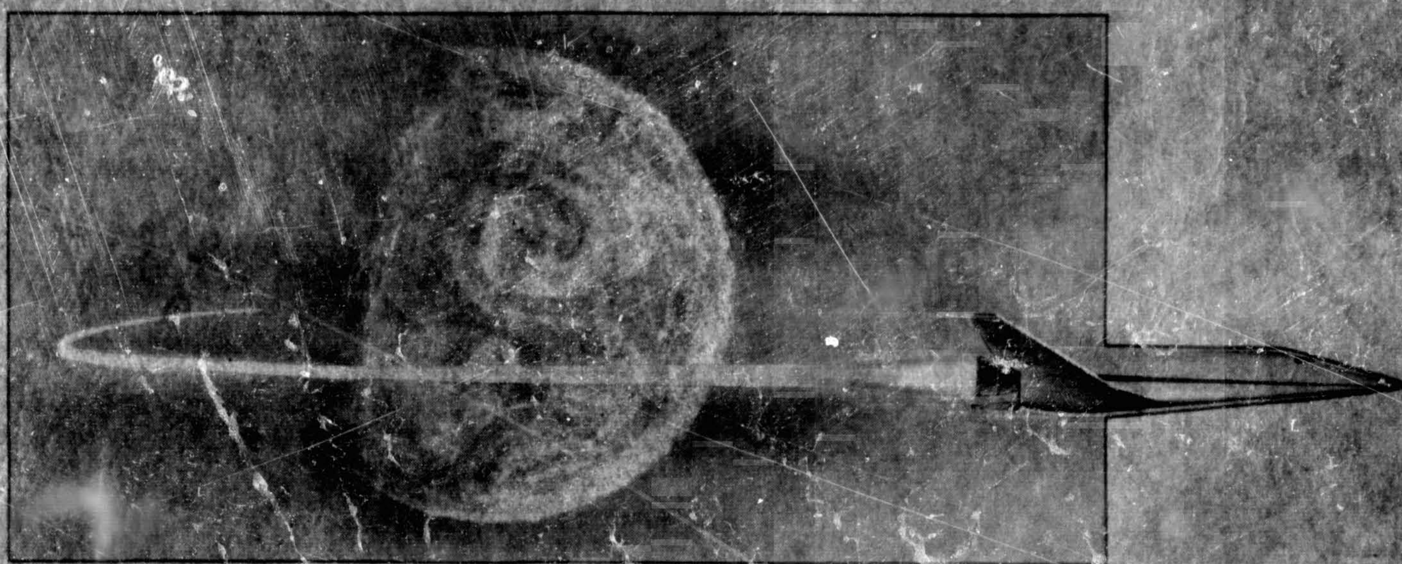
N70-31826
CR-102629

LMSC-A959837
VOL III
DECEMBER 22, 1969



LOCKHEED
LMSC
SPACE SYSTEMS
DIVISION

FINAL REPORT
INTEGRAL LAUNCH AND REENTRY VEHICLE

VOLUME III
SPECIAL STUDIES

PART B – SECTION 4&5
AND APPENDIX B, C & D

FACILITY FORM 602

N70-31826 (ACCESSION NUMBER)	(THRU)
317 (PAGES)	1 (CODE)
NASA CR-102629 (NASA CR OR TMX OR AD NUMBER)	31 (CATEGORY)

FINAL REPORT

INTEGRAL LAUNCH AND REENTRY VEHICLE

Volume III

Special Studies

Prepared by:

Wilson B. Schramm
ILRV Study Manager

Approved by:

J. F. Milton, Program Manager
Manned Space Programs

F. C. E. Oder
Vice President and Assistant General Manager
Space Systems Division

LOCKHEED MISSILES & SPACE COMPANY

LMSC-A959837
Vol.III

FINAL REPORT

INTEGRAL LAUNCH AND

REENTRY VEHICLE

Volume III

Special Studies

Part B - Sections 4 and 5

and Appendixes B, C, and D

FOREWORD

This final report for the Integral Launch and Reentry Vehicle (ILRV) Study, conducted under Contract NAS9-9206 by Lockheed Missiles & Space Company under direction of the NASA Marshall Space Flight Center, is presented in three volumes. Volume I, Configuration Definition and Planning, contains results of the preliminary cost analyses, conceptual design, mission analyses, program planning, cost and schedule analyses, and sensitivity analyses, accomplished under Tasks 1 through 6. Volume II covers Task 7, Technology Identification; and Volume III contains results of the Special Studies conducted under Task 8.

Principal LMSC task leaders and contributors in performance of this study include:

Systems Integration	T. E. Wedge	Primary Engines	A. J. Hief
System Synthesis	J. E. Torrillo	Propulsion	L. L. Morgan
Mission Analysis	D. W. Fellenz	Integrated Avionics	J. J. Herman
Design	G. Havrisik	Safety	J. A. Donnelly
Cost	J. Dippel	Structures	P. P. Plank
Schedule	W. James	Thermodynamics	F. L. Guard
Test	R. W. Benninger	Aerodynamics	C. F. Ehrlich
Operations	K. Urbach	Weights	A. P. Tilley

The three volumes are organized as follows:

Volume I - Configuration Definition and Planning

CONTENTS

Section		Page

Section 4
INTEGRATED ELECTRONICS SYSTEM

"Integrated Electronics" is here considered to be synonymous and interchangeable with the term "Integrated Avionics." As such it encompasses a wide range of functions and equipment. Broadly defined, it may include all Space Shuttle equipment employing electronics to sense, acquire, generate, transmit, process, store, record, or display data required for the operation of any and all vehicle systems, for determining their operational and flight safety status, and for performing onboard launch control and mission control operations. In addition, avionics may include equipment required for electrical power development and energy storage, power control, and power distribution. The software required to accomplish the functions of the individual and inter-related avionics equipment should be considered an integral part of the avionics.

Additional equipment, generally considered part of nonavionic subsystems (e.g., engine controller, propulsion subsystem) should also be treated as avionics equipment in investigations directed at determining the extent of integration that is beneficial. Although some developments may be required or advantageous in individual subsystem areas, the key issue for the Space Shuttle avionics is "How much integration is desirable?"

4.1 OBJECTIVES

The objectives of this study were twofold. Broadly stated, they were as follows:

- For a selected manned Space Shuttle vehicle of the mid-1970s time period, define the system functional requirements for a designated configuration of vehicle subsystems.
- For the designated configuration, identify the extent to which integration of avionics may be beneficially employed. Operational functions of subsystems are to be included in the "integrated system" study.

4.2 SCOPE

It is essential that the scope of this study be clearly understood in order that the conclusions reached are not applied to a more general problem without careful regard for the way in which that problem is structured differently from the one treated in this study. Budgetary and time limitations precluded an exhaustive study of all aspects of the problem of integrating a set of avionics equipment for the Space Shuttle. Ground rules were adopted to reduce the scope of this complex problem and still permit a meaningful preliminary investigation of integration alternatives. The study scope is delineated below:

- The orbiter vehicle of the Space Shuttle was selected, thereby minimizing the dependence of study results on the Shuttle configuration, i.e., the orbiters of the Stage-and-a-Half, Two-Stage, and Triamese configurations were estimated to be functionally similar, with relatively minor variations of detail. Any Shuttle configuration dependence on avionics was to be identified.
- The configuration to be integrated was confined to a single set of vehicle subsystems. The reliability/safety requirements were to be investigated for two cases to indicate the resultant impact.
- The study was confined to technical problems only. Other problems (e.g., maintenance and associated costs) were to be flagged, as time permitted.
- Investigation of the integration of multiple functions into one item of equipment (e.g., multimode radar) was not within the study scope.
- Only major functions of subsystems and major components had to be examined. Subfunctions, tasks, etc., were included as time permitted to better definitize estimates.
- Payload and cargo handling subsystems were specifically excluded from the study.

- The study was directed to the amount of integration that is beneficial. The scope of the study did not include the design of a recommended integrated system.
- A technology freeze date of late 1972 for electronics components was selected to limit the study to considering only those devices or equipment that would be available in time for the Shuttle development program.

4.3 APPROACH

The logistics-resupply mission was selected as most representative for this study, and the mission was partitioned into nine phases plus prelaunch. Nine subsystems were designated, and the functional requirements of each were identified for each mission phase to provide a basis for describing functional interfaces among subsystems and among major functional blocks within each subsystem. The subsystem parameters essential to each function were identified and each was categorized as a control, measure, calculate, or display parameter for each function.

Functional block diagrams were prepared for each subsystem. Interfaces between subsystems and between major blocks of subsystems were tabulated and the signal characteristics of each interface were identified. Test point access requirements (for checkout, fault isolation, and abort warning) were tabulated for each subsystem, and the characteristics of each test point were identified.

Onboard operational support requirements for launch control and for mission control were estimated, with the Apollo/Saturn used as a model from which the orbiter requirement was extrapolated. Computation requirements for vehicle subsystem functions were estimated on the basis of previous or current program requirements for similar functions.

The system functional requirements identified above may be categorized as:

Onboard checkout and fault isolation
Abort warning
Operations support
Interface control
Computation

Three alternative integrated electronics system implementations were investigated. All three implementations employed the same man-machine interface.

The configuration of vehicle subsystems to be integrated was a single-thread configuration, i.e., not configured redundantly to meet reliability and safety requirements. The impact of these requirements was investigated for two specific cases, but study limitations precluded a complete investigation of all subsystems reconfigured redundantly to meet the reliability and safety requirements.

The first alternative IES consisted of conventionally interconnected subsystems and served as a baseline IES configuration. Manual command and control inputs to subsystems were routed directly from control display rather than through a data management subsystem. Configuration control and sequencing functions were performed by respective subsystems upon command, either manual or programmed as appropriate. Also, each subsystem was responsible for its own performance and provided diagnostic information to the data management subsystem. The functions of onboard checkout and fault isolation, abort warning, and operations support were accomplished in Alternative 1.

In the second alternative IES, the functions of Alternative 1 were performed and, in addition, subsystems and major components were interconnected through standardized interfaces and multiplexed data buses, and information and data flow were controlled. The control/display data processing was performed by the data management subsystem.

In the third alternative IES, integration by means of a central computer complex was investigated. All functions performed in Alternative 2, plus computation for subsystem functions, were performed by the centralized system. The only constraint imposed on this alternative IES was the technology of electronic components as projected to the end of 1972. This technology projection was made as part of this study.

4.4 DESIRED CHARACTERISTICS OF AN INTEGRATED ELECTRONICS SYSTEM

The desired characteristics for an integrated electronics system for a manned Space Shuttle are listed below. For brevity, subsystem specific items are not included - only characteristics applicable to the "means of integration" or to the resultant integrated system.

DESIRED IES CHARACTERISTICS

- Safe mission termination capability, including postliftoff intact abort
- Redundant full mission capability
- Ability to fail operational after failure of the two most critical components (for any one function) and fail safe after the third failure
- Designed multiple redundant to minimize or eliminate system transients caused by component failures
- Capable of 30-day missions
- Complete 100 mission cycles with minimum maintenance
- Designed to support "rapid turnaround, minimum ground maintenance" and to use onboard checkout and fault isolation.
- Designed for maximum onboard autonomy (Preflight and inflight checkout capability plus abort warning and mission operations support performed on-board the vehicle.)
- Flexibility to incorporate technology improvements in any area of the system; also, sensitivity of a point design should be low to increased performance requirements or to quantity of avionics equipment to be integrated
- Reduced cabling, use of standard interfaces, and use of standard multiplexed data bus to achieve decreased cable weight, improved interconnect reliability, reduced EMI susceptibility, and ease of incorporating design changes over the program life in equipment with the standard interface
- Reduced complexity of the man-machine interface

- Equipment self-test capability
- Subsystem interfaces defined to aid in system management and to permit independent improvement of any one subsystem without impacting other subsystems.
- Incorporation of multiple functions into one equipment to reduce weight and the number of equipment types
- Best performance attainable through maximum use of latest proven technology

In addition to the characteristics listed above the basic need exists to reduce weight, power, volume, and cost for all avionics equipment.

4.5 VEHICLE SYSTEM FUNCTIONAL REQUIREMENTS

The requirement for safety in manned flight operations dictates the need for redundancy in vehicle subsystem designs and the need for intact abort, much as is the case for commerical and military aircraft. Onboard checkout, fault isolation, and warning of an abort situation must therefore be provided.

In addition, the need for decreased maintenance time on the ground, i.e., rapid turnaround, establishes the need for onboard checkout and fault isolation capability for maintenance purposes. With such capability on board the vehicle, near-autonomous operation becomes an attractive possibility.

Launch operations and mission operations have in the past required the services of many hundreds of skilled personnel and the use of extensive ground facilities. The use of an onboard data management subsystem for operations support is a significant step to achieving the goals of reduced cost and increased efficiency.

The extent to which equipment and functions are to be integrated will be determined on the basis of technical feasibility, reliability, maintainability, flexibility, subsystem autonomy, and the ability to manage interleaved subsystem interfaces and contractor relationships. As discussed in section 4.2, this study was constrained to consideration of a nonredundant configuration of subsystems and to technical problems. The following paragraphs describe the functional requirements of a set of orbiter avionics equipment to be integrated and the methods used to determine these requirements.

4.5.1 Development of Data Requirements

The logistics resupply mission was selected as typical and was divided into the following phases:

- Prelaunch
- Launch and ascent
- Orbit insertion
- Rendezvous
- Docking
- Orbit stay
- Retrograde and deorbit
- Reentry
- Subsonic approach
- Landing

Nine vehicle subsystems were identified, and all equipment was allocated to one of the nine subsystems. These subsystems were:

1.0 Structure/mechanical
2.0 Propulsion
3.0 Electrical power
4.0 Environmental control
5.0 Guidance and Navigation
6.0 Vehicle control
7.0 Communications
8.0 Control/display
9.0 Data management

Subsystem functions were identified for each mission phase, and the parameters essential to each function were characterized as to their need for control, computation, measurement, and display to perform that function. Figure 4.5.1-1 illustrates, for the computer (6.1) of vehicle control subsystem (6.0), the method used to tabulate functions vs mission phase and parameters vs functions.

Subsystem functional block diagrams were prepared at two levels of detail. A second level functional block diagram of part of the vehicle control subsystem (the computer) is illustrated in Fig. 4.5.1-2. Major blocks of each subsystem may be identified in the interconnect diagram (Fig. 4.5.1-11).

Function	Mission Phase																				
	Pre-Launch	Launch and Ascent	Orbit Insertion	Rendezvous	Docking	Orbit Stay	Retrograde and Deorbit	Reentry	Subsonic Approach	Landing	Refurbishment	Rate Error (CMD)	Body Rates p q r	Attack Angle Command	Attack Angle (Computed)	Attack Angle (α)	Bank Angle (CMD)	Bank Angle(Computed)	Side-Slip Angle (CMD)	Side-Slip Angle (β)	Side-Slip Angle
1 6.1 Computer																					
2 Compute Rate Errors		X	X	X	X		X					M	M								
3 Compute Angular Errors								X	X	X				M	D M	D M	M	D M	M	D M	D
4 Provide Stabilization for Rigid Body, Bending & Slosh Modes		X	X	X	X		X	X	X	X			M D								
5 Compute Servo Position CMD's		X	X	X	X		X	X	X	X											
6 Select Operating Modes/ Configuration		X	X	X	X		X	X	X	X											
7 Modulate Reaction Control Valves (PWM)			X	X			X	X													
8 6.4 Control Cont. Surface Servo								X	X	X											
9 6.5 Control Pri Eng Gimbal Servo (TVC)		X	X	X			X														
0 6.7 Control Turbo-Jet Throttle Servo									X	X											
1 6.8 Control Pri Eng Throttle Servo		X	X	X			X														

CP = Compute, D = Display, M = Measure, CO = Control,

FOLDOUT FRAME 1

L

Parameters

Landing	Refurbishment	Rate Error (CMD)	Body Rates p q r	Attack Angle Command	Attack Angle (Computed)	Attack Angle (α)	Bank Angle (CMD)	Bank Angle(Computed)	Side-Slip Angle (CMD)	Side-Slip Angle (β)	Side-Slip Angle	Crew CMD's	Configuration CMD's	Servo Pos. CMD's (δ_c)	Compensation Networks	Gain Schedule CMD's	Gimbal Servo Errors ($\delta\epsilon$)	Pitch Gimbal Pos. Fdbk (δ_{FB})	Yaw Gimbal Pos. Fdbk (δ_{FB})	Gimbal Rate Fdbk (δ_{FB})	Acceleration Error (ϵ_{ax})	Cont. Surface Servo Error (δ_E)	Cont. Surface Pos. (δ_{FB})	Pri Eng Throttle Servo Error ($\delta\epsilon$)	Pri Eng Throttle Pos (δ_{FB})	Turbojet Throttle Servo Error ($\delta\epsilon$)	Turbojet Throttle Pos (δ_{FB})	Reaction Control Valve CMD's ($\delta\epsilon$)	Valve Modulator	Compensated Error Signals	Rate Error	Angular Error	Body Temp
		M	M																														
X				M	D M	D M	M	D M	M	D M	D M																						CP D
X			M D										M		CP	M															M	M	
X																													CP	CP M			
X												M	M																				
												M		M														CO					
X												M		M								CP	M D										
												M		M			CP	M D	M D	M													
X												M		M							CP M					CP	M D						
												M		M					CP	M	CP M			CP	M D								

Fig. 4.5.1-1 Vehicle Control S/S

Parameters

	Servo Pos. CMD's (δ_c)	Compensation Networks	Gain Schedule CMD's	Gimbal Servo Errors ($\delta \epsilon$)	Pitch Gimbal Pos. Fdbk (δ_{FB})	Yaw Gimbal Pos. Fdbk (δ_{FB})	Gimbal Rate Fdbk (δ_{FB})	Acceleration Error (ϵ_{ax})	Cont. Surface Servo Error (δ_E)	Cont. Surface Pos. (δ_{FB})	Pri Eng Throttle Servo Error ($\delta \epsilon$)	Pri Eng Throttle Pos (δ_{FB})	Turbojet Throttle Servo Error ($\delta \epsilon$)	Turbojet Throttle Pos (δ_{FB})	Reaction Control Valve CMD's ($\delta \epsilon$)	Valve Modulator	Compensated Error Signals	Rate Error	Angular Error	Body Temp	Remarks
																				CP M D	
I		CP	M															M	M		
																CP	CP M				
I																					
	M														CO						
	M								CP	M D											
	M			CP	M D	M D	M														
	M							CP	M				CP	M D							
	M					CP	M	CP	M		CP	M D									

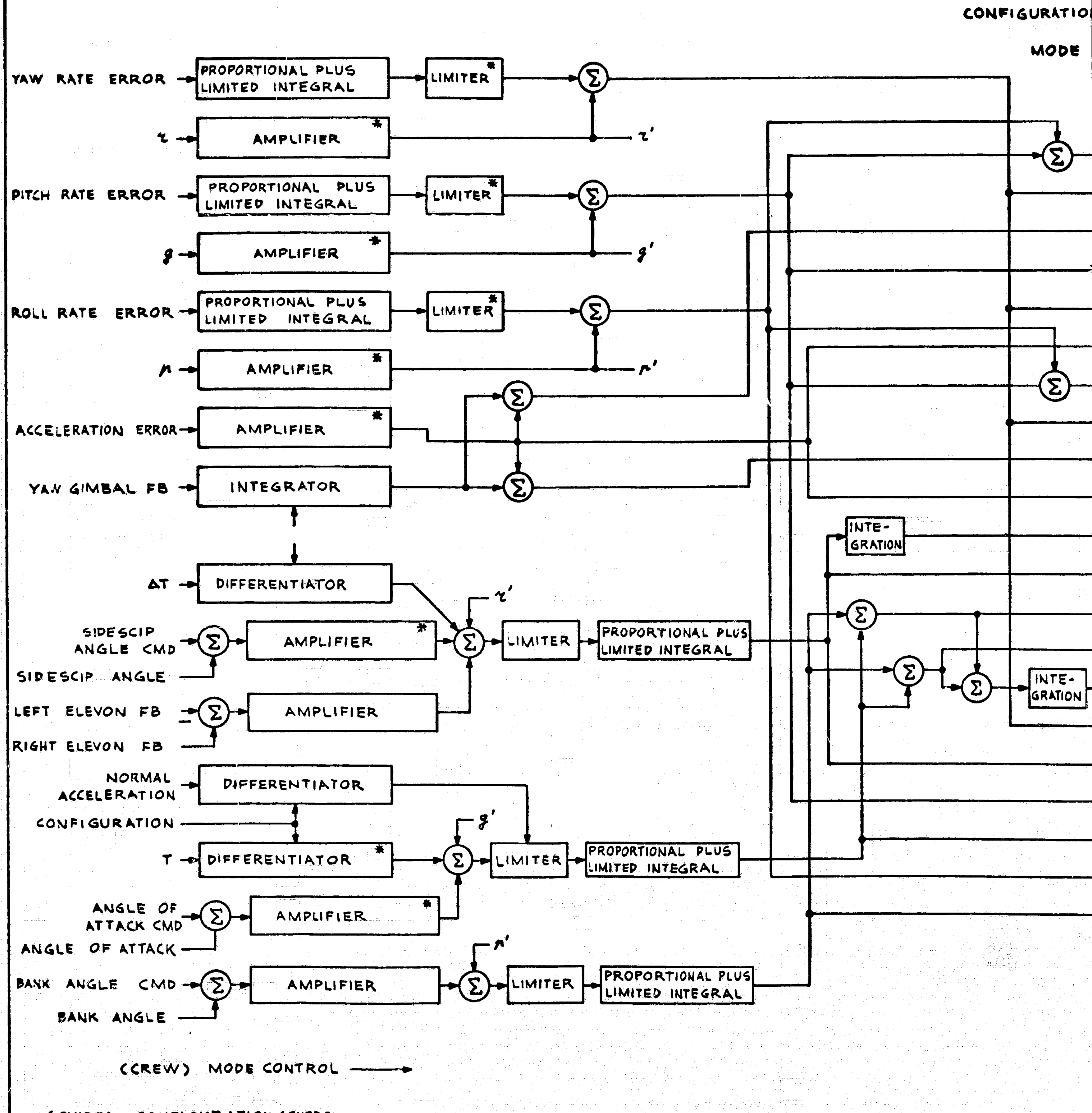
Fig. 4.5.1-1 Vehicle Control S/S Computer

FOLDOUT FRAME 2

FOLDOUT FRAME 3

PRECEDING PAGE BLANK NOT FILMED.

6.1 COMPUTER



✱ DENOTES SCHEDULED GAINS

FOLDOUT FRAME 1

ANK NOT FILMED.

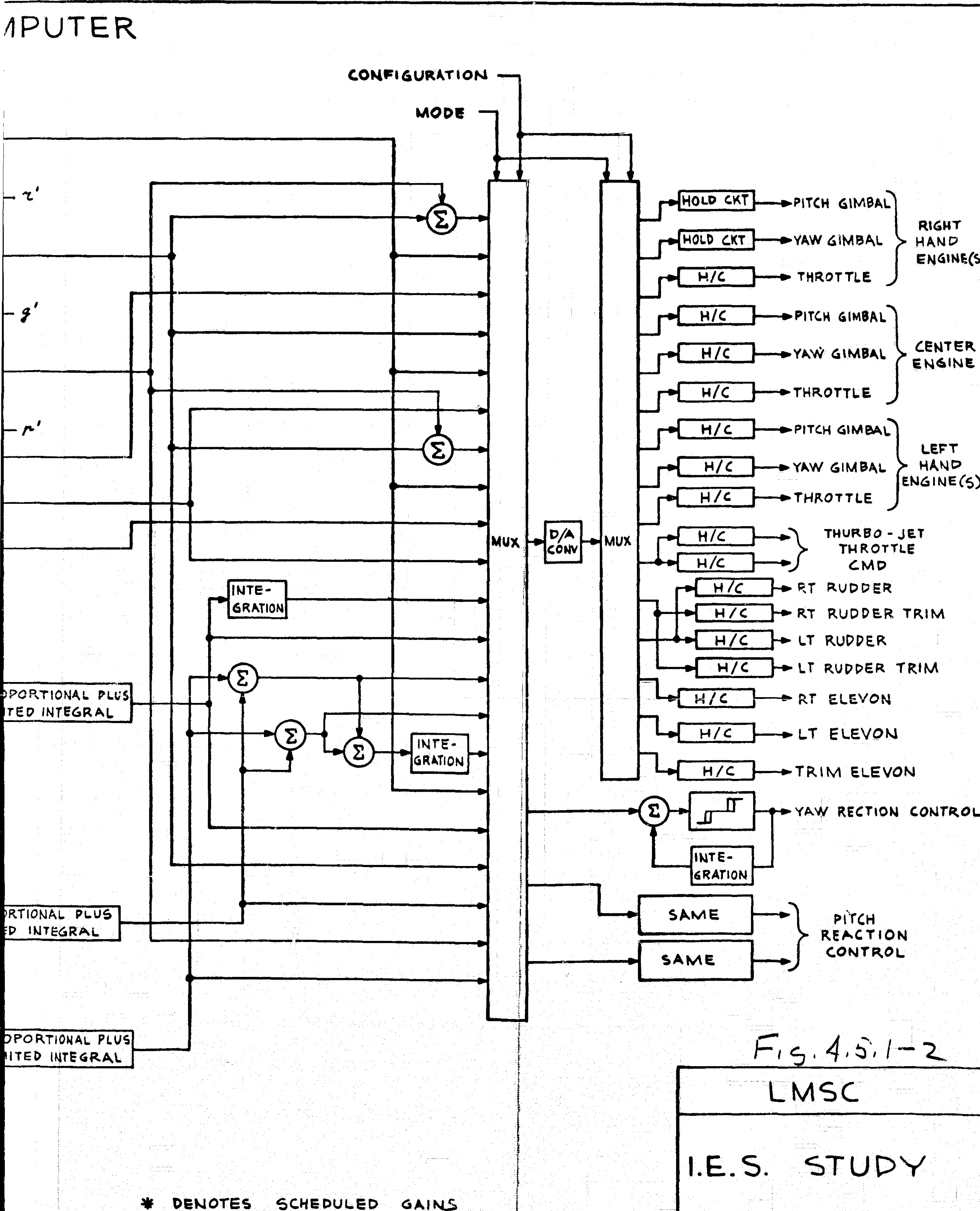


Fig. 4.5.1-2

LMSC

I.E.S. STUDY

Each subsystem was functionally blocked at least to the second level to provide a basis for identifying signal interfaces between major functional blocks within subsystems as well as between subsystems. Test-point access requirements for onboard checkout and fault isolation and for abort warning were identified for each subsystem and for each major functional block. Test points and their signal characteristics were tabulated for each subsystem, as shown in Table 4.5.1-1. (This table closely resembles a similar tabulation prepared for interfaces of each subsystem.)

The abort warning data requirements were considered to be included in the tabulation of test points. The rationale for estimating the number of data points for abort warning is described in the following paragraphs.

Abort warning is a safety-of-flight requirement to alert the crew to a single or probable combination of occurrences that compromise the ability of the system to perform the remaining mission segments within an acceptable factor of risk. The Space Shuttle requirements can be estimated from airplane safety-of-flight data. In advance of a detailed study of the Space Shuttle requirements, the parameter listings employed for airplane crash investigations are selected as being the best source of required data.

Parameters monitored, with the user-vehicle indicated, are summarized below.

SAFETY OF FLIGHT RECORDED DATA

Vehicle	Number of Parameters
C-133 airplane	185
747 airplane	98
F-104G	153
United Kingdom Air Force	48
C-5A crash recorder	67
Average	110

IES STUDY
SUBSYSTEM: Vehicle Control (6.0)

TEST POINT	SUBSYSTEM	LEVEL 1	LEVEL 2	CHARACTERISTICS: TYPE	FREQ (TP)	FREQ (FILTERED)	RANGE	RESOLUTION		
Cmptr Pwr	X			Anal	DC	2HZ	28V	1 bit		
Cmptr Pwr			X	Anal	DC	2HZ	5V	5 bit		
Cmptr Clock			X	Dig.	2 Mhz	1 HZ		10 bit		
Cmptr Self-Test	X			Disc.	1HZ	1 HZ		1 bit		
Cmptr Temp.		X		Anal	DC	0.1HZ	100°C	5 bit		
Parity Prog. Mem.			X	Disc.	2 MHz	R/T	-	1 bit		
Parity X Data			X	"	"	"	-	"		
Parity Y Data			X	"	"	"	-	"		
Parity Oper. Code			X	"	"	"	-	"		
Cmptr Config.	X			Dig.	50 HZ	0.1HZ		30 bits		
Pitch Rate Gain			X	"	"	"		"		
Yaw Rate Gain			X	"	"	"		"		
Roll Rate Gain			X	"	"	"		"		
Pitch Att. Gain			X	"	"	"		"		
Yaw " "			X	"	"	"		"		
Roll " "			X	"	"	"		"		
Pitch Rate Limit			X	"	"	"		"		
Yaw " "			X	"	"	"		"		
Roll " "			X	"	"	"		"		
Pitch Att Limit			X	"	"	"		"		
Yaw " "			X	"	"	"		"		
Roll " "			X	"	"	"		"		
Pitch Mod. Deadband			X	"	"	"		"		RCS Mode
Yaw " "			X	"	"	"		"		" "
Roll " "			X	"	"	"		"		" "
Pitch Mod. Hysteresis			X	"	"	"		"		" "
Yaw " "			X	"	"	"		"		" "
Roll " "			X	"	"	"		"		" "
Prec. Ref. Voltage			X	Anal	DC	0.1Hz	8V	12 bits		
TVC CMD (S/P)		X		"	50 HZ	2 HZ	±10V	10 bits		
" " ([illegible])		X		"	"	"	"	"		

R/T - REAL TIME DISC - DISCRETE

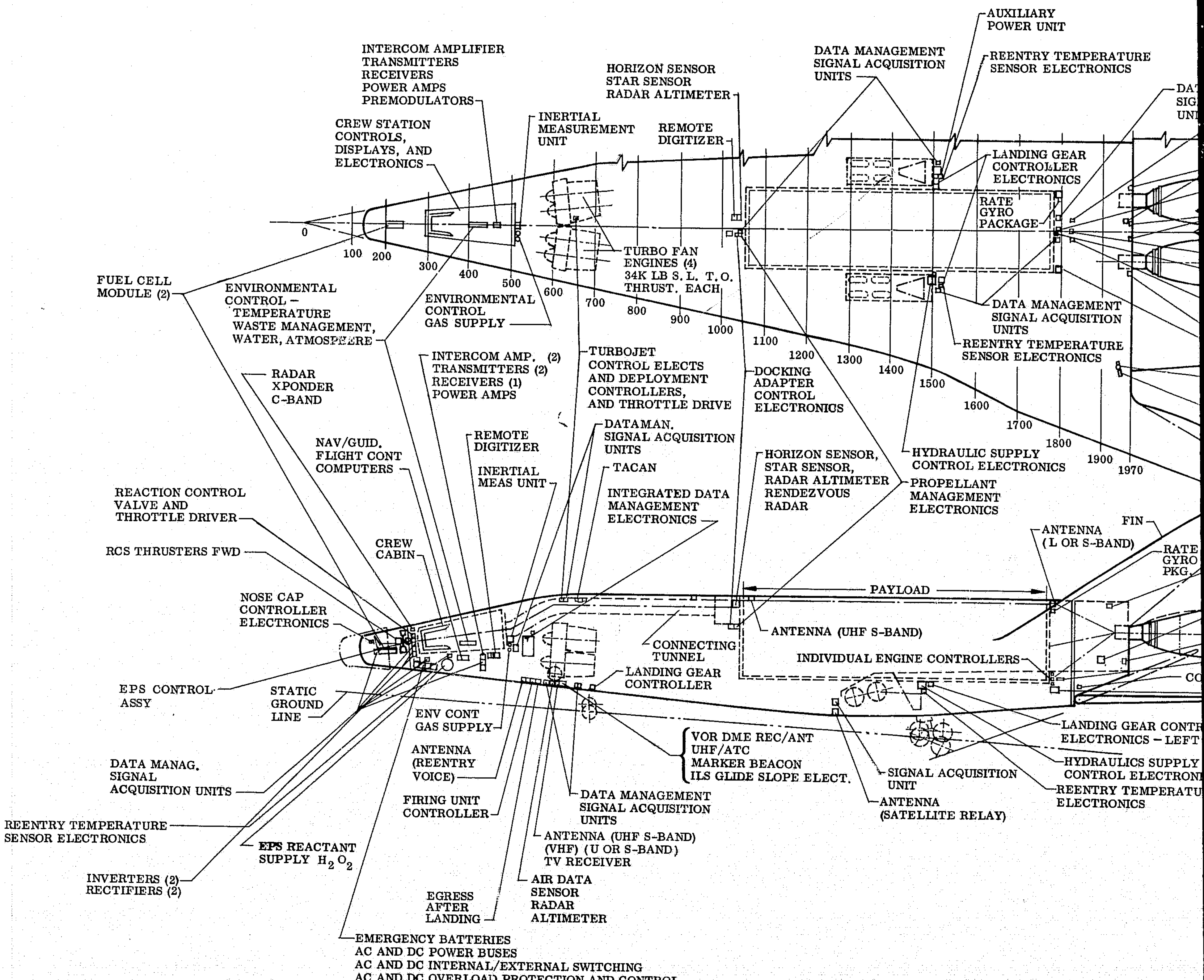
On the basis of this preliminary examination, it is estimated that 150 parameters will suffice for support of the primary decision process referred to as abort warning.

Application of the crash data position indicator recording (CDPIR) subsystem to the C-5A airplane was enhanced by a MADAR data interface. This feature permits, by software routine, any data present in the malfunction detection subsystem to be recorded in the flight recorder of the CDPIR.

In the Space Shuttle data management subsystems considered, the entire accessed data base is available to support the abort warning decision process as required. A compilation of functions vs mission phases and parameters, functional block diagrams, interface tabulations, and test point tabulations for all subsystems constitutes LMSC document A959907 . This compilation of working papers was used as the basis for estimating the requirements that are summarized in the remainder of Section 4.5.

Fig. 4.5.1-3 depicts the location of avionics equipment in a Two-Stage orbiter vehicle. This figure was used to determine cable lengths and weights and also to analyze the geographical distribution of data points and data rates throughout the vehicle. The orbiter avionics signal interconnect diagram (Fig. 4.5.1-4) illustrates the major functional blocks and the number of signal line interfaces between blocks.

PRECEDING PAGE BLANK NOT FILMED.



FOLDOUT FRAME 1

FOLDOUT FRAME 2

G PAGE BLANK NOT FILMED.

