No Output	Incorrect or No Aft Pedal Force Data to Computer	Possible Reduction in Ability to Fly Aircraft from Back Seat	BIT, Inflt Monitor		Inflight Monitoring, Backed Up by Position Transducer. Safety Pilot can Disengage Back Seat and Fly Aircraft	II
Aft Pedal Variable Feel (Torque Motor)			S40.33	Provide Variable Feel to Rear Seat Pedals	Jammed		Rear Pedals Immobilized	No Pedal Position Transducer Output	Possible Reduction in Ability to Fly Aircraft from Back Seat	BIT, Inflt Monitor		BIT, Inflt Monitor Force Transducer Still Operational. Safety Pilot can Disengage Back Seat and Fly Aircraft	II

(Continued Next Page)

System FLT CONTROLS-DIRECTIONAL
Subsystem Single Chan-Parallel Flight ____________
Equipment ____________ GSE ____________
Module ____________

FAILURE MODE AND EFFECTS ANALYSIS
NASA TWO PLACE W/STOL RESEARCH AIRCRAFT

TABLE C-1 CONT.

Date ____________
By ____________
Approved ____________

Item Identification			Reliability Logic Diagram Number	Function	FAILURE MODE	Operation Phase	Failure Effect On			Failure Detection Method	Corrective Action Time Available/Time Required	Design Provisions To Reduce Criticality	Hazard Classification/ Remarks
Name	Ident Number	Drawing Reference Designation					Component/ Functional Assembly	Next Higher Subsystem	Uppermost System				
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)
(Continued) Aft Pedal Variable Feel (Torque Motor)			S40.33 (Continued)	(Continued) Provide Variable Feel to Rear Seat Pedals	Shorted or Open		Little or No Resistance to Pedal Movement	Little or No Force Transducer Output	Little or No Tactile Cues for Pedal Positioning, Tendency to Over-control Via Position Sensor	None Needed-Readily Apparent to Back Seat Pilot		BIT, Inflt Monitor Position Transducer Still Operational Safety Pilot Can Disengage Back Seat and Fly Aircraft	II
Parallel Servo (Electrical)			S40.34E	Move Control Rod Linkage as Commanded by Computer in Response to Aft Pedal Transducer Outputs	Jammed		Won't Move	No Response to Back Seat Pilot Commands	Back Seat Controls Useless, Front Seat Controls Stiffer than Usual, May Trip Circuit Breaker	Readily Apparent to Back Seat Pilot, BIT would Detect if Existing Prior to Flight		BIT, Safety Pilot Can Electrically Disengage Rear Controls and Fly Aircraft by Over-powering Parallel Servo Output Safety Spring Cartridge, Circuit Breaker	III

(Continued Next Page)

FAILURE MODE AND EFFECTS ANALYSIS

NASA TWO PLACE W/STOL RESEARCH AIRCRAFT

System FLT CONTROLS - DIRECTIONAL

Subsystem Single Chan-Parallel Flight________

Equipment________ GSE________

Module________

Date________

By________

Approved________

TABLE C-1 CONT.

Item Identification: Name (1)	Ident Number (2)	Drawing Reference Designation (3)	Reliability Logic Diagram Number (4)	Function (5)	FAILURE MODE (6)	Operation Phase (7)	Failure Effect On: Component/Functional Assembly (8)	Next Higher Subsystem (9)	Uppermost System (10)	Failure Detection Method (11)	Corrective Action Time Available/Time Required (12)	Design Provisions To Reduce Criticality (13)	Hazard Classification/Remarks (14)
(Continued) Parallel Servo (Electrical)			S40.34E		(Continued) Burned Out or Shorted		Little or No Movement	Little or No Response to Back Seat Pilot Commands	Back Seat Controls Virtually Useless, Front Seat Controls May Be Stiffer than Usual If Shorted Will Probably Trip Circuit Breaker	Readily Apparent to Back Seat Pilot, Fwd Pilot Would Probably Notice Change in Feel of Pedals		BIT Prior to Flt, Ckt Breaker to Prevent Overheat, Safety Pilot Can Electrically Disengage Rear Controls and Fly Aircraft by Overpowering Parallel Servo Spring Cartridge or Back-Driving the Parallel Servo	III
					Broken Rod End or Head End Attachment		No Reaction Point for Command Forces	No Output	Rear Seat Controls Useless	Problem Readily Apparent to Rear Pilot		Normal Operation from Front Seat with Rear Seat Controls Electrically Disengaged	III

System FLIGHT CONTROLS - DIRECTIONAL
Subsystem Single Chan-Parallel Flight
Equipment ________ GSE ________
Module ________

FAILURE MODE AND EFFECTS ANALYSIS
NASA TWO PLACE W/STOL RESEARCH AIRCRAFT

TABLE C-1 CONT.

Date ________
By ________
Approved ________

Item Identification			Reliability Logic Diagram Number	Function	FAILURE MODE	Operation Phase	Failure Effect On			Failure Detection Method	Corrective Action Time Available/Time Required	Design Provisions To Reduce Criticality	Hazard Classification/ Remarks
Name	Ident Number	Drawing Reference Designation					Component/ Functional Assembly	Next Higher Subsystem	Uppermost System				
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)
Parallel Servo Safety Spring Cartridge			S40.35	Couple Parallel Servo to Mech Control System and Provide Capabilities for Mech Sys to Override Electrical System	Jammed		Cartridge Won't Collapse or Extend	Direct Link Between Servo and Mech System	Loss of Direct Mechanical Override Protection No Effect Unless Coupled with an Electrical System Failure			Single Jam in Cartridge not Significant a Second Failure in Back Seat Controls Required to Cause Problems.	II
					Broken Spring		No Resistance in One Direction	Servo Could Not Move Linkage in One Direction	Problems Would Be Noted by Pilot as Erratic Response to Back Seat Commands	BIT Prior to Flight Pilot Observation in Flight		Safety Pilot Can Electrically Disengage Back Seat Controls and Fly Aircraft From Front Seat	III
					Broken Attachments		No Resistance in Either Direction	Servo Could Not Move Linkage in Either Direction	No Response to Back Seat Commands	Readily Apparent to Back Seat Pilot		Safety Pilot Can Electrically Disengage Back Seat Controls and Fly Aircraft from Front Seat	III

System FLT CONTROLS-LATERAL/LONG/DIRECTIONAL

Subsystem Dual Chan-Series Flight ____

Equipment ____ GSE ____

Module ____

FAILURE MODE AND EFFECTS ANALYSIS
NASA TWO PLACE V/STOL RESEARCH AIRCRAFT

TABLE C-2 FMEA, DUPLEX SYSTEM

Date ____

By ____

Approved ____

Item Identification: Name (1)	Item Identification: Ident Number (2)	Item Identification: Drawing Reference Designation (3)	Reliability Logic Diagram Number (4)	Function (5)	FAILURE MODE (6)	Operation Phase (7)	Failure Effect On: Component/ Functional Assembly (8)	Failure Effect On: Next Higher Subsystem (9)	Failure Effect On: Uppermost System (10)	Failure Detection Method (11)	Corrective Action Time Available/Time Required (12)	Design Provisions To Reduce Criticality (13)	Hazard Classification/ Remarks (14)
Computers (2)			D40.01	Accept Position and Force Transducer Inputs From Aft Stick/ Pedals Force Link Discretes from Fwd Stick/ Pedals Data Analogs from Motion Sensors and Air Data Computation. Provide Command Outputs for Parallel Servo and Aft Stick/ Pedals Variable Feel; formatted Data for Displays. Provide Storage, Execute, and Decision Capability for BIT and IFIM	Disagreement between Two Computers		Automatic Disengage of DFCS	Loss of Rear Seat Control to Front Seat Safety Pilot	Front Seat Safety Pilot Must Assume Control of Aircraft at Time of Disengage	Comparison of Two Computer Outputs. Pilots are Signaled with Warning Lights that Disengage Has Occurred and Safety Pilot Must Take Control	Safety Pilot Must Immediately Assume Control of Aircraft	Comparison Technique Automatically Disengages of Difference is Detected This Technique Prevents Erroneous Hardover Commands Unless They Are Being Commonly Issued By Both Computers	II (At Altitude) III (Low Level)

FAILURE MODE AND EFFECTS ANALYSIS
NASA TWO PLACE V/STOL RESEARCH AIRCRAFT

TABLE C-2 CONT.

System FLT CONTROLS-LATERAL/LONG/DIRECTIONAL
Sybsystem Dual Chan-Series' Flight ______
Equipment ______ GSE ______
Module ______

Date ______
By ______
Approved ______

Item Identification			Reliability Logic Diagram Number	Function	FAILURE MODE	Operation Phase	Failure Effect On			Failure Detection Method	Corrective Action Time Available/Time Required	Design Provisions To Reduce Criticality	Hazard Classification/ Remarks
Name	Ident Number	Drawing Reference Designation					Component/ Functional Assembly	Next Higher Subsystem	Uppermost System				
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)
Motion Sensors			D40.02	Provide Rotational and Trans- Lational Motion Data to Computer	Incorrect Inputs to Computer Resulting in Auto- matic Disengage- ment of the DFCS		Automatic Disengage of DFCS	Loss of Rear Seat Control to Front Seat Safety Pilot.	Front Seat Safety Pilot Must Assume Control of Aircraft at Time of Dis- engage	Difference in Outputs of Redun- dant Motion Sensors Results in Failure of Comparison Test and Automati- cally Disengages the DFCS	Safety Pilot Must Immedi- ately Assume Control of Aircraft	Comparison Technique Automatic- ally Dis- engages If Differences is Detected This Tech- nique Prevents Erroneous Hardover Commands Unless They Are Being Commonly Issued by Both Computers	II (At Altitude) III (Low Level)

FAILURE MODE AND EFFECTS ANALYSIS
NASA TWO PLACE V/STOL RESEARCH AIRCRAFT

System FLT CONTROLS - LATERAL/LONGITUDINAL
Subsystem Dual Chan-Series
Equipment ____ Flight ____
Module ____ GSE ____

Date ____
By ____
Approved ____

TABLE C-2 CONT.

Item Identification: Name (1)	Ident Number (2)	Drawing Reference Designation (3)	Reliability Logic Diagram Number (4)	Function (5)	FAILURE MODE (6)	Operation Phase (7)	Failure Effect On: Component/Functional Assembly (8)	Next Higher Subsystem (9)	Uppermost System (10)	Failure Detection Method (11)	Corrective Action Time Available/Time Required (12)	Design Provisions To Reduce Criticality (13)	Hazard Classification/Remarks (14)
Rear Stick			S40.03	Evaluation Pilot Input	Jammed at Pivot		Rear Stick Won't Move	No Input to Computer Roll Axis from Rear Seat.	No Lateral /Long. Control from Rear Seat.	None Needed - Readily Apparent to Pilot		Front Seat Can Fly Aircraft	III
					Broken Off at Pivot		Rear Stick Useless, Feel Sys Should Center Transducer	No Rear Seat Input to Computer	No Lateral /Long. Control from Rear Seat.	None Needed Readily Apparent to Pilot		Safety Pilot Can Electrically Disengage Back Seat	III
					Loose at Pivot		No Solid Reaction Point for Rear Seat Input	Erratic Outputs from Rear Stick Transducer	Erratic Lateral/ Long. Control from Rear Seat.	May Feel Loose to Evaluation Pilot - Both Pilots Should Note Erratic Movement of Aircraft		Safety Pilot Can Electrically Disengage Back Seat	III

FAILURE MODE AND EFFECTS ANALYSIS
NASA TWO PLACE V/STOL RESEARCH AIRCRAFT

System FLT CONTROLS – LATERAL/LONGITUDINAL
Subsystem Dual Chan-Series
Equipment ______
Module ______
Flight ______
GSE ______

Date ______
By ______
Approved ______

TABLE C-2 CONT.

Item Identification			Reliability Logic Diagram Number	Function	FAILURE MODE	Operation Phase	Failure Effect On			Failure Detection Method	Corrective Action Time Available/Time Required	Design Provisions To Reduce Criticality	Hazard Classification/ Remarks
Name	Ident Number	Drawing Reference Designation					Component/ Functional Assembly	Next Higher Subsystem	Uppermost System				
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)
Fwd Stick			D40.04	Safety Pilot Override Input	Jammed at Pivot		Stick Won't Move	Input to Computer and to Aileron and Stabilator Control Valves Only From Rear Stick	Loss of Lateral/ Long System Control with Front Stick	None Needed Readily Apparent to Pilots		Back Seat Can Fly Aircraft by Use of "Emergency DFCS" Switch	III
					Broken Off at Pivot		Fwd Stick Useless	No Front Seat Input to Computer or to Aileron Control Valves	No Lat Sys Control from Front Seat	None Needed-Readily Apparent to Front Seat Pilot		Back Seat Can Fly Aircraft by Use of Emergency" SW	III
					Loose at Pivot		No Solid Reaction Point for Fwd Pilot Input	Erratic or Sloppy Inputs to Lateral System	Erratic Lateral Control from Front Seat	None Needed, Readily Apparent to Pilot		Back Seat Can Fly Aircraft by Use of "Emergency" SW	III
Fwd Stick Paddle Switch				Emergency Electrical Disengage of Back Seat Controls	Failed Closed		Can't Break Circuit	No Disengage	Loss of Stick-Mounted Emergency Disengage Capability	Pre-Flt Checks		Emergency Disengage Available Fwd Stick Force Link	II
					Failed Open		Circuit Won't MAke	Engage Solenoid Won't Latch	Constant Disengage of Back Seat Controls	BIT, Pre-Flt Checks		Safety Pilot Can Fly Aircraft	II

PTR

System FLT CONTROLS - LATERAL/LONGITUDINAL

Sybsystem Dual Chan-Series Flight ________

Equipment ________ GSE ________

Module ________

FAILURE MODE AND EFFECTS ANALYSIS
NASA TWO PLACE V/STOL RESEARCH AIRCRAFT

TABLE C-2 CONT.