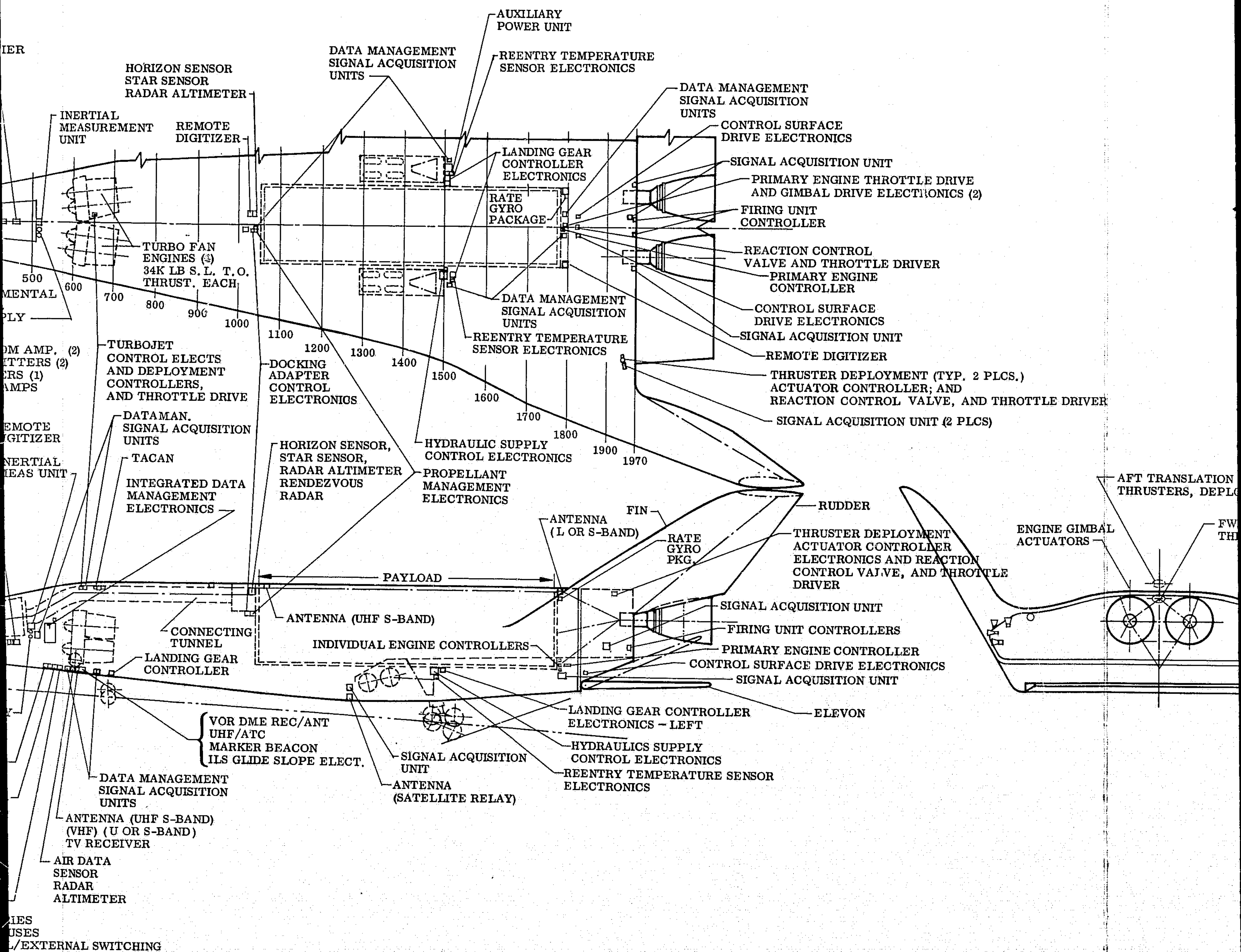


Fig. 4.5.1-3 Av

FOLDOUT FRAME 2

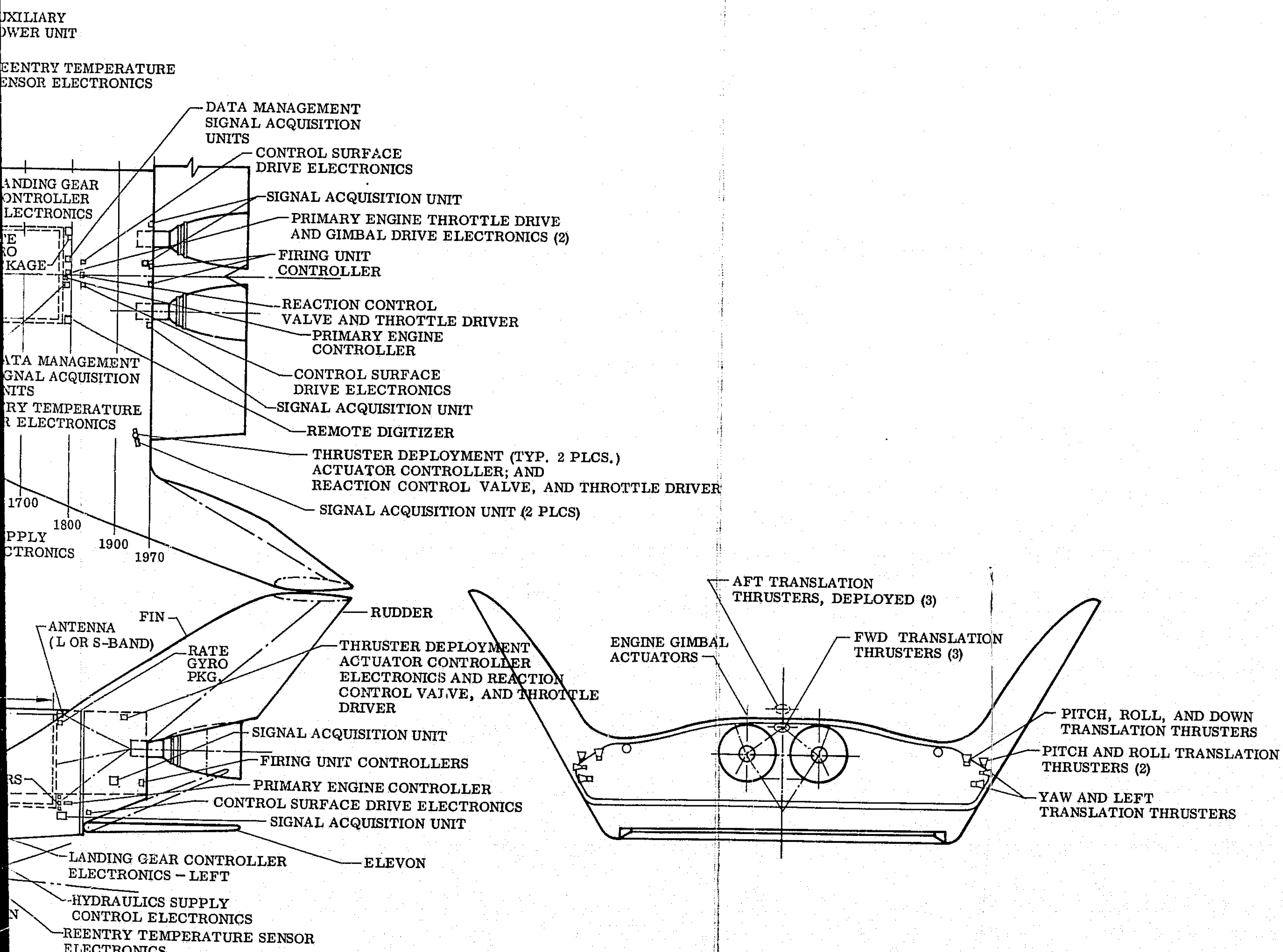
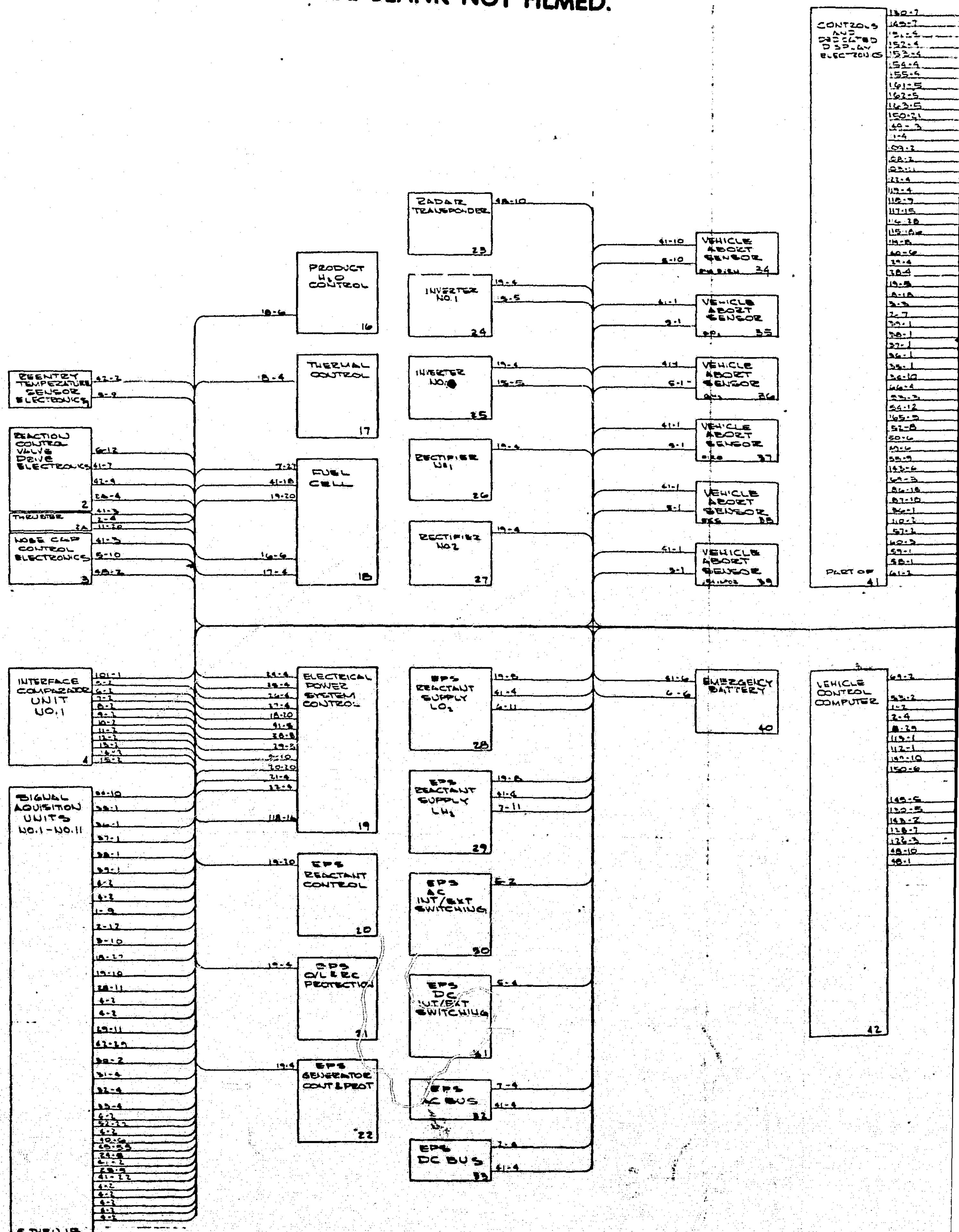


Fig. 4.5.1-3 Avionics Equipment Locations

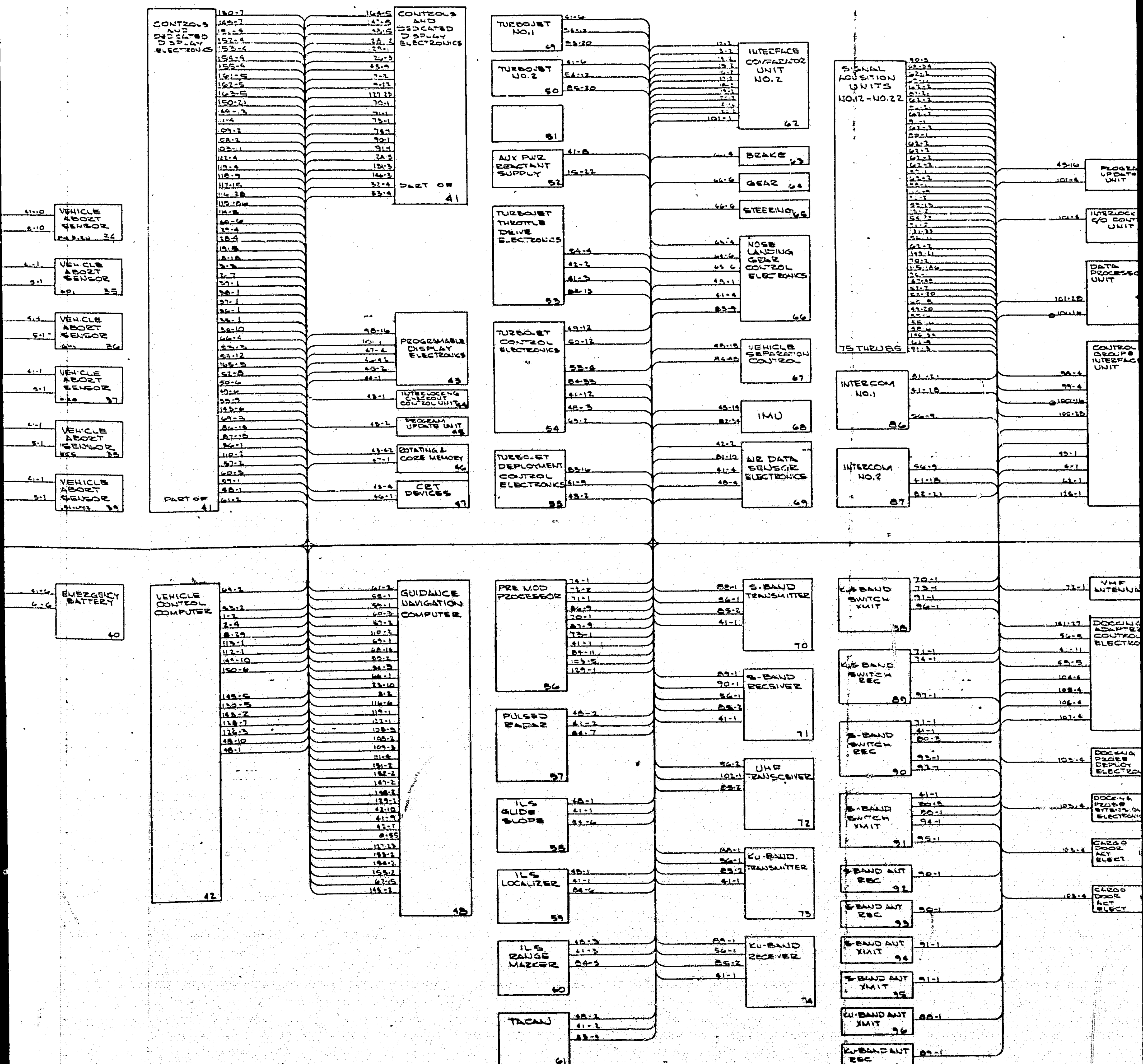
FOLDOUT FRAME 2

FOLDOUT FRAME 3

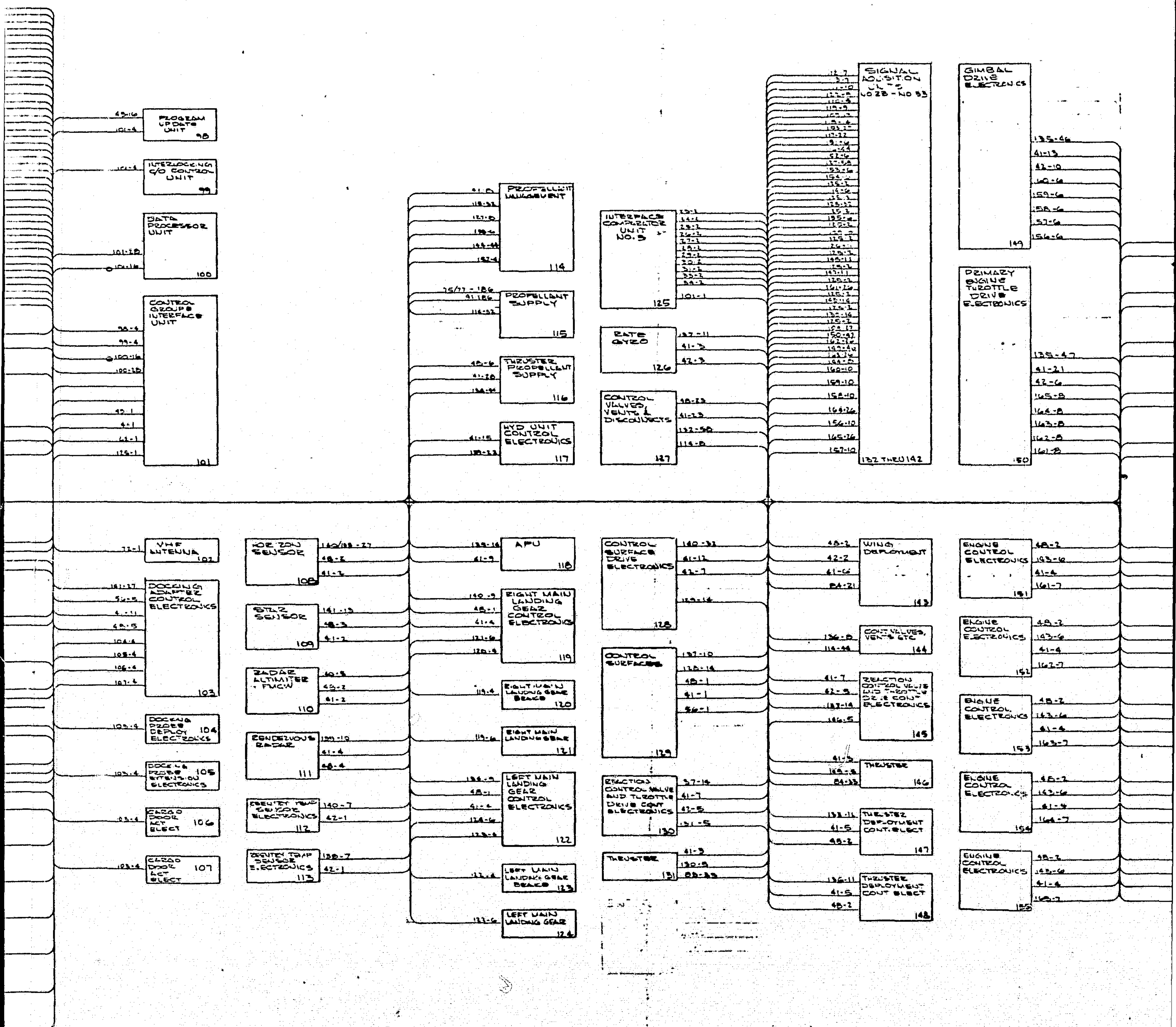
PRECEDING PAGE BLANK NOT FILMED.



FOLDOUT FRAME 1



FOLDOUT FRAME 2


PROGRAM UPDATE UNIT 98
INTERLOCKING C/O CONTROL UNIT 99
DATA PROCESSOR UNIT 100
CONTROL GROUPS INTERFACE UNIT 101
VHF ANTENNA 102
DOCKING ADAPTER CONTROL ELECTRONICS 103
DOCKING PROBE DEPLOY ELECTRONICS 104
DOCKING PROBE EXTENSION ELECTRONICS 105
CARGO DOOR ACT ELECT 106
CARGO DOOR ACT ELECT 107
HORIZON SENSOR 108
STAR SENSOR 109
RADAR ALTIMETER + FMCW 110
RENDEZVOUS RADAR 111
REENTRY TEMP SENSOR ELECTRONICS 112
REENTRY TEMP SENSOR ELECTRONICS 113
PROPELLANT MANAGEMENT 114
PROPELLANT SUPPLY 115
THRUSTER PROPELLANT SUPPLY 116
HYD UNIT CONTROL ELECTRONICS 117
APU 118
RIGHT MAIN LANDING GEAR CONTROL ELECTRONICS 119
RIGHT MAIN LANDING GEAR BRAKE 120
RIGHT MAIN LANDING GEAR 121
LEFT MAIN LANDING GEAR CONTROL ELECTRONICS 122
LEFT MAIN LANDING GEAR BRAKE 123
LEFT MAIN LANDING GEAR 124
INTERFACE COMPUTER UNIT NO. 5 125
RATE GYRO 126
CONTROL VALVES, VENTS & DISCONNECTS 127
CONTROL SURFACE DRIVE ELECTRONICS 128
CONTROL SURFACES 129
REACTION CONTROL VALVE AND THROTTLE DRIVE CONT ELECTRONICS 130
THRUSTER 131
SIGNAL ACQUISITION UNITS NO 28 - NO 33
132 THRU 142
WING DEPLOYMENT 143
CONT VALVES, VENTS ETC 144
REACTION CONTROL VALVE AND THROTTLE DRIVE CONT ELECTRONICS 145
THRUSTER 146
THRUSTER DEPLOYMENT CONT. ELECT 147
THRUSTER DEPLOYMENT CONT ELECT 148
GIMBAL DRIVE ELECTRONICS 149
PRIMARY ENGINE THROTTLE DRIVE ELECTRONICS 150
ENGINE CONTROL ELECTRONICS 151
ENGINE CONTROL ELECTRONICS 152
ENGINE CONTROL ELECTRONICS 153
ENGINE CONTROL ELECTRONICS 154
ENGINE CONTROL ELECTRONICS 155

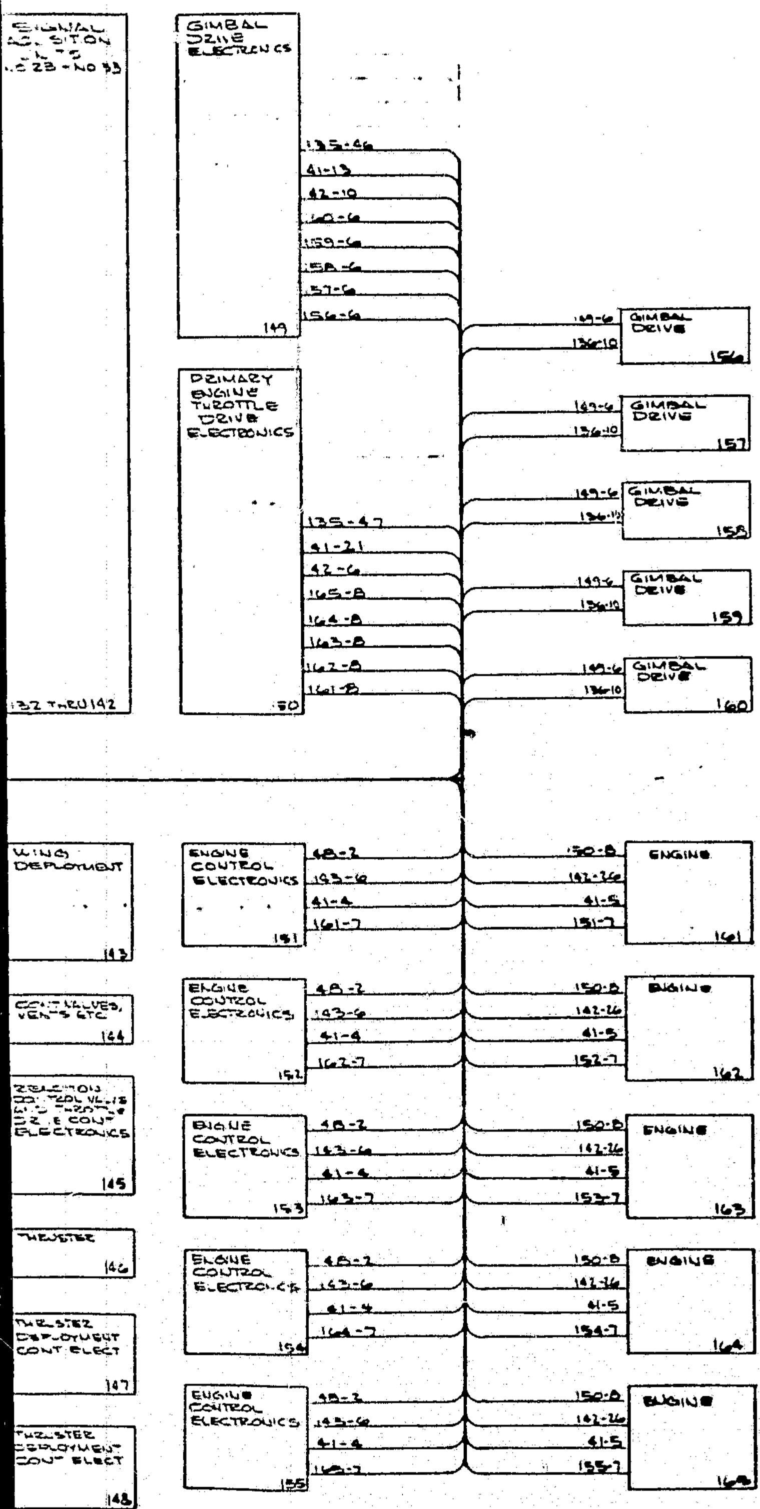


Fig. 4.5.1-4

DATE 8 OCT 1969	LOCKHEED MISSILES & SPACE COMPANY, A GROUP DIVISION OF LOCKHEED AIRCRAFT CORPORATION, SUNNYVALE, CALIFORNIA	
DR		
APPD	SIGNAL INTERCONNECT -ORBITER-	
APPD		
ENGRG		
CHK		
APPD	SIZE J	CODE IDENT	DRAWING NO.	REV O
APPD	SCALE	SHEET

FOLDOUT FRAME 4

PRECEDING PAGE BLANK NOT FILMED.

4.5.2 Summary of Data Requirements

A review of the working level documentation (function lists/test point lists/ interface lists/block diagrams) was completed to define system data requirements. The following guidelines were used to correlate the inputs:

- Power distribution /control function is assumed common to each alternative and is therefore not included.
- Data point lines are assigned to their source.
- Worst case is to be cited.

The last guideline provides a buffer to allow for missed points in subsystem definition and the subsequent compilation. The system and subsystem signals are described with respect to (wrt) quantity and sampling frequency distribution on Fig. 4.5.2-1. System average sample rates are 3.2 samples/sec (sps) for test points and 7.0 sps for interfaces/intrafaces. These values are based on a minimum sample rate of one sps and a maximum of 50 sps. Subsystem sample rates were not computed but are relatively low (most signals are 10 sps or less); exceptions are communications (audio, etc.) and vehicle control parameters, subsystems 5.0 and 6.0 (update rate for these parameters has been established at 50 sps, a conservative, worst-case estimate). Projections of system data rates are included under the discussions of each alternative.

Significant trends in signal classification are, first, that virtually all interfaces and intrafaces are test points. Only 20 points out of more than 800 were not designated as test points; hence, it has been assumed that all would be tested. Second, the parameters to be displayed are, without exception, test points. Third, 30 to 40 percent of the listings are discrete signals. This will appreciably influence the signal bandwidth needed for interface control. As an example, the system test point count is 1189 analog signals and 622 discretes (1811 total). Packing the discrete signals (10 discretes/analog signal) effectively reduces the test point count to 1252 (1189 plus 622/10), which requires only 0.69 times the original bandwidth. Some improvement in error rates and/or power will be realized by this reduction.

Data distribution is presented in Table 4.5.2-1. The total (2820) represents the number of signal conductors that would be required in a conventional system. It is noted that interconnections between subsystems are minimal; each has maintained a high degree of autonomy, with the control/display and test functions providing commonality.

The signal flow is presented pictorially in Fig. 4.5.2-2. This diagram describes the signal class wrt function. The subsystem interfaces are shown only at the point of origin for simplification, except for the control/display and data management subsystems (major common area). The classifications are self-explanatory (as noted earlier, all signal accesses are test points). The "hard-wired control" acknowledges that some level of manual command capability will be required in a final design. This capability will be common to all IES alternatives. Parameters were selected wrt crew safety, e.g., emergency oxygen supply, emergency power, etc. Dedicated displays are parameters that are continuously displayed and are based upon "human engineering" concepts.

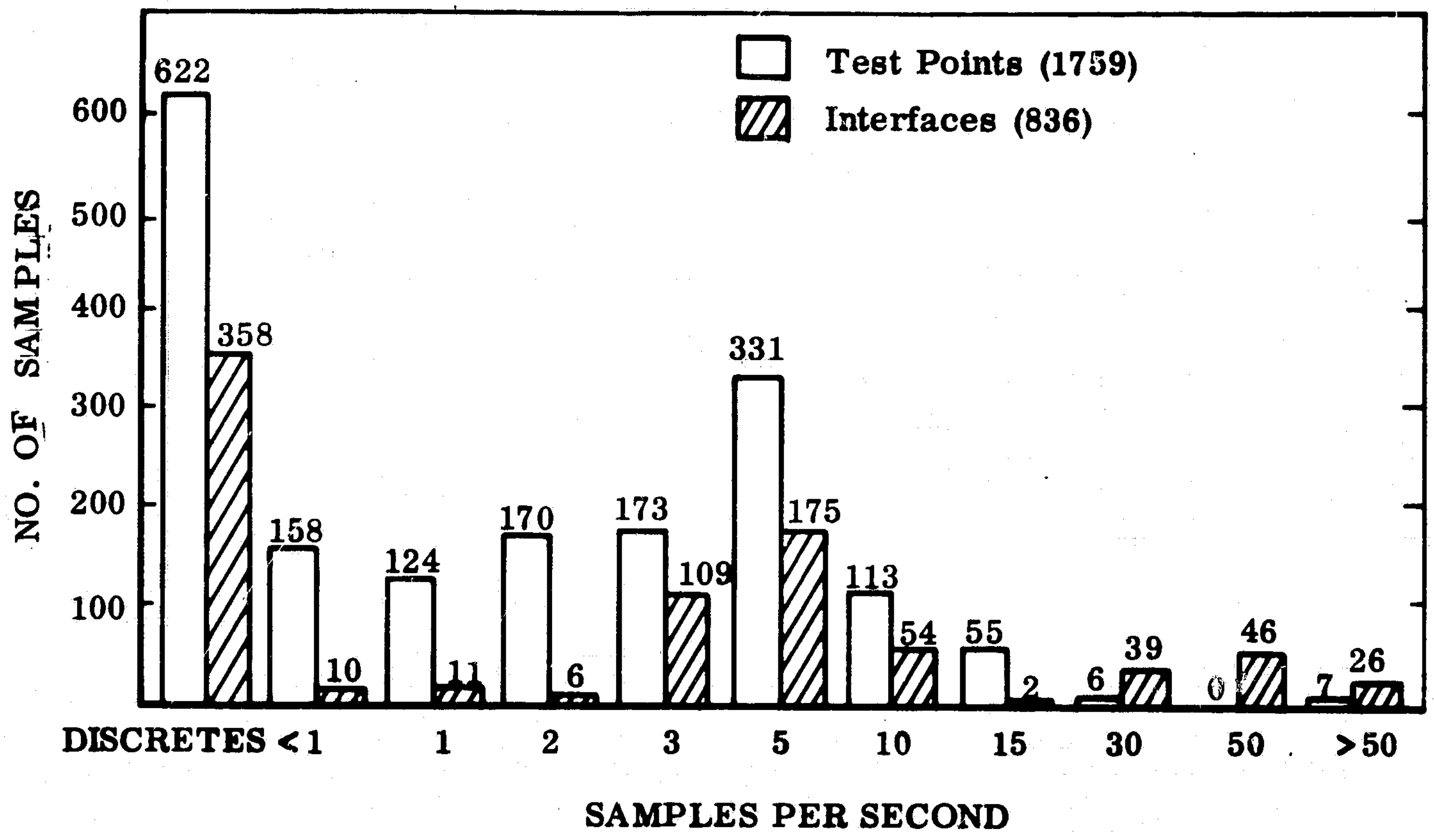
These data requirements have been established as reference values for each of the IES alternatives (levels of integration).

TABLE 4.5.2-1

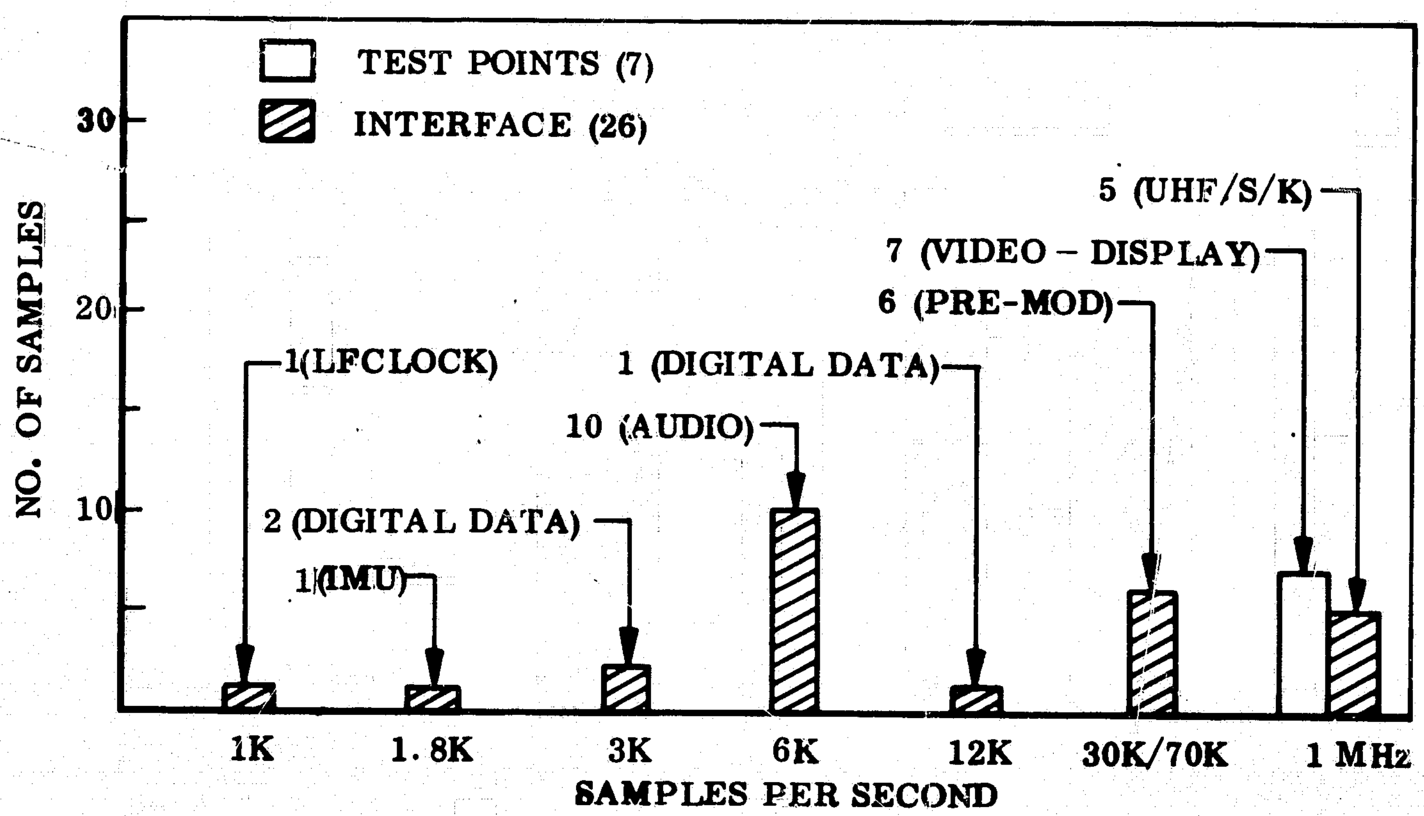
SUBSYSTEM SIGNAL INTERCONNECTION MATRIX

SUB-SYSTEM	1	2	3	4	5	6	7	8	9	Totals
1	48									48
2	0	289								289
3	0	0	86							86
4	0	0	0	0						0
5	47	81	0	0	23					151
6	46	58	0	0	13	51				168
7	0	0	0	0	0	0	31			31
8	19	37	15	29	6	16	43	66		231
9	320 (68)(1)	626 (281)	211 (44)	83 (15)	188 (23)	231 (58)	78 (20)	24	55	1816
Totals	480	1091	312	112	230	298	152	90	55	2820

NOTE: Common parameters for display

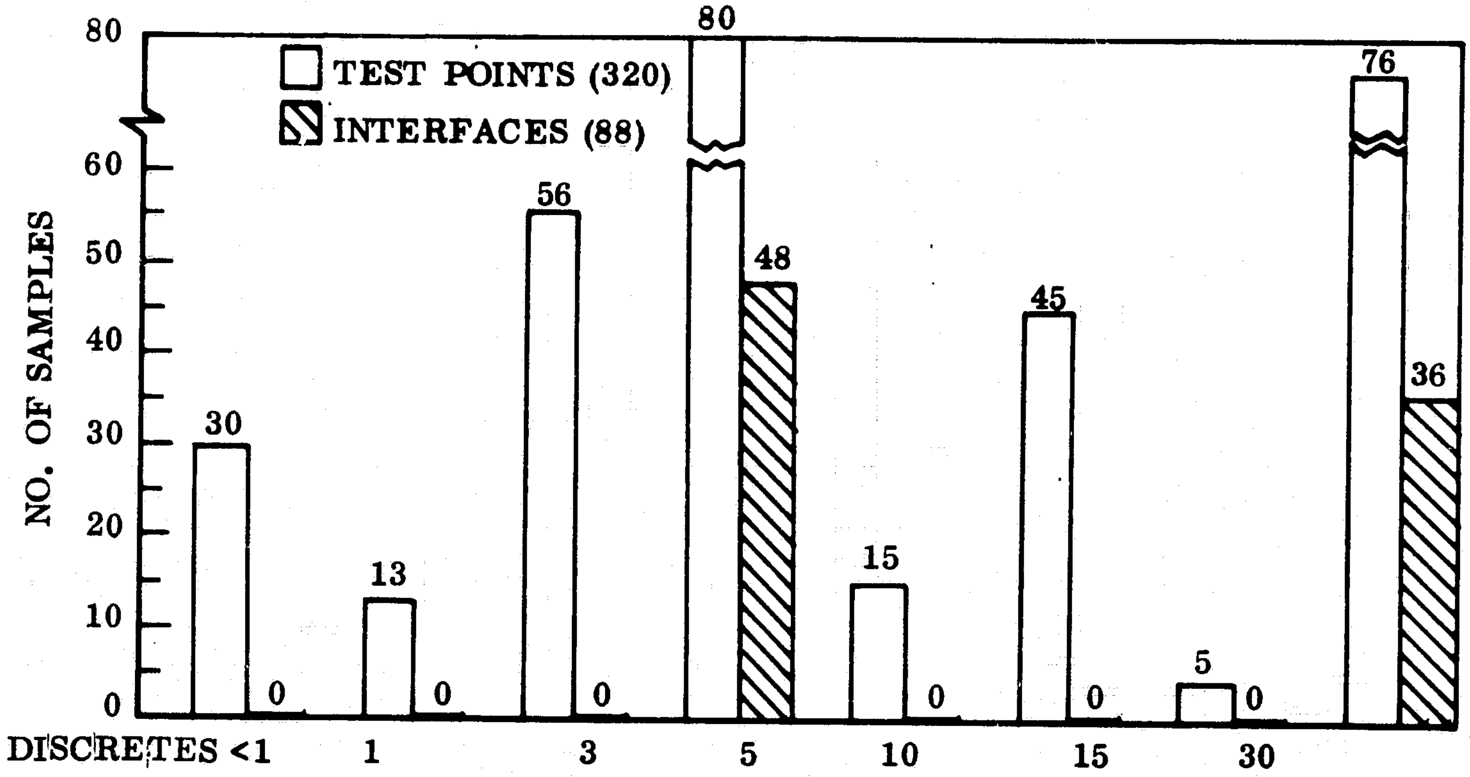


a. System Sampling Rates

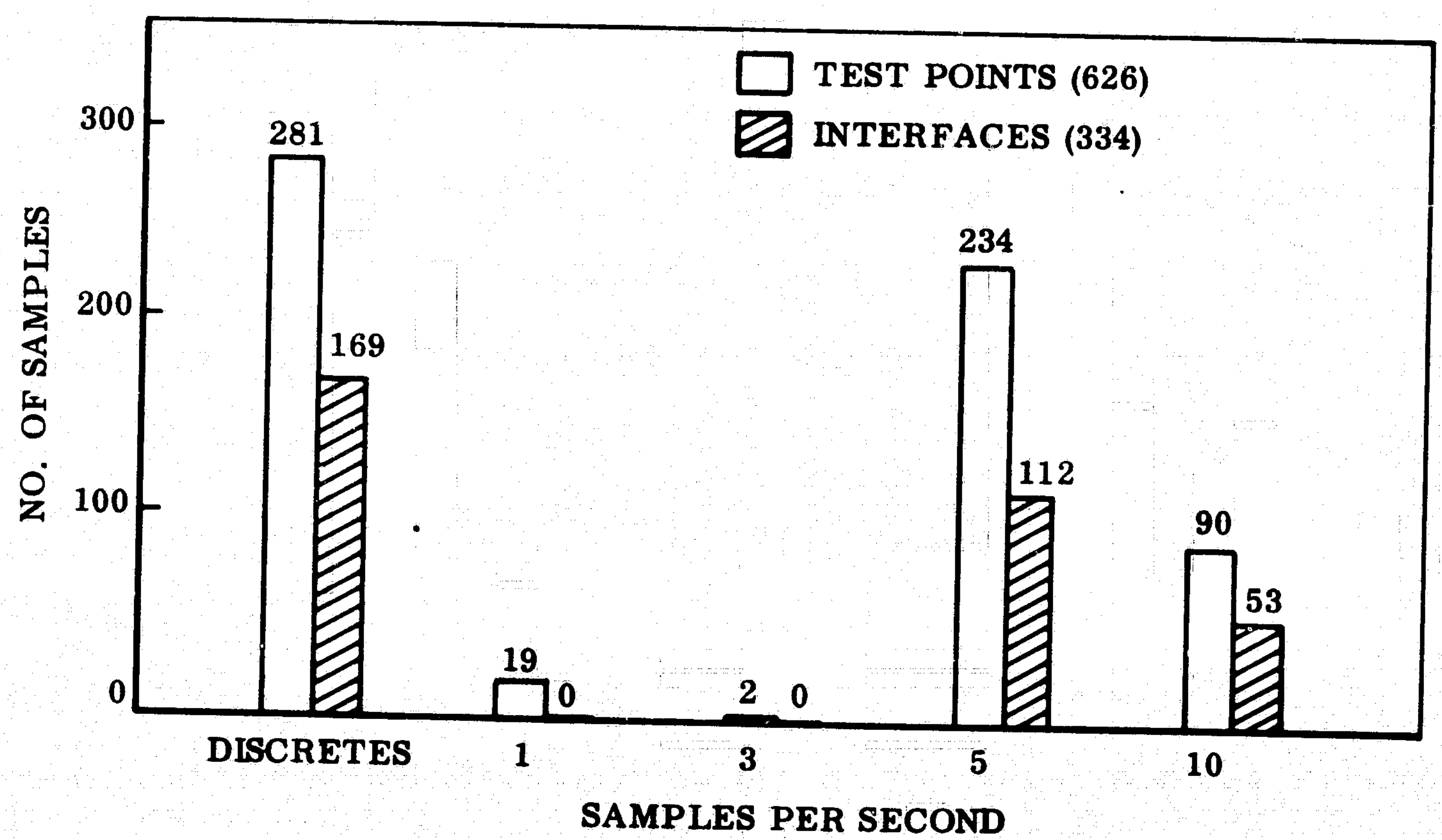


b. System High Frequency Detail

Fig. 4.5.2-1 IES Sampling Spectrum (Sheet 1 of 5)

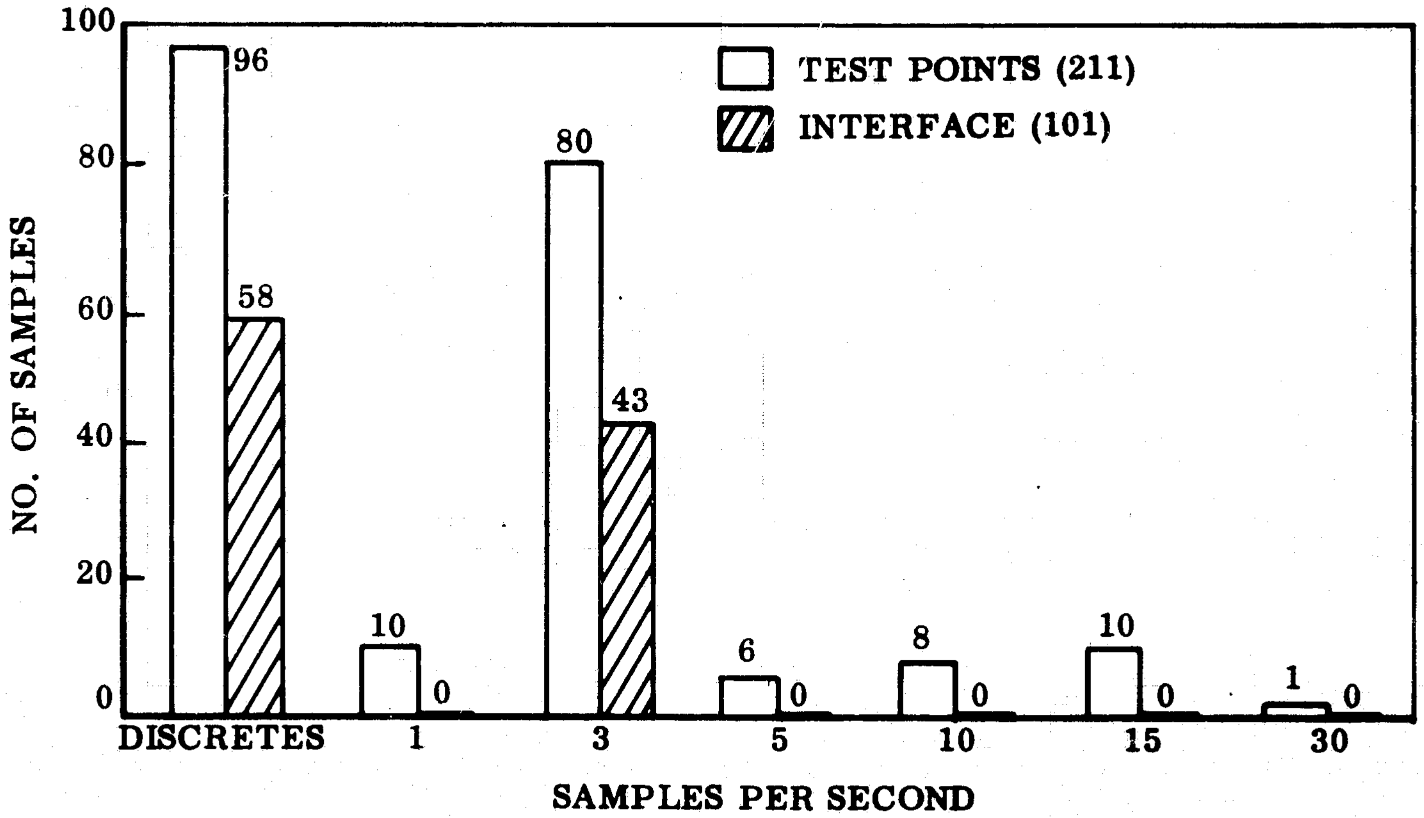


c. Subsystem 1.0 Structure/Mechanical

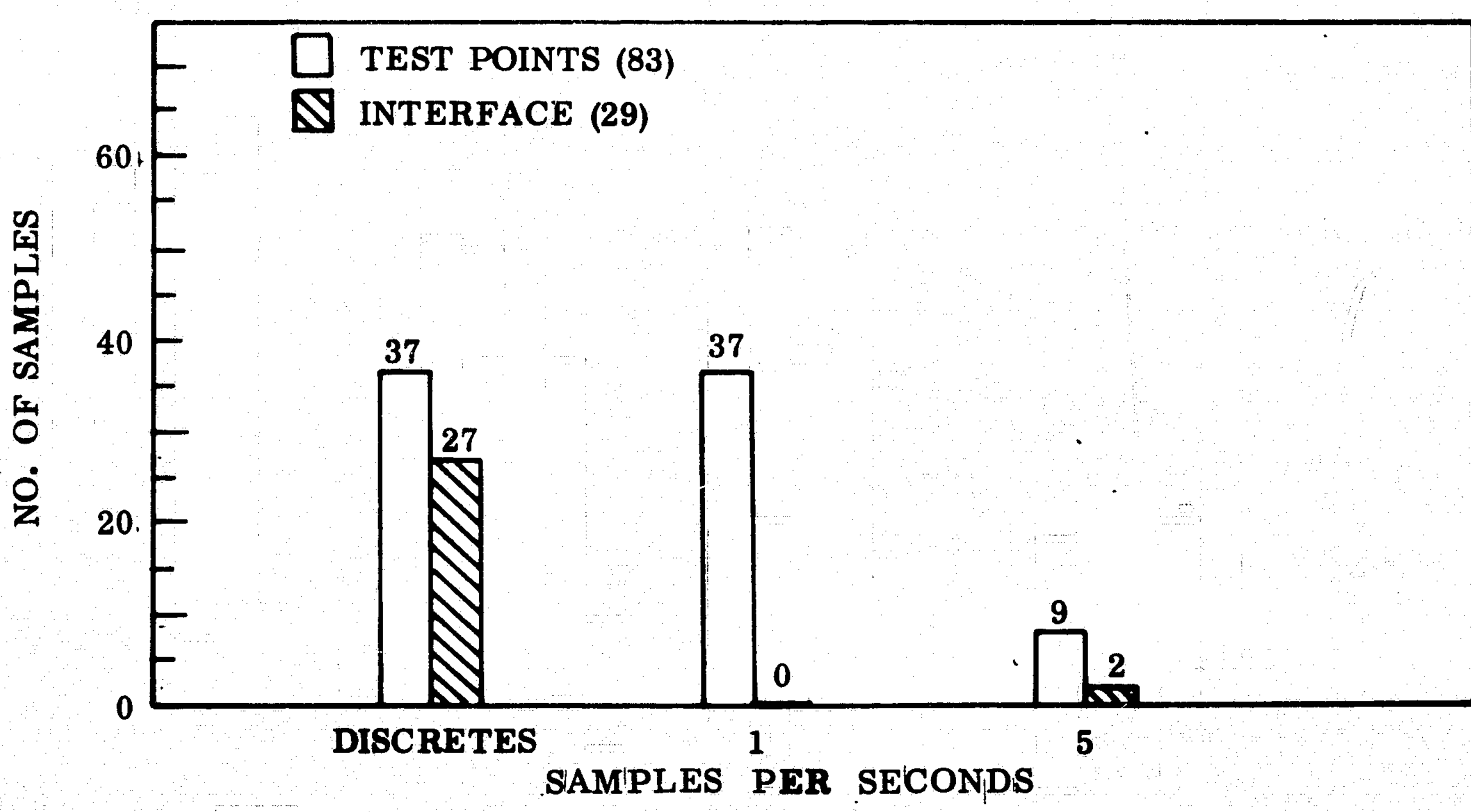


d. Subsystem 2.0 Propulsion

Fig. 4.5.2-1 IES Sampling Spectrum (Sheet 2 of 5)

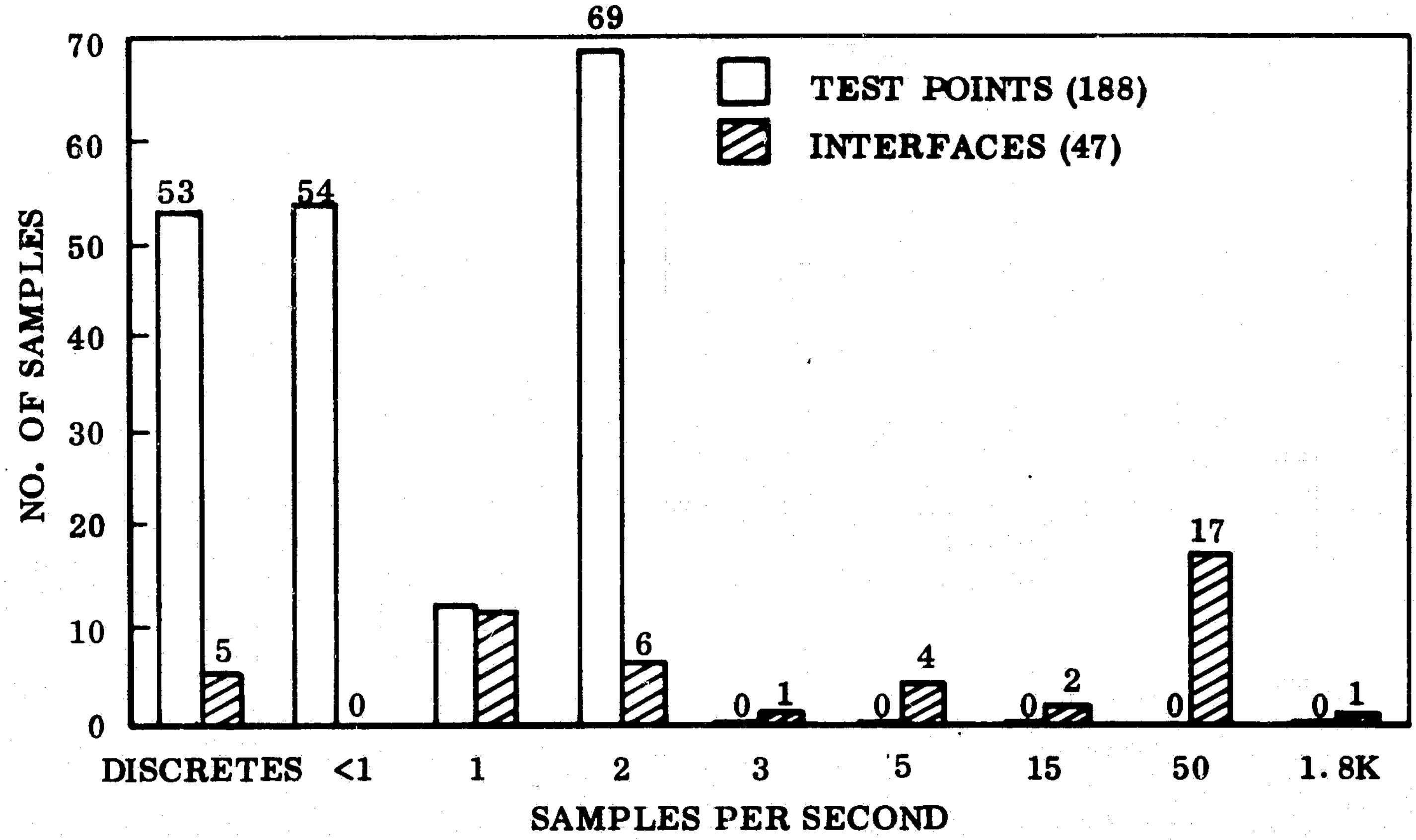


e. Subsystem 3.0 Electrical

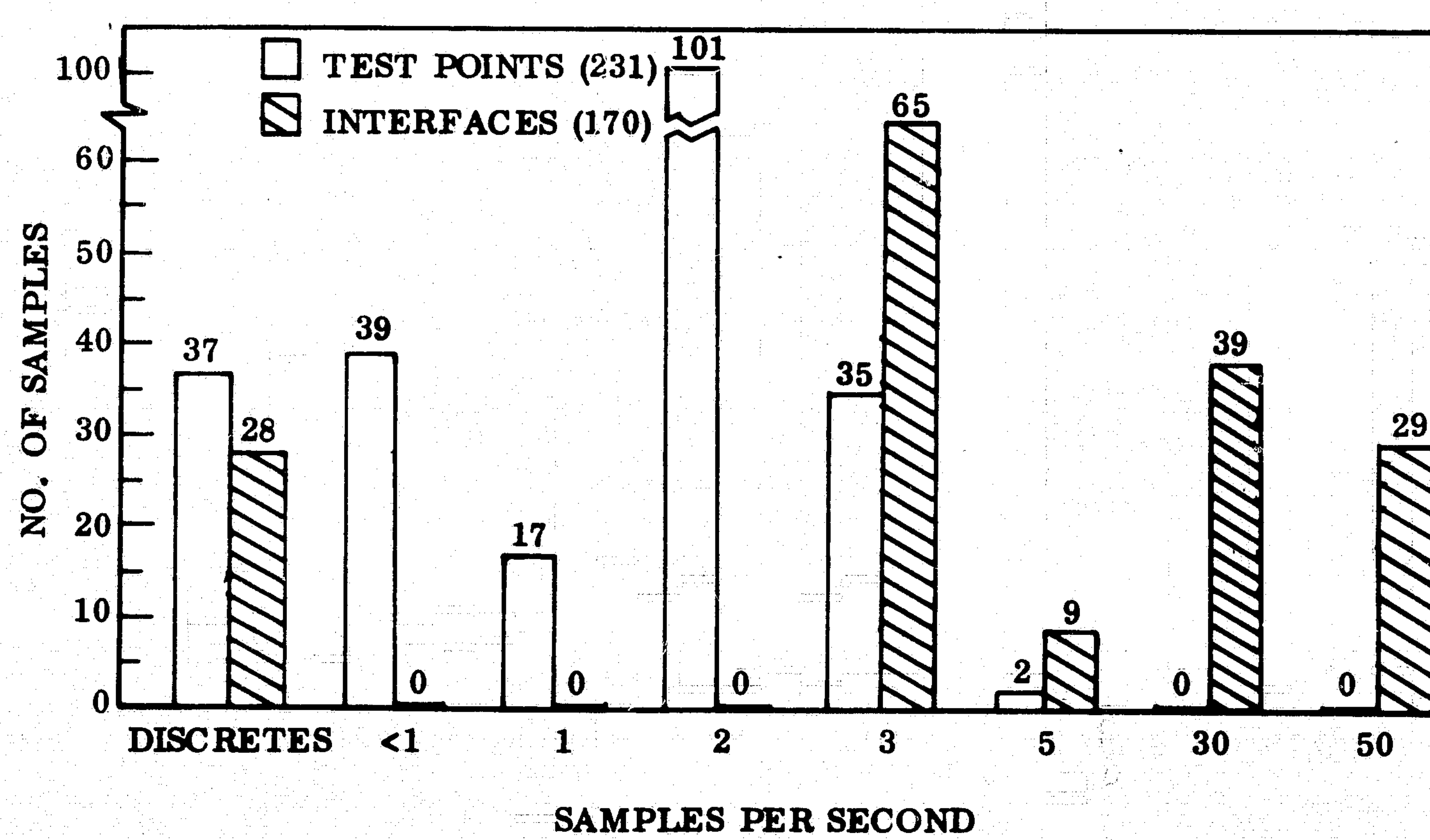


f. Subsystem 4.0 Environmental Control

Fig. 4.5.2-1 IES Sampling Spectrum (Sheet 3 of 5)

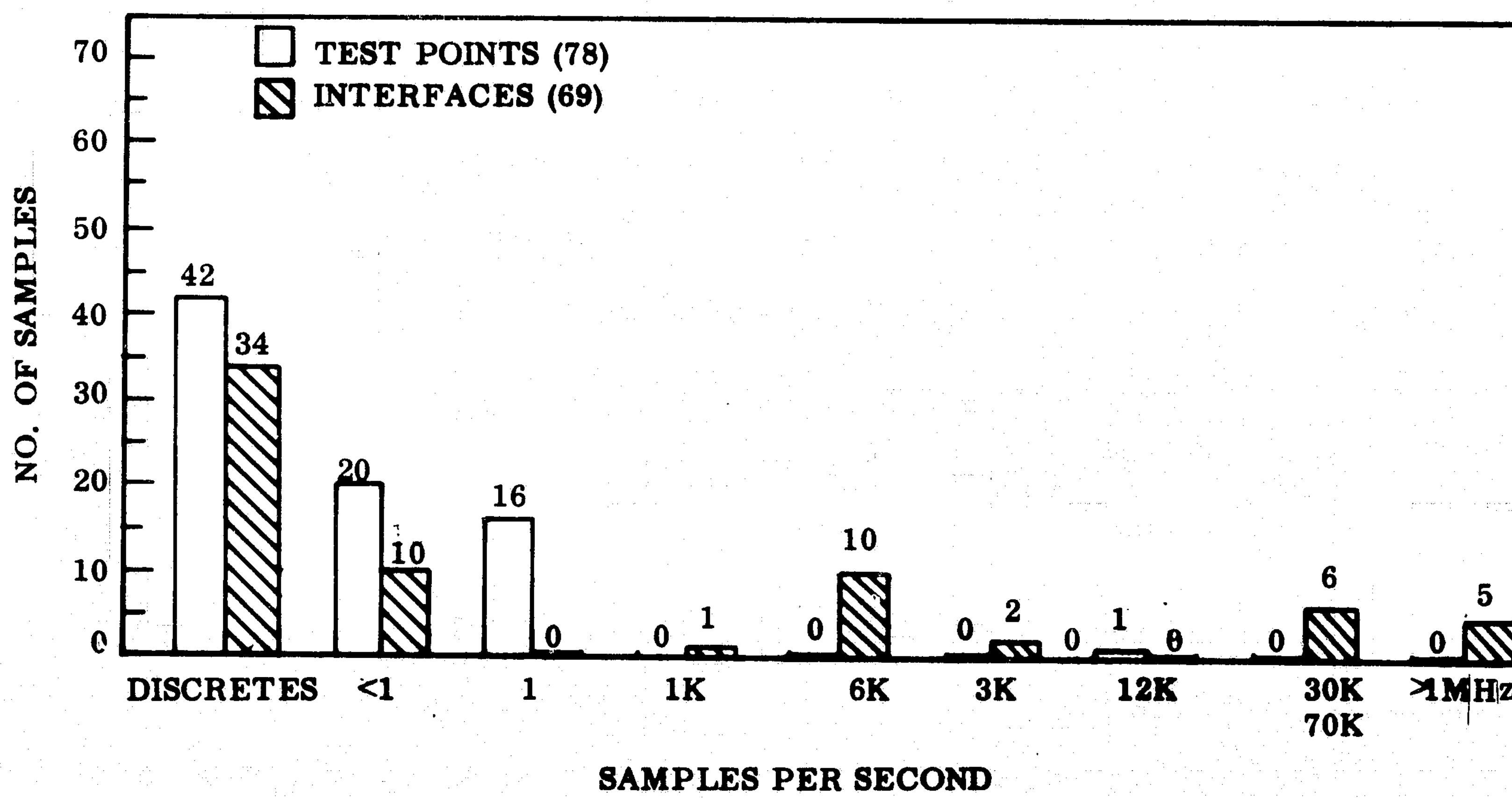


g. Subsystem 5.0 Guidance/Navigation



h. Subsystem 6.0 Vehicle Control

Fig. 4.5.2-1 IES Sampling Spectrum (Sheet 4 of 5)



i. Subsystem 7.0 Communications

Fig. 4.5.2-1 IES Sampling Spectrum (Sheet 5 of 5)

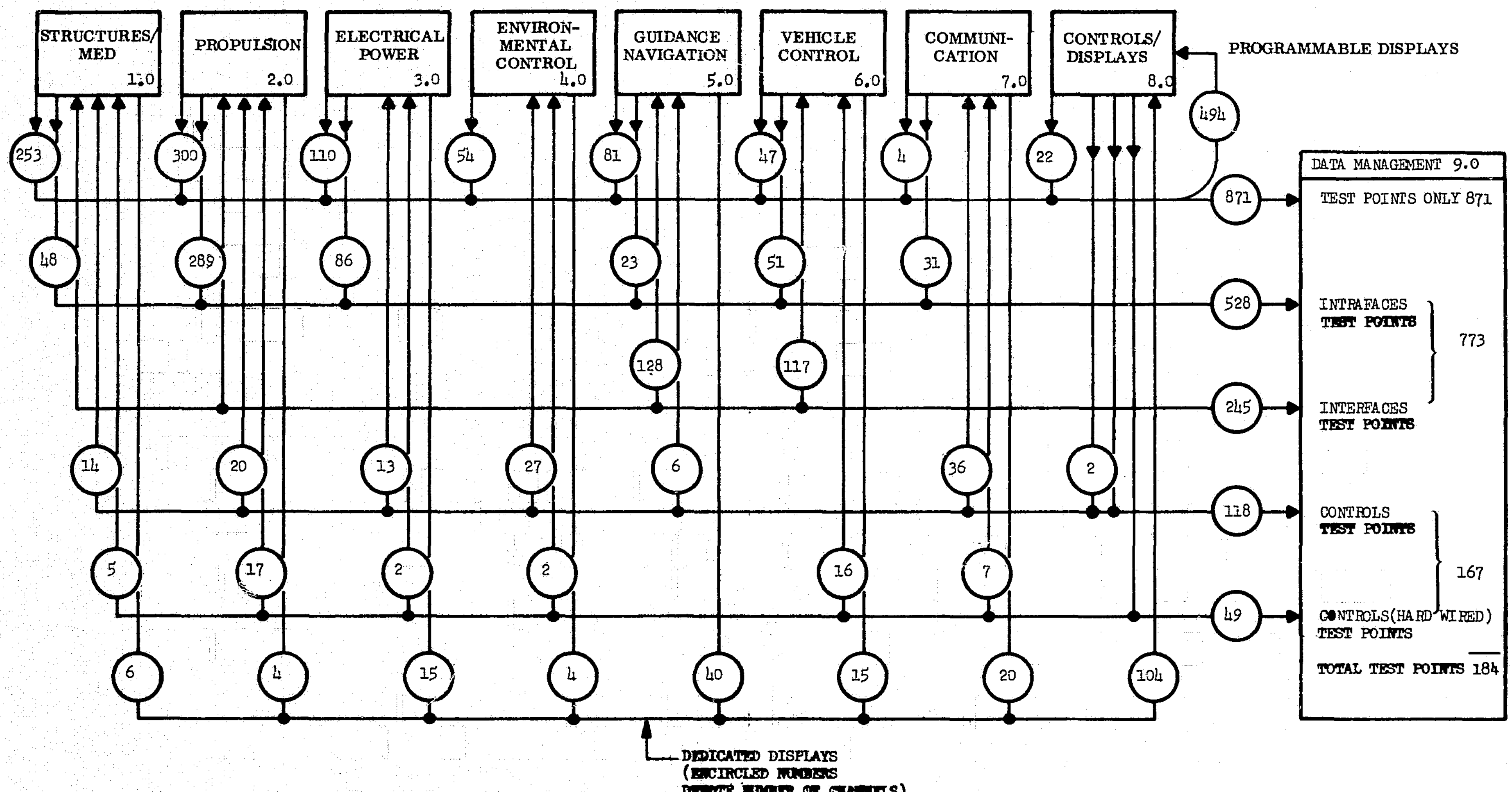


Fig. 4.5.2-2 Data Flow Summary

4.5.3 Operational Support Requirements

Two primary considerations affect the onboard avionics subsystem under the requirement for vehicle autonomy: (1) implementation of the present functions of launch control and (2) satisfying the essential requirements of mission control. The impact of these requirements is strongly dependent upon stored data required, access rates, and methods of implementation. This section discusses the tentative stored data requirements. Access and implementation are addressed in the discussion of alternative configurations.

Operational support data are required in two basic formats:

- Visual data for crew
- Machine data for processor

The content and magnitude of each is discussed below.

4.5.3.1 Visual Data. Graphic data are expected to be required in support of the crew for displaying options available at key decision points. A compilation of anticipated decision points was made from preliminary Space Shuttle time lines and from Apollo/Saturn data. An estimated 1936 points were identified.

A similar investigation was made to estimate the technical order pages required in support of manual troubleshooting. An estimated 3428 pages of data were identified.

On the basis of these studies a 5000-page module was postulated as a standard requirement. Modules are assumed to be available as a library with manual selection. Modules identified are:

- Standard mission
- Contingency fault isolation
- Special mission/payload
- Ground checkout

4.5.3.2 Machine Data. A Space Shuttle processor can perform many of the various tasks of checkout, fault isolation, abort warning, and mission support by using the software involved in a GENERAL LOGIC ROUTINE program. The detection of a malfunction is conceptually identical to detecting an ABORT SITUATION, a NEAR ABORT SITUATION, a PROCEDURE FAULT, or a PROCEDURE REQUIRED. All these different titles for identical operations will hereafter be referred to as LOGIC ROUTINES. The software used in the Space Shuttle can be similar in concept to the proven software used on the C-5A for malfunction detection and mission support logic decisions, supported by Built-In Test Equipment (BITE) at the black-box level. The basic philosophy that makes possible the combination of unrelated tasks in one software package is:

- Create a computer program that will execute a "flow chart" of tests and output the results of these tests.
- Implement these tests in "flow chart" engineering language, and store the flow charts in memory.
- Allow only a small number of different types of "flow chart symbols."

On the basis of this concept, the software need only perform a limited number of types of tests.

The tests and the logic connecting the tests are stored in the computation center as data. Since the tests are created by engineers, there is no need for a translation into computer language by a programmer. This has allowed tests to be changed on the C-5A routinely in 24 hours.

4.5.3.3 Data Assignment. The following normalized increments, which are based upon C-5A software experience, are used for sizing the requirements for identified events (LRs). A 2000-point base was examined to justify the normalized conclusion.

Symbols	5000	2.5/LR
LR starts	1000	.5/LR
Requests	1750	.875/test point
Misc	200	CONSTANT
Total	7950	

On the basis of this evaluation, a general sizing equation is

Data = 200 + 3 (number of LRs) + .875 (number of test points) words

This relation is suitable for words of 24 bits or longer where half-word capability exists in the computer.

4.5.3.4 Computer Speed. This program will execute tests on 2000 LRs in .05 second when programmed on a computer with a 1 μs add time. The execution time is directly proportional to the number of LRs at 25 μs per logic routine.

The above rationale is applied to identified logic routines to develop the total requirement.

In addition to the requirement above, which is dependent on the numbers of discrete logic routine applications identified, the following basic budget is established for the general-purpose computer.

BASIC ROUTINE PROGRAM REQUIREMENTS
(32-Bit Words, One-Half Word Capability)

Basic Routines	No. 32-Bit Words
Frequency conversion	50
Synchro conversion	150
Self-check	150
Executive program	200
Time	30
Input select	30
Initialization	20
Absolute value	250

Basic Routines	No. 32-Bit Words
Square root	20
Polynomial	20
Binary-to-BCD conversion	15
BCD-to-binary conversion	15
Output message	800
Trend routines	600*
Subsystem logic routines	2700
	5050 words basic

*Use requires four memory words per parameter

Automatic checkout/fault isolation requires logic routines for the 2000 signals processed, calculated at 200 + 3 (2000) + 0.875 (2000) = 7950 words.

Prelaunch Operation Support.

The following example is included for information; other segments are summarized.

In the absence of verified countdown and operational sequence data for the Space Shuttle, the Apollo/Saturn is used as a model from which the Shuttle requirement is extrapolated. In general, the development nature of the Apollo imposes more stringent requirements than those expected for the Shuttle.

Review of Apollo/Saturn V S/V Countdown Document KV-OJ13-2, V-40300-501 gives the following:

Check Points	Decision Points	Action Points
527	34	356

Where a check point is a request for observation/correlation, a decision point requires a human evaluation, and an action point requires a physical response from a human.

The period represented is two days and three hours of Saturn/Apollo time and is considered representative of the level of performance expected from the crew. These data, translated into a 3.5 hour (210 minute) launch interval, represent a potential crew action on the average of

$$\frac{34 + 356}{210} = 1.9 \text{ actions/minute}$$

Under a two-hour launch requirement, 3.25 actions/minute are potentially required. Actual response may or may not be required. In most cases it is assumed that check points and action points are preprogrammed man/machine interfaces.

Check point is presentation of information where the crew may or may not elect to take action. Action point is presentation of information to the crew where an action is expected of the crew before further automated action. Decision point is presentation of information to the crew where alternatives exist, such as preplanned hold periods. Failure of the crew to exercise any option results in selection of a preplanned action where total automation is implemented.

The total man/machine interface points are 917. It is assumed that 10 completely automated logic routines are performed for each manual interface. The total computational burden is

$$917 \text{ semiautomatic} + 9170 \text{ automatic} = 10{,}087 \text{ routines}$$

Memory requirements are:

$$200 + 3\,(10{,}087) + .875\,(2000) = 32{,}011 \text{ words}$$

4.5.3.5 Mission Control. In keeping with present terminology, mission control applies to that portion of the mission after liftoff.

Ascent Operations Support. Memory requirements are 2956 words for an estimated 1000 test points.

Orbit Operations Support. The variety and duration of activities expected during orbit vary widely. For the normal logistic mission, it is difficult to imagine a work load requiring work rates more severe than 26 points per hour. For a 7-day mission, memory requirements are 79,262 words.

Entry Operations Support. Crew operations for entry are estimated to require 6157 words.

Rendezvous Operations Support. This requirement is assumed to be equivalent to that for entry operations and is budgeted at 6157 words.

Atmospheric/Landing Operations Support. The entry-of-atmosphere to landing completed time varies between 0.7 hour and 1.4 hours; the memory required is 7909 words.

Abort Warning Mission Support. Approximately 150 test signals are required as a data base for safety of flight/abort condition warning.

The number of logical reasons for abort is expected to be relatively limited, with the mission phase in which the undesired event occurs having a major influence; for example, loss of one level of redundancy during ascent could be tolerated where that same loss on the pad could spell abort.

An estimated level of abort conditions at any one time in the mission is assumed to be 100, and an arbitrary six mission segments (configurations) is postulated for 600 logic routines.

The memory requirement is 2132 words.

Budget Summary

Item - Computer Support	Words
Basic routines	5050*
Checkout/fault isolation	7950*
Prelaunch	32,011
Ascent	2956
Orbit	76,262
Rendezvous	6157
Entry	6157
Approach and landing	7909
Abort warning	2132*
*Required for all operational phases	149,584

Item-Display Support	Visual Frames	Data
Fault isolation	5000	50K half words
Operation support	5000	50K half words*

*Covers the entire mission, prelaunch to rollout

The previous data present a moderately pessimistic assessment of the data requirements for mission support.

4.5.4 Subsystem Computation Requirements

Estimated computation requirements for vehicle subsystems other than data management and control/display are presented in this section.

4.5.4.1 Guidance, Navigation, and Vehicle Control. The estimated word storage requirements and percent use by mission phases are listed in Table 4.5.4-1. The average number of instructions per second for each mission phase are listed in Table 4.5.4-2. The estimated computation requirements support the operational modes and subsystem configurations summarized below.

Prelaunch

- Targeting

  Align and calibrate IMU and initialize transformation matrix

  Generate ascent trajectory and guidance constants

  Generate abort decision criteria and alternate flight plans

  Generate manual display parameters

  Load and verify mission target constants

  Load and verify abort constants and wind data

Ascent

- Atmospheric mode

  Load relief guidance/navigation (IMU, rate gyro pkg., flight control system)

- Exoatmospheric mode

  Explicit guidance/navigation (IMU, flight control system)

- Unpowered coast mode

  Navigation/guidance (IMU, horizon sensor, star sensor, radar altimeter flight control system)
- Powered orbit injection mode

  Explicit guidance/navigation (IMU, flight control system)

On-orbit

- Phasing mode

  Navigation/guidance (IMU, horizon sensor, star sensor, radar altimeter, flight control system)
- Powered transfer orbit injection

  Explicit guidance/navigation (IMU, flight control system)

Rendezvous

- Terminal mode

  Navigation/guidance (IMU, rendezvous radar, flight control system)
- Docking mode

  Guidance/control (IMU, rendezvous radar, flight control system, relative attitude sensor)

Orbit Stay

(Attached to space station - no operation)

Retrograde and Deorbit

- Preretrograde mode

  Targeting, alignment, calibration (all GN&C systems)
- Retrograde mode

  Explicit guidance/navigation (IMU, flight control system)
- Deorbit transfer mode

  Navigation/guidance (IMU, horizon sensor, star sensor, radar altimeter, flight control system)

Reentry

- Explicit guidance/navigation (IMU, temperature sensors, flight controls, rate gyro packages)

Landing

- Cruise and approach mode

  Guidance/navigation (IMU update with ground navaids; DME, VOR, TACAN, LORAN, rate gyro package, flight control system, air data computer)

- Landing mode

  Guidance/control (Automatic GCA involves ground tracking, ground computer data processing, data link to aircraft, tie in to flight control system.) Alternative: modify concept to receive radar data and process on board within the GN&C system)

Table 4.5.4-1

COMPUTER REQUIREMENTS FOR GN&C FUNCTIONS AS A FUNCTION OF MISSION PHASES

Mission Phase	Storage (words)	% Utilization*
Prelaunch		
Mission planning	7800	20(2)
Strapdown algorithm	750	15.6
Ascent (atmospheric)		
Navigation	430	0.3
Guidance	1000	5.0
Attitude control	5260	30.5
Strapdown	750	15.6
Ascent (exoatmospheric)		
Navigation	430	0.3
Guidance	3000	7.0
Attitude control	5260	30.5
Strapdown	750	15.6
Parking orbit and transfer		
Navigation	5400	13.9
Guidance	600	0.5
Attitude control	750	7.9
Mission planning	3000	(all available time)
Strapdown	750	15.6
Terminal rendezvous		
Navigation	430	0.3
Guidance	3000	7.5
Attitude control	730	7.9
Strapdown	750	15.6
Retro/deorbit initialization		
Mission planning	6000	20 **
Strapdown algorithm	750	15.6
Reentry/landing		
Guidance/navigation	1200	4.0
Attitude control	5260	30.5
Strapdown	750	15.6

* Based on IBM 4πEP with floating point arithmetic
** This represents alignment function only.

Table 4.5.4-2

AVERAGE NUMBER OF INSTRUCTIONS PER SECOND BY PHASES

Prelaunch	50,120
Atmospheric ascent	72,400
Ascent (exoatmospheric)	75,200
Parking orbit and transfer	53,340
Terminal rendezvous	44,120
Retro/deorbit initialization	50,120
Reentry/landing	70,530

4.5.4.2 Other Subsystems. Mode control and event scheduling of each subsystem can only be accurately analyzed from a detailed understanding of the phase-by-phase functions. However, it is possible to obtain an approximate estimate of the computer loading required for this task by initially estimating the number of active control points and active data points in each subsystem. Active data aredefined as information that is required to be sensed before making a decision to execute a command. The monitoring of these active data points can be treated similarly to the onboard checkout procedure for malfunction detection and abort warning. However, in this case, the subsystem test routine culminates with a command execution procedure.

From a knowledge of the malfunction detection routines, it can be estimated that an average of 10 computed instructions per test point are required for completing a subsystem checkout procedure. This can be increased by 30 percent to allow for command execution instructions. Thus, an allocation of 13 instructions per active data point can be made. Furthermore, an average iteration rate per test point can be estimated from the average rate of transferring interface data (shown in Fig. 4.8-4 as 7.1 Hz). A worst-case estimate of 3 to 4 times this figure was considered adequate to take care of all control problems within this general category. Thus, 25 Hz was selected as the iteration rate which results in 325 instructions/sec per test point.

The total computer loading per phase for this task is determined by multiplying this figure by the estimated number of active data points required per phase. Table 4.5.4.2-1 gives the complete loading for this task over all mission phases. A memory requirement of 16K words was estimated also by assuming a similar requirement as the onboard checkout system.

Table 4.5.4.2-1

COMPUTER LOADING FOR SUBSYSTEM CONTROL AND EVENT SCHEDULING

	Mission Phase									
	Pre-Launch	Ascent	Orbit	Rendezvous	Docking	Orbit Stay	Retro	Reentry	Subsonic	Landing
Computer loading (thousands of instructions/sec)	103	83	87	90	58	33	78	73	75	65

4.5.5 Controls/Displays

4.5.5.1 General Description. For purposes of this study, the crew station controls and displays were assumed invariant for all three levels of avionics system integration. This was done because the degree of integration at the control panel may be considered independent of the level of integration of the bulk of the avionics subsystem and therefore would not be a contributing factor to the major purpose of the study. In addition, the nature of the programmable display system being considered for the Space Shuttle, with its associated multiformat techniques of information presentation, requires a firmer definition of operational requirements. The following paragraphs describe a preliminary Control/Display configuration.

The approach used to implement Control/Display requirements with candidate hardware included the grouping of requirements by mission phases. Analysis of requirements thus grouped showed that the programmable display techniques would handle the display requirements if properly formatted with well-designed symbology on the CRT or projection devices. However, the need to delineate requirements for dedicated controls and displays also exists. The landing phase contains the bulk of dedicated requirements for jet powered flight. Two electronic attitude director indicators (EADI), one each for pilot and copilot, were chosen to display multiple instrument landing approach parameters. They are used to display vehicle attitude on all other mission phases. In conjunction with two programmable CRT or projection displays (one each for pilot and copilot) all landing phase information may be presented to the crew. The degree to which the CRT or projection and EADI need further dedicated display augmentation is being studied.

Three-axis hand controllers enable vehicle attitude control through the reaction control system while the vehicle is outside the atmosphere and through control of aerodynamic surfaces during the landing phase. Consideration is being given to locating these controllers so that the crewman can operate his controller with either hand. Additional landing phase dedicated controls such as turbojet throttles will be located to permit one-man operation during this phase.

The keyboard devices for making control inputs to the programmable displays and computer switching are located adjacent to the hand controllers at each crewman's right. A static programmable navigation and communications display controlled by keyboard input will be located in the center island between the crewmen.

A proposed solution to the requirements of additional dedicated displays was the inclusion of a mission-phase-oriented CRT or projected display located in the shared panel area between the crewmen. The keyboard for this device would be compatible with that of the other CRTs or projected displays and serve as a redundant controller to both crewmen. In addition to mission phase and sequencing information, this third CRT or projection display may have a dedicated field for the critical caution and warning functions. The requirement for additional dedicated displays is under investigation.

4.5.5.2 Programmable Display Description. The programmable displays represent a major portion of the control/display hardware, and as such require considerable attention. This display technique deviates sharply from past designs by taking advantage of several "human factors" observations. First, a crewman cannot focus his attention in many places at one time; rather, he must concentrate on a small number of areas, or better still, one area. Secondly, display needs vary with time and mission phase, and therefore need not be static. Third, the use of many displays occupies a large surface area within the rather tight confines of a crew compartment. For these reasons a programmable display configuration has been evolved that will overcome these difficulties and provide a legible portrayal of current data status under all conditions. Described herein is a workable, programmable display configuration that has been adapted to the three different levels of integration. The parameters to be displayed will be acquired by the data management subsystem and transferred on request to the C/D subsystem.

The concept of programmable displays encompasses all those parameters that can be presented in alphanumeric form. The concept relies on the fact that the need for particular parameters to be displayed is not continuous and can be programmed, to a large extent, well in advance of a flight. Displays can be programmed to be initiated by time, a single event, or a sequence of events. Override features are built in to allow manual call-up of display parameters by the crew and automatic display of abort warning information by the data management subsystem.

After study of likely crew compartment configurations and investigations of similar compartments in other vehicles, it was determined that three display devices are required. These will be positioned one in front of each crew member and one in the "shared" area between them. In this fashion, each crew man will have to programmable displays available within his scope of vision at all times. With about 50 parameters maximum to be displayed on each device, this will make 100 parameter values available to each crew member.

The display device itself has been chosen as a cathode ray tube, with color capability a desirable feature. This device is more reliable, has better definition, is brighter, and has more versatility than other apparatus. The major drawback to CRT devices is the need to refresh at rates of 40 or more times per second.

It may be possible to use storage tubes or specially selected phosphors to simplify system design, but definition of the exact hardware complement must await further definition of requirements.

The display system is made up of several major devices whose characteristics are defined below:

Keyboard Input. This selection device enables the crew to pick the appropriate display format. It will allow for at least several hundred and as many as a thousand different formats while retaining a very simple, easy-to-use selection keyboard in conjunction with a keyboard "program." Verification of the selected formats will be displayed to the crew by displays that are inherent in the selection device.

Data rates originating in the keyboard are very low; however, the keyboard must be scanned at least several times a second for a change of request. A decode function must be performed (in the keyboard display control) to determine which of the formats has been selected, and a verification signal will be generated to return to the displays associated with the keyboard.