Date ________

By ________

Approved ________

Item Identification			Reliability Logic Diagram Number	Function	FAILURE MODE	Operation Phase	Failure Effect On			Failure Detection Method	Corrective Action Time Available/Time Required	Design Provisions To Reduce Criticality	Hazard Classification/ Remarks
Name	Ident Number	Drawing Reference Designation					Component/ Functional Assembly	Next Higher Subsystem	Uppermost System				
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)
Fwd Stick Force Link (2) Each Axis			D40.042	Provide Automatic Takeover by Safety Pilot	Jammed or Breakout Too High		Won't Switch at Specified Force	Automatic Override Programmed To Operate On Lesser Of Two Force Xducer Outputs	None	Built-In Test Routine/ Ground Checks		Ground Test and Adjustment or Replacement After Identified	I
					Breakout Too Low		Switches Below Spec Force	Undesired Disengagement of Evaluation Pilot's Controls	Nuisance Disengagements Would Invalidate Some Test Results	None Needed - Readily Apparent to Both Pilots		Ground Test and Adjustment or Replacement After Pilot Squawk	II

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FAILURE MODE AND EFFECTS ANALYSIS
NASA TWO PLACE V/STOL RESEARCH AIRCRAFT

TABLE C-2 CONT.

System FLT CONTROLS-LATERAL
Subsystem Dual Chan-Series
Equipment ______
Module ______
Flight ______
GSE ______

Date ______
By ______
Approved ______

Item Identification: Name (1)	Item Identification: Ident Number (2)	Item Identification: Drawing Reference Designation (3)	Reliability Logic Diagram Number (4)	Function (5)	FAILURE MODE (6)	Operation Phase (7)	Failure Effect On: Component/Functional Assembly (8)	Failure Effect On: Next Higher Subsystem (9)	Failure Effect On: Uppermost System (10)	Failure Detection Method (11)	Corrective Action Time Available/Time Required (12)	Design Provisions To Reduce Criticality (13)	Hazard Classification/Remarks (14)
Aft Stick Position Transducers (2)			D40.11	Provide Aft Stick Position Data to Computer	Open or Shorted		Difference in Outputs of 2 Aft Stick Position Xducers	Automatic Disengage of DFCS	Front Seat Safety Pilot Must Assume Control of Aircraft at Time of Dis-engage	Difference in Outputs of 2 Stick Position Xducers is Sensed and Auto Dis-engage Sequence Initiated	Safety Pilot Must Immedi-ately Assume Control of Aircraft	Comparison Technique Automati-cally Disengage if Difference is Detected.	II (At Altitude) III (Low Level)
Aft Stick Force Trans-ducers (2)			D40.12	Provide Aft Stick Force Data to Computer	Open or Shorted		Difference in Outputs of 2 Aft Stick Force Xducers	Automatic Disengage of DFCS	Front Seat Safety Pilot Must Assume Control of Aircraft at Time of Dis-engage	Difference in Outputs of 2 Stick Position Xducers is Sensed and Auto Dis-engage Sequence Initiated	Safety Pilot Must Immedi-ately Assume Control of Aircraft	Comparison Technique Automati-cally Disengage if Difference is Detected.	II (At Altitude) III (Low Level)
Aft Stick Variable Feel (Torque Motor)			D40.13	Provide Variable Feel to Rear Seat Stick	Jammed		Rear Stick Immobil-ized	No Stick Position Transducer Output	Possible Reduction in Ability to Fly Aircraft from Back Seat	BIT, Inflt Monitor		BIT, Inflt Monitor Force Transducer Still Operational; Safety Pilot Can Disengage Back Seat and Fly Aircraft	III

(Continued Next Page)

System FLT CONTROLS - LATERAL
Sybsystem Dual Chan-Series Flight
Equipment GSE
Module

FAILURE MODE AND EFFECTS ANALYSIS
NASA TWO PLACE V/STOL RESEARCH AIRCRAFT

TABLE C-2 CONT.

Page 7 of 26 Pages
Date
By
Approved

Item Identification			Reliability Logic Diagram Number	Function	FAILURE MODE	Operation Phase	Failure Effect On			Failure Detection Method	Corrective Action Time Available/Time Required	Design Provisions To Reduce Criticality	Hazard Classification/ Remarks
Name	Ident Number	Drawing Reference Designation					Component/ Functional Assembly	Next Higher Subsystem	Uppermost System				
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)
(Continued) Aft Stick Variable Feel (Torque Motor)			D40.13 (Continued)	(Continued) Provide Q-programmed Variable Feel to Rear Seat Stick	Shorted or Open		Little or No Resistance to Stick Movement	Little or No Force Transducer Output	Little or No Tactile Cues for Stick Positioning, Tendency to Over-control Via Position Sensor	None Needed-Readily Apparent to Back Seat Pilot		BIT, Inflt Monitor Position Transducer Still Operational Safety Pilot Can Disengage Back Seat and Fly Aircraft	II
Series Servo (Electrical)			D40.14E	Move Control Rod Linkage as Commanded by Computer in Response to Aft Stick Transducer Outputs	Jammed		Decreased Movement	Decreased Response to Back Seat Pilot Commands	Back Seat Controls Degraded Front Seat Controls Stiffer than Usual, May Trip Circuit Breaker	Readily Apparent to Back Seat Pilot, BIT would Detect if Existing Prior to Flight Automatic Disengage	Safety Pilot Must Immediately Assume Control of Aircraft	Comparison Technique Automatically Disengages if Difference is Detected.	II (At Altitude) III (Low Level)

(Continued Next Page)

FAILURE MODE AND EFFECTS ANALYSIS

NASA TWO PLACE V/STOL RESEARCH AIRCRAFT

TABLE C-2 CONT.

System FLT CONTROLS - LATERAL
Subsystem Dual Chan-Series
Flight ______
Equipment ______
GSE ______
Module ______

Date ______
By ______
Approved ______

Item Identification			Reliability Logic Diagram Number	Function	FAILURE MODE	Operation Phase	Failure Effect On			Failure Detection Method	Corrective Action Time Available/Time Required	Design Provisions To Reduce Criticality	Hazard Classification/ Remarks
Name	Ident Number	Drawing Reference Designation					Component/ Functional Assembly	Next Higher Subsystem	Uppermost System				
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)
(Continued) Series Servo (Electrical)					(Continued) Burned Out or Shorted		Little or No Movement	Little or No Response to Back Seat Pilot Commands	Back Seat Controls Virtually Useless, Front Seat Controls May Be Stiffer than Usual If Shorted Will Probably Trip Circuit Breaker	Readily Apparent to Back Seat Pilot, Fwd Pilot Would Probably Notice Change in Feel of Stick		BIT Prior to Flt, Ckt Breaker to Prevent Overheat, Safety Pilot Can Electrically Disengage Rear Controls and Fly Aircraft by Overpowering Series Servo Spring Cartridge or Back-Driving the Series Servo	III
					Broken Rod End or Head End Attachment Common to Both Cylinders or Free Movement of Both Cylinders		No Reaction Point for Command Forces	No Output	Front and Rear Seat Controls Useless	Problem Readily Apparent to Both Pilot		Stress Analysis of Series Servo Common Attachments and Ends. Ensure Failure to Dual Actuator Does Not Allow Both Cylinders to Move Freely.	III

System FLIGHT CONTROLS - LATERAL

Subsystem Single Chan-Parallel Flight ____________

Equipment ____________ GSE ____________

Module ____________

FAILURE MODE AND EFFECTS ANALYSIS

NASA TWO PLACE V/STOL RESEARCH AIRCRAFT

TABLE C-2 CONT.

Date ____________

By ____________

Approved ____________

Item Identification			Reliability Logic Diagram Number	Function	FAILURE MODE	Operation Phase	Failure Effect On			Failure Detection Method	Corrective Action Time Available/Time Required	Design Provisions To Reduce Criticality	Hazard Classification/ Remarks
Name	Ident Number	Drawing Reference Designation					Component/ Functional Assembly	Next Higher Subsystem	Uppermost System				
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)
Series Servo Safety Spring Cartridge (2)				Couple Series Servo to Mech Control System and Provide Capabilities for Mech Sys to Override Electrical System	Jammed		Cartridge Won't Collapse or Extend	Direct Link Between Servo and Mech System	Loss of Direct Mechanical Override Protection No Effect Unless Coupled with an Electrical System Failure			Single Jam in Cartridge not Significant a Second Failure in Back Seat Controls Required to Cause Problems.	II
					Broken Spring		Limited Resistance in One Direction	Degrade Servo Ability to Move Linkage in One Direction	Problems Would Be Noted by Pilot as Erratic Response to Back Seat Commands	BIT Prior to Flight Pilot Observation in Flight		Safety Pilot Can Electrically Disengage Back Seat Controls and Fly Aircraft From Front Seat	III

FAILURE MODE AND EFFECTS ANALYSIS

NASA TWO PLACE V/STOL RESEARCH AIRCRAFT

TABLE C-2 CONT.

System FLIGHT CONTROLS - LATERAL
Subsystem Dual Chan-Series
Equipment
Module
Flight
GSE
Date
By
Approved

Item Identification			Reliability Logic Diagram Number	Function	FAILURE MODE	Operation Phase	Failure Effect On			Failure Detection Method	Corrective Action Time Available/Time Required	Design Provisions To Reduce Criticality	Hazard Classification/ Remarks
Name	Ident Number	Drawing Reference Designation					Component/ Functional Assembly	Next Higher Subsystem	Uppermost System				
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)
Front Seat (Mechanical) Trim			D40.16	Reacts to Pilot Beep Switch Command to Trim Ailerons by Changing Stick Neutral Point	Open		Winding Burnout	Trim Motor Won't Run	No Mechanical Trim Capability, Trim Fixed at One Position			Pilot Can Hold in Trim with Stick, Rear Seat Pilot Can Put in Trim Bias Electrically	II
					Short		Shorted Winding	Sluggish Operation, May Trip Circuit Breaker	Poor or No Mechanical Trip Operation			Circuit Breaker to Prevent Overheat, Rear Seat Pilot Can Put in Trim Bias Electrically	II
					Jammed or Stripped Gearing		Jammed or Stripped	No Trim Output	Loss of Mech Trim Capability, Trim Fixed at One Position			Pilot Can Hold in Trim with Stick, Rear Seat Pilot Can Put in Trim Bias Electrically	II
					Loss of Structural Ground (Attachments Broken)		No Reaction Point for Trim Motor Torque	No Trim Motion	Loss of Mechanical Trim Capability			Pilot Can Hold in Trim with Stick, Rear Seat Pilot Can Put in Trim Bias Electrically	II

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FAILURE MODE AND EFFECTS ANALYSIS
NASA TWO PLACE V/STOL RESEARCH AIRCRAFT

System FLT. CONTROLS - LATERAL
Sybsystem Dual Chan-Series
Equipment
Module
Flight
GSE

TABLE C-2 CONT.

Date
By
Approved

Item Identification			Reliability Logic Diagram Number	Function	FAILURE MODE	Operation Phase	Failure Effect On			Failure Detection Method	Corrective Action Time Available/Time Required	Design Provisions To Reduce Criticality	Hazard Classification/ Remarks
Name	Ident Number	Drawing Reference Designation					Component/ Functional Assembly	Next Higher Subsystem	Uppermost System				
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)
Front Stick Feel Spring			D40.17	Provides Artificial Feel for Fwd Pilot	Jammed		Stick Won't Move	Input to Computer or to Aileron Control Valves Only from Rear Stick	Loss of Lateral System Control with Front Stick	None Needed-Readily Apparent to Pilots		Back Seat Can Fly Aircraft By Use of "Emergency DFCS" Switch	III
					Broken Spring		Absence of Feel and Trim in One Direction	Unbalanced Feel Forces, Erratic Trim	Difficult to Fly Aircraft with Front Stick	None Needed-Readily Apparent to Pilot		With Hands Off in Front Seat, Back Seat Pilot Can Fly Aircraft Electrically. "Emergency" SW Can be Used to Prevent DFCS Disengage.	III
					Broken Head End or Rod End Attachments		Absence of Feel Forces and Trim Position in Both Directions	Loss of Tactile Cues for Proper Stick Positioning	Difficult to Fly Aircraft with Front Stick	None Needed-Readily Apparent to Pilot		With Hands Off in Front Seat, Back Seat Pilot Can Fly Aircraft Electrically. "Emergency" SW Can Be Used to Prevent DFCS Disengage.	III

System FLT CONTROLS-LOGITUDINAL

Sybsystem Dual Chan-Series' Flight________

Equipment________ GSE________

Module________

FAILURE MODE AND EFFECTS ANALYSIS
NASA TWO PLACE V/STOL RESEARCH AIRCRAFT

TABLE C-2 CONT.