Display Parameter Selector. Receives a selection command from either the keyboard input device or from the associated display control processor. This command will enable the selector to select appropriate parameters through the data management interface. The selector must interface with the keyboard display control unit in order to receive crew-initiated requests in the form of a digital word. This feature provides the "manual override" of the automatic display system. It will either be necessary for the selector to scan the input lines repeatedly in order to recognize input changes, or an interrupt feature must be provided to allow a new keyboard message to inform the selector of its presence. The selector must first establish contact through the interface then transmit a series of addresses for the selected parameters to the data management subsystem. Data management in return will send the current digital value of each of the selected parameters. In order to maintain an updated display, the current values must be retransmitted about once per second. A memory function must be incorporated in both the display and data management subsystem in order to remember the parameter list and its correct order. If a maximum of 50 parameters is assumed to be displayed on each display at any one time, and there is a maximum of 500 parameters from which to select the display, the addresses required to select the series of parameters may require about 450 bits to be transmitted to data management per display. The digital value train will require approximately the same number of bits (450), to be transmitted to each display subsystem. For a crew station incorporating three programmable displays, the data flow between the data management subsystem and the displays will be about 1350 bits per second.

For offline maintenance, the number of parameters selectable for display may equal the number of test points (2090) requiring about 600 bits per display and, possibly, a different bit rate.

The display parameter selector must refer to its associated memory to determine the parameter list described above. A memory to house up to 1000 parameter lists, all different, can require as much as 600,000 bits of storage. But this can be divided between a large slow-access memory and a small rapid-access memory. Additionally, by careful coding, the total storage requirement can be reduced.

Display Control Processor. Performs the most complex functions of the programmable displays. These functions include formatting of the displayed material and origination of display requests based upon passage of time or occurrence of certain events. By reference to its associated memory, the processor must determine a screen location for every part of the display, perform the addition of title blocks and format lines as required, and underline or emphasize displays by brightness change or size change as required. Memory requirements for this function will be at least equal to the selector memory and can also be hybrid -- a combination of slow and rapid access memories.

The major task for the processor is the generation of alphanumeric codes to supplement the parameter lists of the selector. Each time a new display is initiated, a coded form of the display must be generated by reference to code tags associated with each parameter. If 5 bits per character, 20 characters per parameter, and 2000 total parameters are assumed, it can be seen that up to 200,000 bits of storage will be required for the code tags. It may be possible to reduce this number substantially because of redundancy in the tags; for example:

PRESSURE, PSI, #1 TANK, XXX
PRESSURE, PSI, #2 TANK, XXX

Before this stored information can be displayed, it is necessary to transform all character codes into a pattern of "on" and "off" lines or dots in the form of alphanumeric characters. Depending upon the scheme selected, up to 50 bits may be required to adequately define each character. It is in this fully defined form that the display must be presented to the refresh memory (or the display device if no refresh is required).

Display Raster Generator. This device, which includes a character generator and a graphics generator, accepts the alphanumeric and other inputs from the processor and converts them to a form suitable for the refresh memory; that is, all displays must be sequenced (if a TV raster display is used) or otherwise organized to be suitable for input to the deflection and brightness modulation circuits. Character, symbol, and line codes must be converted to elemental form prior to loading the refresh memory.

Display Refresh Memory. Stores the complete display picture. It will probably be a rotating device with about 250,000 bits of storage for each complete display. In order to provide a flickerless display on a typical cathode ray tube, a refresh rate of about 40 cycles per second must be used. A tradeoff study must be performed prior to design in order to determine whether a storage tube can be used in place of a standard cathode ray tube, with the attendant reduction in complexity by deletion of the refresh memory. The study must include such parameters as resolution, brightness, required refresh rate (if any), reliability, etc.

Other features of the programmable display system include the following:

Builtin Test Routines. During countdown, or at any other time when the operation of the display equipment is questioned, a builtin test routine can generate a series of patterns to distinguish certain types of malfunction. It may also be possible to include some diagnostic capability.

Recognition of Events or Time. The basis of the automatic operation of the displays is the recognition of events or times and the programming of displays associated with them. The purpose of this configuration is to provide displayed information as it is required throughout a mission and to relieve the crew of the necessity to originate display requests.

Priority Interrupt. This feature is provided in the data management subsystem. It allows warning and abort information and other notification of malfunction to be presented immediately upon occurrence.

Manual Override. Provided by the use of the keyboard input in certain modes of operation. It can be designed as a partial override; that is, part of the programmed display can be retained while a portion is manually selected. The specially designed keyboard provides excellent flexibility and verification to the operator when a selection is made.

4.6 ALTERNATIVE - 1

The first of three progressively more integrated electronic system designs is presented as a baseline.

4.6.1 Introduction

Contemporary electronics systems are predominantly federated designs. Evaluation of the benefits to be derived from integration is desirable against the absolute scale of current practice. The federated design presented as Alternative 1 is employed as the baseline against which more integrated solutions are measured.

Alternative 1 is configured against the same vehicle (orbiter element only) as are the other alternatives. The essential functional requirements of this vehicle system are defined in section 4.5. Integration in Alternative 1 is limited to two subsystem areas:

- Crew Systems - 8.0
- Data Management - 9.0

The rationale behind this decision and the pertinent boundary constraints are provided for use in subsequent evaluation.

4.6.2 Crew Systems - 8.0

A common man-machine interface is postulated as a result of technical coordination during the study. This decision removes configurations of control display from the list of variables dependent upon levels of integration employed.

Meeting the conceptually desirable man-machine interface with a single design actually imposes hardware interface constraints on federated systems, as discussed under options.

The programmable display element of control display is described in section 4.5. The interface between displays and the data management subsystem is discussed in the following paragraphs.

4.6.3 Data Management - 9.0

A data management subsystem is employed as one of several federated subsystems required to satisfy the identified functional requirements. Specific functions included in the Alternative 1 Integrated Electronics System and implemented by the data management subsystem are:

- Onboard checkout
- Fault isolation
- Abort warning
- Operation support (embraces the terms of configuration control for sequencing)

Descriptions of each term are included for clarity.

4.6.3.1 Onboard Checkout. Onboard checkout includes routine fully automatic logical isolation of failures to the level of a box or unit that can be replaced on the vehicle to restore normal operation. The configuration defined assumes online testing (i.e., the unit is evaluated during appropriate periods of normal operation) as the primary evaluation mode.

Offline test requirements in critical areas are recognized to be desirable. This requirement is satisfied by informing the crew of step-by-step procedures for testing and evaluating critical functions through use of manual controls, which are available for backup.

A permanent record of all failures is stored for machine recall along with manually entered flight log data to supply maintenance/administrative information.

4.6.3.2 Fault Isolation. Detailed preprogrammed instructions are included onboard to permit the crew to evaluate any subsystem in detail and isolate faults to a level beyond the capability of a fully automatic program. Simplified, logically arranged data are presented at crew request from a data bank stored in the interlocking control unit of Fig. 4.6-4. This technique permits a minimally trained crewman to evaluate all subsystems under the control of original equipment/subsystem designers, without a requirement for standby personnel during missions.

A fully automatic fault-isolation mode is provided in the onboard checkout concept, limited only to the electronic interface employed (i.e., 2046 test point design). The logical output is the identity of the failed unit.

4.6.3.3 Abort Warning. Abort warning is the output of the subsystem, based on preprogrammed logic, notifying the crew of the occurrence of events that have been defined as potential reasons for termination of the mission or alteration of the mission through the programmed display interface. Sufficient data and instructions are presented with the detected condition of warning to permit the crew to independently evaluate the threat and take appropriate action. This latter function is more appropriately considered a part of operations support.

4.6.3.4 Operation Support. This term is a catch-all for the essential elements of the realtime information and planning service presently rendered to the spacecraft crew by Cape Kennedy and Houston. Presentation of time-line performance, limited configuration control, and automatic access and display of pertinent crew related instructions is included under this heading.

4.6.3.5 Component Technology. A common basis of evaluation is maintained by estimating Alternative 1 physical parameters against 1972 technology (refer to Appendix B .

4.6.3.6 Reliability Configuration. A single thread system design is evaluated, with the impact of redundancy flagged for two examples in section 4.11.

4.6.3.7 Dedicated Control Display Functions. No impact is included in Alternative 1 for this factor, which is common and invariant for the three alternatives examined.

4.6.4 Requirements

Alternative 1 is constrained to satisfy the requirements of the vehicle (section 4.5) but not the desired characteristics of an IES (section 4.4).

The following essential requirements are normalized and eliminated from consideration in each of the three alternatives:

- Control display crew interface
- Builtin test equipment
- Component technology
- Reliability configuration
- Dedicated control display functions

A description of the control display crew interface is provided in section 4.5 for the man-machine concept being supported by all alternatives.

The integrated display portion of the crew systems subsystem and its interface with the data management subsystem is discussed under the baseline configuration.

Builtin test equipment (BITE) has been evaluated (refer in Appendix D) and the suggested implementation made in Alternative 1.

In summary, BITE is employed within a separately stockable, line-replaceable unit to evaluate the health of elements that are not accessible through the normal in-out signal paths (internal power supplies, redundant voting logic, signal path integrity). The results of this evaluation are available, through interrogation, to a common data management subsystem.

Total box performance within the subsystem/system is a logical evaluation performed by the common data management subsystem, with BITE information employed in the process.

4.6.5 Options

Many options exist for virtually any level of integration. Significant options considered for Alternative 1 are presented, with limited arguments for each.

Most subsystem configurations are fixed by the groundrule of federated subsystems adopted for Alternative 1. The subystems are identified as:

1.0 Structure/mechanical
2.0 Propulsion
3.0 Electrical power
4.0 Environmental control/life support
5.0 Guidance/navigation
6.0 Vehicle control
7.0 Communications

The remaining two subsystems within the system are potentially to be treated as an integrated composite, federated as the other subystems are, or to some intermediate design. Three options are presented for the combined concept of crew station and data management subsystems, within the context of Alternative 1.

Both the data management and the programmable displays element of the crew station share a common requirement for interfacing all subsystems. An interdependence exists between these subsystem elements in two ways: first, the data management subsystem, in common with other subystems requires the crew station man/machine interface; second, the crew station is supported in the functions of :

- Subsystem configuration
- Presentation of mission rules
- Presentation of contingency plans

These requirements are invariant within available options and must be met by each.

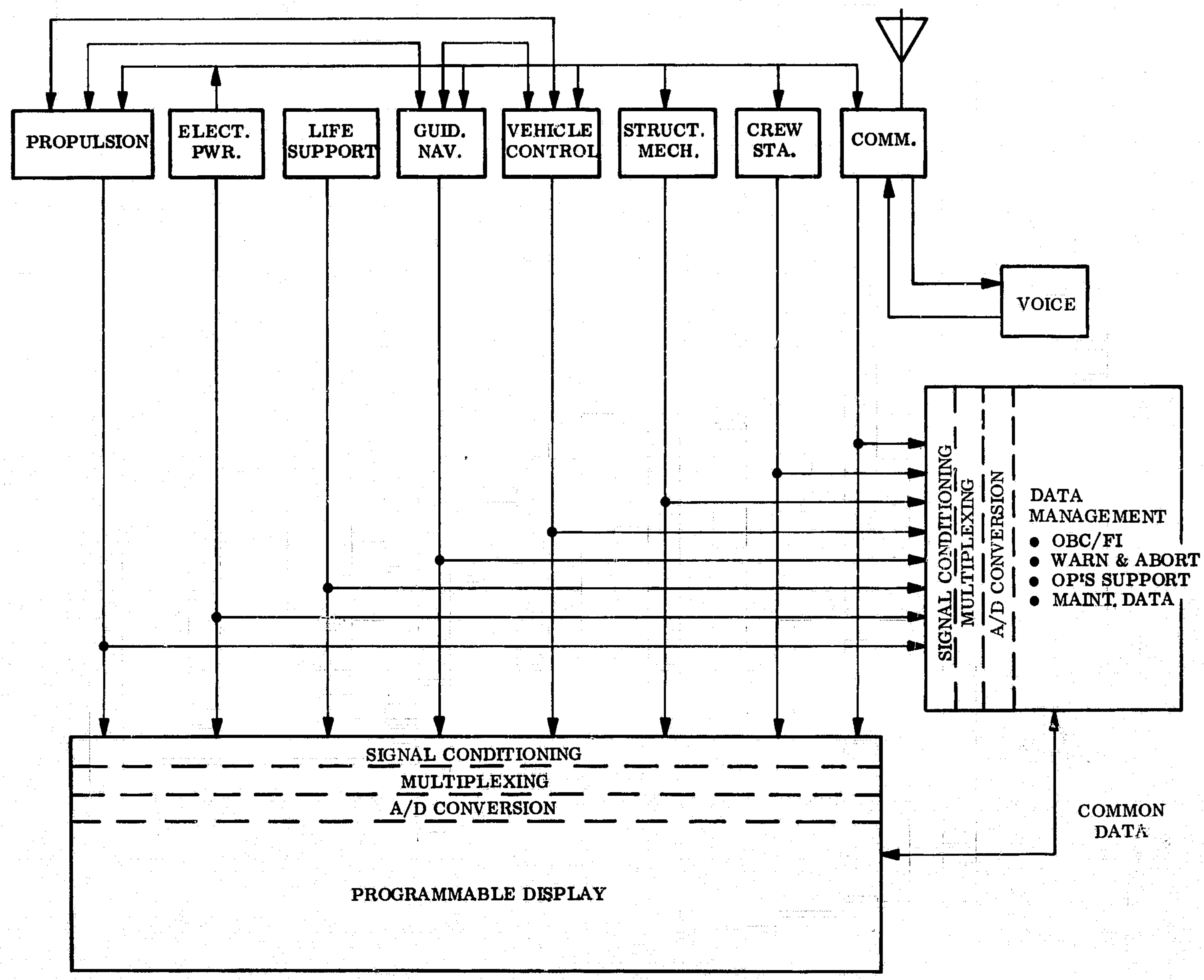
4.6.5.1 Option A. Option A (Fig. 4.6-1) is a completely federated concept where the data management subsystem is considered independent from the crew station except for those interfaces required to meet normal operation, i.e., control and display.

Signal Acquisition. In consonance with the concept of federated autonomy, the 1811 signals required for programmed display and for test will result in

- 494 signals for display
- 1811 signals for test

An examination of the constituency of the signal populations (Fig. 4.5.2-1) shows a 100 percent overlap of data required in the two acquisition systems. Recognition of this duplication within the data population and its proliferation into piece parts, weight, and power required (for duplicate but separate signal conditioning, multiplexing, and conversion) makes the federated approach unattractive, primarily because of the artificial penalty imposed on the programmed display. Separation of the data acquisition into two indepednent subsystems has the potential merit of providing a backup mode in the event of failure in either acquisition complex if provisions are made to share common data between the subsystems.

One of the study constraints is the requirement that each alternative be a single-thread, nonredundant design. Constraining the crew system and data management subsystems to separate and duplicate acquisition of data under reliability considerations is eliminated under this consideration.



KEY CONSIDERATIONS

- CONVENTIONAL S/S DESIGN
- INTEGRATION INTERNAL TO CONTROL/DISPLAY AND DATA MANAGEMENT SUBSYSTEMS

FIG. 4.6-1 OPTION A

Voice Signals. No voice multiplexing is considered in the three alternatives examined; therefore, voice is independent in all three options (Figs. 4.6-1, -2, and -3).

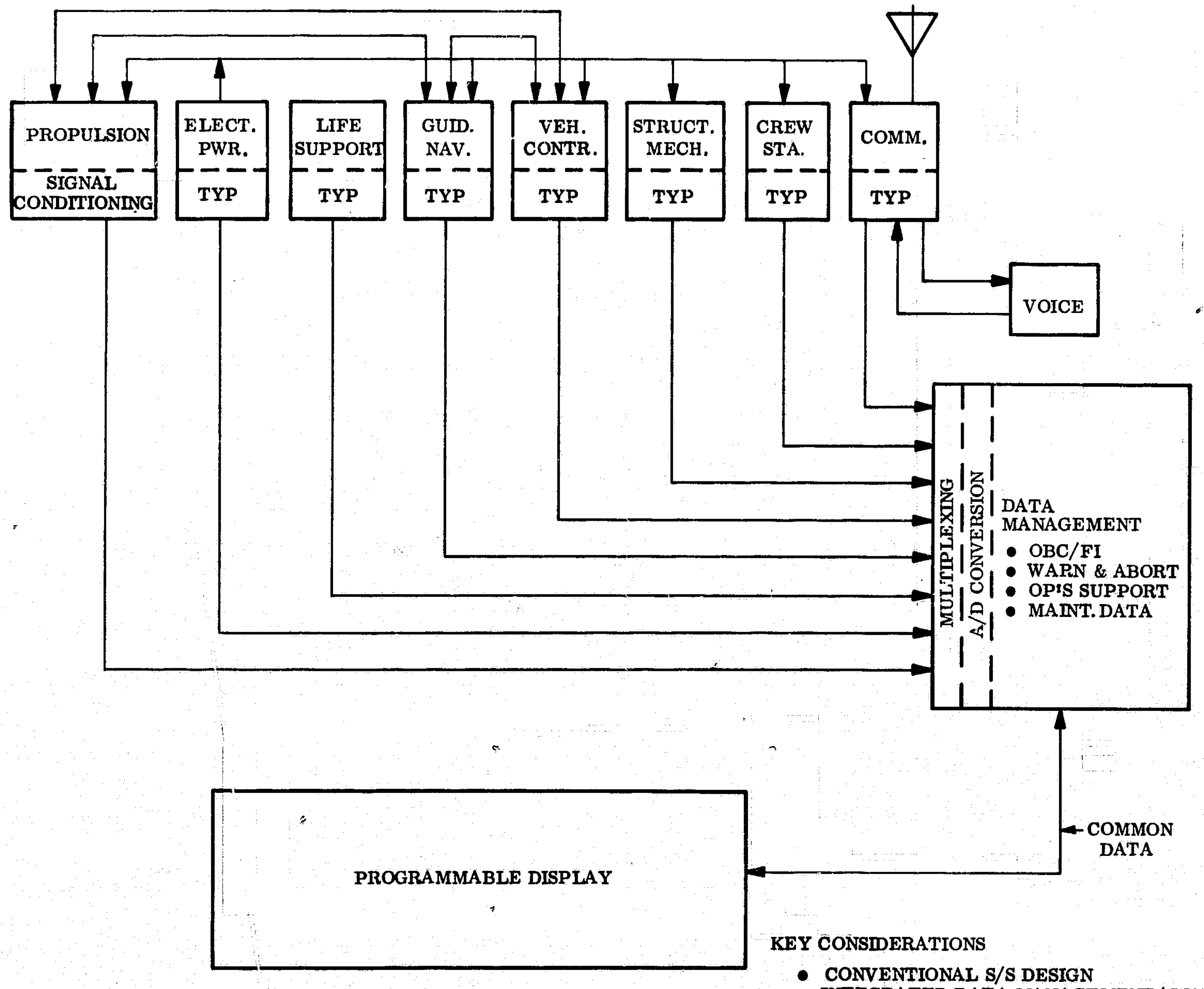
Basic Functions. Independence of the basic functions fulfilled by the separate subsystems is fundamental to the Alternative 1 concept and is maintained in the three options presented.

4.6.5.2 Option B. An integrated data acquisition function is employed in Fig. 4.6-2 on the basis of commonality between input data for the two subsystems (data management and display).

Further integration is attractive between the processing requirements for display data checkout, abort warning, and mission support. The computational requirements in each center are minimal, much of the activity being data transfer and simple logic routines. Implementing integration of these computer centers, while desirable from an end-item view, is avoided for Alternative 1. The Alternative 1 groundrule of federated subsystems would be sufficiently distorted to eliminate some of the incremental difference of Alternative 2, lending an undesired bias to the results.

A fundamental study requirement on Alternative 1 is that subsystem-to-subsystem and unit-to-unit wiring be dedicated (not time-shared). This constraint clearly does not satisfy the desirable characteristic of common interface between elements; however, it is characteristics of existing system design.

Interfacing a data acquisition subsystem to the above-described federated design requires a high level of analog signal interfacing. Reference to the test point listing of Fig. 4.5.2-1a shows a ratio of analog to discrete signals of 1759/622 or 2.8 to 1.



KEY CONSIDERATIONS

- CONVENTIONAL S/S DESIGN
- INTEGRATED DATA MANAGEMENT/CONTROL–DISPLAY SUBSYSTEM (SIGNAL CONDITIONING IN SUBSYSTEM)

FIG. 4.6-2 Option B

Signal conditioning is a generalized term and requires clarification for any particular application. In the context of integrated electronics, the signal conditioning considered at this point is probably more properly referred to as normalization, to simplify the signal acquistion process. Circuitry required for a normalized input may be dedicated to each signal or shared by signals of similar characteristics.

Shared signal conditioning can be used to give an advantage in weight, volume, and power consumption. This technique is employed in operational systems. Three factors lead to deletion of this concept for the baseline. First, a single-point failure mode is introduced by shared signal conditioning (redundancy techniques can eliminate this problem). Second, unless high impedance isolation is provided within the unit being monitored, the prime operating signal exposure to negative environment influences is expanded over the signal path required to reach the shared signal conditioner. Third, if high impedance isolation is employed in the monitored unit, a significant amount of the signal conditioning is already dedicated. A variation is possible if the shared concept is applied at the monitored box level to eliminate exposure of sensitive circuits. The case of a single signal from a box is trivial, since the effect is that of dedicated circuitry. Consideration of the multisignal box is valid although marginally useful, since the probability of multiple signals of similar characteristics is not overly high. The primary difficulty in rationalizing this concept lies in the imposition of a submultiplex requirement within the monitored unit and a clock interface for synchronization.

The above points lead to selection of dedicated normalizing signal conditioning for this study.

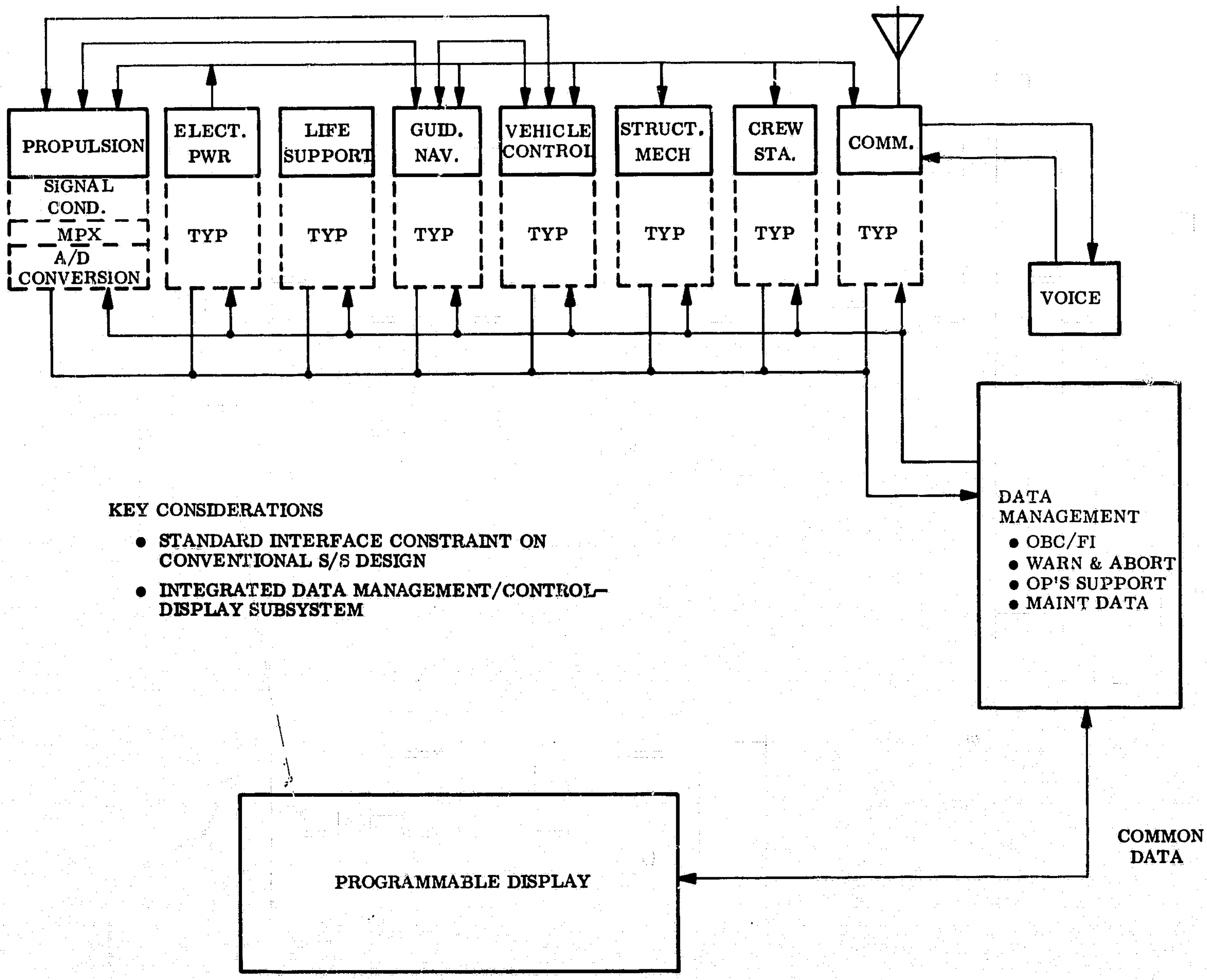
A second alternative exists in the decision regarding location of the dedicated circuitry. It is recommended that the normalization and necessary fault protection of the monitored circuit be provided within the monitored unit. A dual benefit accrues from this decision: maximum protection from environment is obtained and the original equipment designer can assure his design integrity in the presence of the interfaced data management subsystem.

Experience to date suggests that advanced hybrid signal conditioning can be implemented for an average of approximately 0.3 ounce and 300 milliwatts per signal. Hybrid technology is required for high ohmic resistors (megohm range), which are not within the limits of thin-film techniques.

4.6.5.3 Option C. Inclusion of local multiplexing and analog to digital conversion in the monitored units is the distinguishing feature of Option C (Fig. 4.6-3).

This concept is attractive, and integrated circuit technology can support it. Two significant considerations are cause for its rejection for the Alternative 1 baseline. First the circuit design considerations required for a digital interface in the traditionally analog, federated subsystem approach (guidance/navigation and vehicle control excepted) creates a significant impact on the power and weight of the subsystem elements of the baseline configuration. Eliminating this factor from the Alternative 1 data management subsytem will then permit a more equitable evaluation of the incremental changes in weight and power between Alternatives 1, 2, and 3. Also, application of the digital interface to subsystems that consist of many small elements, such as valves and gages, will often create a weight and power impact greater than the circuit (unit) being monitored.

4.6.5.4 Baseline Option. The option selected as an element of the Alternative 1 baseline is Option C. The essential features of conventional subsystem design concepts and minimizing the negative impact of a common display are overriding considerations.



KEY CONSIDERATIONS

- STANDARD INTERFACE CONSTRAINT ON CONVENTIONAL S/S DESIGN
- INTEGRATED DATA MANAGEMENT/CONTROL-DISPLAY SUBSYSTEM

FIG. 4.6-3 OPTION C

4.6.6 Baseline Configuration

The rationale supporting the need for a baseline configuration in general and the system concept selected was presented in **section** 4.6.5. All subsystems are the same for each alternative except the data management subsystem. The Alternative 1 data management factors considered in the selected design are discussed in the following paragraphs.

4.6.6.1 Design Factors. The data base accessed by the data management subsystem consists of 1816 signals, with an average sampling requirement of 3.2 samples per second for full frequency restitution. (Reference Fig. 4.5.2-1a and Table 4.5.2-1).

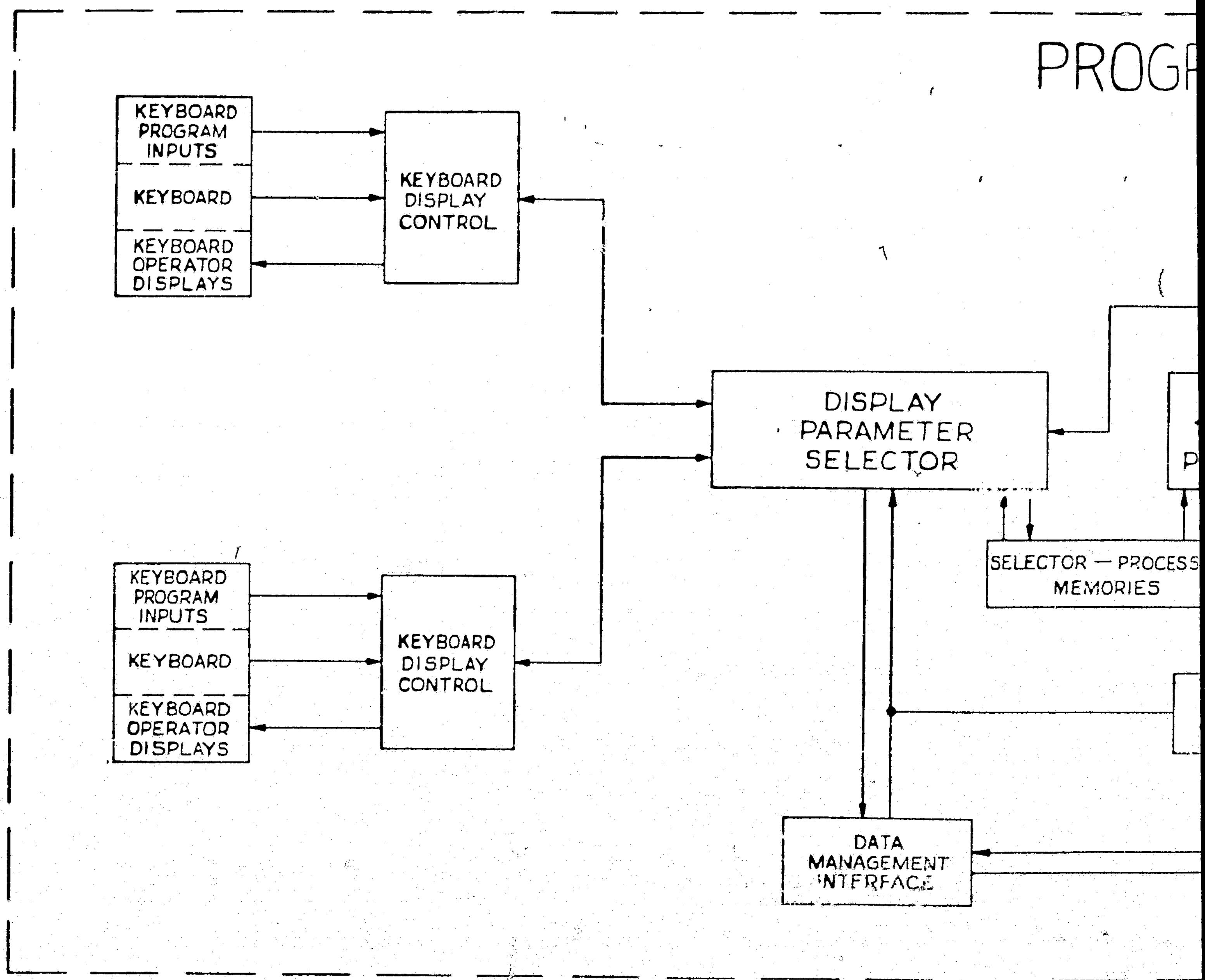
The function requirements of section 4.5 for

- Programmable display
- Onboard checkout
- Fault isolation
- Abort warning
- Operational support
- Maintenance support

must be satisfied within the previously outlined constraints. In addition, as many of the desirable features of section 4.4 are to be incorporated as practicable.

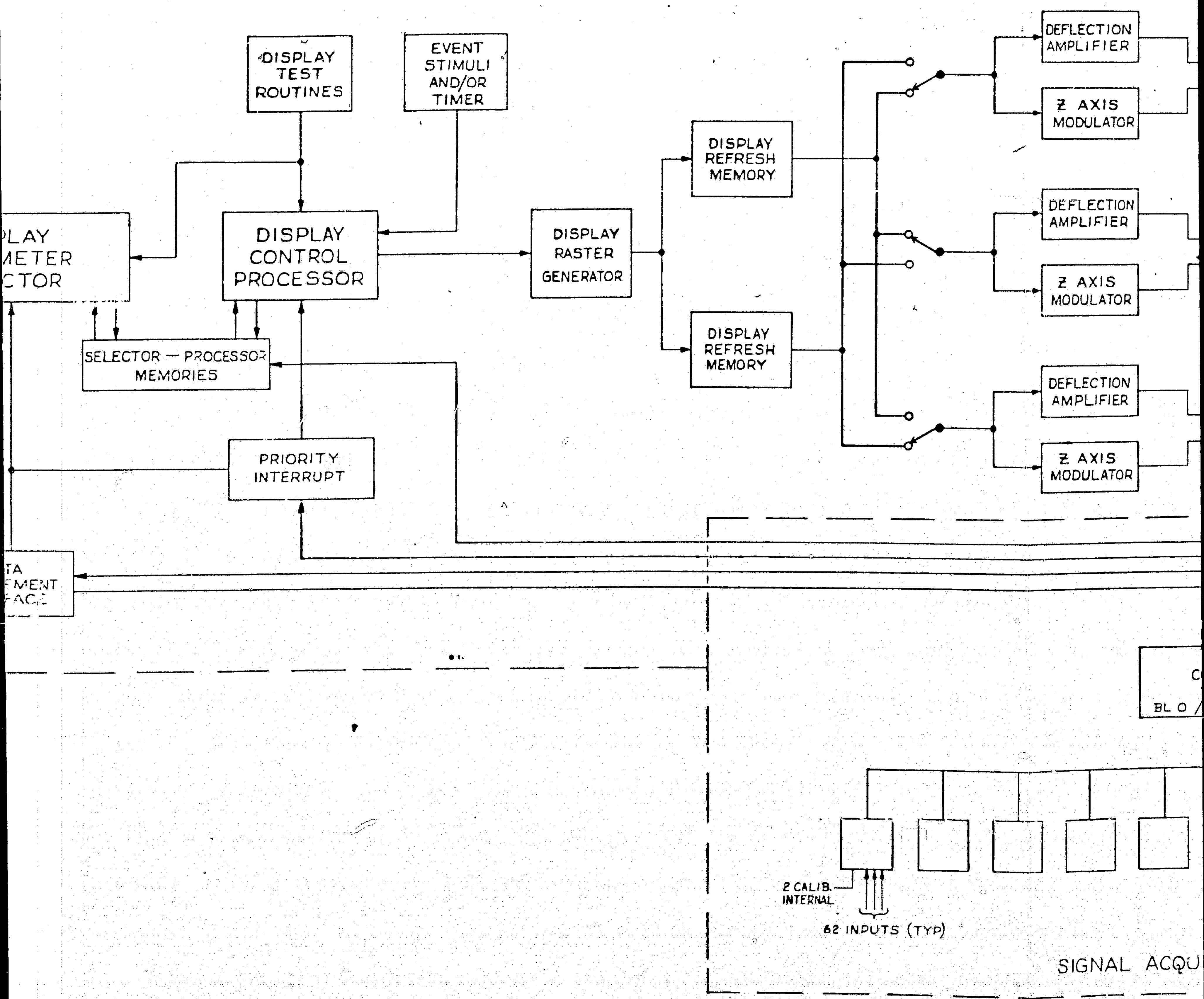
The following paragraphs discuss the mechanization employed, the significant alternatives considered, and **the function of the elements comprising the baseline of Fig. 4.6-4.**

4.6.6.2 Data Management Subsystem. A prime function of the data management subsystem is to collect data for presentation and decision making. A system of remotely located signal acquisition units is employed for a common interface with the federated subystems.

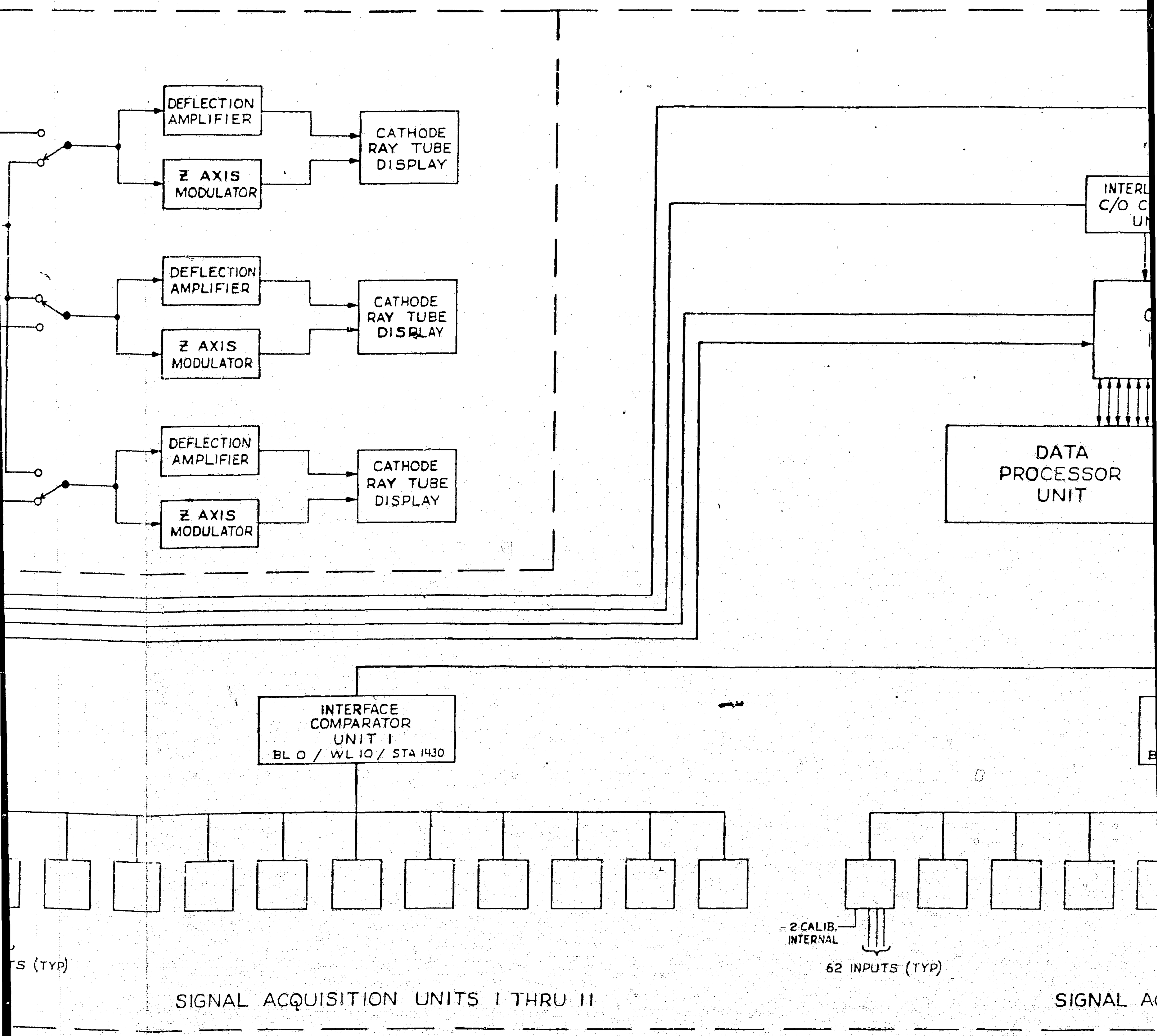


FOLDOUT FRAME (

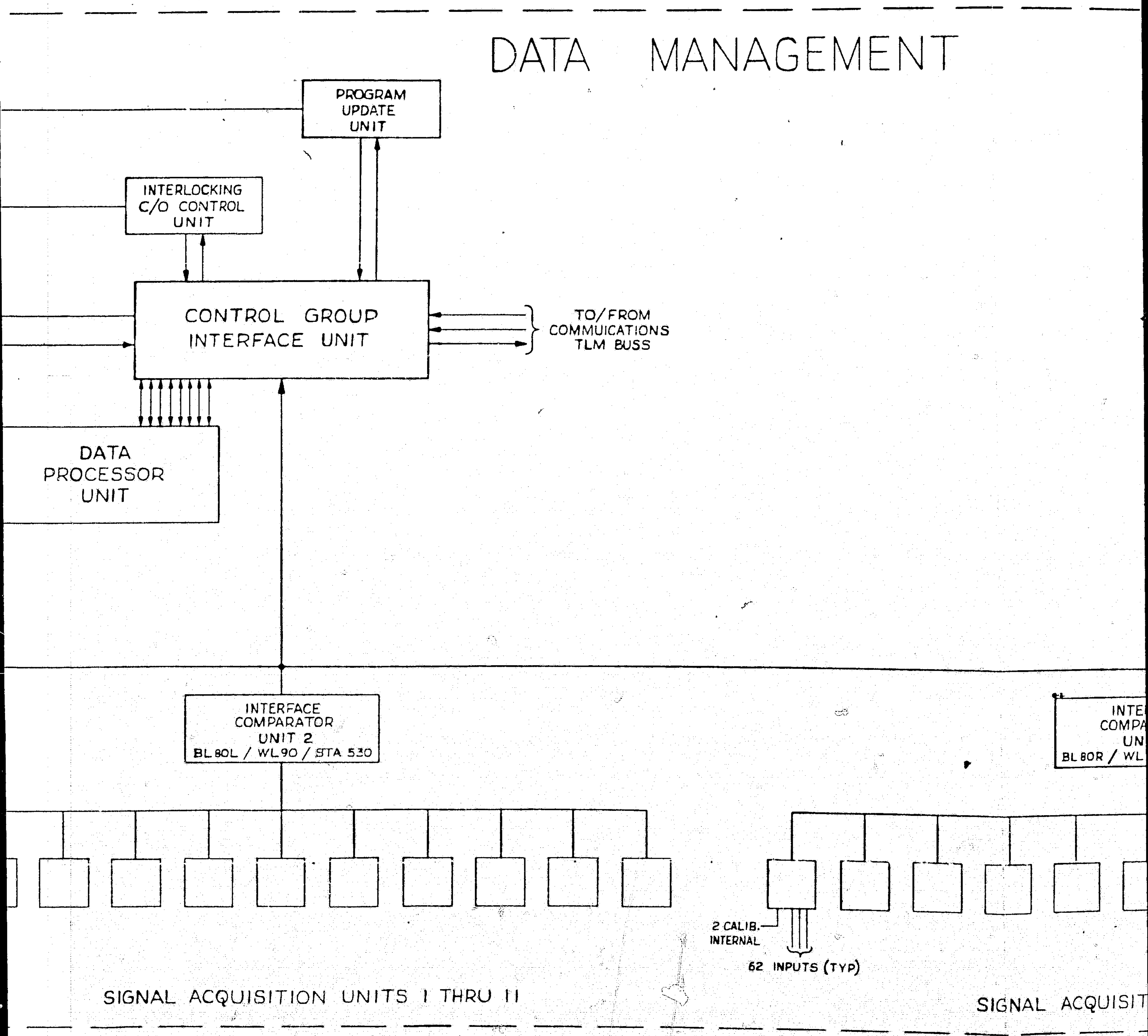
PROGRAMMABLE DISPLAYS



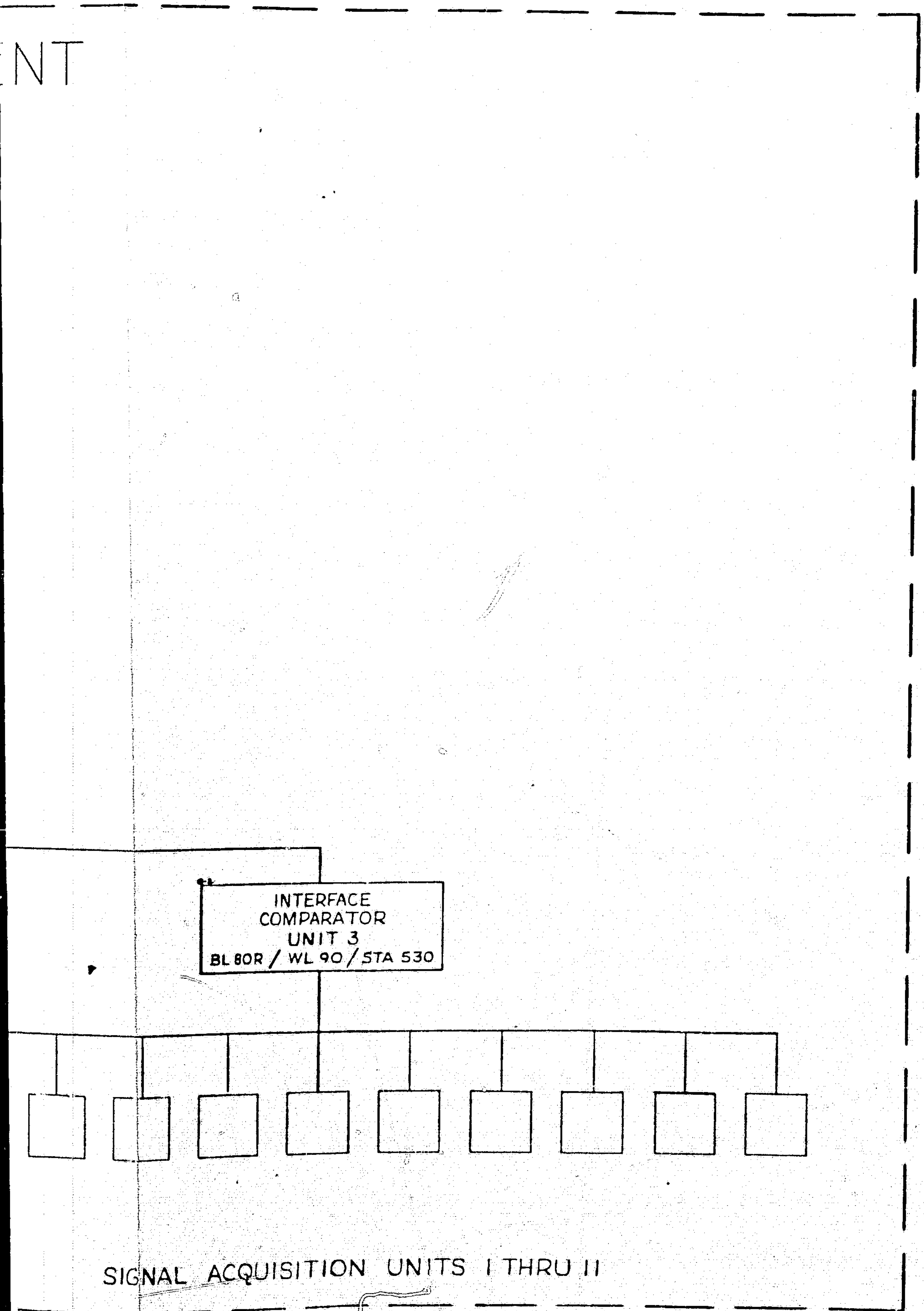
FOLDOUT FRAME 2



FOLDOUT FRAME 3



FOLDOUT FRAME 4



INTERGRATED AVIONICS
ALTERNATIVE 1, OPTION
BLOCK DIAGRAM
FIGURE

Fig 4.6-4

4.6.6.3 Signal Acquisition. A common, normalized analog interface is used as discussed under options considered. The sizing of remotes is conveniently done in powers of two. Several companies have developed integrated multiplex chips, usually based on 16 inputs.

Sizing of the units is a geographical problem, since the primary consideration is minimizing wire weight.

Fig. 4.8-3 shows an isolated concentration of test points around station 850. For the basis of this study, a single acquisition unit of 64 inputs is selected to service this low-signal-density area, and is applied as a standard throughout Alternative 1. The absence of precise signal coordinate information precludes a more vigorous solution. This decision, as implemented, provides a comfortable 11.4 percent growth factor. A trade study should be conducted on any specific design to justify the sizing of the remote access units.

4.6.6.4 A/D Signal Conversion. Two potentially conflicting requirements influence the location within the data stream for conversion from analog to digital. First, it is desirable to minimize the number of digitizers to one centrally located, since this is feasible (normalized signal characteristics) and gives reduced weight and acquisition costs. Second, the effects of ambient noise are minimized by digitizing as near the signal source as possible to reduce exposure of the normalized analog signal. This latter desirable feature is maximized with separate digitizers for each signal. The compromise solution selected is to digitize the multiplexed signals in each remote acquisition unit. Digitizing at a lower level involves either dedicated digitizers for each signal, or for each box serviced. The latter course would demand inclusion of multiplexing within the box and either external sample control or buffer storage for the external acquisition system interface. This approach was ruled out under Option C. The weight associated with individual converters (200 pounds at an estimated

0.1 pound per signal for a LSI converter) is sufficiently high to raise a serious question of its advisability in a nonredundant design.

Conversion at each of the remote units minimizes exposure of the relatively noise-susceptible analog signals to the environment localized around their source and gives the noise immunity of digital data for the transmission link to the processing center.

The average sample rate required is 3.2 samples per second. This data rate (based on data of 400 Hz or less) is quite low, well within available technology.

Signal paths from the remote units to the comparator may be either single or multiple bus structures. The low rates are compatible with single bus techniques, and a single bus is employed in the baseline. In the event higher data rates are desirable for growth, a two-bus concept is recommended to permit addressing during data transmission.

Signal access sequencing can be under control of a central processor (as in MADAR) or a separate programmable control unit, or it may be local control with adequate synchronizing from a master clock.

A locally controlled (comparator unit), sequential sampling scheme is selected to reduce data rates on the transmission line and permit growth changes with minor hardware alterations. Growth provisions impact the address word length to permit the addition of more comparators.

4.6.6.5 Signal Acquisition Unit. The selected remote unit receives normalized analog information from the monitored subystems through dedicated signal conditioning. The remote unit provides

- Multiplexing
- Calibration (self test)

- A-to-D conversion

Self-test is possible by processing a signal of known value through the end-to-end path and verifying its arrival at the processor. Gating elements are simple and reliable, and addition of calibrated signal sources or added multiplexing to permit a calibrated input to each signal source is considered impractical. The impact in increased part count (lower reliability) and priority interrupt requirements to permit insertion of the calibrate signal are sufficiently severe to raise a question of the validity of such an approach.

A technique employed rather widely is to devote one or more of the input signals to a calibrated source (such as a zener) of known input within the data frame.

With alternative and a self-contained reference, two of the 64 inputs are assumed to be used to exercize the signal path at a value inside the end point of the digitizing range.

Sequence control should not be a problem, from a moding requirement, in view of the low sampling rates. A worst-case design can reproduce the expected frequency content well within the state-of-the-art. The design considered is brute forced to the extent of providing full signal reproduction capability to a buffer storage in each comparator unit (Fig. 4.6-4) on a single-mode sequence. The buffered data can then be accessed as desired without the requirement to change sampling rates.

Growth considerations for sampling mode changes can be satisfied by the use of read only memory in the comparator unit to control the sample sequence. A common LSI array could be designed with discretionary wiring prior to encapsulation to provide any desired sequence if technology is available. Core rope will meet this need today. Inclusion of two or more of these arrays in the design will give the program an option that can be selected as needed. The selection, being initiated in the processor program, would include instructions for the reconfiguration required for buffer identification.

The primary reason for varying sample rates is power conservation.

On the basis of the requirements of section 4.5, data rates into and out of the access unit are tabulated for reference:

- Input to acquisition unit - 62 analog inputs at an average bandwidth of 1 Hz
- Input to digitizer, 64 signals at 3.2 samples/second for 205 samples/second
- Output from digitizer, 205 samples/second at 10 bits/sample for 2050 bits/second per acquisition unit

For sizing, it is convenient to consider the bit rate generated at the digitizer to be 2.5 kb/s.

The basic data rate required to sequentially sample the data base is the sum of the address and data requirements. The address requires 4 bits for acquisition unit identification plus 6 bits for the 64 signals. With a potential of 704 signals to be sampled 3.2 times per second, each sample requires a total 20-bit request-reply, or approximately 4.1 kb/s for each acquisition unit. This requirement imposes a request-reply rate demand on the comparator of 45 kb/s for the acquisition units.

4.6.6.6. Interface Comparator. The comparator decouples the asynchronous, demand-oriented data users (processor and programmable display) from the cyclic acquisition portion of the subsystem.

Incoming data are submitted to the logic examination of Fig. 4.6-5.

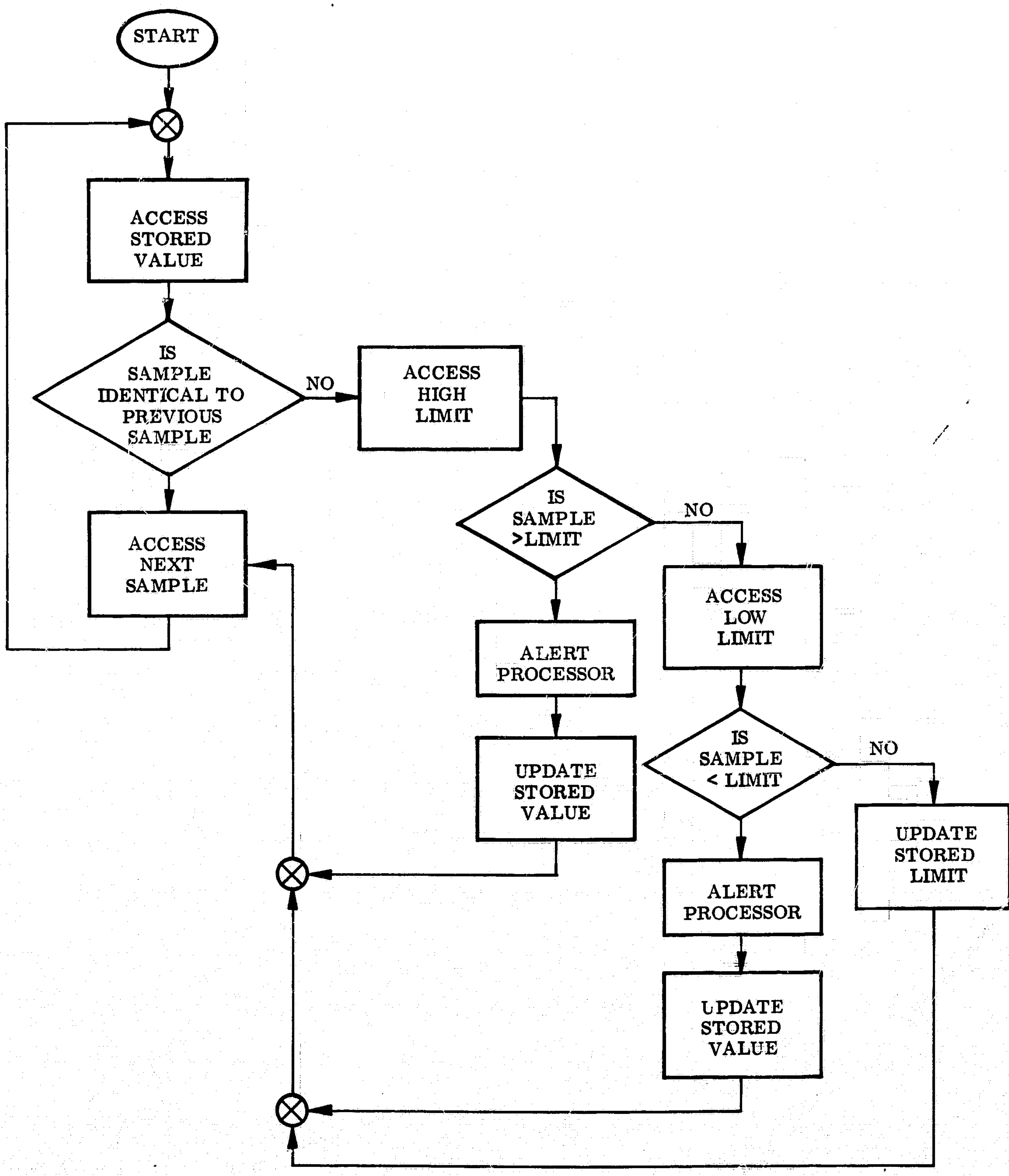


Fig. 4.6-5 Interface Comparator Logic Diagram

This function can be performed in the data processor quite simply; however, as seen in the discussion on busing loads in section 4.7, it is desirable to facilitate growth through performing this function remotely. An added benefit is the ability to access any signal for display as a ground checkout feature.

Worst-case design would suggest a memory of 64 30-bit words. Each word consists of the following subwords:

Present value	10 bits
Upper limit	10 bits
Lower limit	10 bits

Examination of the test point data shows approximately one-third of the points are discrete; therefore, considerable reduction is realized in a final design by using this fact to reduce bit requirements.

Roughly, the worst-case reduces to

21 words at 3 bits + 43 words at 30 bits = 1353 bit memory

The comparator memory thus requires

$$\frac{1353 \text{ bits/acquisition unit} \times 11 \text{ units}}{32 \text{ bits/word}} = 466 \text{ equivalent words}$$

Data flow between the comparator and the remainder of the system is relatively heavy, yet still below that level presenting a challenge to the anticipated state-of-the-art; approximately 157 kb/s are required for full data fidelity at the processor.

Requirements for the programmable display are discussed under that heading, and data needs may be supplied either by time sharing with the data processor needs or by priority interrupt. It is recommended that an interleaved time sharing be employed with a backup priority slaved to the model of operation, either manual or automatic.

The entire data base can be accessed by 24 bits per parameter with full words or in a 50.7K bit frame which, at 3.2 frames/sec, represents only 324 milliseconds at a 500 KHz clock rate. The projected 1350 bit/sec display requirement only adds 2.7 milliseconds, if satisfied serially.

The implementation of an alert regarding the exceedance of any stored limit is by an addressable buffer, which indicates the occurrence with a bit change and includes the address of the parameter that penetrated a limit. A five-tiered register should be an adequate buffer, permitting 15 faults in the interval between strobes.

Memory for the comparator could be solid-state, since the intelligence for re-initializing after loss of power exists in the processor. The initializing process would require each limit memory to be refreshed, resulting in a 64-millisecond interrupt. A more conservative approach that uses core or plated wire is suggested to eliminate this problem.

4.6.6.7 Control Group Interface. Interconnection of the data management, programmable displays, and telemetry bus is provided. The limited time for the study prevented detailing the mechanization. No significant problems are anticipated, the unit acting as a hard-wired programmer to effect data flow.

4.6.6.8 Data Processor. Existing general-purpose processors are more than adequate for the proposed baseline configuration. System decoupling provided by the comparator permits the processing elements to operate at a higher multiple of the acquisition clock to minimize priority problems. Use of 1 or 2 microsecond add time machines will permit growth in application well beyond initial requirements. A question exists on the method of storing the data required in support of the previously listed functions.

The data storage summary for Alternative 1 is as follows:

Budget Summary	32-Bit Words	Average Use Rate B/S
Basic routines	5050*	-
Checkout/fault isolation	7950*	-
Prelaunch (2 hr)	32,011	141
Ascent (436 sec)	2956	224
Orbit (7 days)	79,262	7
Rendezvous (15 min)	6157	219
Entry (2000 sec)	6157	156
Approach and landing (45 min)	7909	96
Warning and abort	2132*	-
Trend data	2000*	0

*Independent of mission segment

The internal computer memory includes the 17,132 words asterisked. With a conventional arrangement of 4000 memory blocks, a 20,000 word capacity is adequate. Approximately 3000 words are employed as a scratch pad to buffer the mission control data from external storage.

The most severe requirement is during ascent.

The configuration control information and operations support data supplied to the crew are at very low rates (reference section 4.5), with most of the activity consisting of logic routines performed in the data processor by the use of programming information stored in the program update and interlocking checkout units. These logic routines are executed on data from the acquisition element of the data management subsystem.

The most severe bit-rate requirement is during ascent. It is also a mission segment where the entire support package should be entered rather than updated during operation.

A basic design is indicated which consists of

Scratch pad	3000 words
Main memory	17,000 words

with the remainder of the serially required data in supplemental storage. It appears feasible to consider read-only memory for the basic routines and warning and abort, although warning and abort requirements could alter with design changes during vehicle life.

The preferred memory is plated wire, on the assumption that the aging problem will be solved.

The concept selected will read into temporary storage in 96K-bit bytes in 192 milliseconds.

Two types of data storage are required to support the operation

- Machine data
- Visual data

Considerations for each are discussed.

4.6.6.9 Program Update Unit. The previous budget of stored data indicates a requirement for approximately 150K words of machine data. Several methods of storage are available; core, which is the most expensive in space and weight; plated wire, which is relatively bulky; MOS arrays, which suffer from power interruptions when employed in a read/wr te configuration, but which are quite dense; holography and laser technology, which could potentially store the machine and visual data requirements in volumes comparable to the previous considerations.

In keeping with the relatively conservative approach of Alternative 1, a magnetic storage device is postulated. Technology is well developed and the use of such devices are demonstratably compatible with long-term use by relatively unsophisticated personnel.

An incremental read-write capability is required and a read-on-write feature is employed to check the validity of the data when the tape is loaded. Use of electronic buffering is employed to minimize the effects of word-to-word start-stop requirements.

Bit packing has been demonstrated feasible in excess of 2 Kb/inch and 100 tracks/inch. An 800-bpi linearly and 16 tracks lead to 32 feet of tape for the entire requirement. An attractive approach is to package related mission segments in modular containers.

This concept would require a capacity to accept the entry and landing segments without module clearage; a 10-foot tape is more than adequate.

The mission support routine would be programmed to alert the crew to the requirement for a new module through the programmable display. Crew response is evaluated by verifying the proper module through decoding the leader of the unit installed. Either an OK or corrective statement would be displayed.

A secondary source of machine data exists in the stored words in the interlocking control unit.

4.6.6.10 Interlocking Checkout Control. Requirements for preprogrammed visual data for operations support are met in this unit. The data are expected in

- Text
- Graphics

Development in holography may impact this area significantly by the 1972 time frame. Existing laboratory demonstrations show promise; however, the modest data requirement appears to keep film storage in a competitive position.

The display of film stored data can be by direct projection. Problems encountered in direct projection are glare and the desirability of projecting from the general location of the crewman's head. Rear projection circumvents the latter problem, but both compete with the primary display tube for critical panel space. Rear projection using the display tube face is technically possible; however, considerations of parallax and skew require the tube to meet undesirable design constraints. Either final optics must be on the rear centerline, built into the tube, or a thick, optically flat, window must be provided. Surmounting these obstacles still leaves the difficulties of registration and light intensity variations of the projected and electronically painted information.

The technique selected for Alternative 1 is to use a remote television monitor to scan the film frame. Low light intensities are required as compared to direct projection (front or rear) and the information is compatible with the selected display.

Synchronization of the visual data and the machine processed operations support are provided through a data field adjacent to each film frame. Half words are used, ten per frame. One word is the frame address. The remaining words are used to request actions such as changing data display formats or requesting a subroutine action from the processor. The dual requirement of the configuration control and verification of crew response can be implemented by judicious use of the capabilities provided in the baseline configuration.

A desirable feature that is not included in the design is automatic configuration control.

The baseline configuration, in keeping with the groundrule of conventional federated subsystems, has omitted the electronic interface that would issue commands decoded from the crew-requested actions to automate configuration control. Addition of this feature would reslt in a minimal impact, primarily in a small (1K to 2K) memory, demultiplexing, and D/A conversion for an estimated 600 control signals.

4.6.7 Physical Characteristics

The characteristics of weight and electrical power requirements for Alternative 1 are listed in section 4.10. The incremental change for Alternatives 2 and 3 are tabulated where applicable.

4.6.8 Technical Risk

No technology requirement in either element or system was identified that exceeded anticipated 1972 technology. In general, the techniques employed are presently being used on production contracts.

An estimated 85 to 90 percent of the elements required will be production configurations requiring either selective quality control or minor redesign and selective quality control to support the program.

The remaining 10 to 15 percent of the elements are expected to be new design, based on techniques and capabilities that support similar production items. Technical risk of Alternative 1 is commensurate with any high-performance avionic system employing technology at the forefront of production capability.

4.6.9 Software

(See section 4.9)

4.7 IES ALTERNATIVE 2

Alternative 2 of the IES study differs from Alternative 1 by integrating into the data management the control display subsystem and the interface control function. The interface control function is concerned with multiplexing or time sharing the transfer of data between "boxes" and subsystems in the vehicle. By performing this function within the data management subsystem, it is feasible to devise a truly integrated control, display, and data transfer subsystem for the total vehicle. Therefore, Alternative 2 will include in data management the following functions:

- Onboard checkout/fault isolation (OBC/FI)
- Abort warning (AW)
- Operation support (OS)
- Configuration control and sequencing (CC/S)
- Interface control (IC)

Alternative 2 will permit totally integrated configuration control and checkout of the vehicle and will also allow each operational subsystem to remain autonomous. With this concept, it is possible to attain the objectives of:

- Cable weight reduction
- Reduced number and complexity of interfacing devices
- Enhanced reliability by reduced parts count
- Ease of maintenance
- Reduced EMI problems
- Growth capability and flexibility for future modifications

The major function that Alternative 2 will perform that Alternative 1 did not is the transfer of operational data. Therefore, the data management subsystem in this alternative becomes an active part of the operational subsystems. This is an important factor in the system design with regard to reliability, safety, and mission success. The onboard checkout information is now the operational information, and is acted upon as directed by the routines stored in the data processor (whether distributed or centralized).

The data acquisition part of the data management subsystem may be common for all operations, and the actual operational procedure on those data can be controlled by the processing part of the data management subsystem. The traffic studies of data and test points show that this commonality is true; only the operating speeds or times for checkout or subsystem operation are different. As long as the data acquisition part is capable of operating at the maximum speed or rate necessary for any function, it can be common for all functions. This allows for standardization of hardware (interfaces), data formatting, and software. Therefore, the data acquisition is common for all functions and the control of the data becomes the major factor.

4.7.1 Options

The data acquisition will be common for all options considered. A common interface circuit will be located within each "box" or subsystem function, as shown in Fig. 4.7-1. This interface circuit will be standardized for every interface that is in the data management subsystem and will format all data to the standard format needed for the data management multiplexed data transfer function. Each signal will be formatted to a digital serial bit stream and passed to other "boxes" or subsystems by a standardized coax twisted-pair cable. This circuit can functionally correspond to the signal acquisition unit used in Alternative 1. Depending on the option of processing discussed below, the detail hardware will change slightly but the basic functions will be the same in the standard interface circuit.

4.7.1.1 Option A. Option A centralizes all the control of the data transfer and data processing as shown in Fig. 4.7-2. The standard interface circuit is used in all peripheral equipment (subsystem functional equipment) and interfaces directly with the central computer. All of the data management functions (OBC/FI, AW, OS, CC/S, and IC) are processed by this computer. One coax twisted-pair cable is routed to every item of peripheral equipment in the entire vehicle.

Even though this option saves weight and may save some cost in hardware, especially in a redundant system, there are many technical problems. The

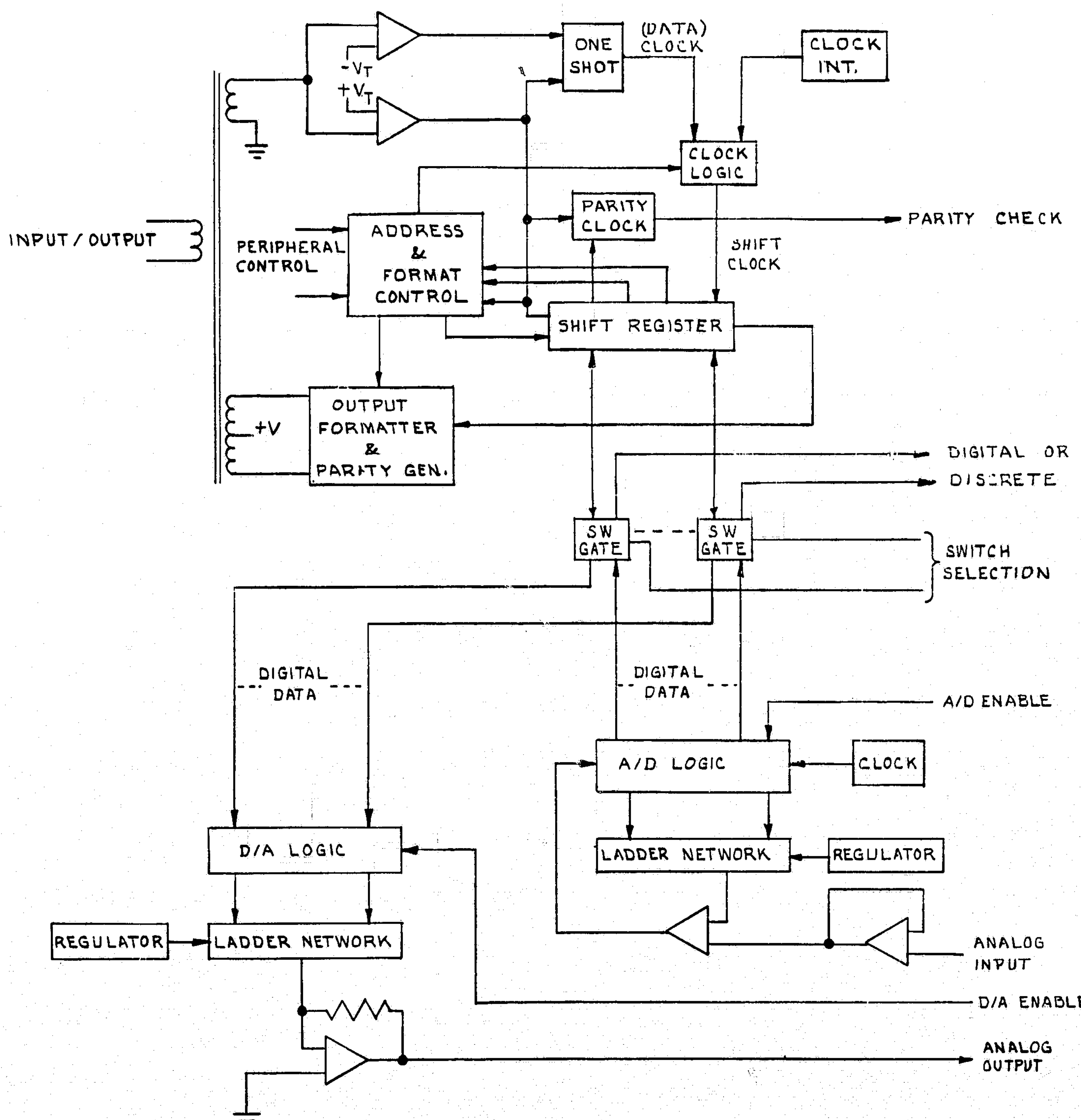
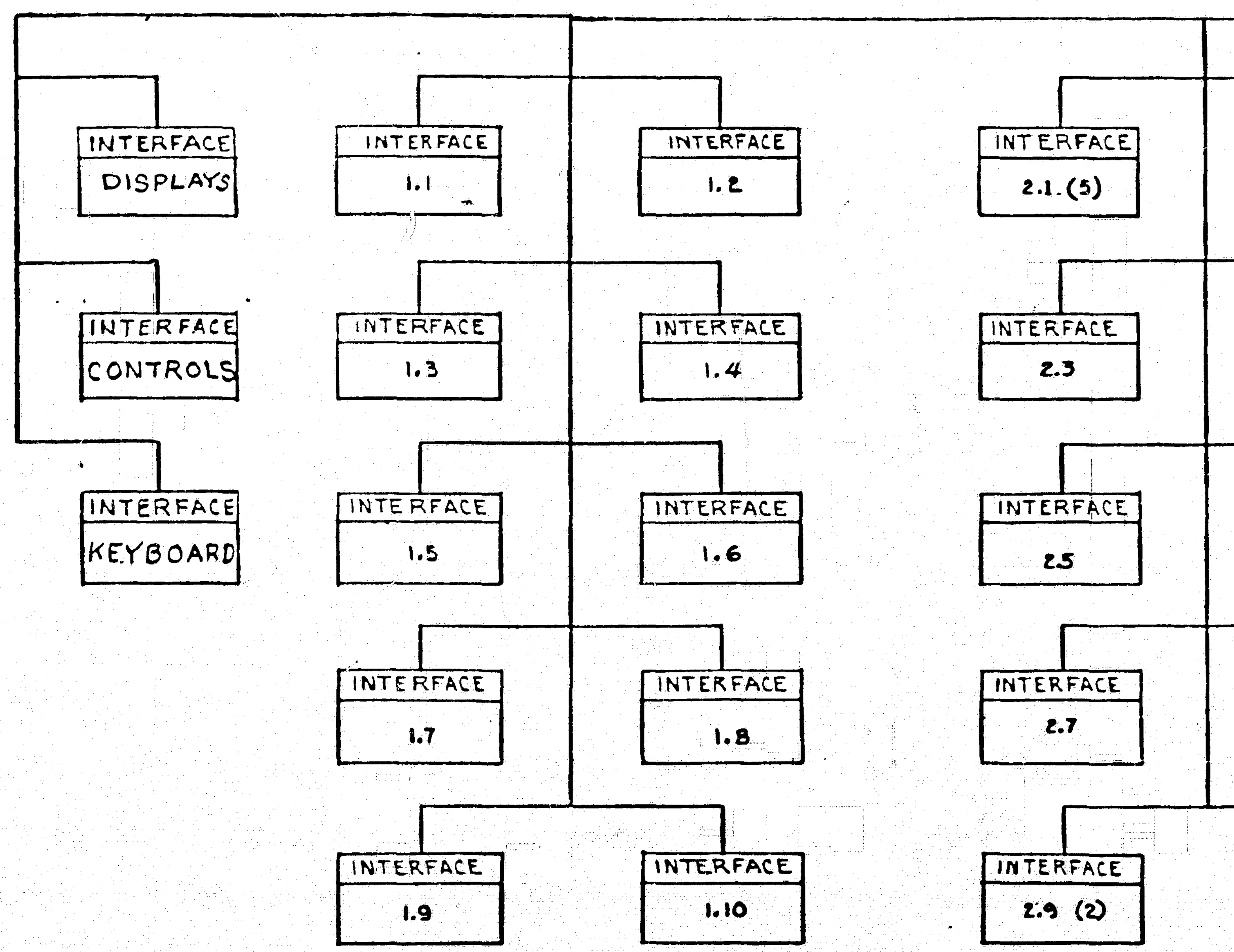
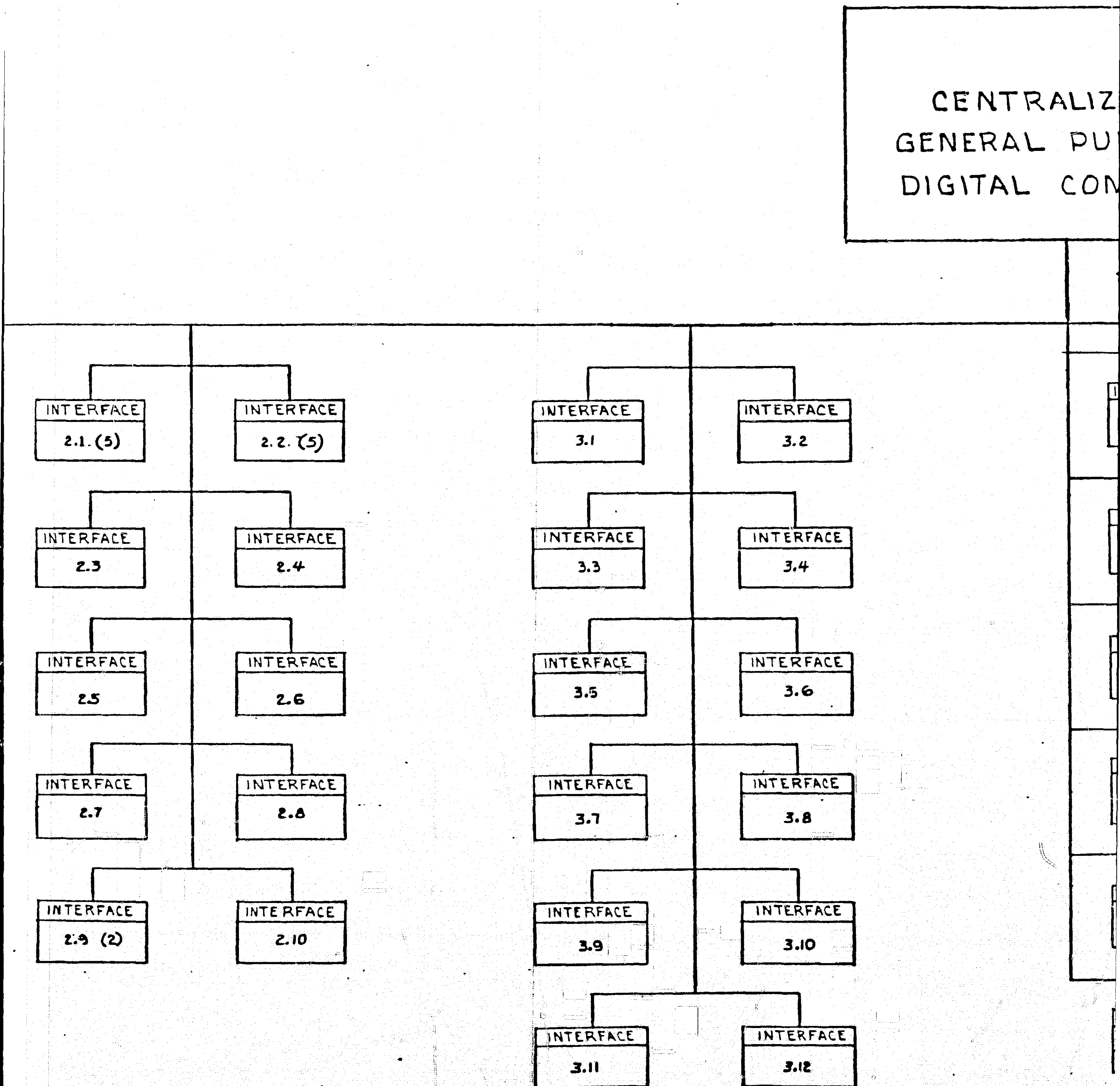