Date________

By________

Approved________

Item Identification			Reliability Logic Diagram Number	Function	FAILURE MODE	Operation Phase	Failure Effect On			Failure Detection Method	Corrective Action Time Available/Time Required	Design Provisions To Reduce Criticality	Hazard Classification/ Remarks
Name	Ident Number	Drawing Reference Designation					Component/ Functional Assembly	Next Higher Subsystem	Uppermost System				
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)
Aft Stick Position Transducer (2)			D40.21	Provide Aft Stick Position Data to Computer	Open or Shorted		Difference in Outputs of 2 Aft Stick Position Xducers	Automatic Disengage of DFCS	Front Seat Safety Pilot Must Assume Control of Aircraft at Time of Disengage	Difference in Outputs of 2 Stick Position Xducers Is Sensed and Auto Disengage Sequence Is Initiated.	Safety Pilot Must Immediately Assume Control of Aircraft	Comparison Technique Automatically Disengages if Difference is Detected	II (At Altitude) III(Low Level)
Aft Stick Force Transducer				Provide Aft Stick Force Data to Computer	Open or Shorted		Difference in Outputs of 2 Aft Stick Force Xducers	Automatic Disengage of DFCS	Front Seat Safety Pilot Must Assume Control of Aircraft at Time of Disengage	Difference in Outputs of 2 Stick Position Xducers Is Sensed and Auto Disengage Sequence Is Initiated.	Safety Pilot Must Immediately Assume Control of Aircraft	Comparison Technique Automatically Disengages if Difference is Detected	II (At Altitude) III (Low Level)
Aft Stick Variable Feel (Torque Motor)				Provide Q-Programmed Variable Feel to Rear Seat Stick	Jammed		Rear Stick Immobilized	No Stick Position Transducer Output	Possible Reduction in Ability to Fly Aircraft from Back Seat	BIT, Inflt Monitor		BIT, Inflt Monitor Force Transducer Still Operational Safety Pilot Can Disengage Back Seat and Fly Aircraft	III

(Continued Next Page)

FAILURE MODE AND EFFECTS ANALYSIS
NASA TWO PLACE V/STOL RESEARCH AIRCRAFT

System FLT CONTROLS - LONGITUDINAL
Sybsystem Dual Chan-Series   Flight ____________
Equipment ____________   GSE ____________
Module ____________

TABLE C-2 CONT.

Date ____________
By ____________
Approved ____________

Item Identification			Reliability Logic Diagram Number	Function	FAILURE MODE	Operation Phase	Failure Effect On			Failure Detection Method	Corrective Action Time Available/Time Required	Design Provisions To Reduce Criticality	Hazard Classification/ Remarks
Name	Ident Number	Drawing Reference Designation					Component/ Functional Assembly	Next Higher Subsystem	Uppermost System				
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)
(Continued) Aft Stick Variable Feel (Torque Motor)			D40.23 (Continued)	(Continued) Provide Q-programmed Variable Feel to Rear Seat Stick	Shorted or Open		Little or No Resistance to Stick Movement	Little or No Force Transducer Output	Little or No Tactile Cues for Stick Positioning, Tendency to Overcontrol Via Position Sensor	None Needed-Readily Apparent to Back Seat Pilot		BIT, Inflt Monitor Position Transducer Still Operational Safety Pilot Can Disengage Back Seat and Fly Aircraft	II
Series Servo (Electrical)			D40.24E	Move Control Rod Linkage as Commanded by Computer in Response to Aft Stick Transducer Outputs	Jammed		Won't Move	No Response to Back Seat Pilot Commands	Back Seat Controls Useless, Front Seat Controls Stiffer than Usual, May Trip Circuit Breaker	Readily Apparent to Back Seat Pilot, BIT would Detect if Existing Prior to Flight		BIT, Safety Pilot Can Electrically Disengage Rear Controls and Fly Aircraft by Overpowering Series Servo Output Safety Spring Cartridge, Circuit Breaker	III

(Continued Next Page)

FAILURE MODE AND EFFECTS ANALYSIS
NASA TWO PLACE V/STOL RESEARCH AIRCRAFT

System FLT CONTROLS – LONGITUDINAL
Subsystem Dual Chan-Series   Flight
Equipment   GSE
Module

Date
By
Approved

TABLE C-2 CONT.

Item Identification			Reliability Logic Diagram Number	Function	FAILURE MODE	Operation Phase	Failure Effect On			Failure Detection Method	Corrective Action Time Available/Time Required	Design Provisions To Reduce Criticality	Hazard Classification/ Remarks
Name	Ident Number	Drawing Reference Designation					Component/ Functional Assembly	Next Higher Subsystem	Uppermost System				
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)
(Continued) Series Servo (Electrical)			D40.14E		(Continued) Burned Out or Shorted		Little or No Movement	Little or No Response to Back Seat Pilot Commands	Back Seat Controls Virtually Useless, Front Seat Controls May Be Stiffer than Usual If Shorted Will Probably Trip Circuit Breaker	Readily Apparent to Back Seat Pilot, Fwd Pilot Would Probably Notice Change in Feel of Stick		BIT Prior to Flt, Ckt Breaker to Prevent Overheat, Safety Pilot Can Electrically Disengage Rear Controls and Fly Aircraft by Overpowering Series Servo Spring Cartridge or Back-Driving the Series Servo	III
					Broken Rod End or Head End Attachment Common to Both Cylinders or Free Movement of Both Cylinders		No Reaction Point for Command Forces	No Output	Front and Rear Seat Controls Useless	Problem Readily Apparent to Both Pilot		Stress Analysis of Series Servo Common Attachments and Ends. Ensure Failure to Dual Actuators Does Not Allow Both Cylinders to Move Freely.	IV

System FLIGHT CONTROLS-LONGITUDINAL
Subsystem Dual Chan-Series Flight____
Equipment____ GSE____
Module____

FAILURE MODE AND EFFECTS ANALYSIS
NASA TWO PLACE V/STOL RESEARCH AIRCRAFT

Date____
By____
Approved____

TABLE C-2 CONT.

Item Identification: Name (1)	Ident Number (2)	Drawing Reference Designation (3)	Reliability Logic Diagram Number (4)	Function (5)	FAILURE MODE (6)	Operation Phase (7)	Failure Effect On: Component/Functional Assembly (8)	Next Higher Subsystem (9)	Uppermost System (10)	Failure Detection Method (11)	Corrective Action Time Available/Time Required (12)	Design Provisions To Reduce Criticality (13)	Hazard Classification/Remarks (14)
Series Servo Safety Spring Cartridge			D40.25	Couple Series Servo to Mech Control System and Provide Capabilities for Mech Sys to Override Electrical System	Jammed		Cartridge Won't Collapse or Extend	Direct Link Between Servo and Mech System	Loss of Direct Mechanical Override Protection No Effect Unless Coupled with an Electrical System Failure			Single Jam in Cartridge not Significant a Second Failure in Back Seat Controls Required to Cause Problems.	II
					Broken Spring		Limited Resistance in One Direction	Degrade Servo Ability to Move Linkage in One Direction	Problems Would Be Noted by Pilot as Erratic Response to Back Seat Commands	BIT Prior to Flight Pilot Observation in Flight		Safety Pilot Can Electrically Disengage Back Seat Controls and Fly Aircraft From Front Seat	III

FAILURE MODE AND EFFECTS ANALYSIS
NASA TWO PLACE V/STOL RESEARCH AIRCRAFT

System FLIGHT CONTROLS - LONGITUDINAL
Sybsystem Dual Chan-Series Flight ______
Equipment ______ GSE ______
Module ______

Date ______
By ______
Approved ______

TABLE C-2 CONT.

Item Identification: Name (1)	Item Identification: Ident Number (2)	Item Identification: Drawing Reference Designation (3)	Reliability Logic Diagram Number (4)	Function (5)	FAILURE MODE (6)	Operation Phase (7)	Failure Effect On: Component/ Functional Assembly (8)	Failure Effect On: Next Higher Subsystem (9)	Failure Effect On: Uppermost System (10)	Failure Detection Method (11)	Corrective Action Time Available/Time Required (12)	Design Provisions To Reduce Criticality (13)	Hazard Classification/ Remarks (14)
Front Seat (Mechanical) Trim			D40.26	Reacts to Pilot Beep Switch Command to Trim Stabilator by Changing Stick Neutral Point	Open		Winding Burnout	Trim Motor Won't Run	No Mechanical Trim Capability, Trim Fixed at One Position			Pilot Can Hold in Trim with Stick, Rear Seat Pilot Can Put in Trim Bias Electrically	II
					Short		Shorted Winding	Sluggish Operation, May Trip Circuit Breaker	Poor or No Mechanical Trip Operation			Circuit Breaker to Prevent Overheat, Rear Seat Pilot Can Put in Trim Bias Electrically	II
					Jammed or Stripped Gearing		Jammed or Stripped	No Trim Output	Loss of Mech Trim Capability, Trim Fixed at One Position			Pilot Can Hold in Trim with Stick, Rear Seat Pilot Can Put in Trim Bias Electrically	II
					Loss of Structural Ground (Attachments Broken)		No Reaction Point for Trim Motor Torque	No Trim Motion	Loss of Mechanical Trim Capability			Pilot Can Hold in Trim with Stick, Rear Seat Pilot Can Put in Trim Bias Electrically	II

System FLT. CONTROLS – LONGITUDINAL
Sybsystem Dual Chan-Series
Equipment
Module

Flight
GSE

FAILURE MODE AND EFFECTS ANALYSIS
NASA TWO PLACE V/STOL RESEARCH AIRCRAFT

TABLE C-2 CONT.

Date
By
Approved

Item Identification			Reliability Logic Diagram Number	Function	FAILURE MODE	Operation Phase	Failure Effect On			Failure Detection Method	Corrective Action Time Available/Time Required	Design Provisions To Reduce Criticality	Hazard Classification/ Remarks
Name	Ident Number	Drawing Reference Designation					Component/ Functional Assembly	Next Higher Subsystem	Uppermost System				
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)
Front Stick Feel Spring			D40.27	Provides Artificial Feel for Fwd Pilot	Jammed		Stick Won't Move	Input to Computer or to Stabilator Control Valves Only from Rear Stick	Loss of Pitch System Control with Front Stick	None Needed-Readily Apparent to Pilots		Back Seat Can Fly Aircraft By Use Of "Emergency DFCS" Switch	III
					Broken Spring		Absence of Feel and Trim in One Direction	Unbalanced Feel Forces, Erratic Trim	Difficult to Fly Aircraft with Front Stick	None Needed-Readily Apparent to Pilot		With Hands Off in Front Seat, Back Seat Pilot Can Fly Aircraft Electrical-ly. "Emergency" SW Can be Used to Prevent DFCS Dis-engage.	III
					Broken Head End or Rod End Attach-ments		Absence of Feel Forces and Trim Position in Both Directions	Loss of Tactile Cues for Proper Stick Positioning	Difficult to Fly Aircraft with Front Stick	None Needed-Readily Apparent to Pilot		With Hands Off in Front Seat, Back Seat Pilot Can Fly Air-craft Electrical-ly. "Emergency" SW Can Be Used to Prevent DFCS Dis-engage.	III

System FLT CONTROLS-LONGITUDINAL
Subsystem Dual Chan-Series Flight______
Equipment Pitch Axis GSE______
Module______

FAILURE MODE AND EFFECTS ANALYSIS
NASA TWO PLACE V/STOL RESEARCH AIRCRAFT

TABLE C-2 CONT.

Date______
By______
Approved______

Item Identification			Reliability Logic Diagram Number	Function	FAILURE MODE	Operation Phase	Failure Effect On			Failure Detection Method	Corrective Action Time Available/Time Required	Design Provisions To Reduce Criticality	Hazard Classification/ Remarks
Name	Ident Number	Drawing Reference Designation					Component/ Functional Assembly	Next Higher Subsystem	Uppermost System				
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)
Forward RCS Valve Series Actuator (2)			D40.28	Provide SAS Capability on front RCS Valve	Jammed to bellcrank		Limited Movement relative to Bellcrank	Stifened Link Stick TO RCS Valve	Partial Loss of Front RCS SAS, Valve Still Operational	BIT, in-flt monitor		A/C Still Completely Flyable With Increase Pilot Work-load. Some Pitch RCS SAS Still Available Via Rear RCS and Front RCS Valve. No Interferences with Conventional Landing.	II
					broken, burned out or shorted Which Allows Free Movement of Cylinder		Servo Degraded	Limited Transmittal of Pilot Command	Partial Loss of Use of Front RCS Valve VTOL Operations Difficult with Fwd C.G. Aircraft	BIT, in-flight monitor		BIT or in-flt monitor warns pilot not to attempt vertical operations with failed actuator. No interference with conventional landing. Safety Pilot Disengage DFCS and Land.	II

FAILURE MODE AND EFFECTS ANALYSIS
NASA TWO PLACE V/STOL RESEARCH AIRCRAFT

System FLT CONTROLS-DIRECTIONAL
Sybsystem Dual Chan-Series Flight________
Equipment________ GSE________
Module________

Date________
By________
Approved________

TABLE C-2 CONT.

Item Identification			Reliability Logic Diagram Number	Function	FAILURE MODE	Operation Phase	Failure Effect On			Failure Detection Method	Corrective Action Time Available/Time Required	Design Provisions To Reduce Criticality	Hazard Classification/ Remarks
Name	Ident Number	Drawing Reference Designation					Component/ Functional Assembly	Next Higher Subsystem	Uppermost System				
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)
Fwd Pedals			D40.38	Safety Pilot Override Input	Jammed at Pivot		Pedals Won't Move	No Input to Computer or to Rudder from Front Pedals	Only Directional System Control with Rear Pedals	None Needed Readily Apparent to Pilots		Back Seat Can Fly Aircraft By Use of "Emergency DFCS" Switch	III
					Broken Off at Pivot		Fwd Pedals Useless	No Front Seat Input to Computer or to Rudder	No Directional Sys Control from Front Seat	None Needed - Readily Apparent to Front Seat Pilot		Back Seat Can Fly Aircraft by Use of "Emergency" SW	III
					Loose at Pivot		No Solid Reaction Point for Fwd Pilot Input	Erratic or Sloppy Inputs to Directional System	Erratic Control from Front Seat	None Needed, Readily Apparent to Pilot		Back Seat Can Fly Aircraft by Use of "Emergency" SW	III

FAILURE MODE AND EFFECTS ANALYSIS
NASA TWO PLACE V/STOL RESEARCH AIRCRAFT

TABLE C-2 CONT.

System FLT CONTROLS-DIRECTIONAL
Sybsystem Dual Channel-Series Flight ______
Equipment ______ GSE ______
Module ______

Date ______
By ______
Approved ______

Item Identification			Reliability Logic Diagram Number	Function	FAILURE MODE	Operation Phase	Failure Effect On			Failure Detection Method	Corrective Action Time Available/Time Required	Design Provisions To Reduce Criticality	Hazard Classification/ Remarks
Name	Ident Number	Drawing Reference Designation					Component/ Functional Assembly	Next Higher Subsystem	Uppermost System				
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)
Fwd Pedal Force Link (2)			D40.39	Provide Automatic Takeover by Safety Pilot	Jammed or Breakout Too High		Won't Switch at Specified Force	Automatic Override Programmed to Operate on Lesser of Two Force Xducer Outputs.	None	Built-In Test Routine/ Ground Checks		Ground Test and Adjustment or Replacement After Identified.	I
					Breakout Too Low		Switches Below Spec Force	Undesired Disengagement of Evaluation Pilot's Controls	Nuissance Disengagements Would Invalidate Some Test Results	None Needed - REadily Apparent to Both Pilots		Ground Test and Adjustment or Replacement After Pilot Squawk	II

FAILURE MODE AND EFFECTS ANALYSIS
NASA TWO PLACE V/STOL RESEARCH AIRCRAFT

TABLE C-2 CONT.

System FLT. CONTROLS - DIRECTIONAL
Subsystem Dual Chan-Series
Equipment
Module
Flight
GSE

Date
By
Approved

Item Identification: Name (1)	Ident Number (2)	Drawing Reference Designation (3)	Reliability Logic Diagram Number (4)	Function (5)	FAILURE MODE (6)	Operation Phase (7)	Failure Effect On: Component/Functional Assembly (8)	Next Higher Subsystem (9)	Uppermost System (10)	Failure Detection Method (11)	Corrective Action Time Available/Time Required (12)	Design Provisions To Reduce Criticality (13)	Hazard Classification/Remarks (14)
Front Pedal Feel Spring			D40.37	Provides Artificial Feel for Fwd Pilot	Jammed		Pedals Won't Move	No Input to Computer or to Rudder from Front Pedals	Only Directional System Control with Rear Pedals	None Needed-Readily Apparent to Pilots		Back Seat Can Fly Aircraft By Use Of "Emergency DFCS" Switch	III
					Broken Spring		Absence of Feel in One Direction	Unbalanced Feel Forces	Difficult to Fly Aircraft with Front Pedals	None Needed-Readily Apparent to Pilot		With Hands Off in Front Seat, Back Seat Pilot Can Fly Aircraft Electrically. "Emergency" SW Can be Used to Prevent DFCS Disengage.	III
					Broken Head End or Rod End Attachments		Absence of Feel Forces in Both Directions	Loss of Tactile Cues for Proper Pedal Positioning	Difficult to Fly Aircraft with Front Pedals	None Needed-Readily Apparent to Pilot		With Hands Off in Front Seat, Back Seat Pilot Can Fly Aircraft Electrically. "Emergency" SW Can Be Used to Prevent DFCS Disengage.	III

FAILURE MODE AND EFFECTS ANALYSIS
NASA TWO PLACE V/STOL RESEARCH AIRCRAFT

System FLT CONTROLS-DIRECTIONAL
Subsystem Dual Chan-Series Flight ____
Equipment ____ GSE ____
Module ____

Date ____
By ____
Approved ____

TABLE C-2 CONT.

Item Identification: Name (1)	Ident Number (2)	Drawing Reference Designation (3)	Reliability Logic Diagram Number (4)	Function (5)	FAILURE MODE (6)	Operation Phase (7)	Failure Effect On: Component/Functional Assembly (8)	Next Higher Subsystem (9)	Uppermost System (10)	Failure Detection Method (11)	Corrective Action Time Available/Time Required (12)	Design Provisions To Reduce Criticality (13)	Hazard Classification/Remarks (14)
Rear Pedals			D40.38	Evaluation Pilot Input	Jammed at Pivot		Rear Pedals Won't Move	No Input to Computer Yaw Axis from Rear Seat.	No Directional Control from Rear Seat.	None Needed - Readily Apparent to Pilot		Front Seat Can Fly Aircraft	III
					Broken Off at Pivot		Rear Pedals Useless, Feel Sys Should Center Transducer	No Rear Seat Input to Computer	No Directional Control Seat	None Needed Readily Apparent to Pilot		Safety Pilot Can Electrically Disengage Back Seat	III
					Loose at Pivot		No Solid Reaction Point for Rear Seat Input	Erratic Outputs from Rear Pedal Transducer	Erratic Directional Control from Rear Seat	May Feel Loose to Evaluation Pilot - Both Pilots Should Note Erratic Movement of Aircraft		Safety Pilot Can Electrically Disengage Back Seat	III

System FLT CONTROLS-DIRECTIONAL
Subsystem Dual Chan-Series Flight____
Equipment____ GSE____
Module____

FAILURE MODE AND EFFECTS ANALYSIS
NASA TWO PLACE V/STOL RESEARCH AIRCRAFT

Date____
By____
Approved____

TABLE C-2 CONT.

Item Identification			Reliability Logic Diagram Number	Function	FAILURE MODE	Operation Phase	Failure Effect On			Failure Detection Method	Corrective Action Time Available/Time Required	Design Provisions To Reduce Criticality	Hazard Classification/ Remarks
Name	Ident Number	Drawing Reference Designation					Component/ Functional Assembly	Next Higher Subsystem	Uppermost System				
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)
Aft Pedal Position Transducer (2)			D40.31	Provide Aft Pedal Position Data to Computer	Open or Shorted		Difference in Outputs of 2 Aft Pedal Position Xducers	Automatic Disengage of DFCS	Front Seat Safety Pilot Must Assume Control of Aircraft At Time of Dis-engage	Difference in Outputs of 2 Stick Position Xducers Is Sensed and Auto Dis-engage Sequence Initiated	Safety Pilot Must Immedi-ately Assume Control of Aircraft	Comparison Technique Automati-cally Dis-engaged if Difference is Detected	II (At Altitude) III (Low Level)
Aft Pedal Force Transducer (2)				Provide Aft Pedal Force Data to Computer	Open or Shorted		Difference in Outputs of 2 Aft Pedal Force Xducers	Automatic Disengage of DFCS	Front Seat Safety Pilot Must Assume Control of Aircraft At Time of Dis-engage	Difference in Outputs of 2 Stick Position Xducers Is Sensed and Auto Dis-engage Sequence Initiated	Safety Pilot Must Immedi-ately Assume Control of Aircraft	Comparison Technique Automati-cally Dis-engaged if Difference is Detected	II (At Altitude) III(Low Level)
Aft Pedal Variable Feel (Torque Motor)				Provide Variable Feel to Rear Seat Pedals	Jammed		Rear Pedals Immobil-ized	No Pedal Position Transducer Output	Possible Reduction in Ability to Fly Aircraft from Back Seat	BIT, Inflt Monitor		BIT, Inflt Monitor Force Transducer Still Operational Safety Pilot can Dis-engage Back Seat and Fly Air-craft	II

(Continued Next Page)

FAILURE MODE AND EFFECTS ANALYSIS
NASA TWO PLACE V/STOL RESEARCH AIRCRAFT

System FLT CONTROLS - DIRECTIONAL
Sybsystem Dual Chan-Series Flight ____
Equipment ____ GSE ____
Module ____

Date ____
By ____
Approved ____

TABLE C-2 CONT.

Item Identification			Reliability Logic Diagram Number	Function	FAILURE MODE	Operation Phase	Failure Effect On			Failure Detection Method	Corrective Action Time Available/Time Required	Design Provisions To Reduce Criticality	Hazard Classification/ Remarks
Name	Ident Number	Drawing Reference Designation					Component/ Functional Assembly	Next Higher Subsystem	Uppermost System				
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)
(Continued) Aft Pedal Variable Feel (Torque Motor)			D40.33 (Continued)	(Continued) Provide Variable Feel to Rear Seat Pedlas	Shorted or Open		Little or No Resistance to Pedal Movement	Little or No Force Transducer Output	Little or No Tactile Cues for Pedal Positioning, Tendency to Over-control Via Position Sensor	None Needed- Readily Apparent to Back Seat Pilot		BIT, Inflt Monitor Position Transducer Still Operational Safety Pilot Can Disengage Back Seat and Fly Aircraft	II
Series Servo (Electrical)			D40.34E	Move Control Rod Linkage as Commanded by Computer in Response to Aft Pedal Transducer Outputs	Jammed		Decreased Movement	Decreased Response to Back Seat Pilot Commands	Back Seat Controls Degraded, Front Seat Controls Stiffer than Usual, May Trip Circuit Breaker	Readily Apparent to Back Seat Pilot, BIT would Detect if Existing Prior to Flight Automatic Disengage	Safety Pilot Must Immediately Assume Control of Aircraft	Comparison Technique Automatically Disengages if Difference is Sensed.	II (At Altitude) III (Low Level)

(Continued Next Page)

FAILURE MODE AND EFFECTS ANALYSIS
NASA TWO PLACE V/STOL RESEARCH AIRCRAFT

System FLT CONTROLS-DIRECTIONAL
Subsystem Dual Chan-Series'   Flight ____
Equipment ____   GSE ____
Module ____

Date ____
By ____
Approved ____

TABLE C-2 CONT.

Item Identification: Name (1)	Ident Number (2)	Drawing Reference Designation (3)	Reliability Logic Diagram Number (4)	Function (5)	FAILURE MODE (6)	Operation Phase (7)	Failure Effect On: Component/Functional Assembly (8)	Next Higher Subsystem (9)	Uppermost System (10)	Failure Detection Method (11)	Corrective Action Time Available/Time Required (12)	Design Provisions To Reduce Criticality (13)	Hazard Classification/Remarks (14)
(Continued) Series Servo (Electrical)			D40.34E		(Continued) Burned Out or Shorted		Little or No Movement	Little or No Response to Back Seat Pilot Commands	Back Seat Controls Virtually Useless, Front Seat Controls May Be Stiffer than Usual. If Shorted Will Probably Trip Circuit Breaker	Readily Apparent to Back Seat Pilot, Fwd Pilot Would Probably Notice Change in Feel of Pedals		BIT Prior to Flt, Ckt Breaker to Prevent Overheat, Safety Pilot Can Electrically Disengage Rear Controls and Fly Aircraft by Overpowering Series Servo Spring Cartridge or Back-Driving the Series Servo	III
					Broken Rod End or Head End Attachment Common to Both Cylinders or Free Movement of Both Cylinders		No Reaction Point for Command Forces	No Output	Front and Rear Seat Controls Useless	Problem Readily Apparent to Both Pilot		Stress Analysis or Series Servo Common Attachments and Ends. Ensure Failure to Dual Actuator Does Not Allow Both Cylinders to Move Freely	IV

System FLIGHT CONTROLS - DIRECTIONAL
Subsystem Dual Chan-Series
Equipment
Module
Flight
GSE

FAILURE MODE AND EFFECTS ANALYSIS
NASA TWO PLACE V/STOL RESEARCH AIRCRAFT

TABLE C-2 CONT.