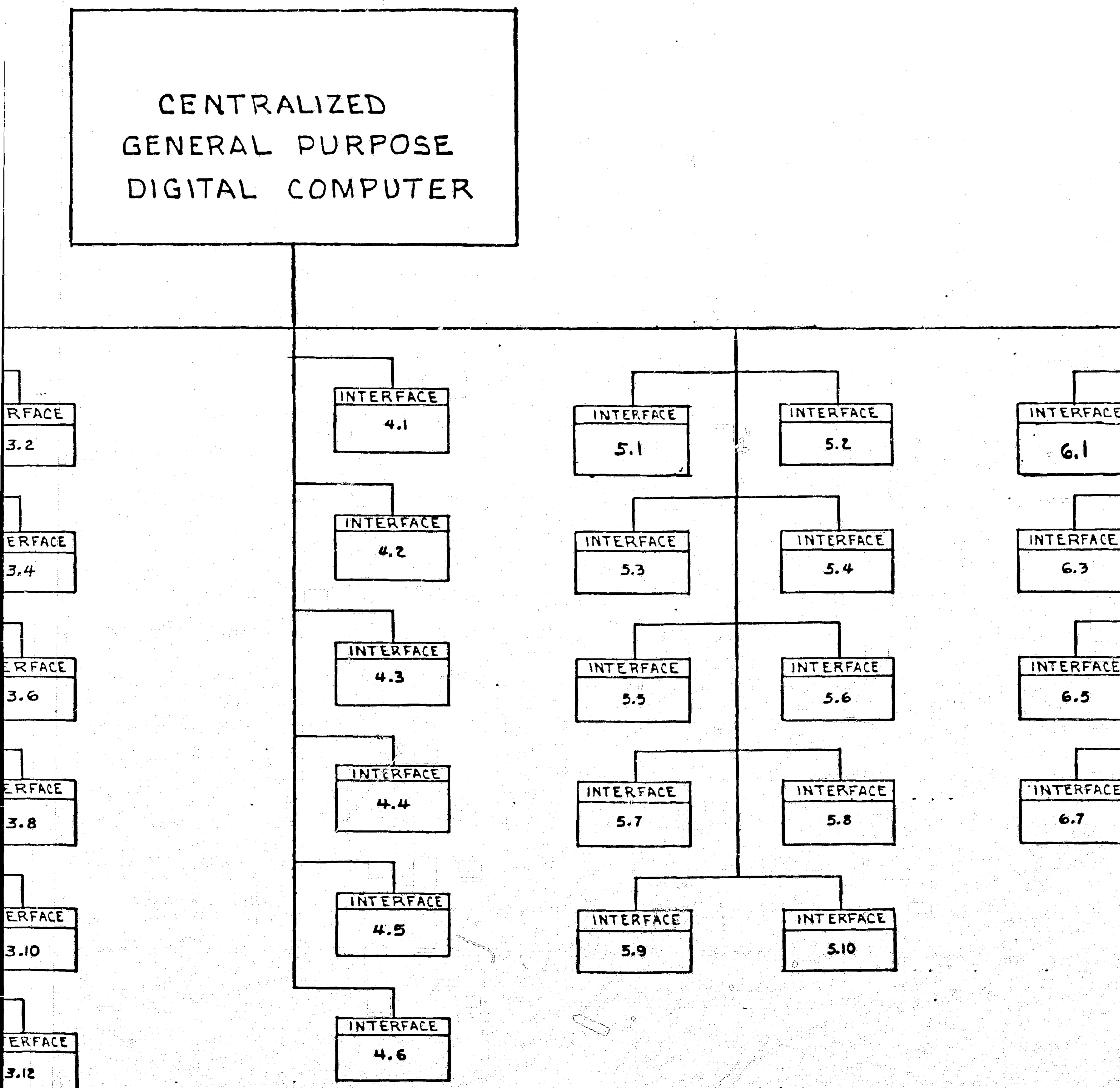
Fig. 4.7-1 Alternative 2 Interface Circuit



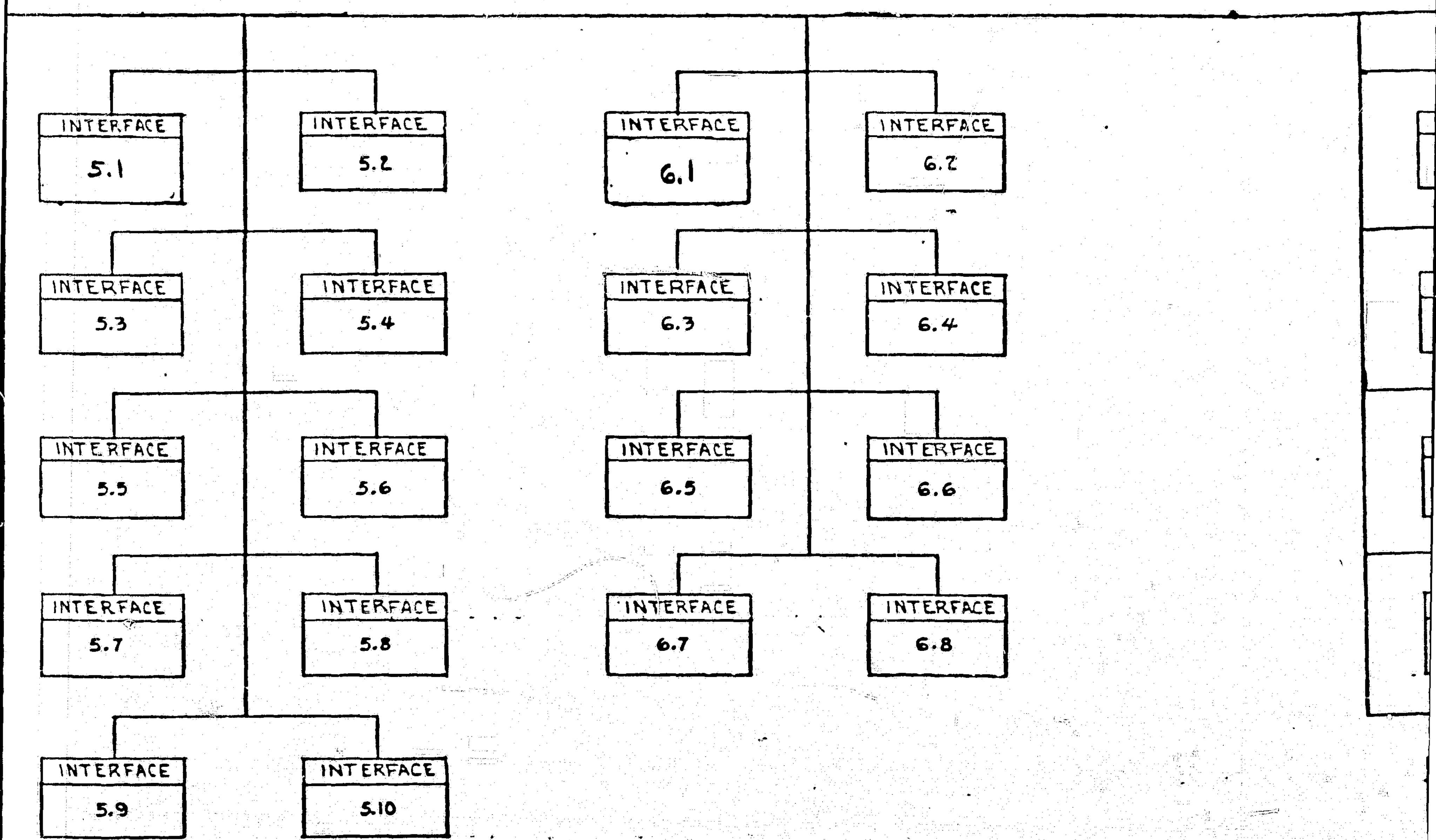
FOLDOUT FRAME 1



FOLDOUT FRAME 2



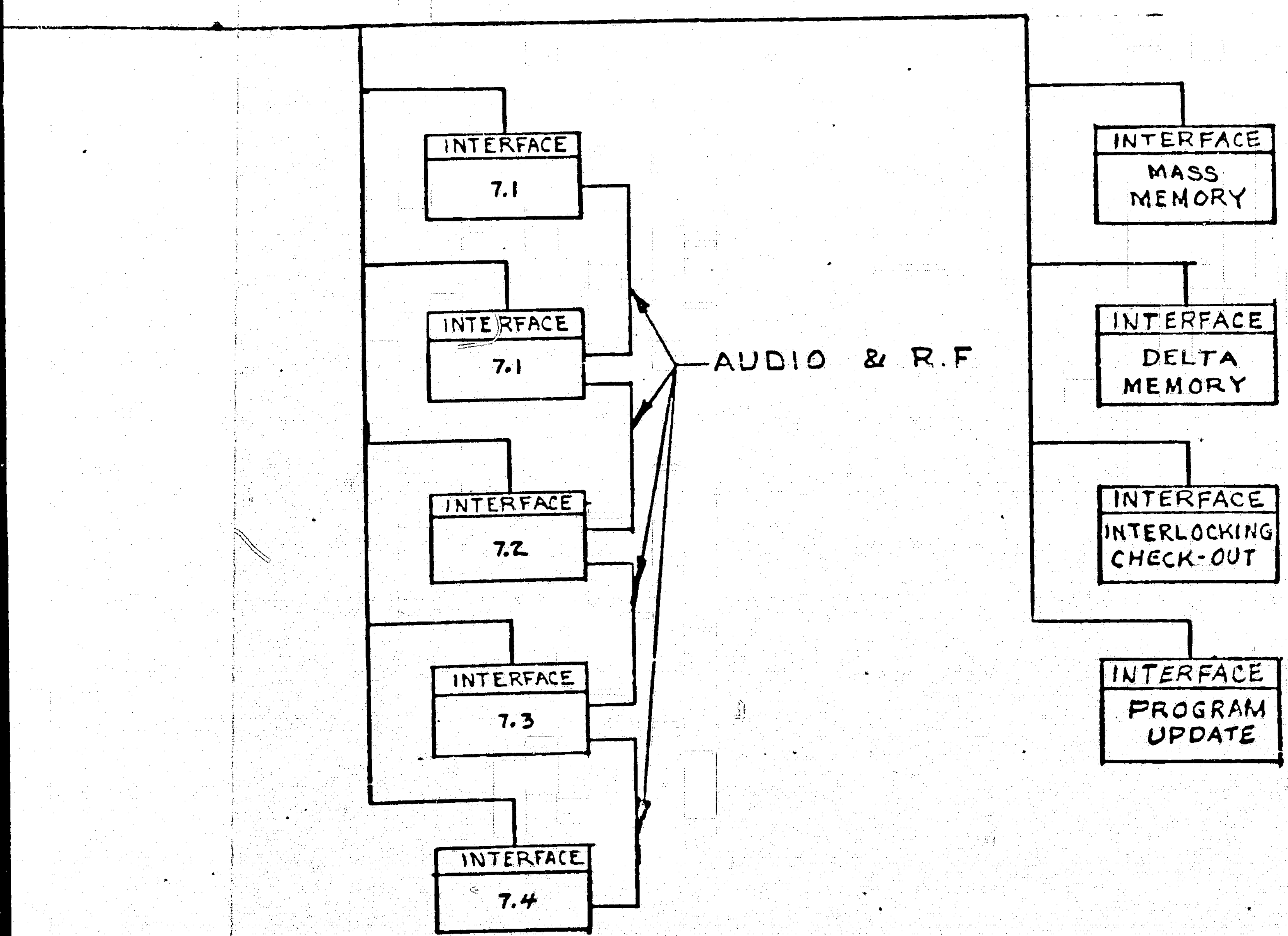
FOLDOUT FRAME 3



AME 3

FOLDOUT FRAME 4

LMSC-4956837
Vol III



FOLDOUT FRAME 5

INTEGRATED AVIONICS
ALTERNATIVE 2, OPTION A
BLOCK DIAGRAM
FIGURE 4.7-2

PRECEDING PAGE BLANK NOT FILMED.

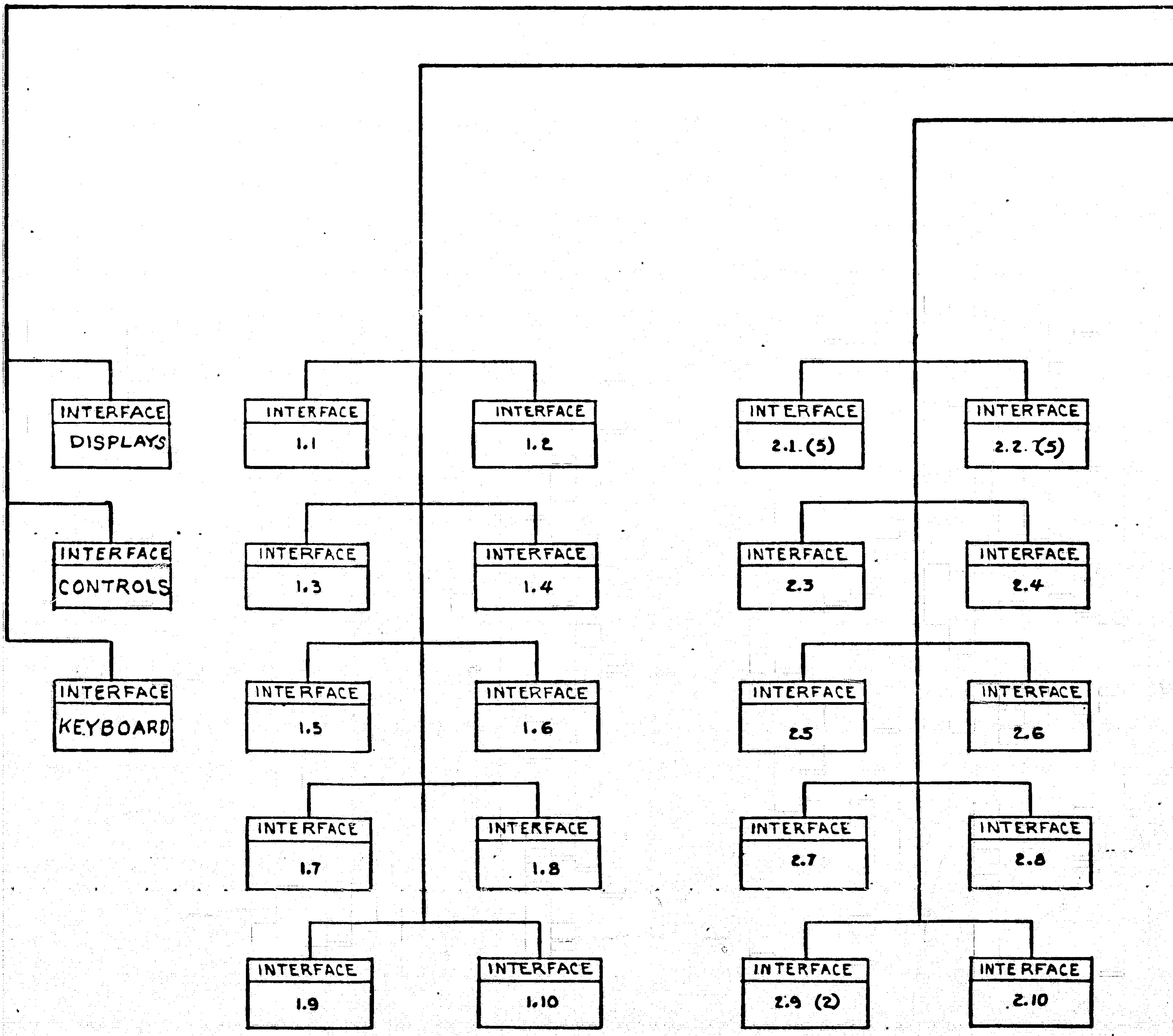
major problem is electrical transmission line matching and termination due to the large number of interface circuits on one line. Each of the connections will cause a discontinuity in the line and, due to the high bit rate (1.5 MHz to 3 MHz), the discontinuity signal reflections will build up in the line and may cause unacceptable bit errors. The reliability of a single line is poor. The centralized system has all data flowing on the same line, and errors in the data are more likely to cause major errors in the operation of the vehicle subsystems.

The technical risks for this option are higher than for other options with respect to software, reliability, and bit errors due to transmission line termination problems.

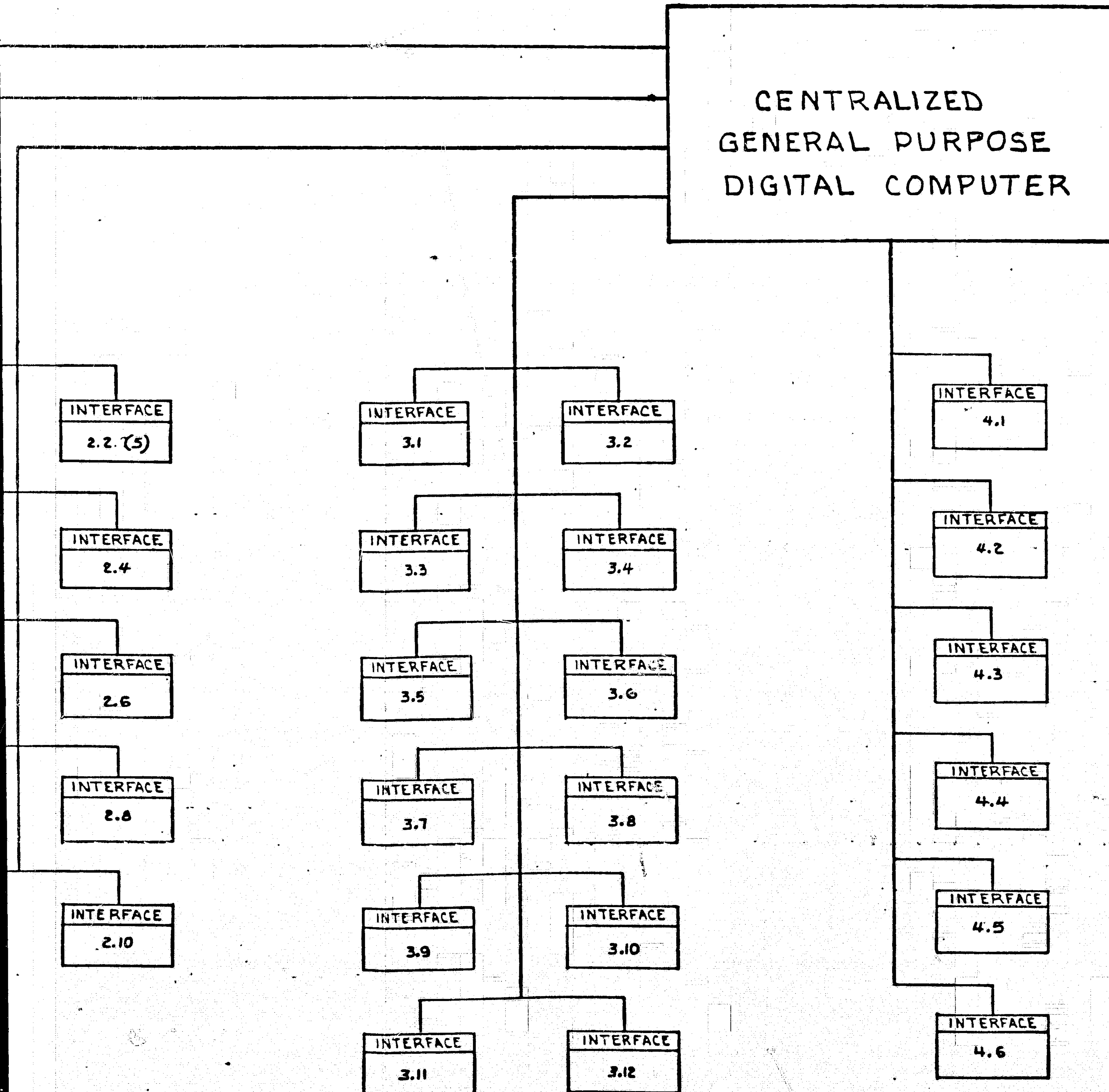
4.7.1.2 Option B. This option is a logical result of the above discussion of Option A. The control of the data transfer and data processing are still centralized, but the actual transfer of data is broken into several logical subgroupings, each connected to the central computer. Fig. 4.7-3 shows the grouping by subsystems as a representation of this option. Data lines or bus information rates are reduced by approximately a factor of ten (to 150 or 300 KHz from 1.5 or 3.0 MHz). Also, the number of interfaces on each line has been reduced, and therefore, the termination problems have been reduced. With the reduced speed, the high frequency reflections may be filtered, reducing the chances of bit errors due to reflections. The major problem here is twofold in that the computer interface hardware increases and all information must return to the central computer to be transferred to another subsystem, increasing the number of routines handled by the computer, thereby increasing the hardware requirements in the computer. The technical risks from the software problems are increased as a result of the increased routines and traffic.

4.7.1.3 Option C. In Options A and B, the data control and processing for the data management subsystem are in a centralized computer. This arrangement does not lend itself to having each subsystem or logical grouping of functions acting as an entity. Option C, as shown in Fig. 4.7-4 is a configuration that includes some of the desirable characteristics of Option B

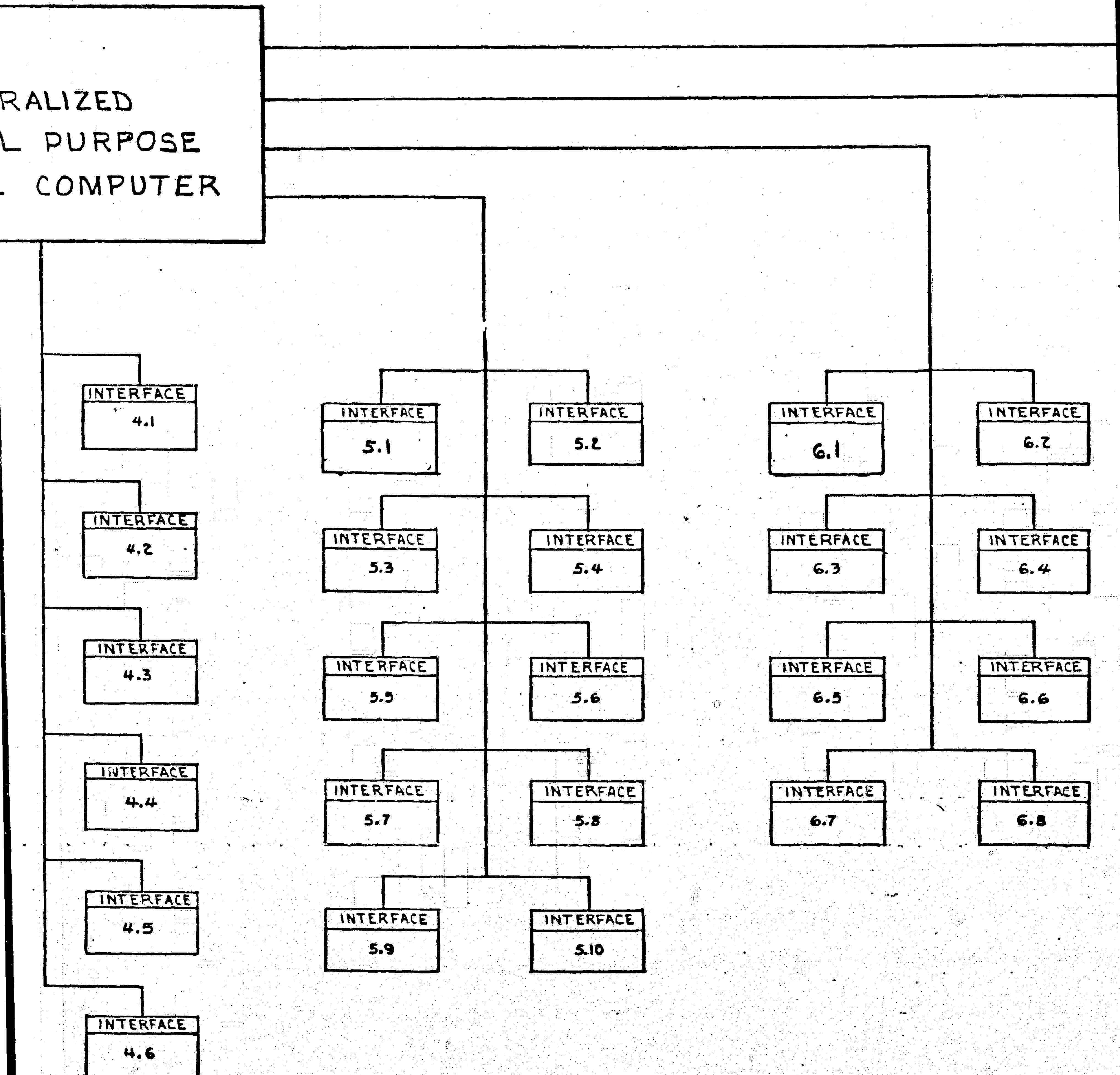
PRECEDING PAGE BLANK NOT FILMED.



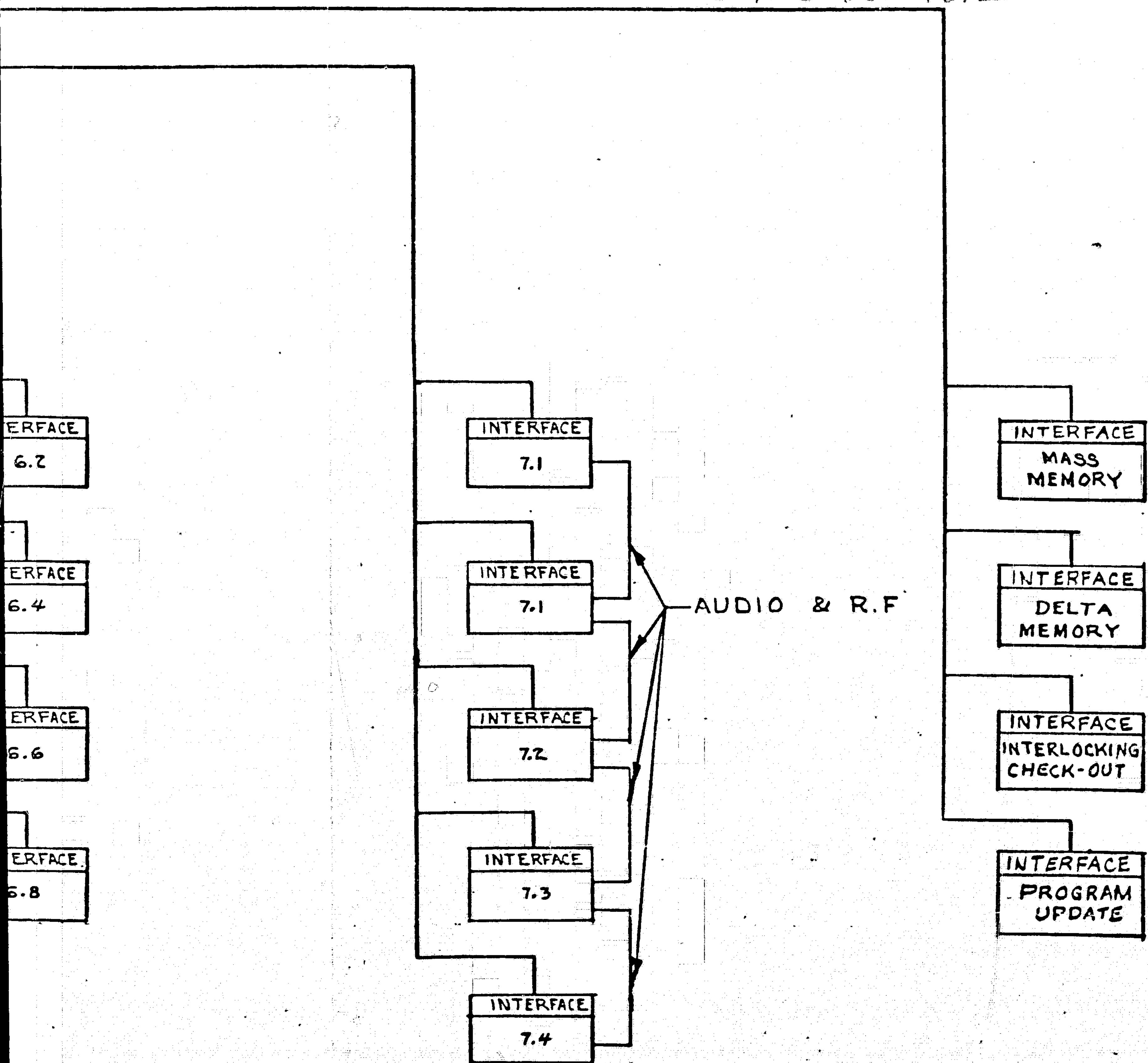
FOLDOUT FRAME 1



FOLDOUT FRAME 2

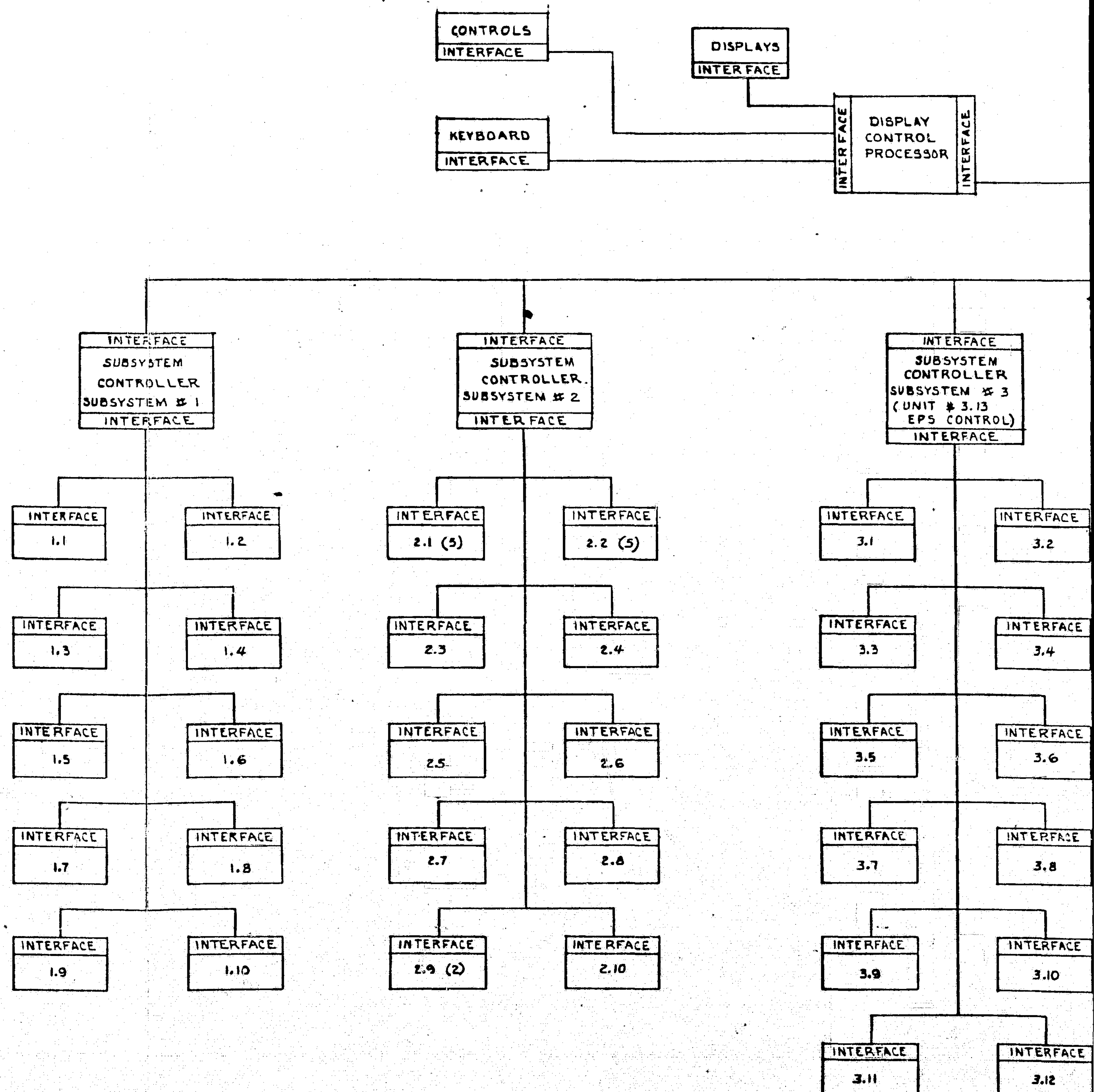


FOLDOUT FRAME 3

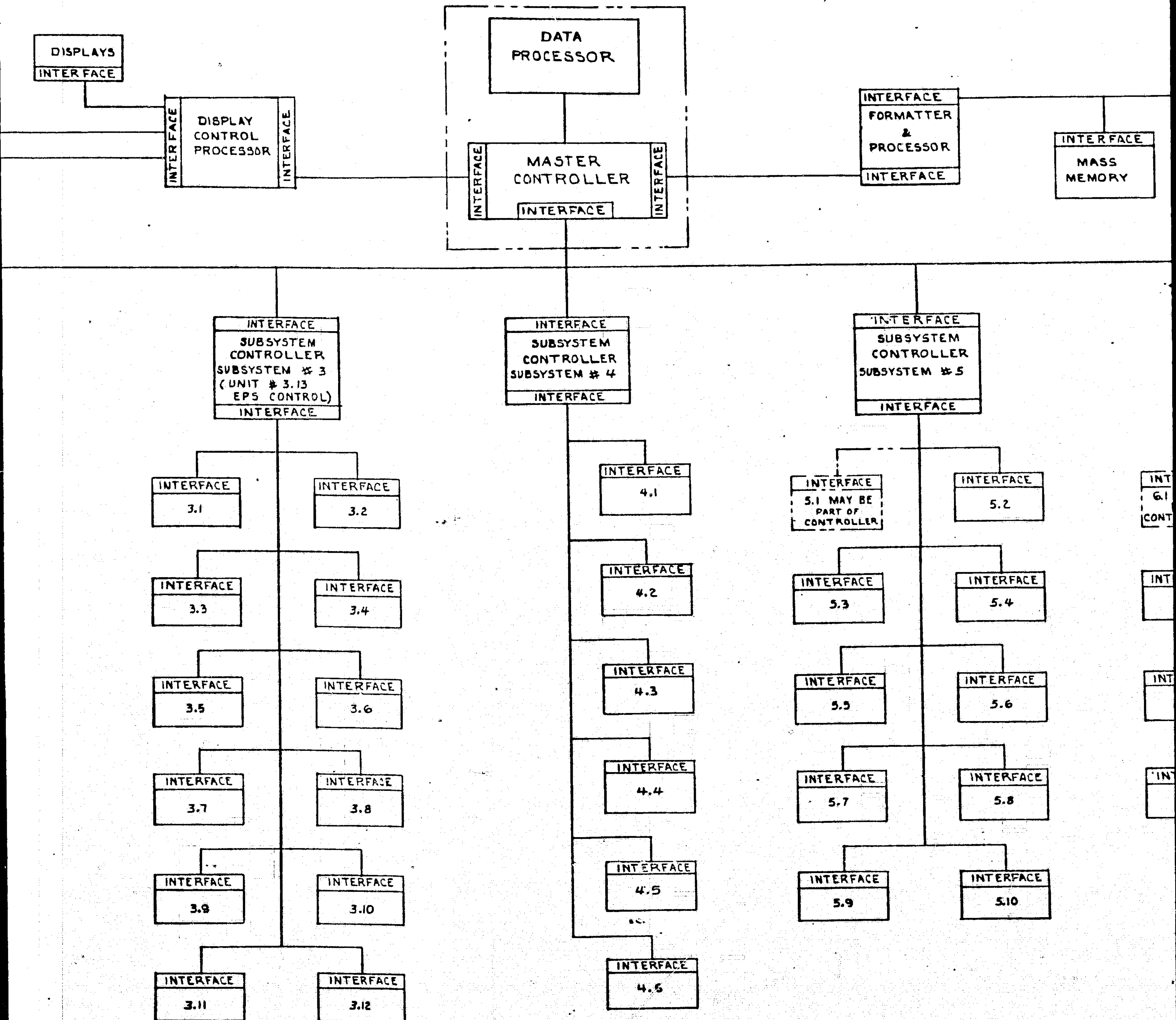


INTEGRATED AVIONICS
ALTERNATIVE 2, OPTION B
BLOCK DIAGRAM
FIGURE 4.7-3

PRECEDING PAGE BLANK NOT FILMED.

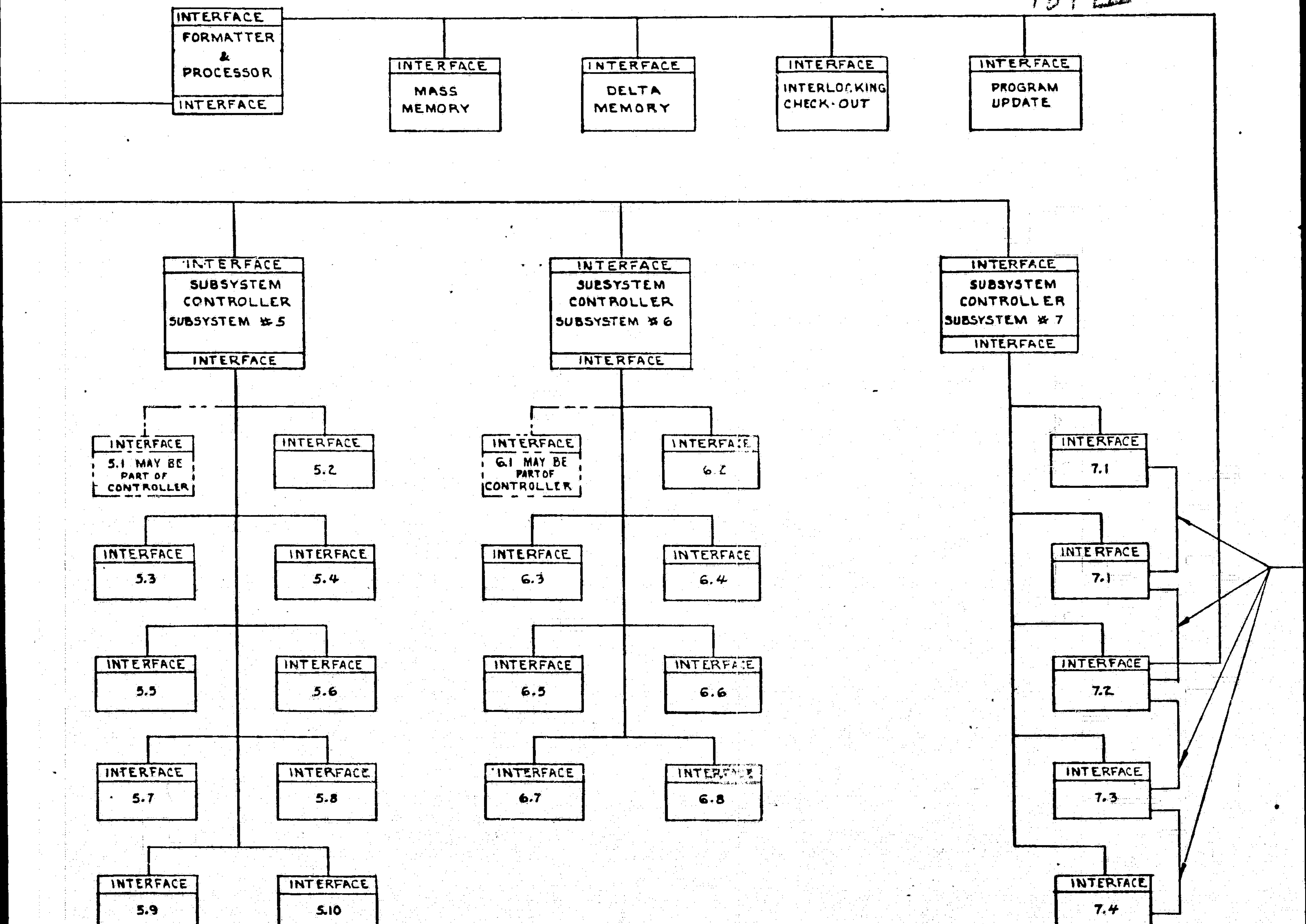


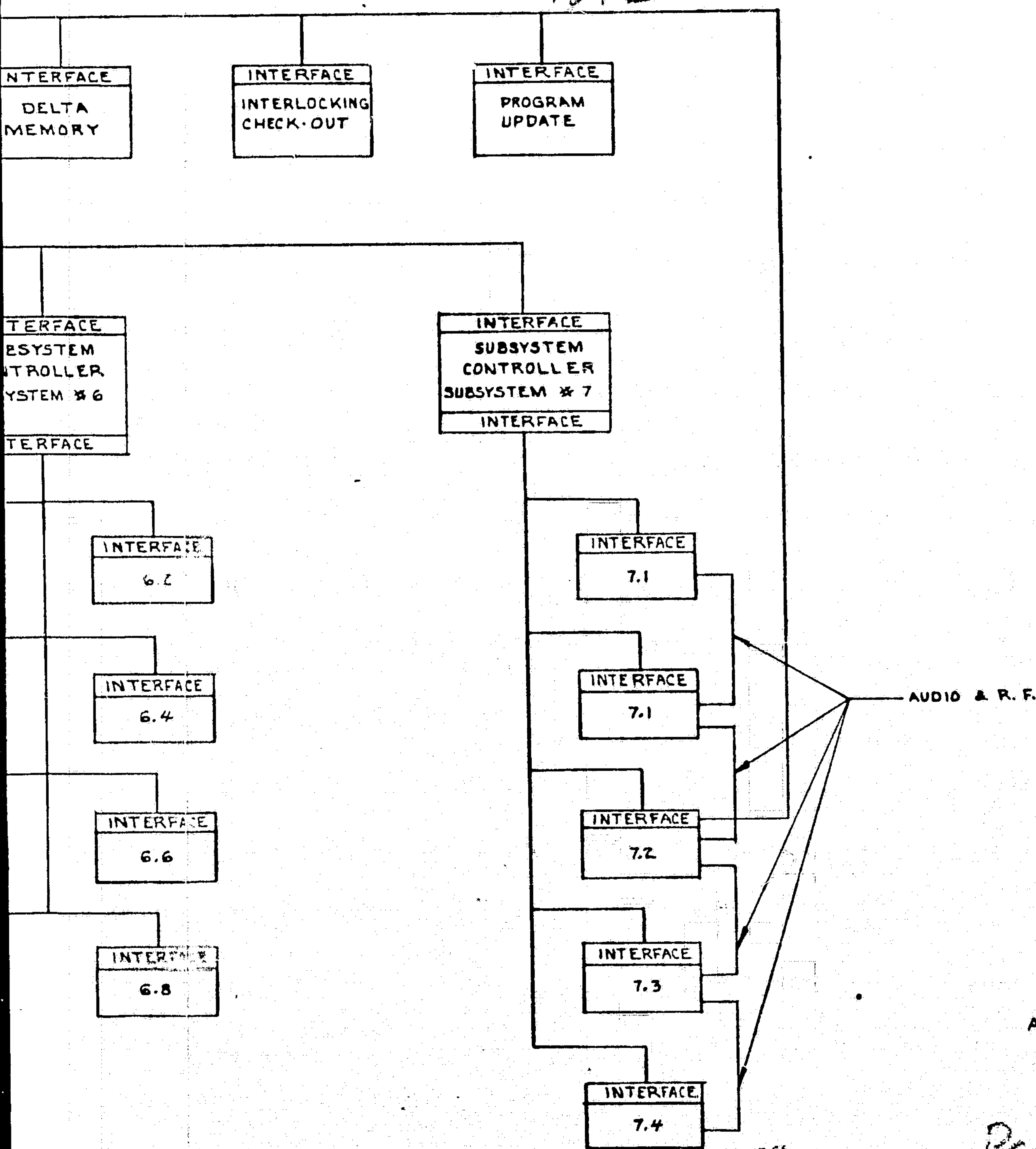
FOLDOUT FRAME 1



FOLDOUT FRAME 2

LMSC-A959837
Vol III





INTEGRATED AVIONICS
ALTERNATIVE 2, OPTION C
BLOCK DIAGRAM
FIGURE 4.7-4

but allows the grouping of functions by subsystems. This is representative of such a configuration. Studies of the total vehicle configuration show that it may be desirable to distribute some subsystem functions in a different groupings, both by function and physical locations. For example, the main engine propellants, main engine fuel control, and main engine gimbaling and positioning control could be a desirable grouping. This would cut sections out of subsystems 1.0, 2.0, and 5.0 and put them into a different grouping. The RCS, landing engines, landing aids, and guidance/navigational/vehicle control sensors are examples of other groupings. For the present, the basic subsystem configuration will be used as the example for Option C.