Date
By
Approved

Item Identification: Name (1)	Ident Number (2)	Drawing Reference Designation (3)	Reliability Logic Diagram Number (4)	Function (5)	FAILURE MODE (6)	Operation Phase (7)	Failure Effect On: Component/Functional Assembly (8)	Next Higher Subsystem (9)	Uppermost System (10)	Failure Detection Method (11)	Corrective Action Time Available/Time Required (12)	Design Provisions To Reduce Criticality (13)	Hazard Classification/Remarks (14)
Series Servo Safety Spring Cartridge(2)			D40.35	Couple Series Servo to Mech Control System and Provide Capabilities for Mech Sys to Override Electrical System	Jammed		Cartridge Won't Collapse or Extend	Direct Link Between Servo and Mech System	Loss of Direct Mechanical Override Protection No Effect Unless Coupled with an Electrical System Failure			Single Jam in Cartridge not Significant a Second Failure in Back Seat Controls Required to Cause Problems.	II
					Broken Spring		Limited Resistance in One Direction	Degraded Servo Ability to Move Linkage in One Direction	Problems Would Be Noted by Pilot as Erratic Response to Back Seat Commands	BIT Prior to Flight Pilot Observation in Flight		Safety Pilot Can Electrically Disengage Back Seat Controls and Fly Aircraft From Front Seat	III

System FLT CONTROLS-LATERAL/LONG/DIRECTIONAL
Sybsystem Three Chan-Parallel Flight
Equipment ____ GSE ____
Module ____

FAILURE MODE AND EFFECTS ANALYSIS

NASA TWO PLACE W/STOL RESEARCH AIRCRAFT

TABLE C-3 FMEA, TRIPLEX SYSTEM

Date ____
By ____
Approved ____

Item Identification			Reliability Logic Diagram Number	Function	FAILURE MODE	Operation Phase	Failure Effect On			Failure Detection Method	Corrective Action Time Available/Time Required	Design Provisions To Reduce Criticality	Hazard Classification/ Remarks
Name	Ident Number	Drawing Reference Designation					Component/ Functional Assembly	Next Higher Subsystem	Uppermost System				
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)
Computers (3)			T40.01	Accept Position and Force Transducer Inputs From Aft Stick/ Pedals Force Link Discretes From Fwd Stick/ Pedals Data Analogs from Motion Sensors And Air Data Computation. Provide Command Outputs for Parallel Servo and Aft Stick/ Pedals Variable Feel; formatted Data for Displays. Provide Storage, Execute, and Decision Capability for BIT and IFIM	One Computer Produces Incorrect Output to Parallel Servo, Variable Feel Device, BIT, or Display		Voted Outputs Identifies the Computer Output as Incorrect. Malfunctioning Computer Output is Identified and Disregarded.	Output From Two Computers Which Are In Agreement is Utilized.	Light Is Lit Warning Both Pilots That a First Failure Has Occurred at One of the Voting Planes.	Warning Light to Pilots.	At Evaluation Safety Pilot's Convenience DFCS is Disengaged and Control Returned to the Safety Pilot.	Voting Technique (Best Two of Three) Prevents Single Computer Failure From Causing Erroneous Outputs.	I

System FLT CONTROLS-LATERAL/LONG/DIRECTIONAL
Sybsystem Three Chan-Parallel Flight
Equipment ______ GSE ______
Module ______

FAILURE MODE AND EFFECTS ANALYSIS
NASA TWO PLACE W/STOL RESEARCH AIRCRAFT

TABLE C-3 CONT.

Page 2 of 26 Pages
Date ______
By ______
Approved ______

Item Identification			Reliability Logic Diagram Number	Function	FAILURE MODE	Operation Phase	Failure Effect On			Failure Detection Method	Corrective Action Time Available/Time Required	Design Provisions To Reduce Criticality	Hazard Classification/ Remarks
Name	Ident Number	Drawing Reference Designation					Component/ Functional Assembly	Next Higher Subsystem	Uppermost System				
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)
Motion Sensors (3) Each Chan.			T40.02	Provide Rotational and Translational Motion Data to Computer	Incorrect Inputs to Computer Resulting in Incorrect Input Being Voted Out.		Computers Vote the Inputs from the Motion Sensors and Input Identified as Disagreeing with the Other Two is Disregarded.	Output from Two Channels Which Are In Agreement is Utilized.	Light is Lit Warning Both Pilot's that a First Failure Has Occurred at One of the Voting Planes.	Warning Light to Pilots.	At Evaluation /Safety Pilot's Convenience DFCS Is Disengaged and Control Returned to the Safety Pilot.	Voting Technique (Best Two of Three) Prevents Single Sensor Failure from Causing Erroneous Outputs	I

System FLT CONTROLS - LATERAL/LONGITUDINAL
Subsystem Three Chan-Parallel Flight
Equipment ____ GSE ____
Module ____

FAILURE MODE AND EFFECTS ANALYSIS
NASA TWO PLACE W/STOL RESEARCH AIRCRAFT

TABLE C-3 CONT.

Page 3 of 26 Pages
Date ____
By ____
Approved ____

Item Identification			Reliability Logic Diagram Number	Function	FAILURE MODE	Operation Phase	Failure Effect On			Failure Detection Method	Corrective Action Time Available/Time Required	Design Provisions To Reduce Criticality	Hazard Classification/ Remarks
Name	Ident Number	Drawing Reference Designation					Component/ Functional Assembly	Next Higher Subsystem	Uppermost System				
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)
Rear Stick			T40.03	Evaluation Pilot Input	Jammed at Pivot		Rear Stick Won't Move	No Input to Computer Roll Axis from Rear Seat.	No Lateral /Long. Control From Rear Seat.	None Needed - Readily Apparent to Pilot		Front Seat Can Fly Aircraft	III
					Broken Off at Pivot		Rear Stick Useless, Feel Sys Should Center Transducer	No Rear Seat Input to Computer	No Lateral /Long. Control from Rear Seat.	None Needed Readily Apparent to Pilot		Safety Pilot Can Electrically Disengage Back Seat	III
					Loose at Pivot		No Solid Reaction Point for Rear Seat Input	Erratic Outputs from Rear Stick Transducer	Erratic Lateral/ Long Control from Rear Seat	May Feel Loose to Evaluation Pilot - Both Pilots Should Note Erratic Movement of Aircraft		Safety Pilot Can Electrically Disengage Back Seat	III

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System FLT CONTROLS - LATERAL/LONGITUDINAL

Subsystem Three Chan-Parallel Flight

Equipment ______ GSE ______

Module ______

FAILURE MODE AND EFFECTS ANALYSIS

NASA TWO PLACE W/STOL RESEARCH AIRCRAFT

TABLE C-3 CONT.

Page 4 of 26 Pages

Date ______

By ______

Approved ______

Item Identification: Name (1)	Ident Number (2)	Drawing Reference Designation (3)	Reliability Logic Diagram Number (4)	Function (5)	FAILURE MODE (6)	Operation Phase (7)	Failure Effect On: Component/Functional Assembly (8)	Next Higher Subsystem (9)	Uppermost System (10)	Failure Detection Method (11)	Corrective Action Time Available/Time Required (12)	Design Provisions To Reduce Criticality (13)	Hazard Classification/Remarks (14)
Fwd Stick			T40.04	Safety Pilot Override Input	Jammed at Pivot		Stick Won't Move	No Input to Computer or to Aileron and Stabilator Control Valves	No Long/Lateral System Control with Either Stick	None Needed Readily Apparent to Pilots		Pilot Has Considerable Mechanical Advantage to Push through Some Potential Jams	IV
					Broken Off at Pivot		Fwd Stick Useless	No Front Seat Input to Computer or to Aileron/Stab. Control Valves	No Lat/Long Sys Control from Front Seat	None Needed-Readily Apparent to Front Seat Pilot		Back Seat Can Fly Aircraft by Use of "Emergency" SW	III
					Loose at Pivot		No Solid Reaction Point for Fwd Pilot Input	Erratic or Sloppy Inputs to Long/Lateral System	Erratic Lateral/Long Control from Front Seat	None Needed, Readily Apparent to Pilot		Back Seat Can Fly. Aircraft by Use of "Emergency" SW	III
Fwd Stick Paddle Switch			T40.041	Emergency Electrical Disengage of Back Seat Controls	Failed Closed		Can't Break Circuit	No Disengage	Loss of Stick-Mounted Emergency Disengage Capability	Pre-Flt Checks		Emergency Disengage Available Fwd Stick Force Link	II
					Failed Open		Circuit Won't Make	Engage Solenoid Won't Latch	Constant Disengage of Back Seat Controls	BIT, Pre-Flt Checks		Safety Pilot Can Fly Aircraft	II

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System FLT CONTROLS - LATERAL/LONGITUDINAL
Sybsystem Three Chan-Parallel Flight
Equipment ______ GSE ______
Module ______

FAILURE MODE AND EFFECTS ANALYSIS
NASA TWO PLACE W/STOL RESEARCH AIRCRAFT

TABLE C-3 CONT.

Date ______
By ______
Approved ______

Item Identification: Name	Item Identification: Ident Number	Item Identification: Drawing Reference Designation	Reliability Logic Diagram Number	Function	FAILURE MODE	Operation Phase	Failure Effect On: Component/ Functional Assembly	Failure Effect On: Next Higher Subsystem	Failure Effect On: Uppermost System	Failure Detection Method	Corrective Action Time Available/Time Required	Design Provisions To Reduce Criticality	Hazard Classification/ Remarks
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)
Fwd Stick Force Links (3)			T40.042	Provide Automatic Takeover by Safety Pilot	Jammed or Breakout Too High Breakout Too Low		Computers Note the Inputs from the Force Links and Force Link Input Identified as Dia-greeing with Other Two is Disre-garded	Output from Two Force Links which are in Agree-ment is Utilized.	Light is Lit Warning Both Pilots that a First Failure Has Occurred at One of Voting Planes	Warning Light to the Pilots	At Evaluation, Safety Pilots Convenience DFCS is Disengaged and Control Returned to Safety Pilot.	Voting Technique (Best Two of Three) Prevents Single Force Link Failure from Causing Erroneous Outputs	I

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FAILURE MODE AND EFFECTS ANALYSIS

NASA TWO PLACE V/STOL RESEARCH AIRCRAFT

TABLE C-3 CONT.

System FLT CONTROLS
Subsystem Three Chan-Parallel Flight
Equipment ______ GSE ______
Module ______

Date ______
By ______
Approved ______

Item Identification			Reliability Logic Diagram Number	Function	FAILURE MODE	Operation Phase	Failure Effect On			Failure Detection Method	Corrective Action Time Available/Time Required	Design Provisions To Reduce Criticality	Hazard Classification/ Remarks
Name	Ident Number	Drawing Reference Designation					Component/ Functional Assembly	Next Higher Subsystem	Uppermost System				
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)
Aft Stick Position Transducer (3)			T40.11	Provide Aft Stick Position Data to Computer	Open or Shorted		Distorted or No Output	Voting Technique Utilizes Output from Two Position Transducers which are in agreement	Light is Lit Warning Both Pilots that a First Failure Has Occurred at One of the Voting Planes	Warning Light to the Pilots	At Evaluation Safety Pilots Convenience DFCS is Disengaged and Control Returned to Safety Pilot	Voting Technique (Best Two of Three) Prevents Single Failure from Causing Erroneous Outputs	I
Aft Stick Force Transducer (3)			T40.12	Provide Aft Stick Force Data to Computer	Open or Shorted		Distorted or No Output	Voting Technique Utilizes Output from Two Force Transducers which are in agreement	Light is Lit Warning Both Pilots that a First Failure Has Occurred at One of the voting Planes	Warning Light to the Pilots	At Evaluation Safety Pilots Convenience DFCS is Disengaged and Control Returned to Safety Pilot	Voting Technique (Best Two of Three) Prevents Single Failure from Causing Erroneous Outputs	I
Aft Stick Variable Feel (Torque Motor)			T40.13	Provide Q-programmed Variable Feel to Rear Seat Stick	Jammed		Rear Stick Immobilized	No Stick Position Transducer Output	Possible Reduction in Ability to Fly Aircraft from Back Seat	BIT, Inflt Monitor		BIT, Inflt Monitor Force Transducer Still Operational. Safety Pilot can Disengage Back Seat and Fly Aircraft	III

(Continued Next Page)

FAILURE MODE AND EFFECTS ANALYSIS
NASA TWO PLACE V/STOL RESEARCH AIRCRAFT

TABLE C-3 CONT.