This option still has the central processor (or computer) and the standard interface circuit on each subfunction in the subsystem, but each subsystem now has a central controller (called a subsystem controller - SSC), which controls the traffic of data within the subsystem, controls the subsystem OBC/FI and the configuration control and sequencing, and interfaces the subsystem with the master controller in the central computer. The SSC allows the subsystem to operate at its "natural frequency" without disturbing, or being disturbed by, other vehicle subsystems, except on a master configuration and control setup command. The reporting to the central master control or transfer of data from subsystem to subsystem can take place without "tying up" the subsystem operation. The information bit rates within the subsystems range from approximately 10 KHz to 200 KHz, with the master controller/subsystem interface rates in the range of 500 KHz. The master controller/subsystem interface bus is a single line for this study but could be a multiple bus if reliability and safety requirements so dictate. This arrangement of a single bus simplifies the interface at the master controller, and the data rates are not excessive for the state-of-the-art. Of the three options, Option C presents the least technical risk since each subsystem is autonomous but allows the master control and configuration to have a major control in its operation within the total vehicle mission requirements.

4.7.2 Baseline Configuration

The baseline configuration for Alternative 2 in the IES Study will be Option C. The rationale for this selection is many faceted and will be discussed at this time.

Option C meets the requirements of having a common interface. The interface can be well defined, and details can be given to each subsystem for using this interface. (This has been done in several major programs already in operation and others now being designed.) The operational subsystem can be separated and operated as an entity; this includes the major parts of fault detection and isolation, abort warning, and configuration control and sequencing. The subsystem controller can be integrated with the control functions of the operational subsystem and save power and weight over a separate subsystem controller. The self-test and warning function will be a "hybrid" type function with the "box" having its own builtin test equipment (BITE) and reporting its condition to the subsystem controller, and the system check being performed by the subsystem controller. This information can be part of processing done at the subsystem controller and reported to the master controller when needed. As shown in the separate tradeoff study summarized in Appendix D, the combination of BITE internal to "boxes" (reporting to a central controller) and centralized system end-to-end check is better than the total centralized testing and checkout. The master controller will have the responsibility of total configuration control in conjunction with the man-machine interface. The operations support routine will be under the control of the central processor and will work in conjunction with the master controller configuration control. The master controller will inform each subsystem as to its particular mode, and the subsystem control will configure the subsystem for that particular phase or mode. Therefore, the configuration control and sequence function will be distributed throughout the subsystems, with an overall master at the central controller.

The interface with control and display (C/D) will be through the master controller. This will be compatible with the configuration control and

and sequencing function as discussed above. Some dedicated controls and displays will not enter the data management subsystem; these are not considered here because of the commonality of all alternatives. Including the controls of the crew station in the data management subsystem allows some of the storage and routines in the C/D electronics to be combined with the data processor storage and routines. This display parameter selector logic and memory and the priority interrupt subfunctions can be combined with the same subfunction operations in the data processor.

With the standardized interface, standard formats, buffering, and transfer rates could be used. Also, with the standard interface, there is flexibility for growth and changes as technological improvements are made over the life of the program. Also, the subsystem controllers could each use standardized modules along with the master controller and processor and other logic/storage functions throughout the vehicle systems. The flexibility then becomes a matter of software changes and having enough additional address capability for growth.

There are few if any performance compromises because of the IES Alternative 2, Option C. It may even help the end performance by overseeing the total vehicle systems operation early in the program and then making the requirements known to the subsystems early in their design cycle. This will help avoid incompatibility when the system integration phase is reached.

As a result of the distributed nature and autonomous operation of each subsystem, the effects of catastrophic failure of a simple component are less likely to cause major perturbation in the overall performance of the vehicle. The Alternative 2, Option C is not constrained by the 1972 electronic component technology. It will depend on the scheduled developments between today and 1972 of such things as:

- MOS/bipolar - LSI techniques, especially in the operating speeds (in the 1 to 5 MHz range) and quantity of circuit elements per chip
- Main memory techniques - in the area of plate-wire memories for bit packing densities and aging effects

- Scratch-pad memories - ROM semiconductor, in the area of bit packing and wiring techniques.

No major breakthroughs are needed in the data bus or data processing techniques to use this Alternative. The major effort is needed in the systems organization and management of this organization. New directional thinking in the software area could be of benefit, especially in coordination of the subsystem control within a total master control.

The weight and power increment summaries are shown in section 4.10 of this report. The major consideration for this alternative is a savings in cabling weight; this is due to the multiplexing/common data bus arrangement. The power increment is insignificant in the overall picture, as expected, in that no functions of the subsystems have been centrally located. The interface subsystem control functions are contained in federated systems but have been standardized in Alternative 2 and will be a managed and standardized function in the alternative.

Technical risks involved in this alternative are not necessarily increased, and they may be decreased as a result of forced management of the total vehicle operations. The subsystems are still autonomous with respect to their own functional operation; but as far as operation within the overall vehicle functional operation, they are under the control of the master configuration control and sequencer (master controller). This factor should lend itself to less technical risk.

Traffic studies were made from the test point and interface listings and summarized in section 4.5. The studies show that the sampling-rate requirements are well within today's state-of-the-art without the use of special packing or multiplexing techniques. The highest rates are in the guidance navigation/vehicle control subsystem (GN/VC). These are in the order of 2 to 3 KHz for each subsystem. The transfer of vehicle control information accounts for this sample-rate level. Without this transfer the sample rate for these subsystems would be approximately 10 times less. All other subsystems are almost 10 times less than these. If the communication subsystem were required to digitize voice for transmission, the sample rate for this particular function would be in the order of 8 KHz.

At the master controller interface, the data traffic problems are somewhat different. The data transfer exclusively between subsystems is minimal and mainly in the vehicle control area. The rest of the data transfer is between the subsystems and control/display-data management. Again, the vehicle control rates are similar to the rates discussed above in the subsystems. Transfer of data to the control/display and the data management control processor is at the same 2 to 3 KHz rate as mentioned in the subsystem discussion above. There is also some command transfer to the subsystem controller (to be added to the data sampling rates), but again this is slow in regard to the vehicle control sample rates. Combining the sample rates from all subsystems gives a bit rate of approximately 500 KHz without the use of any special techniques to reduce data rate handling requirements. The data management system operating rates are well within the state-of-the-art, and no special requirements are shown from the standpoint of data transfer and multiplexing.

Detail studies were made for the data processing requirements of the data management subsystem in the functional areas of OBC/FI, AW, OS, and CC/S; the results were summarized for Alternative 1. The same functional areas will be used in Alternative 2 with no increase or decrease in requirements. There may be a slight regrouping and redistribution of these functional areas, but they will remain essentially the same.

The interface control function is the area to be detailed in this section. This function includes the transfer of commands from control/display through the data management subsystem instead of hardwired as in Alternative 1. As shown in the interconnecting diagram of section 4.5, the number of controls to be multiplexed is 118 of the total of 167. Because of the criticality of the controls to the subsystem operation, the remaining 49 controls will be dedicated wires and not multiplexed.

The interface control function includes the transfer of commands and data from "box" to "box" and subsystem to subsystem. Again, the information in section 4.5 shows a total of 723 operational interfaces, with 195 of the interfaces between subsystems (as mentioned before, these are almost exclusively restricted to the GN/VC operations of the vehicle). A total of 841 new instructions must be included in the Data Processor over what was required in Alternative 1. In regard to the total requirement of the instructions needed as shown in Alternative 1 (150K), this is minor and is within the estimated accuracy of Alternative 1.

One other factor has to be considered beyond Alternative 1; that is, an additional instruction has to be included in the command word to each interface. The additional instructions tell the interface where to send the requested information. This can be included in the subsystem by adding Transmit-to-Address bits to the instruction word sent to every "box" from the subsystem controller. For the configuration of Option C this, will be only 4 bits, since there are no more than 15 "boxes" or 15 subsystems.

Therefore, the data processor requirements of Alternative 2 are well within the estimates of Alternative 1, and no further increase or decrease is needed to do the data processing functions within the data management of Alternative 2.

4.7.3 Functional Description

The functional description of IES Alternative 2, Option C, which has been selected as the configuration for Alternative 2, will start with the subsystem subfunctions. These subfunctions within the subsystem have been treated as representing "boxes" within the IES study and will be so treated in this discussion.

The subfunctions will continue to do their operations within the subsystem in Alternative 2. The subfunction will also have BITE to be able to check itself. Whenever information is needed to be transferred from or to the subfunction from other functional areas within the subsystem, it will become part of the data management subsystem-data acquisition and transfer function. The interface circuit will be contained with the subfunction "box" to format the data for transfer out or recognize the incoming information and format it for use within the subfunction.

The control of the information flow is des gned into the subsystem controller for each subsystem. Each information transfer point in the subsystem is given a "time slot" within the total frame of information to be transferred. Within the "time slot" are instructions that will tell what to do with the information, i.e., transfer to another "box" within subsystem, transfer to subsystem controller for status, OBC operations, or transfer to another subsystem. Also, there is an address instruction as to where the information is to go. All of the instruction routines and addresses are stored in the SSC memory. This memory (and in turn the subsystem) is sequenced in time at the "natural frequency" of the subsystem information points (test points, interface, and commands) as designed and programmed into the executive routine of the subsystem controller as determined by the original design of the subsystem. Also designed into the subfunction word or information transfer is the capability to indicate a priority interrupt, to the subsystem controller, that special routines are needed in a critical time cycle and all other routines will be recognized by the subsystem controller and acted upon accordingly.

The subsystem controller must also interface with the master controller to transfer information between subsystems and to the central control. Also, the subsystem controller will receive commands from the master controller for such things as data transfer, configuration control, etc. The interface with the master controller will be asychronous and therefore will require storage of the subsystem information to be transferred on the master controller interface. By doing this the information will be continuously available, on call from the master controller, without disturbing the subsystem operation.

The subsystem controller will have routines stored to do the subsystem end-to-end checkout. This is above and beyond the BITE in each "box" and supplements this function on the subsystem basis. Also, routines will be stored for the configuration control and sequencing for each subsystem, as discussed earlier.

The subsystem controller performs a significant role for the operational subsystem and in fact, may be absorbed into the subsystem's computer or controller/sequencer. As an example, the guidance, navigation, and vehicle control computers can absorb these subsystem controller functions without major design impact. This has been configured in this design. Usually, these computers have the multiplexing and digital interface already included in them and their memories have spare storage capability. The subsystem controller logic circuit operations are well within the capability of such computers. This same concept may be used in other subsystems, as mentioned in the Alternative 2 discussion. The configuration discussed here has used the GNC computers but has used separate subsystem controllers for the other subsystems.

The master controller-subsystem controller interface operation is very similar to the subsystem-subfunction interface. The number of information points is different, but the same basic functions are accomplished within the master controller, only on a "higher level." The master controller operates at the "natural frequency" of the overall vehicle systems requirements, as does the subsystem controller with the subsystem. The same "time slotting," memory, and logic routine methods can be used. Therefore, in this operation, the master controller becomes a "subsystem controller."

There are other functions that the master controller must perform. It must, interface with the control display subsystem. This interface function has become an integral part of the data management subsystem in Alternative 2. Only the major display formatters and drivers and control interface receivers and drivers are part of the control/display subsystem. This includes the detail formatting routines, refresh memory, internal self-check, etc. The actual control of the information to be displayed and the "call up" of routines is stored in the data processor of the data management subsystem. Since many of these routines are similar for automatic control and sequencing, a common storage can be used.

Another functional operation of the master controller is to transfer information to and from the mass storage, delta storage, program update, and interlocking checkout. These units are mainly used for program or data storage for playback at some later time. The information must be formatted, prepared, and time tagged for storage and playback. This is similar to formatting and preparing data for transfer over a communication link. Since the voice is not digitized in this configuration, the data to be transmitted over the RF link are also prepared in this "subsystem" grouping. This requires an extra interface with the communication subsystem. If the voice is digitized in the intercommunication subsystem, the sample rate is increased by approximately 100 to 1. The actual rates are not extremely high (approximately 500 KHz) nor difficult to handle with today's technology. Savings could be made in the premodulation hardware and the extra interface hardware. This was not done for Alternative 2, because the increase in subsystem data rates may cause increase in the error rates. Further detail tradeoffs should be performed, beyond the scope of this study, to completely answer this question.

In summary, the major impact of the Alternative 2 configuration on the vehicle is the multiplexing and management of data transfer between subsystems and subfunctions of the vehicle. A major savings in cable weight will be realized, but other equipment and functions are essentially the same. Also, the common interface hardware and system design will result in major savings in cost (this has been shown in studies done on present day operational programs).

4.8 IES ALTERNATIVE 3

This portion of the study addresses the problem of combining all of the electronic functions into a single integrated system. In this alternative, the integrated system embraces both the vehicle health monitoring functions and all vehicle subsystem operating functions. These functions encompass:

- Vehicle status and maintenance monitoring
- Abort warning
- Fault isolation
- Data acquisition and processing associated with vehicle subsystem functions
- Configuration and mode control of all vehicle subsystems
- Guidance, navigation, and flight control
- Interface control between control/displays and vehicle subsystems
- Display data processing

For this study, the sensors have been defined to include the signal conditioner, and the vehicle subsystem elements defined to include the driver amplifiers required for subsystem control.

The only limitation placed on the design study is that the results are to be based on the 1972 state of the art, which places a limitation on hardware availability rather than on system design philosophy.

In this configuration, all electronic processing is performed centrally by time sharing a centralized processor and is justified on two basic grounds:

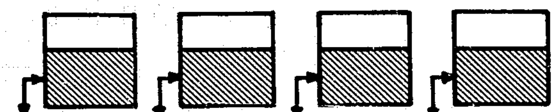
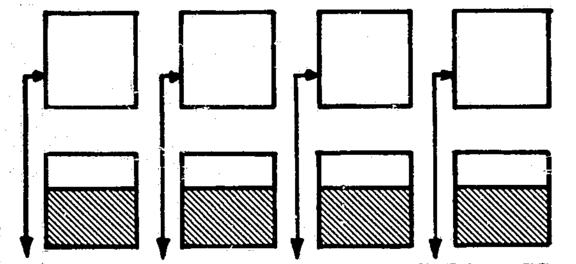
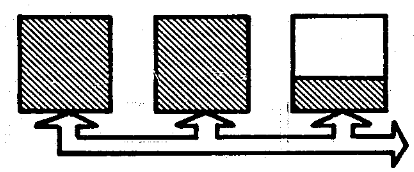
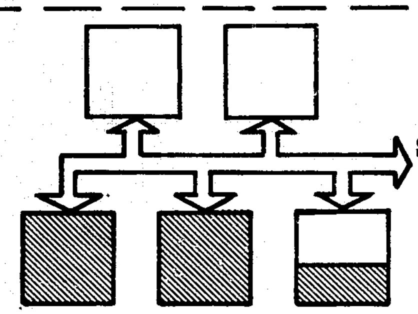
- A single processor can be efficiently time shared between all vehicle functions.
- Only a single set of redundant processing equipment is needed to provide the desired reliability.

Onboard computers for most space-vehicle applications operate well below their designed speed capacity. The guidance and navigation routines generally require less than 100,000 basic operations (such as ADD) per second. By adding housekeeping and other specific procedural functions, the operational load may be increased to as much as 200,000 per second, which would require add times of about 5 microseconds. Today's space computer is generally designed to perform additions in 2 to 3 μs. By 1972, add times of 1 and 2 μs will be standard through the use of parallel processing and memory cycle times of less than 1 μs. Thus, the resulting loading on the individual processor for virtually all applications is often less than 60 percent of maximum capability. Other tasks such as onboard checkout and crew display processing have relatively low demands that require possibly 20 to 30 percent of a computer duty cycle.

Figure 4.8-1 illustrates typical use factors of the arithmetic processing systems in most vehicle computers. The top diagram assumes a hypothetical nonredundant set of four such subsystem processors, each loaded at 60 percent duty cycle. The inefficiency of equipment use inherent in this approach is magnified when redundancy is incorporated as shown.

The centralized scheme illustrates the use of multiple processors located within one machine, which allows more efficient use of processing equipment. This is particularly applicable in a redundant configuration. The redundant configuration shown in Fig. 4.8-1 sustains any two failures (as opposed to the more constrained reliability statement which must be made when the redundancy is incorporated at a subsystem level). A full reliability analysis is required to fully demonstrate these aspects. Such a study was made under an LMSC study program (Ref. 1) dealing with complex unmanned vehicles in which it was determined that centralized processing can simultaneously reduce power and weight, while improving reliability.

PROCESSOR COMPARISONS

DESCRIPTION	PROCESSOR ARRANGEMENT	NUMBER OF UNITS	UTILIZATION FACTOR	ALLOWABLE MAJOR FAILURE
INDIVIDUAL SUBSYSTEM (SS) (NONREDUNDANT)		4	60%	NONE
INDIVIDUAL SUBSYSTEM (SS) (REDUNDANT)		8	30%	ONE IN EACH SUBSYSTEM
CENTRALIZED (NONREDUNDANT)		3	80%	NONE
CENTRALIZED (REDUNDANT)		5	48%	TWO

NOTES: 1. EACH BLOCK REPRESENTS EQUAL PROCESSING CAPABILITY

2. ☐ REPRESENTS CAPABILITY AND ▨ REPRESENTS REQUIRED DUTY CYCLE

Fig. 4.8.1 Processor Comparisons

Figure 4.8-2 shows the basic concept of centralized processing used in Alternative 3.

The central processor, here termed the data coordination system (DCS), is one part in the fully integrated configuration. The other vital part of the system is the network interconnecting the DCS with all other vehicle subsystems. Because of the centralization of control inherent in this concept, the subsystem interface problem reduces to basically two simple tasks: data retrieval and command execution. Both tasks require a minimum of remote electronics, which can be distributed throughout the vehicle and can be implemented in lightweight microcircuit modules. The combination of the distribution network, called the data distribution system (DDS), and the DCS results in a totally integrated electronic processing and control system. Within the basic definition of a fully integrated system, as discussed above, there are various ways to implement the DCS and the DDS. With the central processor (DCS), the options include:

- General purpose uniprocessor
- General purpose multiprocessor
- Multicomputer systems (only necessary to achieve redundancy)
- Cellular processor (distributed logic)
- Memory organized processors, including list-processors and associative-processors

The last two were eliminated on the basis of current technology. The remaining three can be discussed effectively only by considering the reliability required by the overall system. This is dealt with in more detail in section 4.8.3.

Optional organizations for the distribution network (DDS) include all concepts from the one-channel-per-wire technique to the fully multiplexed twin-wire system. The first technique results in an extensive weight penalty with respect to a multiplexed system. The other extreme, however, may not

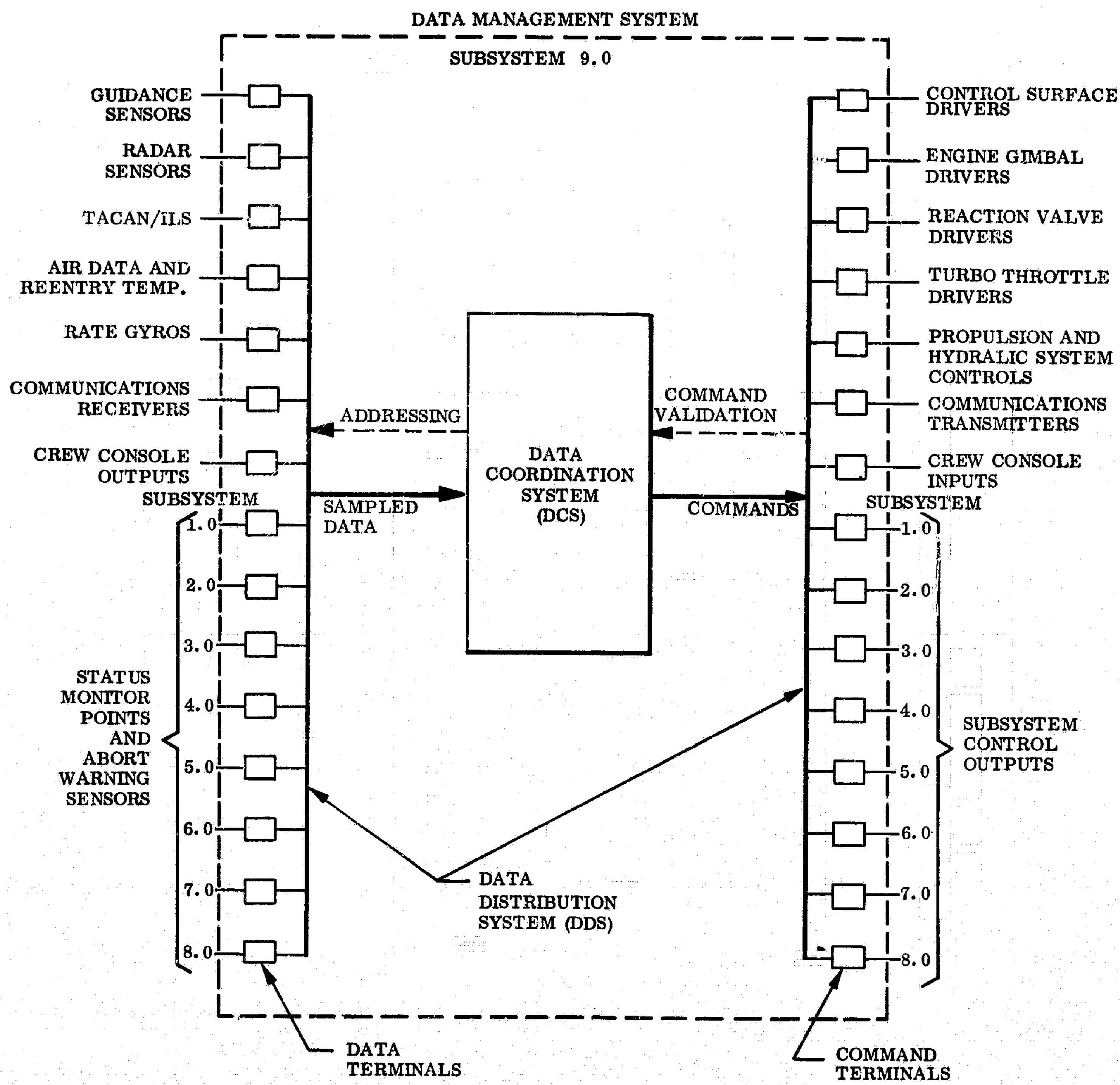


Fig. 4.8-2 Alternative 3, Centralized Processing

be optimum, when all aspects of weight and complexity are considered. The DDS discussed here attempts to minimize both weight and complexity by exploiting the advantages of modern cabling techniques and modern microcircuitry. The system is fully discussed in section 4.8.2.

4.8.1 Interpretation of the Orbiter Requirements

The requirements defined in section 4.5 are used as the basis for sizing both the DDS and the DCS. It is first necessary to make the following definitions concerning the information flowing into and out of the DCS. Information flowing along the DDS bus structure can be categorized into the following groups:

- Commands (information flowing out of the DCS)

  Discrete commands, each of which will result in a single on-off action

  Serial digital commands, each of which will result in a "proportional" action

- Data (information flowing into the DCS)

  Discrete data (output from two-state sensors)

  Serial digital data (output from proportional (analog) sensors)

A final categorization is necessary to complete the definitions.

- Active data - Refers to information, obtained from a subsystem element, that is required by the processor in order to complete the computation procedure associated with one of the subsystem control loops. Such computational procedures result in a command that is sent out to the appropriate subsystem control element. The combination of the DCS and the DDS forms a sampled data and control system, which is time shared among all onboard functions.

- Passive data - Refers to information, obtained from a subsystem element, that is required for monitoring the health of that subsystem. It does not form part of a closed loop control.

In general a serial digital data channel is either active or passive, and only rarely is one data channel used for both active and passive purposes. However, a discrete channel may well play an active role during short periods of time and a passive role for the remainder of the time.

The reason for the quoted definitions is twofold. First, it allows the DDS to be sized from purely a hardware standpoint; second, it allows an approximate sizing of the computer loading requirements in performing the tasks of malfunction detection, abort warning, and configuration control.

Table 4.8-1 deals with the first part of this task. This chart includes a 20 percent contingency and shows that there are just over 2000 data points to be sampled, and about 800 command points. In Alternative 3 the data management subsystem comprises the DCS and the DDS. The points allocated in Fig. 4.8-2 for the data management system are simply test points on the power supply lines associated with the DCS and the DDS. This subsystem provides its own test procedure internally by continually executing self-test routines.

The crew station subsystem (8.0) is treated similarly to other subsystems, even though this information does not strictly fall into the previously described categories. Manual crew controls are treated similarly to subsystem data points and are sampled rapidly; the appropriate commands are executed when necessary. Display data are treated similarly to commands issued by the DCS, by continuously updating the particular block of data required by the display.

Table 4.8-2, which is a subdivision by mission phase of active data channels, passive data channels, and discrete data channels, shows the proportions of the total number of channels that must be sampled in each phase to satisfy the complete checkout task and the complete vehicle control task. The detailed requirements for the DDS and the DCS are discussed in more depth in sections 4.8.2 and 4.8.3.

Table 4.8-1

COMMAND AND MONITOR POINT REQUIREMENTS

Subsystem	Command* Channels		Monitor* Channels	
	Discretes	Serial Digital	Discretes	Serial Digital
1.0 Structure/Mechanical	52	6	54	253
2.0 Propulsion	243	23	291	457
3.0 Electrical Power	86	10	66	183
4.0 Environmental Control	57	0	60	83
5.0 Guidance and Navigation	31	3	38	129
6.0 Vehicle Control	118	59	52	61
7.0 Communications	47	2	40	67
8.0 Crew Station	55	25	229	85
9.0 Data Management	-	-	11	11
Subtotals*	689	128	863	1329
TOTALS*	817		2192	

*20% contingency included

TABLE 4.8-

SUBSYSTEM MONITOR CHA

Phase →	Pre-Launch			Launch & Ascent			Orbit Insert.			Rendezvous			Dock		
	Discretes	Ser Dig Passive	Ser Dig Active	Discretes	Ser Dig Passive	Ser Dig Active	Discretes	Ser Dig Passive	Ser Dig Active	Discretes	Ser Dig Passive	Ser Dig Active	Discretes	Ser Dig Passive	
1.0 STRUCTURE/MECHANICAL	39	161	45	36	126	11	30	116	6	30	110	20	12	16	2
2.0 PROPULSION	243	258	13	238	290	5	226	290	5	226	290	5	184	190	
3.0 ELECTRICAL POWER	69	94	41	45	61	37	45	61	33	45	61	33	45	61	3
4.0 ENV. CONTROL	48	15	51	48	15	51	48	15	51	51	15	51	48	15	5
5.0 GUIDANCE/NAVIGATION	31	57	51	7	17	16	25	38	37	24	39	38	24	39	3
6.0 VEHICLE CONTROL	40	21	29	6	9	8	35	9	8	33	9	3	31	6	
7.0 COMMUNICATION	40	20	2	40	20	2	40	20	2	40	20	2	40	20	
8.0 CREW STATION	180		67	180		67	180		67	180		67	180		6
9.0 DATA MANAGEMENT		10			10			10			10			10	
TOTAL DISCRETES	690			600			629			639			563		
TOTAL S.D. PASSIVES		636			548			559			554			357	
TOTAL S.D. ACTIVES			299			197			158			219			21

FOLDOUT FRAME |

TABLE 4.8-2

IONITOR CHANNELS PER PHASE

Discretes	Ser Dig Passive	Ser Dig Active	Discretes	Ser Dig Passive	Ser Dig Active	Discretes	Ser Dig Passive	Ser Dig Active	Discretes	Ser Dig Passive	Ser Dig Active	Discretes	Ser Dig Passive	Ser Dig Active	Discretes	Ser Dig Passive	Ser Dig Active
Dock			Orbit Stay			Retro & Deorbit			Reentry			Subsonic Approach			Landing		
12	16	20	11	32	20	13	60	7	6	48	2	11	65	20	11	65	20
84	190	0	153	190	0	214	358	13	214	358	13	83	194	8	83	194	8
45	61	31	45	61	31	45	61	31	55	95	43	55	95	43	55	95	45
48	15	51	48	15	51	48	15	51	48	15	51	48	15	51	48	15	51
24	39	38	23	35	34	18	35	32	18	35	32	22	41	40	12	32	26
31	6	3	31	0	5	34	9	20	5	18	20	7	18	24	6	18	7
40	20	2	40	20	2	40	20	2	38	18	2	34	20	2	34	20	2
80		67	180		67	180		67	180		67	180		67	180		67
	10			10			10			10			10			10	
63			531			592			564			440			429		
	357			363			568			597			458			449	
		212			210			223			230			255			226

PRECEDING PAGE BLANK NOT FILMED.

4.8.2 Sizing the Data Distribution System

The first task is to decide how to distribute the data retrieval and command system throughout the Space Shuttle.

Figure 4.8-3 shows the distribution of monitoring and command channel locations throughout the vehicle (corresponding to the subsystem equipment distribution shown in Fig. 4.8-6). Two heavy density areas exist, one in the general locale of the crew station, and the other in the aft end of the vehicle, in the general area of the engines and the control surfaces.

4.8.2.1 Defining the DDS bus structure. The following points were considered in selecting the most suitable bus structure:

- Data rates
- Bus driving power requirements
- Subsystem distribution throught the vehicle

Data Rates. Figure 4.8-4 shows a data rate profile derived from the requirements analysis. The test-point rate profile for these configurations can be almost directly equated with the profile expected in Alternative 3. The average for this group is 3.2 samples per second. With a reasonable contingency, an average passive data sampling rate of 5 samples per second can be assumed.

Although the active data for Alternative 3 are derived directly from sensors and used in closed loop control functions, in Alternatives 1 and 2 this is not true, as the control functions remain at a subsystem level. Thus, it was felt necessary to increase the sampling rate of active data channels somewhat over the average rate defined for interface control (7.1 samples per second); 25 samples per second was selected as being an adequate average format rate for closing all vehicle control loops. Loops with higher speed than this would either be supermultiplexed within this sampling format or would remain local within the subsystem equipment.

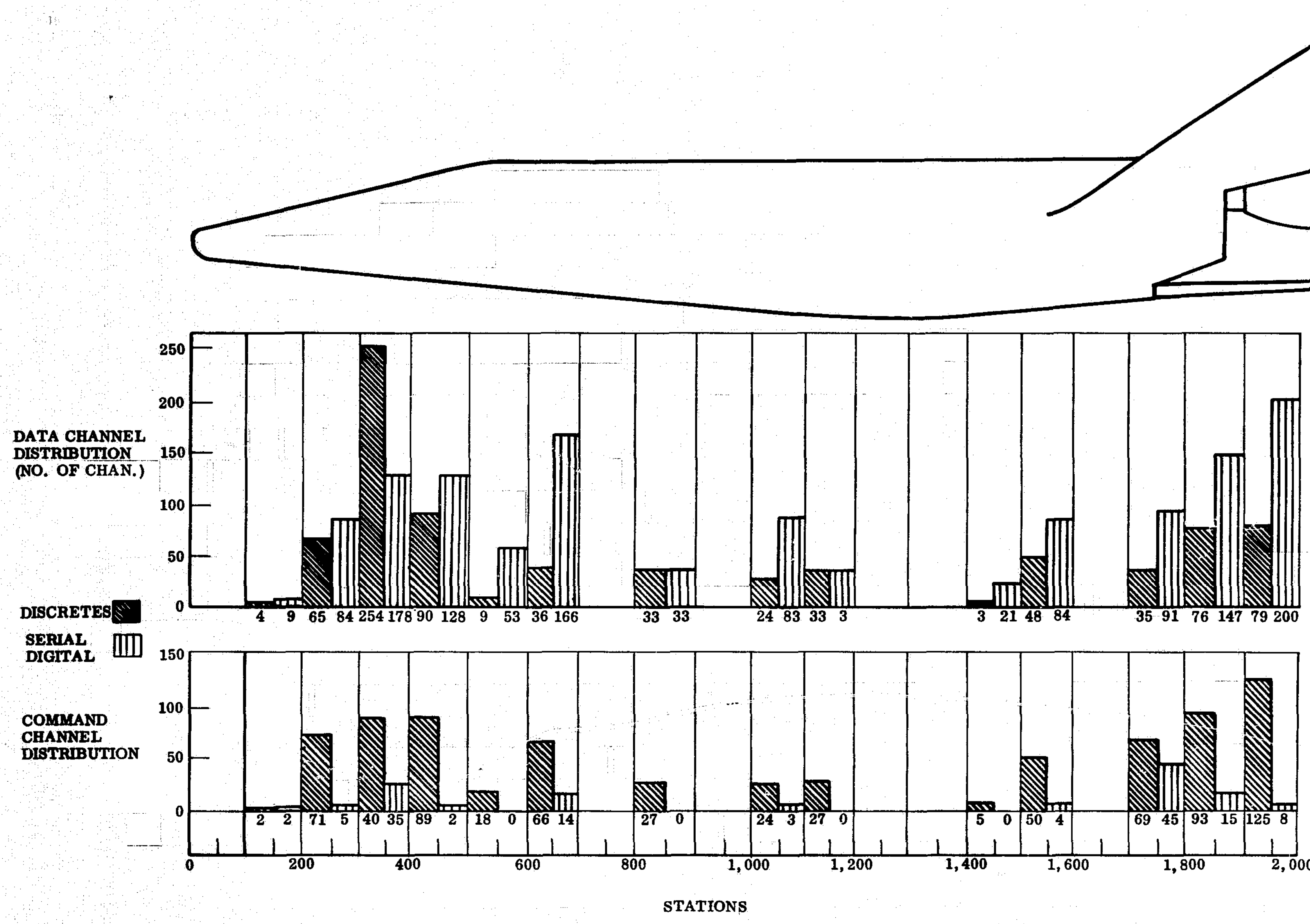


Fig. 4.8-3 Vehicle Distribution of Data and Command Channels

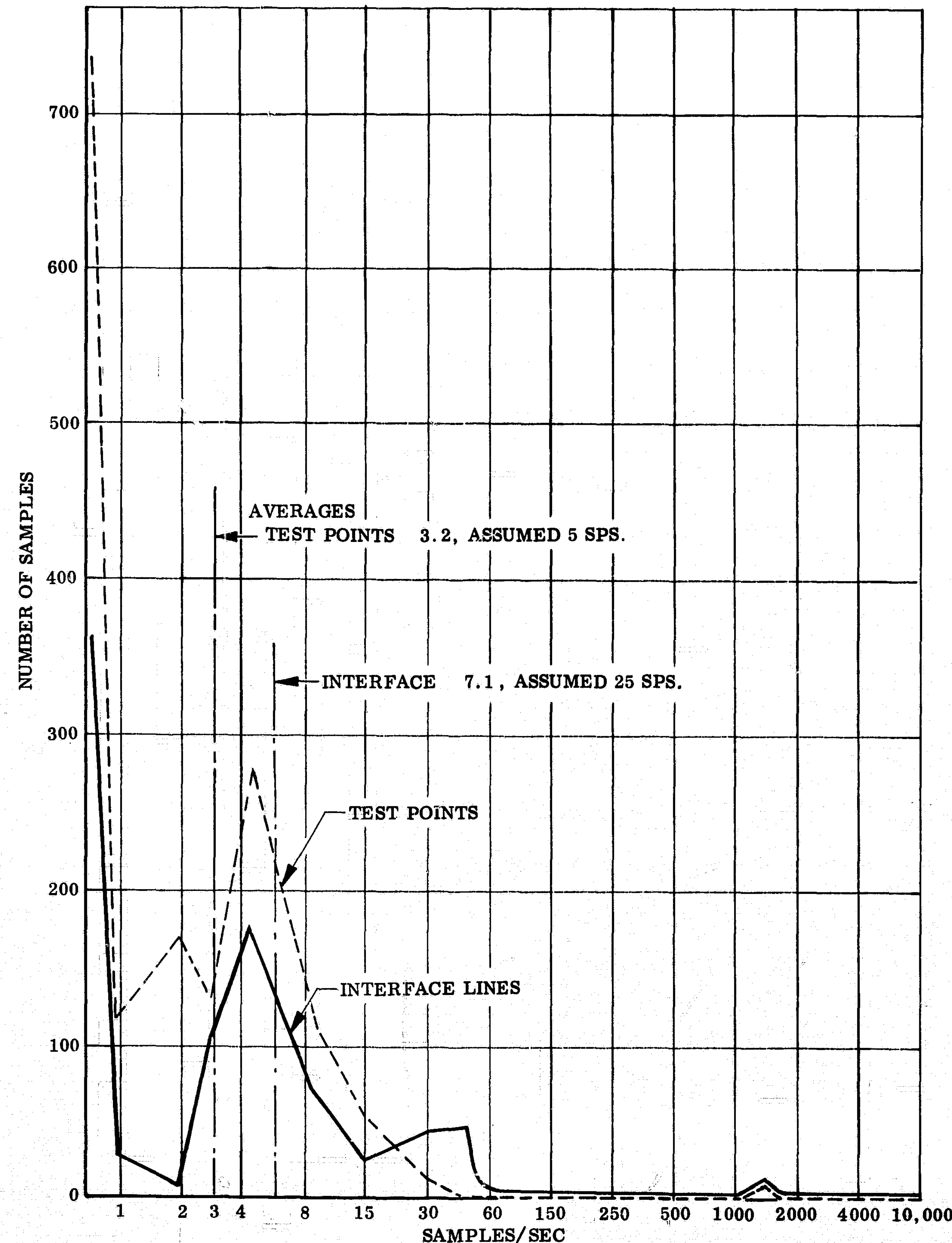


Fig. 4.8-4 Data Rate Profile

These rates must be translated into an overall data transmission requirement; first, however, it is necessary that assumptions be made about the method of data transmitted. It is assumed that a 10-bit data word provides sufficient accuracy for most analog measurements; thus a 12-bit word is sufficient to encompass data, sign, and parity. The occasional requirements for greater accuracy than this can be performed by allocating two 12-bit words to such channels. It is further assumed that each data channel and command channel must have a unique address, which is randomly accessible by the Data Coordination System, to satisfy the multifunction sampled data and control system requirements. The cumulative total, from Table 4.8-1, is 3009 channels, again requiring a maximum of 12 bits of address for each channel; this assumes a single bus structure for the whole system. Thus, every data channel must be addressed with a 12-bit word and will respond with a second 12-bit word. Effectively, each analog channel requires 24 bits.