System FLT CONTROLS - LATERAL
Sybsystem Three Chan-Parallel Flight ____
Equipment ____ GSE ____
Module ____

Date ____
By ____
Approved ____

Item Identification			Reliability Logic Diagram Number	Function	FAILURE MODE	Operation Phase	Failure Effect On			Failure Detection Method	Corrective Action Time Available/Time Required	Design Provisions To Reduce Criticality	Hazard Classification/ Remarks
Name	Ident Number	Drawing Reference Designation					Component/ Functional Assembly	Next Higher Subsystem	Uppermost System				
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)
(Continued) Aft Stick Variable Feel (Torque Motor)			T40.13 (Continued)	(Continued) Provide Q-programmed Variable Feel to Rear Seat Stick	Shorted or Open	---	Little or No Resistance to Stick Movement	Little or No Force Transducer Output	Little or No Tactile Cues for Stick Positioning, Tendency to Over-control Via Position Sensor	None Needed-Readily Apparent to Back Seat Pilot		BIT, Inflt Monitor Position Transducer Still Operational Safety Pilot Can Disengage Back Seat and Fly Aircraft	II
Parallel Servos (3) (Electrical)			T40.14E	Move Control Rod Linkage as Commanded by Computer in Response to Aft Stick Transducer Outputs	Jammed (1)		Won't Move	None-Differential Output Assures Output will Be Unaffected by Jam of One Servo	None	In-Flight Monitoring Detects Jammed Servo and Lights Light Warning Safety and Evaluation Pilots of a Failure in the System.		Differential Output Assures Output Will Be Unaffected By Jam of One Servo. Safety Pilot Can Disengage Back Seat and Fly Aircraft.	II

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System FLT CONTROLS - LATERAL
Subsystem Three Chan-Parallel Flight_____
Equipment_____ GSE_____
Module_____

FAILURE MODE AND EFFECTS ANALYSIS

NASA TWO PLACE V/STOL RESEARCH AIRCRAFT

TABLE C-3 CONT.

Date_____
By_____
Approved_____

Item Identification			Reliability Logic Diagram Number	Function	FAILURE MODE	Operation Phase	Failure Effect On			Failure Detection Method	Corrective Action Time Available/Time Required	Design Provisions To Reduce Criticality	Hazard Classification/ Remarks
Name	Ident Number	Drawing Reference Designation					Component/ Functional Assembly	Next Higher Subsystem	Uppermost System				
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)
(Continued) Parallel Servos (3) (Electrical)			T40.014E		(Continued) Burned Out or Shorted		Little or No Movement in One Channel	Loss of One Channel	Loss of Redundancy	Motor Tach		Brake to Prevent Open Drive Train	I
					Stripped Gears or Broken Shaft-Motor to Differential		No Output or Brake Action	Open Drive Train	Loss of Back Seat Control in Affected Axis	No Response to Back Seat Controls in Affected Axis		Safety Pilot Can Take Over Control	III
					Broken Rod End or Head End Attachment (Common)		No Reaction Point for Command Forces	No Output	Rear Seat Controls Useless	Problem Readily Apparent to Rear Pilot		Normal Operation from Front Seat with Rear Seat Controls Electrically Disengaged Design Margin in Excess of 100%	III

System FLIGHT CONTROLS - LATERAL
Subsystem Three Chan-Parallel Flight ____
Equipment ____ GSE ____
Module ____

FAILURE MODE AND EFFECTS ANALYSIS

NASA TWO PLACE V/STOL RESEARCH AIRCRAFT

TABLE C-3 CONT.

Date ____
By ____
Approved ____

Item Identification			Reliability Logic Diagram Number	Function	FAILURE MODE	Operation Phase	Failure Effect On			Failure Detection Method	Corrective Action Time Available/Time Required	Design Provisions To Reduce Criticality	Hazard Classification/ Remarks
Name (1)	Ident Number (2)	Drawing Reference Designation (3)	(4)	(5)	(6)	(7)	Component/ Functional Assembly (8)	Next Higher Subsystem (9)	Uppermost System (10)	(11)	(12)	(13)	(14)
Parallel Servo Safety Spring Cartridges (3)			T40.15	Couple Parallel Servo to Mech Control System and Provide Capabilities for Mech Sys to Override Electrical System	Jammed (1)		Cartridge Won't Collapse or Extend	Direct Link Between Servo and Mech System	Loss of Direct Mechanical Override Protection No Effect Unless Coupled with an Electrical System Failure			Single Jam in Cartridge not Significant a Second Failure in Back Seat Controls Required to Cause Problems.	II
					Broken Spring		No Resistance in One Direction	Servo Could Not Move Linkage in One Direction	Problems Would Be Noted by Pilot as Erratic Response to Back Seat Commands	BIT Prior to Flight Pilot Observation in Flight		Safety Pilot Can Electrically Disengage Back Seat Controls and Fly Aircraft From Front Seat	III
					Broken Attachments		No Resistance in Either Direction	Servo Could Not Move Linkage in Either Direction	No Response to Back Seat Commands	Readily Apparent to Back Seat Pilot		Safety Pilot Can Electrically Disengage Back Seat Controls and Fly Aircraft from Front Seat	III

System FLIGHT CONTROLS - LATERAL
Subsystem Three Chan-Parallel Flight
Equipment ____ GSE ____
Module ____

FAILURE MODE AND EFFECTS ANALYSIS

NASA TWO PLACE V/STOL RESEARCH AIRCRAFT

TABLE C-3 CONT.

Date ____
By ____
Approved ____

Item Identification: Name (1)	Item Identification: Ident Number (2)	Item Identification: Drawing Reference Designation (3)	Reliability Logic Diagram Number (4)	Function (5)	FAILURE MODE (6)	Operation Phase (7)	Failure Effect On: Component/ Functional Assembly (8)	Failure Effect On: Next Higher Subsystem (9)	Failure Effect On: Uppermost System (10)	Failure Detection Method (11)	Corrective Action Time Available/Time Required (12)	Design Provisions To Reduce Criticality (13)	Hazard Classification/ Remarks (14)
Front Seat (Mechanical) Trim			T40.16	Reacts to Pilot Beep Switch Command to Trim Ailerons by Changing Stick Neutral Point	Open		Winding Burnout	Trim Motor Won't Run	No Mechanical Trim Capability, Trim Fixed at One Position			Pilot Can Hold in Trim with Stick, Rear Seat Pilot Can Put in Trim Bias Electrically	II
					Short		Shorted Winding	Sluggish Operation, May Trip Circuit Breaker	Poor or No Mechanical Trip Operation			Circuit Breaker to Prevent Overheat, Rear Seat Pilot Can Put in Trim Bias Electrically	II
					Jammed, or Stripped Gearing		Jammed or Stripped	No Trim Output	Loss of Mech Trim Capability, Trim Fixed at One Position			Pilot Can Hold in Trim with Stick, Rear Seat Pilot Can Put in Trim Bias Electrically	II
					Loss of Structural Ground (Attachments Broken)		No Reaction Point for Trim Motor Torque	No Trim Motion	Loss of Mechanical Trim Capability			Pilot Can Hold in Trim with Stick, Rear Seat Pilot Can Put in Trim Bias Electrically	II

FAILURE MODE AND EFFECTS ANALYSIS

NASA TWO PLACE V/STOL RESEARCH AIRCRAFT

TABLE C-3 CONT.

System FLT. CONTROLS - LATERAL

Sybsystem Three Chan-Parallel Flight

Equipment ______ GSE ______

Module ______

Page 11 of 26 Pages

Date ______

By ______

Approved ______

Item Identification			Reliability Logic Diagram Number	Function	FAILURE MODE	Operation Phase	Failure Effect On			Failure Detection Method	Corrective Action Time Available/Time Required	Design Provisions To Reduce Criticality	Hazard Classification/ Remarks
Name	Ident Number	Drawing Reference Designation					Component/ Functional Assembly	Next Higher Subsystem	Uppermost System				
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)
Front Stick Feel Spring			T40.17	Provides Artificial Feel for Fwd. Pilot	Jammed		Stick Won't Move	No Input to Computer or to Aileron Control Valves	No Lateral System Control with Either Stick	None Needed-Readily Apparent to Pilots		Pilot Has Considerable Mechanical Advantage to Push through Some Potential Jams	IV
					Broken Spring		Absence of Feel and Trim in One Direction	Unbalanced Feel Forces, Erratic Trim	Difficult to Fly Aircraft with Front Stick	None Needed-Readily Apparent to Pilot		With Hands Off in Front Seat, Back Seat Pilot Can Fly Aircraft Electrically. "Emergency" SW Can be Used to Prevent DFCS Disengage.	III
					Broken Head End or Rod End Attachments		Absence of Feel Forces and Trim Position in Both Directions	Loss of Tactile Cues for Proper Stick Positioning	Difficult to Fly Aircraft with Front Stick	None Needed-Readily Apparent to Pilot		With Hands Off in Front Seat, Back Seat Pilot Can Fly Aircraft Electrically. "Emergency" SW Can Be Used to Prevent DFCS Disengage.	III

FAILURE MODE AND EFFECTS ANALYSIS
NASA TWO PLACE V/STOL RESEARCH AIRCRAFT

System FLT CONTROLS - LONGITUDINAL
Sybsystem Three Chan-Parallel Flight
Equipment ______ GSE ______
Module ______

Date ______
By ______
Approved ______

TABLE C-3 CONT.

Item Identification: Name (1)	Ident Number (2)	Drawing Reference Designation (3)	Reliability Logic Diagram Number (4)	Function (5)	FAILURE MODE (6)	Operation Phase (7)	Failure Effect On: Component/ Functional Assembly (8)	Next Higher Subsystem (9)	Uppermost System (10)	Failure Detection Method (11)	Corrective Action Time Available/Time Required (12)	Design Provisions To Reduce Criticality (13)	Hazard Classification/ Remarks (14)
Aft Stick Position Transducer (2)			T40.21	Provide Aft Stick Position Data to Computer	Open or Shorted		Distorted or No Output	Voting Technique Utilizes Output from Two Position Transducers which are in Agreement	Light is Lit Warning Both Pilots that a First Failure Has Occurred at One of the Voting Planes	Warning Light to the Pilots	At Evaluation /Safety Pilots' Convenience DFCS is Disengaged and Control Returned to Safety Pilot	Voting Technique (Best Two of Three) Prevents Single Failure from Causing Erroneous Outputs	I
Aft Stick Force Transducer			T40.22	Provide Aft Stick Force Data to Computer	Open or Shorted		Distorted or No Output	Voting Technique Utilizes Output from Two Position Transducers which are in Agreement	Light is Lit Warning Both Pilots that a First Failure Has Occurred at One of the Voting Planes	Warning Light to the Pilots	At Evaluation /Safety Pilots' Convenience DFCS is Disengaged and Control Returned to Safety Pilot	Voting Technique (Best Two of Three) Prevents Single Failure from Causing Erroneous Outputs	I
Aft Stick Variable Feel (Torque Motor)			T40.23	Provide Q-programmed Variable Feel to Rear Seat Stick	Jammed		Rear Stick Immobilized	No Stick Position Transducer Output	Possible Reduction in Ability to Fly Aircraft from Back Seat	BIT, Inflt Monitor		BIT, Inflt Monitor Force Transducer Still Operational. Safety Pilot can Disengage Back Seat and Fly Aircraft	III

(Continued Next Page)

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FAILURE MODE AND EFFECTS ANALYSIS

NASA TWO PLACE V/STOL RESEARCH AIRCRAFT

System FLT CONTROLS - LONGITUDINAL
Subsystem Three Chan-Parallel Flight ______
Equipment ______ GSE ______
Module ______

Date ______
By ______
Approved ______

TABLE C-3 CONT.

Item Identification: Name (1)	Ident Number (2)	Drawing Reference Designation (3)	Reliability Logic Diagram Number (4)	Function (5)	FAILURE MODE (6)	Operation Phase (7)	Failure Effect On: Component/ Functional Assembly (8)	Next Higher Subsystem (9)	Uppermost System (10)	Failure Detection Method (11)	Corrective Action Time Available/Time Required (12)	Design Provisions To Reduce Criticality (13)	Hazard Classification/ Remarks (14)
(Continued) Aft Stick Variable Feel (Torque Motor)			T40.23 (Continued)	(Continued) Provide Q-programmed Variable Feel to Rear Seat Stick	Shorted or Open		Little or No Resistance to Stick Movement	Little or No Force Transducer Output	Little or No Tactile Cues for Stick Positioning, Tendency to Over-control Via Position Sensor	None Needed-Readily Apparent to Back Seat Pilot		BIT, Inflt Monitor Position Transducer Still Operational Safety Pilot Can Disengage Back Seat and Fly Aircraft	II
Parallel Servo (3) (Electrical)			T40.24E	Move Control Rod Linkage as Commanded by Computer in Response to Aft Stick Transducer Outputs	Jammed (1)		Won't Move	None-Differential Output Assures Output Will Be Unaffected by Jam of One Servo	None	In-Flight Monitoring Defects Jammed Servo and Lights Light Warning Safety and Evaluation Pilots of a Failure in the System.		Differential Output Assures Output Will Be Unaffected by Jam Of One Servo. Safety Pilot Can Disengage Back Seat and Fly Aircraft	II

(Continued Next Page)

FAILURE MODE AND EFFECTS ANALYSIS
NASA TWO PLACE V/STOL RESEARCH AIRCRAFT

TABLE C-3 CONT.

System FLT CONTROLS - LONGITUDINAL
Subsystem Three Chan-Parallel Flight ______
Equipment ______ GSE ______
Module ______

Date ______
By ______
Approved ______

Item Identification			Reliability Logic Diagram Number	Function	FAILURE MODE	Operation Phase	Failure Effect On			Failure Detection Method	Corrective Action Time Available/Time Required	Design Provisions To Reduce Criticality	Hazard Classification/ Remarks
Name	Ident Number	Drawing Reference Designation					Component/ Functional Assembly	Next Higher Subsystem	Uppermost System				
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)
(Continued) Parallel Servos (3) (Electrical)			T40.14E		(Continued) Burned Out or Shorted		Little or No Movement	Brake is Set to Allow Differential to Operate	None-Brake Setting Allows Normal Output.	In-Flight Monitoring Detects Fault and Lights Light Indicating Failure in System.		Brake on Servo Output to Differential Prevents Loss of Output with Loss (free-wheeling) of Single Servo. Safety Pilot Can Disengage Back Seat and Fly Aircraft	II
					Broken Rod End or Head End Attachment (Common)		No Reaction Point for Command Forces	No Output	Rear Seat Controls Useless	Problem Readily Apparent to Rear Pilot		Normal Operation from Front Seat with Rear Seat Controls Electrically Disengaged Design Margins in Excess of 100%	III

System FLIGHT CONTROLS - LONGITUDINAL
Sybsystem Three Chan-Parallel Flight
Equipment ____________ GSE ____________
Module ____________

FAILURE MODE AND EFFECTS ANALYSIS

NASA TWO PLACE V/STOL RESEARCH AIRCRAFT

TABLE C-3 CONT.

Date ____________
By ____________
Approved ____________

Item Identification			Reliability Logic Diagram Number	Function	FAILURE MODE	Operation Phase	Failure Effect On			Failure Detection Method	Corrective Action Time Available/Time Required	Design Provisions To Reduce Criticality	Hazard Classification/ Remarks
Name	Ident Number	Drawing Reference Designation					Component/ Functional Assembly	Next Higher Subsystem	Uppermost System				
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)
Parallel Servo Safety Spring Cartridge			T40.25	Couple Parallel Servo to Mech Control System and Provide Capabili- ties for Mech Sys to Override Electrical System	Jammed		Cartridge Won't Collapse or Extend	Direct Link Between Servo and Mech System	Loss of Direct Mechanical Override Protection No Effect Unless Coupled with an Electrical System Failure			Single Jam in Cartridge not Significant a Second Failure in Back Seat Controls Required to Cause Problems.	II
					Broken Spring		No Resistance in One Direction	Servo Could Not Move Linkage in One Direction	Problems Would Be Noted by Pilot as Erratic Response to Back Seat Commands	BIT Prior to Flight Pilot Observa- tion in Flight		Safety Pilot Can Electrical- ly Disen- gage Back Seat Controls and Fly Aircraft From Front Seat	III
					Broken Attachments		No Resistance in Either Direction	Servo Could Not Move Link- age in Either Direction	No Response to Back Seat Commands	Readily Apparent to Back Seat Pilot		Safety Pilot Can Electrical- ly Disen- gage Back Seat Controls and Fly Aircraft from Front Seat	III

FAILURE MODE AND EFFECTS ANALYSIS

NASA TWO PLACE V/STOL RESEARCH AIRCRAFT

TABLE C-3 CONT.

System FLIGHT CONTROLS- LONGITUDINAL
Subsystem Three Chan-Parallel Flight
Equipment GSE
Module

Date
By
Approved

Item Identification			Reliability Logic Diagram Number	Function	FAILURE MODE	Operation Phase	Failure Effect On			Failure Detection Method	Corrective Action Time Available/Time Required	Design Provisions To Reduce Criticality	Hazard Classification/ Remarks
Name (1)	Ident Number (2)	Drawing Reference Designation (3)	(4)	(5)	(6)	(7)	Component/ Functional Assembly (8)	Next Higher Subsystem (9)	Uppermost System (10)	(11)	(12)	(13)	(14)
Front Seat (Mechanical) Trim			T40.26	Reacts to Pilot Beep Switch Command to Trim Stabilator by Changing Stick Neutral Point	Open		Winding Burnout	Trim Motor Won't Run	No Mechanical Trim Capability, Trim Fixed at One Position			Pilot Can Hold in Trim with Stick, Rear Seat Pilot Can Put in Trim Bias Electrically	II
					Short		Shorted Winding	Sluggish Operation, May Trip Circuit Breaker	Poor or No Mechanical Trip Operation			Circuit Breaker to Prevent Overheat, Rear Seat Pilot Can Put in Trim Bias Electrically	II
					Jammed or Stripped Gearing		Jammed or Stripped	No Trim Output	Loss of Mech Trim Capability Trim Fixed at One Position			Pilot Can Hold in Trim with Stick, Rear Seat Pilot Can Put in Trim Bias Electrically	II
					Loss of Structural Ground (Attachments Broken)		No Reaction Point for Trim Motor Torque	No Trim Motion	Loss of Mechanical Trim Capability			Pilot Can Hold in Trim with Stick, Rear Seat Pilot Can Put in Trim Bias Electrically	II

System FLT. CONTROLS - LONGITUDINAL

Subsystem Three Chan-Parallel Flight

Equipment ____ GSE ____

Module ____

FAILURE MODE AND EFFECTS ANALYSIS

NASA TWO PLACE V/STOL RESEARCH AIRCRAFT

TABLE C-3 CONT.

Date ____

By ____

Approved ____

Item Identification			Reliability Logic Diagram Number	Function	FAILURE MODE	Operation Phase	Failure Effect On			Failure Detection Method	Corrective Action Time Available/Time Required	Design Provisions To Reduce Criticality	Hazard Classification/ Remarks
Name	Ident Number	Drawing Reference Designation					Component/ Functional Assembly	Next Higher Subsystem	Uppermost System				
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)
Front Stick Feel Spring			T40.27	Provides Artificial Feel for Fwd Pilot	Jammed		Stick Won't Move	No Input to Computer or to Stabilator Control Valves	No Pitch System Control with Either Stick	None Needed-Readily Apparent to Pilots		Pilot Has Considerable Mechanical Advantage to Push through Some Potential Jams	IV
					Broken Spring		Absence of Feel and Trim in One Direction	Unbalanced Feel Forces, Erratic Trim	Difficult to Fly Aircraft with Front Stick	None Needed-Readily Apparent to Pilot		With Hands Off in Front Seat, Back Seat Pilot Can Fly Aircraft Electrically. "Emergency" SW Can be Used to Prevent DFCS Disengage.	III
					Broken Head End or Rod End Attachments		Absence of Feel Forces and Trim Position in Both Directions	Loss of Tactile Cues for Proper Stick Positioning	Difficult to Fly Aircraft with Front Stick	None Needed-Readily Apparent to Pilot		With Hands Off in Front Seat, Back Seat Pilot Can Fly Aircraft Electrically. "Emergency" SW Can Be Used to Prevent DFCS Disengage.	III

FAILURE MODE AND EFFECTS ANALYSIS

NASA TWO PLACE V/STOL RESEARCH AIRCRAFT

TABLE C-3 CONT.

System FLT CONTROLS - LONGITUDINAL
Subsystem Three Chan-Parallel Flight ______
Equipment Pitch Axis GSE ______
Module ______

Date ______
By ______
Approved ______

Item Identification			Reliability Logic Diagram Number	Function	FAILURE MODE	Operation Phase	Failure Effect On			Failure Detection Method	Corrective Action Time Available/Time Required	Design Provisions To Reduce Criticality	Hazard Classification/ Remarks
Name	Ident Number	Drawing Reference Designation					Component/ Functional Assembly	Next Higher Subsystem	Uppermost System				
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)
Forward RCS Valve Series Actuator			T40.28	Provide SAS Capability on front RCS Valve	Jammed to bellcrank		Won't move relative to bellcrank	Solid link Stick to RCS Valve	Loss of front RCS SAS. Valve still operational	BIT, in-flt monitor		A/C still completely flyable with increase pilot work-load. Some pitch RCS SAS still available via rear RCS Valve. No interferences with conventional landing.	II
					broken, burned out or shorted Which Allows Free Movement of Cylinder		Servo inop	No transmittal of pilot commond	loss of use of front RCS Valve. VTOL operations difficult with fwd C.G. Aircraft	BIT, in-flight monitor		BIT or in-flt monitor warns. pilot not to attempt vertical operations with failed actuator. No interference with conventional landing.	II

FAILURE MODE AND EFFECTS ANALYSIS

NASA TWO PLACE V/STOL RESEARCH AIRCRAFT

TABLE C-3 CONT.

System FLT CONTROLS – DIRECTIONAL
Subsystem Three Chan-Parallel Flight
Equipment ______ GSE ______
Module ______

Date ______
By ______
Approved ______

Item Identification			Reliability Logic Diagram Number	Function	FAILURE MODE	Operation Phase	Failure Effect On			Failure Detection Method	Corrective Action Time Available/Time Required	Design Provisions To Reduce Criticality	Hazard Classification/ Remarks
Name	Ident Number	Drawing Reference Designation					Component/ Functional Assembly	Next Higher Subsystem	Uppermost System				
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)
Fwd Pedal			T40.38	Safety Pilot Override Input	Jammed at Pivot		Pedals Won't Move	No Input to Computer or to Rudder	No Directional System Control	None Needed Readily Apparent to Pilots		Pilot Has Considerable Mechanical Advantage to Push through Some Potential Jams	III (At Altitude) IV (Low Level)
					Broken Off at Pivot		Fwd Pedals Useless	No Front Seat Input to Computer or to Rudder	No Directional Sys Control from Front Seat	None Needed- Readily Apparent to Front Seat Pilot		Back Seat Can Fly Aircraft by Use of "Emergency" SW	III
					Loose at Pivot		No Solid Reaction Point for Fwd Pilot Input	Erratic or Sloppy Inputs Directional System	Erratic Directional Control from Front Seat	None Needed, Readily Apparent to Pilot		Back Seat Can Fly Aircraft by Use of "Emergency" SW	III

PTR

FAILURE MODE AND EFFECTS ANALYSIS

NASA TWO PLACE V/STOL RESEARCH AIRCRAFT

TABLE C-3 CONT.

System FLT CONTROLS - DIRECTIONAL
Subsystem Three Chan-Parallel
Equipment
Module
Flight
GSE

Date
By
Approved

Item Identification			Reliability Logic Diagram Number	Function	FAILURE MODE	Operation Phase	Failure Effect On			Failure Detection Method	Corrective Action Time Available/Time Required	Design Provisions To Reduce Criticality	Hazard Classification/ Remarks
Name	Ident Number	Drawing Reference Designation					Component/ Functional Assembly	Next Higher Subsystem	Uppermost System				
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)
Fwd Pedal Force Link			T40.39	Provide Automatic Takeover by Safety Pilot	Incorrect Input to Computer Resulting in Incorrect Input Being Voted Out.		Computers Vote the Inputs from the Motion Sensors and Input Identified as Disagreeing with the Other Two is Disregarded.	Output from Two Channels Which Are In Agreement is Utilized.	Light is Lit Warning Both Pilot's that a First Failure Has Occurred at One of the Voting Planes.	Warning Light to Pilots.	At Evaluation/ Safety Pilot's Convenience DFCS Is Disengaged and Control Returned to the Safety Pilot.	Voting Technique (Best Two of Three) Prevents Single Sensor Failure from Causing Erroneous Outputs	I

System FLT. CONTROLS - DIRECTIONAL
Subsystem Three Chan-Parallel Flight
Equipment GSE
Module

FAILURE MODE AND EFFECTS ANALYSIS
NASA TWO PLACE V/STOL RESEARCH AIRCRAFT

TABLE C-3 CONT.

Date
By
Approved

Item Identification			Reliability Logic Diagram Number	Function	FAILURE MODE	Operation Phase	Failure Effect On			Failure Detection Method	Corrective Action Time Available/Time Required	Design Provisions To Reduce Criticality	Hazard Classification/ Remarks
Name	Ident Number	Drawing Reference Designation					Component/ Functional Assembly	Next Higher Subsystem	Uppermost System				
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)
Front Pedal Feel Spring			T40.37	Provides Artificial Feel for Fwd Pilot	Jammed		Pedals Won't Move	No Input to Computer or to Rudder Control Valves	No Directional System Control with Either Stick	None Needed-Readily Apparent to Pilots		Pilot Has Considerable Mechanical Advantage to Push through Some Potential Jams	III (At Altitude) IV (Low Level)
					Broken Spring		Absence of Feel in One Direction	Unbalanced Feel Forces	Difficult to Fly Aircraft with Front Pedals	None Needed-Readily Apparent to Pilot		With Hands Off in Front Seat, Back Seat Pilot Can Fly Aircraft Electrically. "Emergency" SW Can be Used to Prevent DFCS Dis-engage.	III
					Broken Head End or Rod End Attachments		Absence of Feel Forces in Both Directions	Loss of Tactile Cues for Proper Pedal Positioning	Difficult to Fly Aircraft with Front Pedals	None Needed-Readily Apparent to Pilot		With Hands Off in Front Seat, Back Seat Pilot Can Fly Aircraft Electrically. "Emergency" SW Can Be Used to Prevent DFCS Dis-engage.	III

FAILURE MODE AND EFFECTS ANALYSIS

NASA TWO PLACE V/STOL RESEARCH AIRCRAFT

TABLE C-3 CONT.

System FLT CONTROLS - DIRECTIONAL

Subsystem Three Chan-Parallel Flight ________

Equipment ________ GSE ________

Module ________

Date ________

By ________

Approved ________

Item Identification			Reliability Logic Diagram Number	Function	FAILURE MODE	Operation Phase	Failure Effect On			Failure Detection Method	Corrective Action Time Available/Time Required	Design Provisions To Reduce Criticality	Hazard Classification/ Remarks
Name	Ident Number	Drawing Reference Designation					Component/ Functional Assembly	Next Higher Subsystem	Uppermost System				
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)
Rear Pedals			T40.38	Evaluation Pilot Input	Jammed at Pivot		Rear Pedals Won't Move	No Input to Computer Yaw Axis from Rear Seat.	No Directional Control From Rear Seat.	None Needed - Readily Apparent to Pilot		Front Seat Can Fly Aircraft	III
					Broken Off at Pivot		Rear Pedals Useless, Feel Sys Should Center Transducer	No Rear Seat Input to Computer	No Directional Control from Rear Seat.	None Needed Readily Apparent to Pilot		Safety Pilot Can Electrically Disengage Back Seat	III
					Loose at Pivot		No Solid Reaction Point for Rear Seat Input	Erratic Outputs from Rear Pedal Transducer	Erratic Directional Control from Rear Seat	May Feel Loose to Evaluation Pilot - Both Pilots Should Note Erratic Movement of Aircraft		Safety Pilot Can Electrically Disengage Back Seat	III

System FLT CONTROLS- DIRECTIONAL
Subsystem Three Chan-Parallel Flight
Equipment
GSE
Module

FAILURE MODE AND EFFECTS ANALYSIS
NASA TWO PLACE V/STOL RESEARCH AIRCRAFT

TABLE C-3 CONT.

Date
By
Approved

Item Identification			Reliability Logic Diagram Number	Function	FAILURE MODE	Operation Phase	Failure Effect On			Failure Detection Method	Corrective Action Time Available/Time Required	Design Provisions To Reduce Criticality	Hazard Classification/ Remarks
Name	Ident Number	Drawing Reference Designation					Component/ Functional Assembly	Next Higher Subsystem	Uppermost System				
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)
Aft Pedal Position Transducer			T40.31	Provide Aft Pedal Position Data to Computer	Incorrect Inputs to Computer Resulting in Incorrect Input Being Voted Out.		Computers Vote the Inputs from the Motion Sensors and Input Identified as Disagreeing with the Other Two is Disregarded.	Output from Two Channels Whith Are in Agreement is Utilized.	Light is Lit Warning Both Pilot's that a First Failure Has Occurred at One of the Voting Planes.	Warning Light to Pilots.	At Evaluation/Safety Pilot's Convenience DFCS Is Disengaged and Control Returned to the Safety Pilot.	Voting Technique (Best Two of Three) Prevents Single Sensor Failure from Causing Erroneous Outputs.	I
Aft Pedal Force Transducer			T40.32	Provide Aft Pedal Force Data to Computer	Incorrect Inputs to Computer Resulting in Incorrect Input Being Voted Out		Computers Vote the Inputs from the Motion Sensors and Input Identified as Disagreeing with the Other Two is Disregarded.	Output from Two Channels Which Are in Agreement is Utilized.	Light is Lit Warning Both Pilot's that a First Failure Has Occurred at One of the Voting Planes.	Warning Light to Pilots.	At Evaluation/Safety Pilot's Convenience DFCS Is Disengaged and Control Returned to the Safety Pilot.	Voting Technique (Best Two of Three) Prevents Single Sensor Failure from Causing Erroneous Outputs.	I
Aft Pedal Variable Feel (Torque Motor)			T40.33	Provide Variable Feel to Rear Seat Pedals	Jammed		Rear Pedals Immobilized	No Pedal Position Transducer Output	Possible Reduction in Ability to Fly Aircraft from Back Seat	BIT, Inflt Monitor		BIT, Inflt Monitor Force Transducer Still Operational. Safety Pilot can Disengage Back Seat and Fly Aircraft	III

(Continued Next Page)

FAILURE MODE AND EFFECTS ANALYSIS

NASA TWO PLACE V/STOL RESEARCH AIRCRAFT

TABLE C-3 CONT.

System FLT CONTROLS - DIRECTIONAL
Subsystem Three Chan-Parallel Flight
Equipment ______ GSE ______
Module ______

Date ______
By ______
Approved ______

Item Identification: Name (1)	Item Identification: Ident Number (2)	Item Identification: Drawing Reference Designation (3)	Reliability Logic Diagram Number (4)	Function (5)	FAILURE MODE (6)	Operation Phase (7)	Failure Effect On: Component/Functional Assembly (8)	Failure Effect On: Next Higher Subsystem (9)	Failure Effect On: Uppermost System (10)	Failure Detection Method (11)	Corrective Action Time Available/Time Required (12)	Design Provisions To Reduce Criticality (13)	Hazard Classification/Remarks (14)
(Continued) Aft Pedal Variable Feel (Torque Motor)			T40.33 (Continued)	(Continued) Provide Variable Feel to Rear Seat Pedals	Shorted or Open		Little or No Resistance to Pedal Movement	Little or No Force Transducer Output	Little or No Tactile Cues for Pedal Positioning, Tendency to Over-control Via Position Sensor	None Needed-Readily Apparent to Back Seat Pilot		BIT, Inflt Monitor Position Transducer Still Operational Safety Pilot Can Disengage Back Seat and Fly Aircraft	II
Parallel Servo (3) (Electrical)			T40.34E	Move Control Rod Linkage as Commanded by Computer in Response to Aft Pedal Transducer Outputs	Jammed		Won't Move	None-Differential Output Assures Output Will Be Unaffected by Jam of One Servo	None	In-flight Monitoring Detects Jammed Servo and Lights Light Warning Safety and Evaluation Pilots of a Failure in the System.		Differential Output Assures Output Will Be Unaffected by Jam of One Servo Safety Pilot Can Disengage Back Seat and Fly Aircraft.	II

(Continued Next Page)

System FLT CONTROLS – DIRECTIONAL
Subsystem Three Chan-Parallel Flight
Equipment GSE
Module

FAILURE MODE AND EFFECTS ANALYSIS
NASA TWO PLACE V/STOL RESEARCH AIRCRAFT

TABLE C-3 CONT.

Date
By
Approved

Item Identification			Reliability Logic Diagram Number	Function	FAILURE MODE	Operation Phase	Failure Effect On			Failure Detection Method	Corrective Action Time Available/Time Required	Design Provisions To Reduce Criticality	Hazard Classification/ Remarks
Name	Ident Number	Drawing Reference Designation					Component/ Functional Assembly	Next Higher Subsystem	Uppermost System				
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)
(Continued) Parallel Servos (3) (Electrical)			T40.034E		(Continued) Burned Out or Shorted		Little or No Movement in One Channel	Loss of One Channel	Loss of Redundancy	Motor Tach		Brake to Prevent Open Drive Train	II
					Stripped Gears or Broken Shaft-Motor to Differential		No Output of Brake Action	Open Drive Train	Loss of Back Seat Control in Affected Axis	No Response to Back Seat Controls in Affected Axis		Safety Pilot Can Take Over Control	III
					Broken Rod End or Head End Attachment (Common)		No Reaction Point for Command Forces	No Output	Rear Seat Controls Useless	Problem Readily Apparent to Rear Pilot		Normal Operation from Front Seat with Rear Seat Controls Electrically Disengaged Design Margin in Excess of 100%	III

FAILURE MODE AND EFFECTS ANALYSIS
NASA TWO PLACE V/STOL RESEARCH AIRCRAFT

TABLE C-1 CONT.

System FLIGHT CONTROLS - DIRECTIONAL
Subsystem Three Chan-Parallel Flight ______
Equipment ______ GSE ______
Module ______

Date ______
By ______
Approved ______

Item Identification: Name (1)	Ident Number (2)	Drawing Reference Designation (3)	Reliability Logic Diagram Number (4)	Function (5)	FAILURE MODE (6)	Operation Phase (7)	Failure Effect On: Component/ Functional Assembly (8)	Next Higher Subsystem (9)	Uppermost System (10)	Failure Detection Method (11)	Corrective Action Time Available/Time Required (12)	Design Provisions To Reduce Criticality (13)	Hazard Classification/ Remarks (14)
Parallel Servo Safety Spring Cartridge			T40.35	Couple Parallel Servo to Mech Control System and Provide Capabilities for Mech Sys to Override Electrical System	Jammed		Cartridge Won't Collapse or Extend	Direct Link Between Servo and Mech System	Loss of Direct Mechanical Override Protection No Effect Unless Coupled with an Electrical System Failure			Single Jam in Cartridge not Significant a Second Failure in Back Seat Controls Required to Cause Problems.	II
					Broken Spring		No Resistance in One Direction	Servo Could Not Move Linkage in One Direction	Problems Would Be Noted by Pilot as Erratic Response to Back Seat Commands	BIT Prior to Flight Pilot Observation in Flight		Safety Pilot Can Electrically Disengage Back Seat Controls and Fly Aircraft From Front Seat	III
					Broken Attachments		No Resistance in Either Direction	Servo Could Not Move Linkage in Either Direction	No Response to Back Seat Commands	Readily Apparent to Back Seat Pilot		Safety Pilot Can Electrically Disengage Back Seat Controls and Fly Aircraft from Front Seat	III

APPENDIX D

ACTION ITEM SUMMARY

During the course of this research program four technical coordination meetings were held:

- 13 October 1977 - Program "Kick Off" Meeting
  (NASA Ames Research Center)

- 14 November 1977 - Design Coordination Meeting
  (NASA Ames Research Center)

- 13 December 1977 - Option Selection Meeting
  (McDonnell Aircraft Company)

- 16 January 1978 - Final Oral Presentation
  (NASA Ames Research Center)

A number of MCAIR and NASA action items were identified at each meeting. Figures D-1 through D-4 list these action items and their disposition.

FIGURE D-1 ACTION ITEM SUMMARY

13 October 1977 Program "Kick Off" Meeting

Responsibility	Subject	Status
NASA	Identify Redundant Control System Concepts to be Studied	Complete
MCAIR	Study Feasibility of Retaining Rear Cockpit Mechanical Controls Connected Until Confidence in FBW is Gained	Complete
MCAIR	MCAIR/NASA Pilots Begin Dialogue	Complete
MCAIR	Can YAV-8B Roll Nozzle Design be Used to Improve Roll Control?	Complete
MCAIR	Determine Control System Frequency Response and Velocities	Complete
MCAIR	Contact NASA to Identify Telemetry Interface Required	Complete
MCAIR	Determine Time Identification of Data Which is not Time Tagged	Not Required
MCAIR	Can Low Speed Sensor be Mounted on Flight Test Boom?	Complete
MCAIR	Contact NASA for LTN-51 INS Data and Usage	Complete
MCAIR	Add Four Weeks Simulation Support at NASA	Complete
MCAIR	Study Feasibility and Cost of Software Verification Prior to Flight	Complete
MCAIR	Select Computer Which Uses HAL/S Higher Order Language and Size Software Accordingly	Complete

FIGURE D-2 ACTION ITEM SUMMARY

14 November 1977 Design Coordination Meeting

Responsibility	Subject	Status
MCAIR	Add Disengage Button to Forward Cockpit Stick and Throttle	Complete
MCAIR	Put Computer Control in Aft Cockpit, Avionics Controls in Forward Cockpit	Complete
MCAIR	Keep Aft Cockpit Stick and Rudder Pedals Mechanically Connected to Control System in Dual Series Configuration	Complete
MCAIR	Compute Frequency Response Between Servo and Forward Stick	Complete
MCAIR	Keep Throttle and Nozzle Levers Mechanically Connected to Present Systems in Dual Series Configuration	Complete

FIGURE D-3 ACTION ITEM SUMMARY

13 December 1977 Option Selection Meeting

Responsibility	Subject	Status
MCAIR	Investigate Feasibility of Using Ferranti INS	Complete
MCAIR	Review Word Number Estimate for Computer Program	Complete
NASA	Provide Measurand List	Incomplete
MCAIR	Add Second Low Speed Air Data Sensor	Complete
MCAIR	Provide Instrument Design Aids for Ron Gerdes Trip to U.K.	Complete
MCAIR	Include NASA Engineering Safety Review and Flight Readiness Review in Schedule	Complete
MCAIR	Revise Support Plan to Four Years/One Site/ 120 Hours Per Year	Complete
MCAIR	Prioritize Aircraft Mods to be Done at NARF	Deferred

C-4

FIGURE D-4 ACTION ITEM SUMMARY

16 January 1978 - Final Oral Presentation

Responsibility	Subject	Status
MCAIR	Include a Computer Cost Comparison in the Final Report	Complete
MCAIR	Provide Technical Explanation of Engine Limitations in Hover	Data Being Obtained, Will be Provided by April, 1978.
MCAIR	Provide Model for Simulating Dynamics Relating Longitudinal Parallel Servo Input to Actuator Valve Input and Forward Stick	Complete
NASA	Initiate review of AV-8A Simulation Data Package currency	Pending
MCAIR	Provide Descriptions of Theory of Operation of Low Speed Air Data Sensor	Complete