Discretes can usually be handled in blocks, and each 12-bit word can carry 10 discrete information channels. With the same basic 24-bit structure, 12-bit address would again be needed for each block of 10 discretes. The same figures generally hold true for command channels, except that the 12-bit response word now takes the form of a command validation or "echo-check". A further assumption is that, in order to minimize cabling weight, all data transmission would be in serial format.

The most active phase of the mission (see Table 4.8-2), which is the prelaunch phase, has a peak sampling loading of 690 discrete channels, 636 passive analogs, and 299 active analogs. A worst case command loading would be to assign one command channel for each active analog channel.

The total loading on the bus can be determined from Table 4.8-3 which shows a word rate 24K words/sec and the bit rate = 576 K bits/sec.

Table 4.8-3

DDS BUS LOADING

Channels	Number of Words	Sampling Rate	Bus Loading Words/Second
690 discretes	69	5 Hz	345
636 passive analogs	636	5 Hz	3,180
299 active analogs	299	25 Hz	7,475
299 commands	299	25 Hz	7,575

Total 18,475 words/sec

This represents a worst-case in data transmission rate along the DDS bus structure; nevertheless, it is indicative of the types of rates expected. For the sake of circuit simplicity, the remote multiplexers and decoders should ideally work at the bus transfer rate. One result of a 500 K bit/sec bus data rate is that moderately high-speed circuitry must be used. However, if the bus is partitioned into two or more groups, the operating rate becomes low enough to use MOS technology, with its inherent power and weight advantages.

Bus Driving Power Requirements. The maximum length of cable from the DCS to the most remote point could possibly be 250 feet. Such a cable length requires a significant amount of driving power unless it is properly terminated. Figure 4.8-3 shows that approximately half the channels are in the remote areas and that the other half are in the forward end of the vehicle close to the DCS, which is at station 500.

To conserve overall driving power, it is necessary to consider partitioning the bus into at least two sections, a centralized section and a distributed section.

Subsystem Distribution Throughout Vehicle. A final aspect that was considered was the distribution of the equipment at each subsystem throughout the vehicle. Figure 4.8-5 illustrates this distribution. Subsystems 1, 2, and 6 are distributed throughout the entire length of the vehicle, whereas the remaining subsystems are predominantly localized around the crew station area.

Subsystems 1, 2, and 6 represent the prime subsystems dealing with vehicle contol; the remainder have a relatively low significance in exercising primary control over the vehicle. This distribution suggests a possible bus-positioning approach.

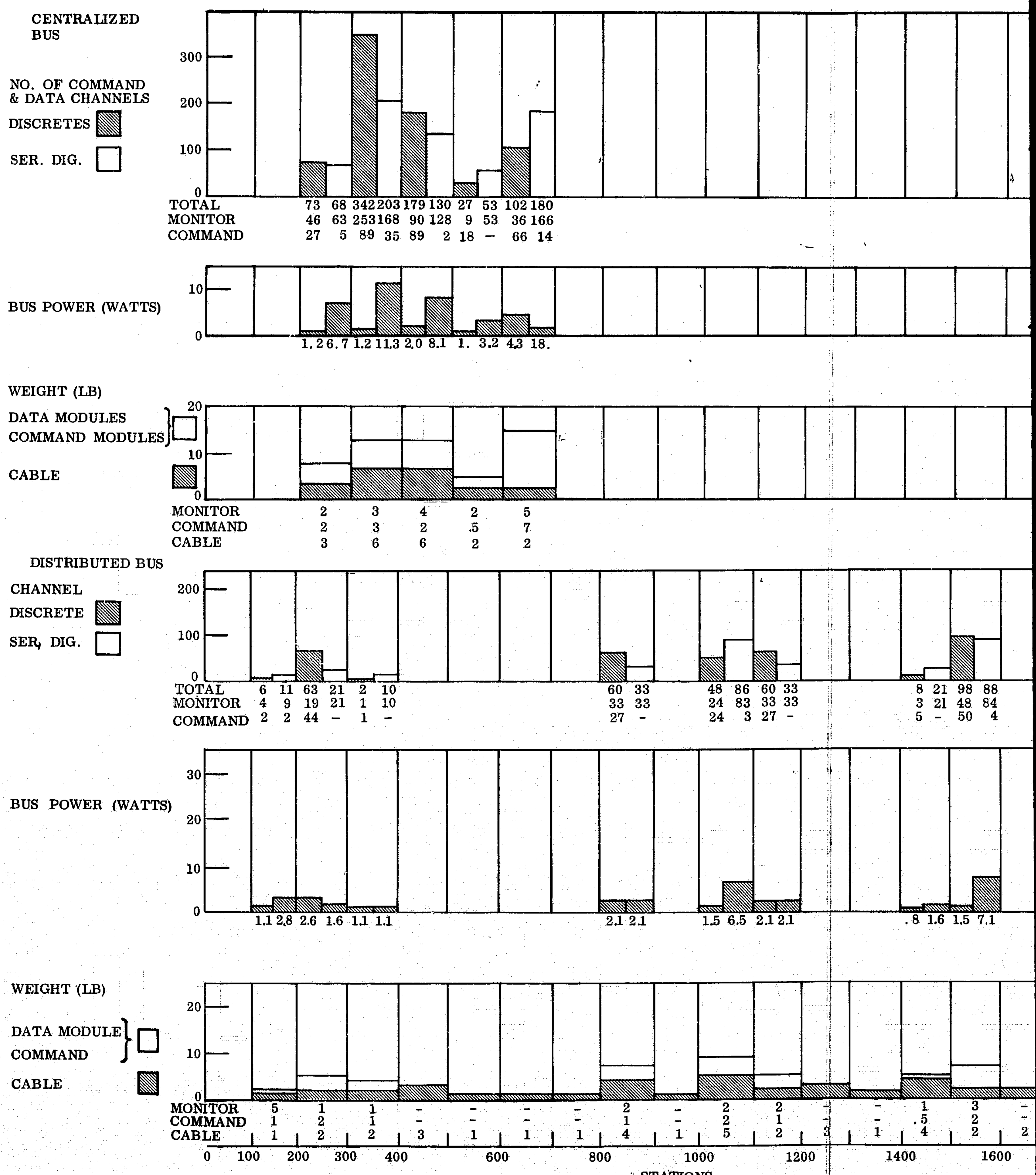


Fig. 4.8-5 IES Alt

FOLDOUT FRAME |

FOLDO

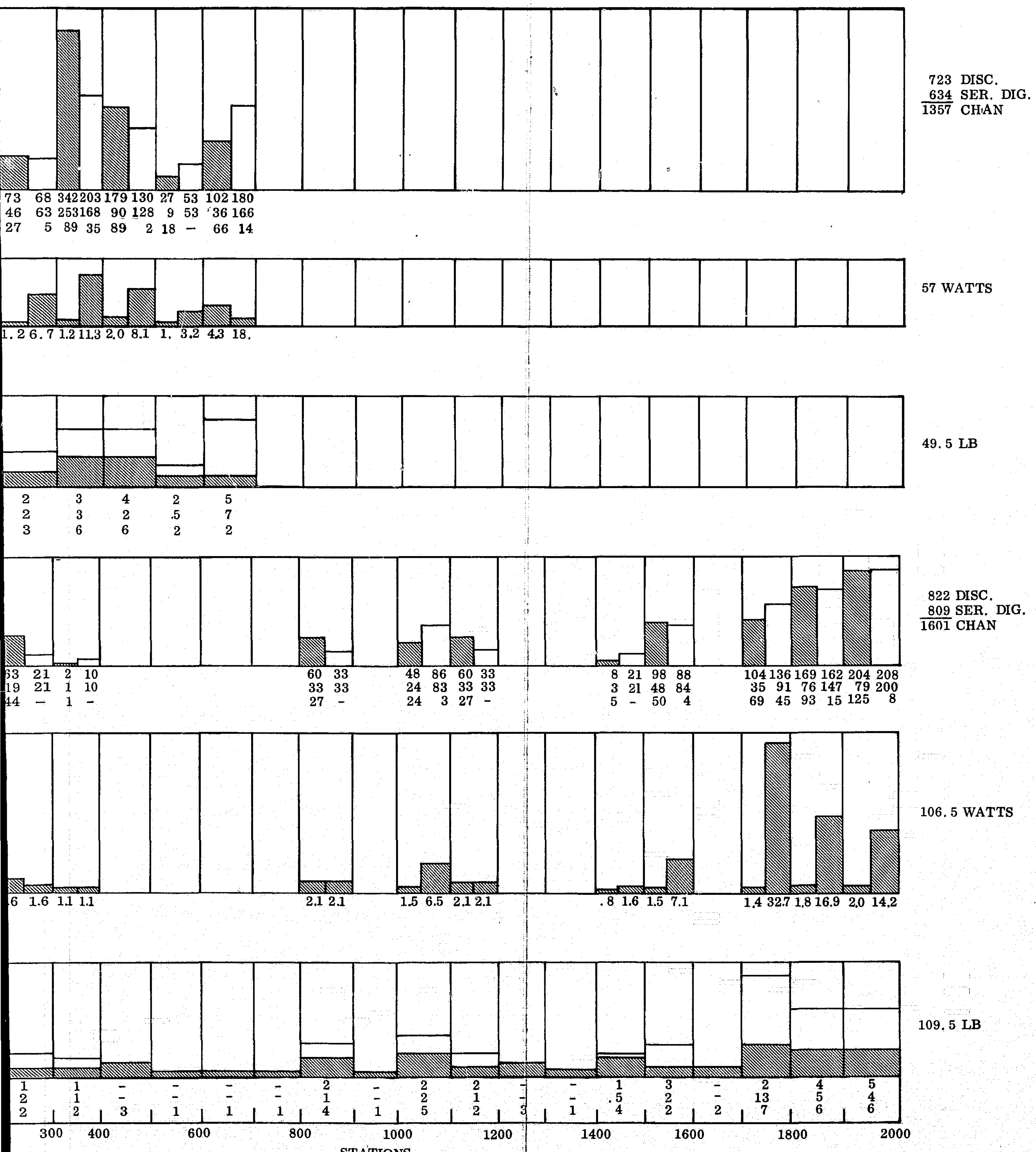


Fig. 4.8-5 IES Alternative 3 Block Diagram

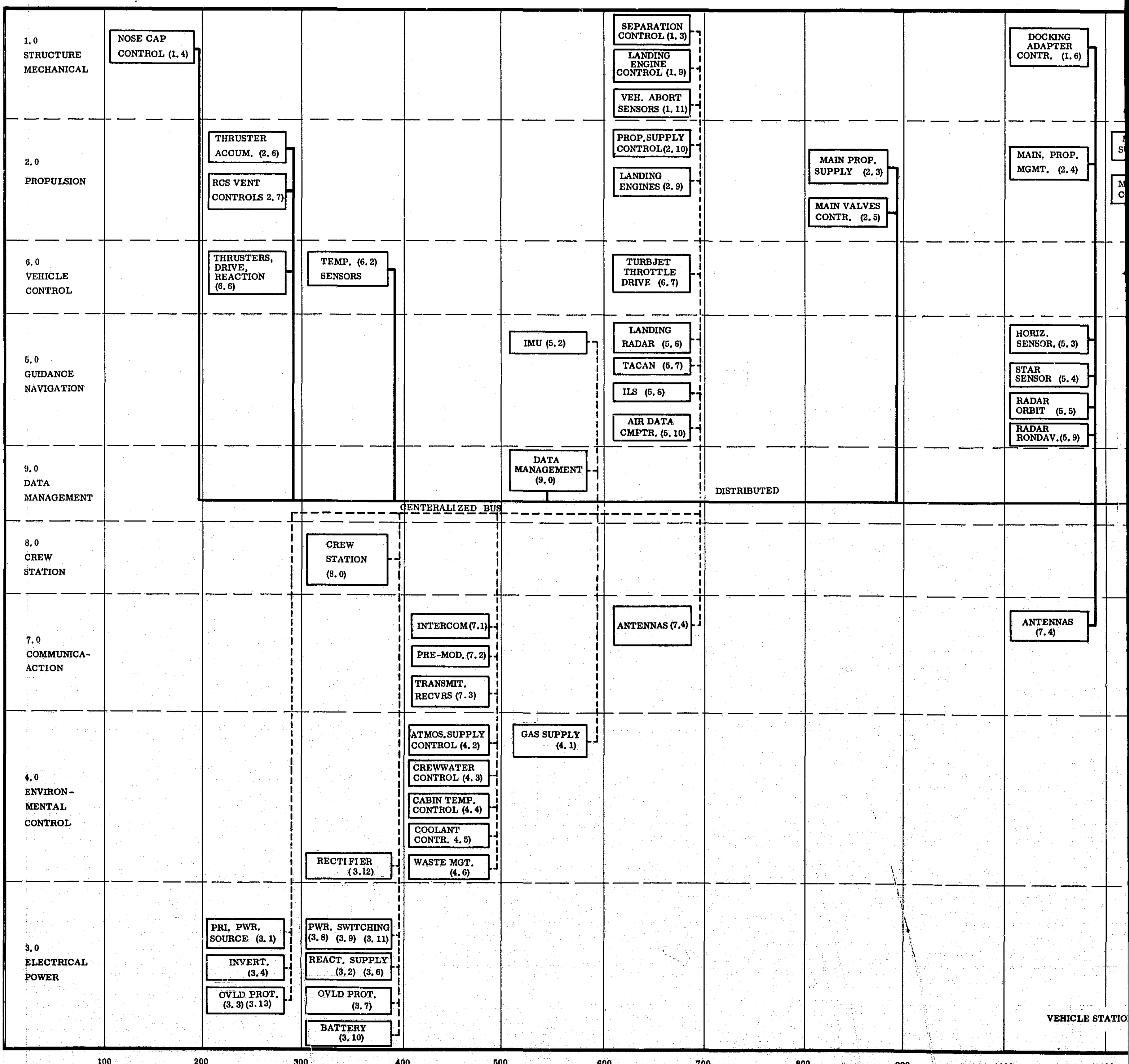
4.8.2.2 The Twin Bus Approach. On the basis of the previous discussion, it was decided to provide two buses, as shown in Fig. 4.8-5. The "centralized bus" as a maximum length of about 85 feet and is predominatly concerned with collecting data from the crew associated subsystems crew station, environmental control, and communications, and the electrical power subsystem 4.0. It also partially alleviates the loading of the second bus by interfacing with other subsystem equipment in the immediate vicinity of the DCS. Figure 4.8-6 shows the weight and power profile of each bus throughout the vehicle.

The "distributed" bus has a maximum length of about 250 feet and is mostly concerned with transferring prime vehicle control information between the DCS and the hydraulics, mechanical, and propulsion systems in the aft end of the vehicle. Because of the critical nature of the information travelling on this bus, extensive command protection would be used, as described in section 4.8.2.3.

It was further decided to divide both buses into a "command" bus and a "data" bus, primarily because of the difference between the data rates and the command rate and the necessity for effective transient protection on all commands. Section 4.8.2.3 describes the hardware design precautions that have been included to minimize the RFI, EMI, and noise transient problem to prevent command failures. A further failure mechanism is the incorrect interpretation of addresses. The data retrieval rates are typically in the order of 5 to 10 thousand words a second; the maximum command rate will probably be considerably less than this. Thus, if the command and data buses are separated, the chance of incorrect addressing is also considerably reduced.

Partitioning each bus into two main sections reduces the bit rate on any one bus to less than 200 K bits/sec. It is therefore practical to consider MOS circuitry for all DDS electronic modules; this allows extensive weight, power, and cost savings and a potentially higher reliability than bipolar circuitry permits.

PRECEDING PAGE BLANK NOT FILMED.



FOLDOUT FRAME 1

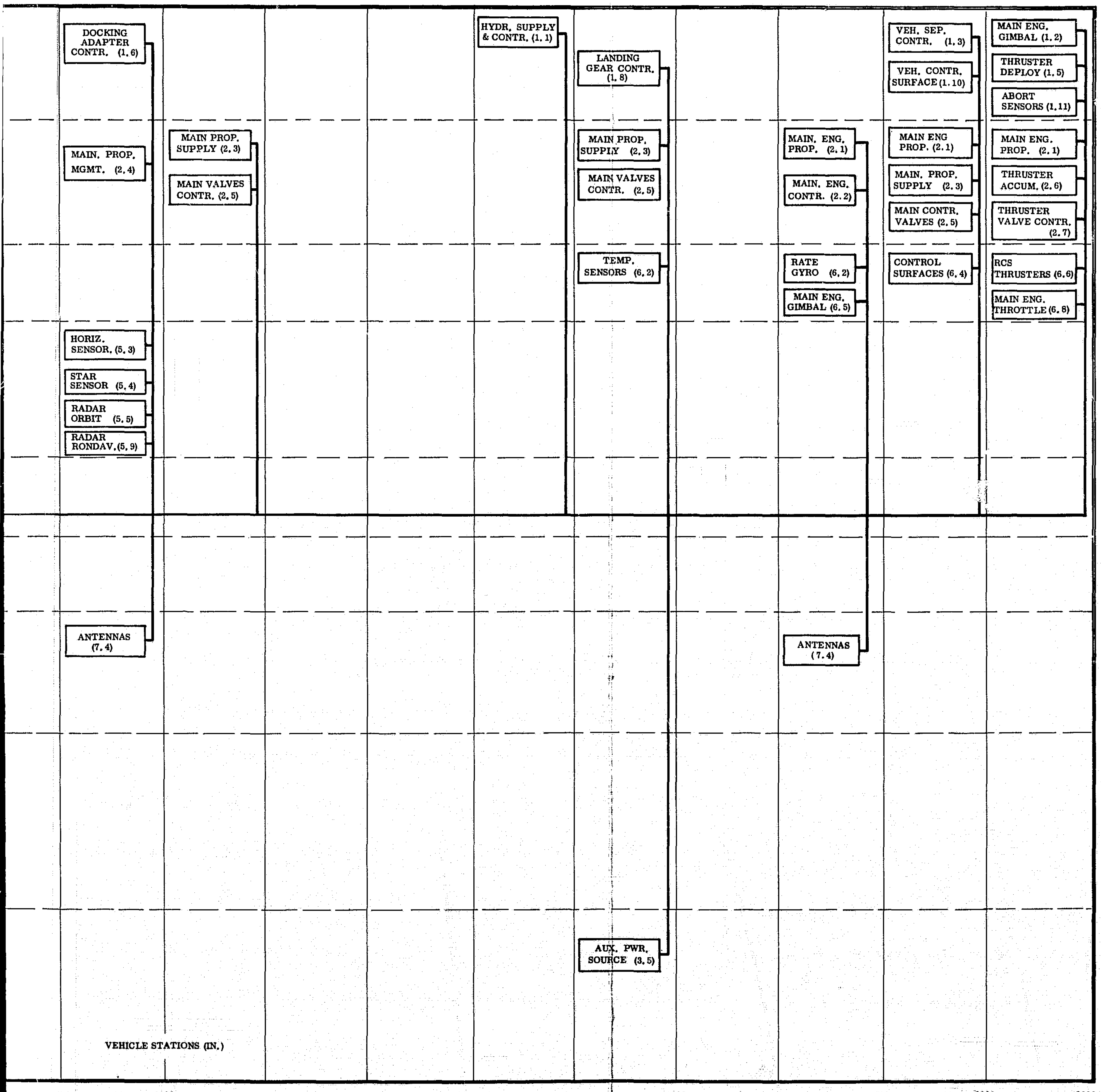


Fig. 4.8-6 Centralized and Distributed Bus Structure

FOLDOUT FRAME 2

PRECEDING PAGE BLANK NOT FILMED.

4.8.2.3 Data Distribution System Electronics. This section presents the standardized modular approach that will be within the state-of-the-art by 1972.

The basic module is called the data terminal which, simply described, is a remote-addressable, random-access, 16-channel multiplexer. The multiplexer is a microcircuit module. The modules can be used individually or grouped to provide additional localized channels. They can be arranged in a group of four to be compatible with the 64 channels of Alternate 1. Addressable 16-channel miltiplexers are currently available and should have a proven reliability by 1972.

The data terminal receives and decodes the serial addresses sent by the DCS and routes the appropriate analog data into a local A/D converter. The data are digitized and returned to the DCS in a serial word format.

In a slightly different form the same module, called a command terminal, is used to decode and execute up to 16 discrete on-off commands. Alternately, serial digital commands can be sent from the DCS, through the command terminal, to a D/A converter to effect the proportional control required by mechanical control elements such as elevons.

The command terminal (and, in fact, the data terminal) employs a validation procedure that prevents noise or EMI transients from inadvertently operating control devices. In this procedure the address received by the command terminal is parity-tested and loaded into the appropriate module address decoder. The address is then returned along an echo-check line to the DCS, where it is compared against the address previously sent. If correct, and EXECUTE condition is relayed to the appropriate command terminal, where it is logically "anded" with the channel address to execute the command.

A further degree of protection can be offered by using a "code-protection" module. In this system the command word includes a unique 10-bit code, which is sent through a command terminal to the "code-protection" module, wherein the code is compared against a code stored in a hard-wired register. The two codes must correspond exactly, before the command can be executed. Such a technique completely eliminates random noise firings and is used on all the highly critical control functions associated with the main engines and the main propellant tanks.

One method of using the "code-protection" technique would be to interconnect two command terminal modules on the main distribution bus with two modules mounted on a redundant bus, through an "interlock" module. This combines the use of serial-parallel redundancy with a fully interlocked system to effect an absolutely fail-operational remote control system.

4.8.2.4 Summary of the Data Distribution System. The weight, power, and size summary of the data distribution system of IES Alternate 3 for an orbiter vehicle is contained in Table 4.8-5 of Section 4.8.4.

4.8.3 Sizing the Data Coordination System

As indicated in section 4.8.1, the central processing part of this configuration can be considered as a high-speed digital computer. However, considering the critical importance of this unit, special attention must be given to the design in order to achieve the necessary reliability.

One solution to improving the reliability of a CP computer is to stack one or more redundant computers with the prime computer. These machines can either operate in standby (passive) redundancy, in which case each requires its own self-checking capability, or they can operate in parallel (active) redundancy and incorporate a voter system to determine the correct computational results. The standby system conservs power but requires an extensive "warm-up" period to allow the machine to establish the appropriate program section in order to take over with minimum impact on the operation of the vehicle. The parallel system is expensive in terms of power, but has the advantage of immediate takeover. The many problems inherent in such a multicomputer concept can be alleviated to a degree by use of a special-purpose executive control system that manages the operation of multiple general-purpose processors. Such arrangements are more properly termed multiprocessors, of the master-slave variety, and are being extensively investigated (Ref. 2). Another type of multiprocessor is one in which identical general-purpose processing modules are connected to form a "bank" of processors. Any processor can take over any task that is next on the list, or can remain in standby until extra processing capability is required. The executive control "floats" among the processors and is not dedicated to any one as in the master-slave concept. This technique is highly flexible and exhibits the least dramatic failure characteristics of any configuration. However, it invariably costs more in software.

This section discusses the nonredundant sizing of the DCS and is not involved in multiprocessing concepts; nevertheless, the "floating executive" concept is preferred as the most optimal form of multiprocessing to meet the diverse functional requirements and very high reliability of the orbiter vehicle, as discussed in section 4.11.

4.8.3.1 Computational requirements of DCS. Table 4.8-4 shows the estimated computational loading, by phase, for a typical mission. Each computational function is discussed below.

Operational Support. This function is primarily associated with mission planning operations--either before launch or during flight--that are required for long-term planning. Operations support embraces any of the programs that are required for the real-time operation of the vehicle. The crew, who will actively participate in the operations support procedure, are involved in this task. The computational loading figures are intended to represent peak loading estimates for each phase. The maximum figure of 50,000 operations per second is shown in the prelaunch and in the orbit stay period and would undoubtedly occur during the closing countdown stages before ascent to reentry. An extensive memory allocation is made for this function to allow for the diversity of mission plans that would have to be stored onboard. Only small sections of this memory need be active at any one time.

Malfunction Detection. This is the major onboard checkout program and is primarily a limit testing procedure on all testpoints. From a knowledge of the MADAR computer loading, it was determined that the average computer loading is 10 operations/test point. This routine is repeated at 5 Hz. The cumulative totals of the discretes plus the passive serial-digital channels were taken from Table 4.8-2; these were then multiplied by the loading figures above to derive the total computer loading. The memory allocation was again estimated on the basis of knowledge gained from the MADAR system.

Abort Warning. This abort warning was treated similarly to malfunction detection, except that the numbers of test points allocated to this

Table 4.8-4
CENTRAL PROCESS REQUIR

		Required Processor Operat Instructions/Sec (Thousands)					
Subsystem	Function	Prelaunch	Ascent	Orbit	Rendezvous	Docking	Orbit Stay
All	Operations Support	50	10	20	30	10	50
1 thru 8	Malfunction Detection	77	58	58	58	47	47
1 thru 8	Abort Warning	39	29	29	29	24	24
1 thru 8	Subsystem Control	103	83	87	90	58	63
6 & 7	Prelaunch Alignment, etc.	50					
6 & 7	Ascent, Boost Coast, and Injection		75				
6 & 7	On-orbit Transfer Injection			53			
6 & 7	Rendezvous Terminal Mode & Docking				44	44	
6 & 7	Retro & Deorbit						
6 & 7	Reentry						
6 & 7	Landing Approach & Landing						
8	Display & Command Processing	50	50	50	50	50	50
9	Executive	50	50	50	50	50	50
	Mission Computation Profile	419	355	347	351	353	284

FOLDOUT FRAME

Table 4.8-4
RAL PROCESS REQUIREMENTS

ed Processor Operating Rate ons/Sec (Thousands) Per Phase						Required Memory (Thousands of Words)	
Docking	Orbit Stay	Retro	Reentry	Subsonic	Landing	Total Memory	Max on-line Memory
10	50	10	10	20	10	148	8.0
47	47	57	57	42	41	16.0	12.5
24	24	29	29	21	21	Included Above	
58	63	78	73	75	65	16.0	4.0
						8.5	
						16.0	16.0
						10.5	10.5
44						5.0	5.0
		50				7.0	
			70			7.0	
				70	70	Included Above	
50	50	50	50	50	50	27.0	4.0
50	50	50	50	50	50	20.0	4.0
353	284	324	339	328	307	281.0 (Total)	64.0 (Total)

function were assumed to be 10 percent of the malfunction detection function requirement. Also, the sampling rate was increased to 25 Hz to give rapid warning of an imminent abort condition.

Subsystem Control. This function includes all subsystem control functions not under the general heading of guidance, navigation, and flight control computations. It deals mostly with sequencing the events and testing and verifying their correct operation. It handles all computations associated with the control of the environmental control subsystem and the electrical power subsystem. It handles the major tasks of propellant loading during prelaunch and assures that all onboard resources are appropriately allocated to meet the forthcoming operational requirements (i.e., the configuration control task).

Without a detailed software study, it is impossible to size this task with any accuracy. However, a rough estimate can be made by assigning all active data points from Table 4.8-2 to the task and assuming a loading factor corresponding roughly to that of the malfunction detection routines. This results in the computational and memory loading shown in Table 4.8-4.

Guidance, Navigation, and Control. These functions, which predominantly involve subsystems 6 and 7, are discussed in more detail in section 4.4. The estimated computer loading figures are based on software generated for a 32-bit floating-point machine.

Display and Command Processing. The important feature of the crew display system is simplicity. A combination of a small selection of dedicated displays in conjunction with a flexible programmable display should achieve this basic goal. The bulk of the display presentation techniques reduces to either alphanumeric CRT presentations or alphanumeric lamp displays. The first group must be performed by special-purpose character generation equipment and second by BCD conversion equipment. This equipment is assumed to be part of the display console.

It was assumed that rates of 5 Hz and 10 Hz would be maximums required for updating the programmable and dedicated displays respectively. The display function would be performed by storing the list of display addresses in DCS memory and modifying these when required either by a crew member or a programmed display routine. The result is a maximum computer requirement of about 15,000 ops per second for these two functions. Also, the information from the crew controls is multiplexed in the same way as vehicle sensors at a 25-Hz rate. This results in a computer loading of 27,000 operations per second. A 20 percent contingency results in the 50,000 operations per second associated with the total crew systems.

The on-line display formats are assumed to be stored in the computer memory either in active or bulk memory, which accounts for the relatively large computer loading required for this task.

Executive. This function includes all aspects of task scheduling, computer self-testing, and any internal rode control. Because of the critical nature of the DCS, an extensive rapid self-test is required, involving a large section of memory dedicated to this task.

4.8.3.2 Mission Computational Profile. From the cumulative profile of Table 4.8-4 an indication can be derived of the requirement for at least a 2.5 microsecond computer; i.e., an add time, including memory access, should be no more than 1.5 microseconds for a worst-case operating duty cycle of about 70 percent.

Figures 4.8-7 and 4.8-8 present two parametric studies that were carried out under a Lockheed study program (Ref. 1). They show that an inter-register add time of 1 microsecond is possible with T^2L logic (1969 technology). It is apparent that a 32-bit carry-lookahead arithmetic unit is required, which would consume possibly 12 watts if the standard T^2L were used or about 3 watts if low power T^2L were used. From these figures a worst case CPU power of 15 watts was estimated.

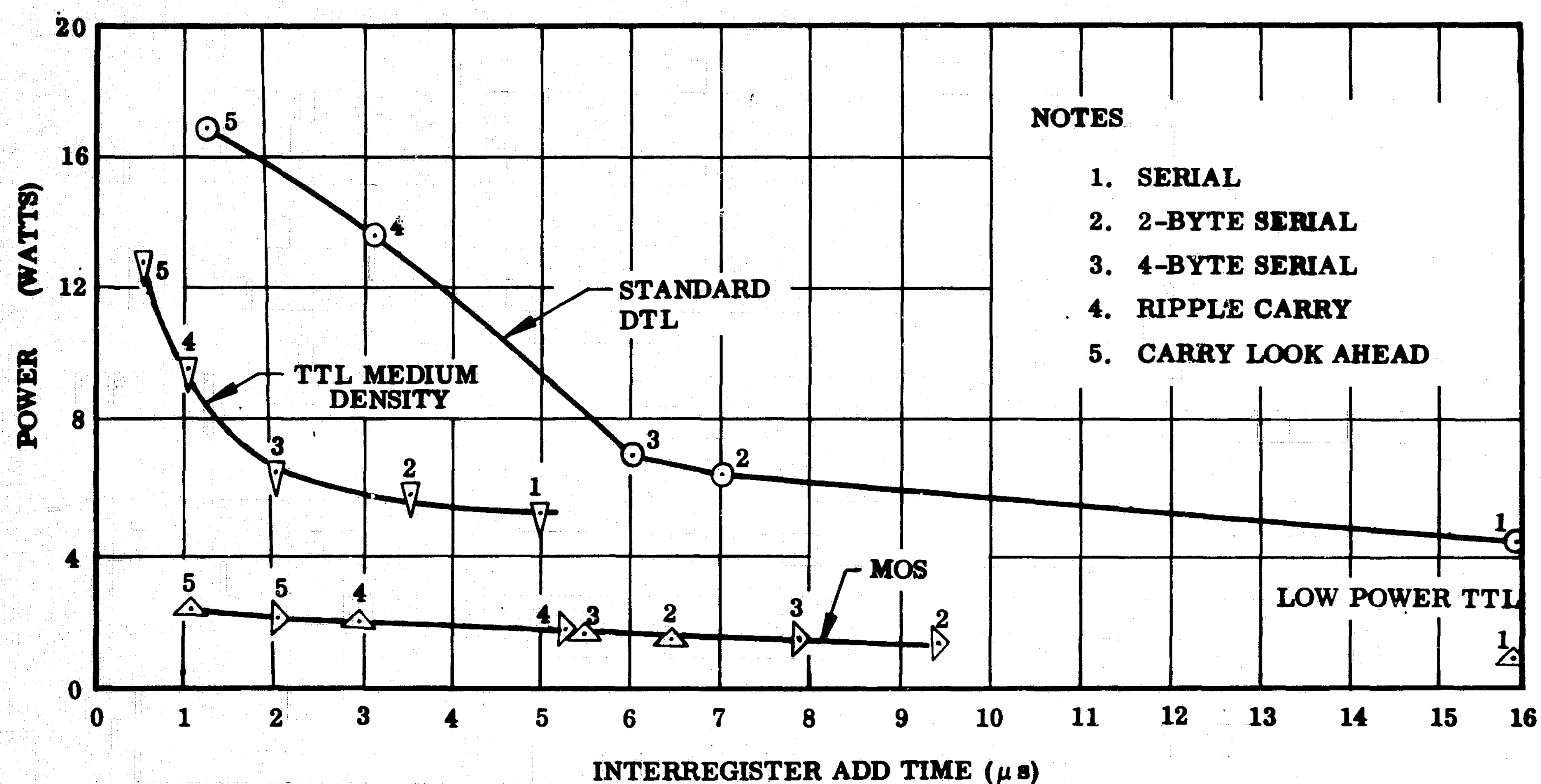


Fig. 4.8-7 ARITHMETIC UNIT POWER VS. SPEED FOR 32-BIT WORD

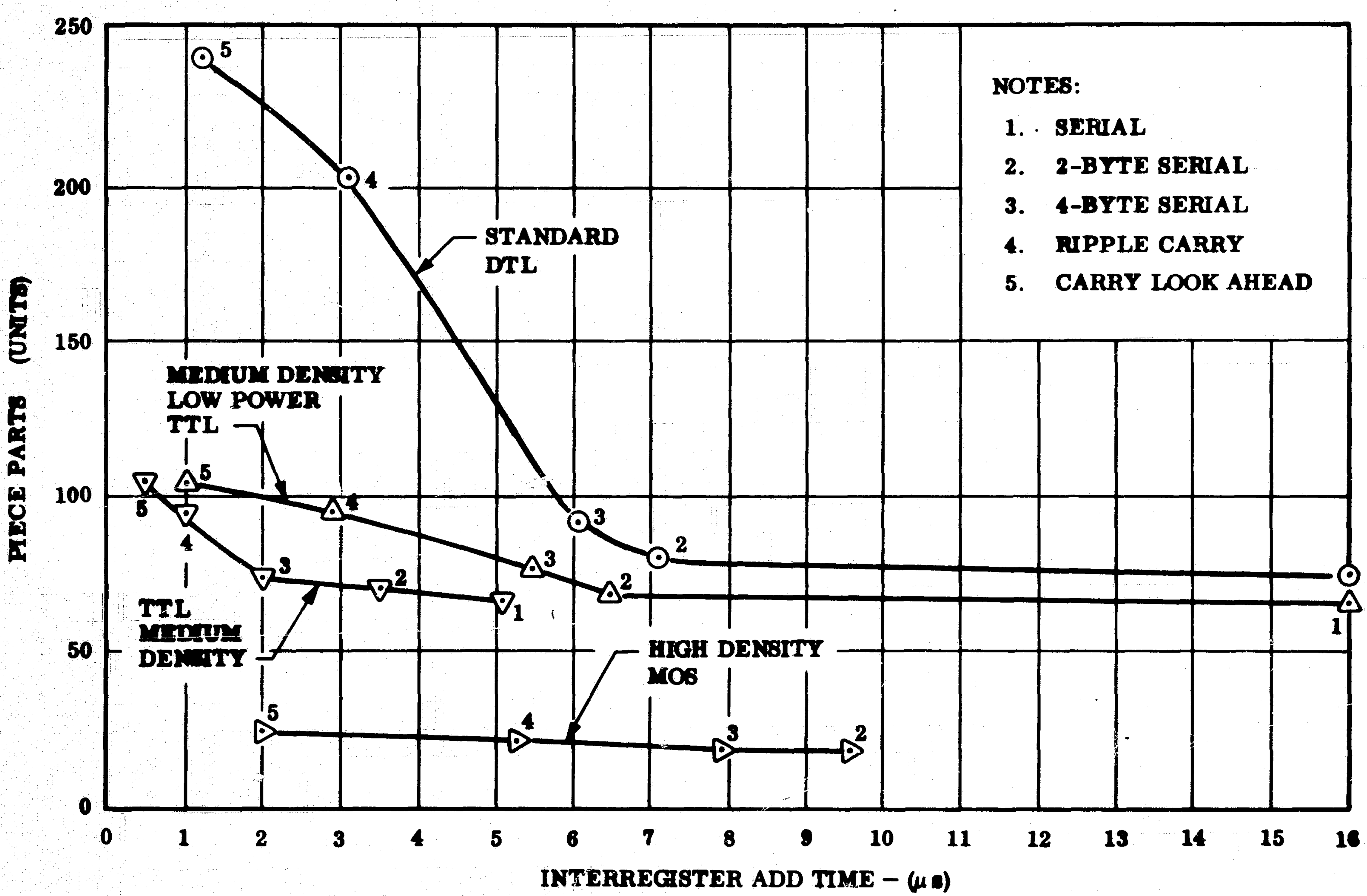


Fig. 4.8-8 ARITHMETIC UNIT PIECE PART VS. SPEED FOR 32-BIT WORD

LMSC-A959837
Vol. III

4.8.3.3 Memory Requirement. The active (on-line) memory requirement is sized on the basis of the ascent, injection, and rendezvous phases. As shown in Fig. 4.8-3, 64K words are required.

The average number of memory access cycles for each basic computer operation is 0.4 when averaged over many types of space vehicle function. This number was derived during work carried out under a Lockheed space program (Ref. 1). If a maximum computational loading of 500K ops/sec is assumed, the equivalent memory accesses would be 200,000 per sec.

Figure 4.8-9 shows the estimated memory power/speed curve based on 8K word x 32-bit core stacks; 20 mil cores are used. The standby power is high as a result of the individual addressing electronics required by each stack. This power could be reduced by either increasing the stack size or by strobing the sense amplifiers. Nevertheless, the curve is reasonably representative and indicates the linear relationship between speed and transient power. The total operating power, at the assumed access rate, is seen to be 95 watts.

4.8.3.4 Input-Output Requirements. The I/O section of the data coordination system is relatively simple because of the decentralization of the multiplexing and command decoding tasks. The I/O tasks performed centrally are the following:

- Formatting and sequencing of data channel addresses
- Line driving of the DDS buses
- Interfacing with DCS internal data bus
- Formatting and validation of commands
- Buffer storage to allow data rate differential between DDS and DCS rates
- Buffer storage and interrupt control for bulk memory interface
- Special-purpose task associated with the guidance system

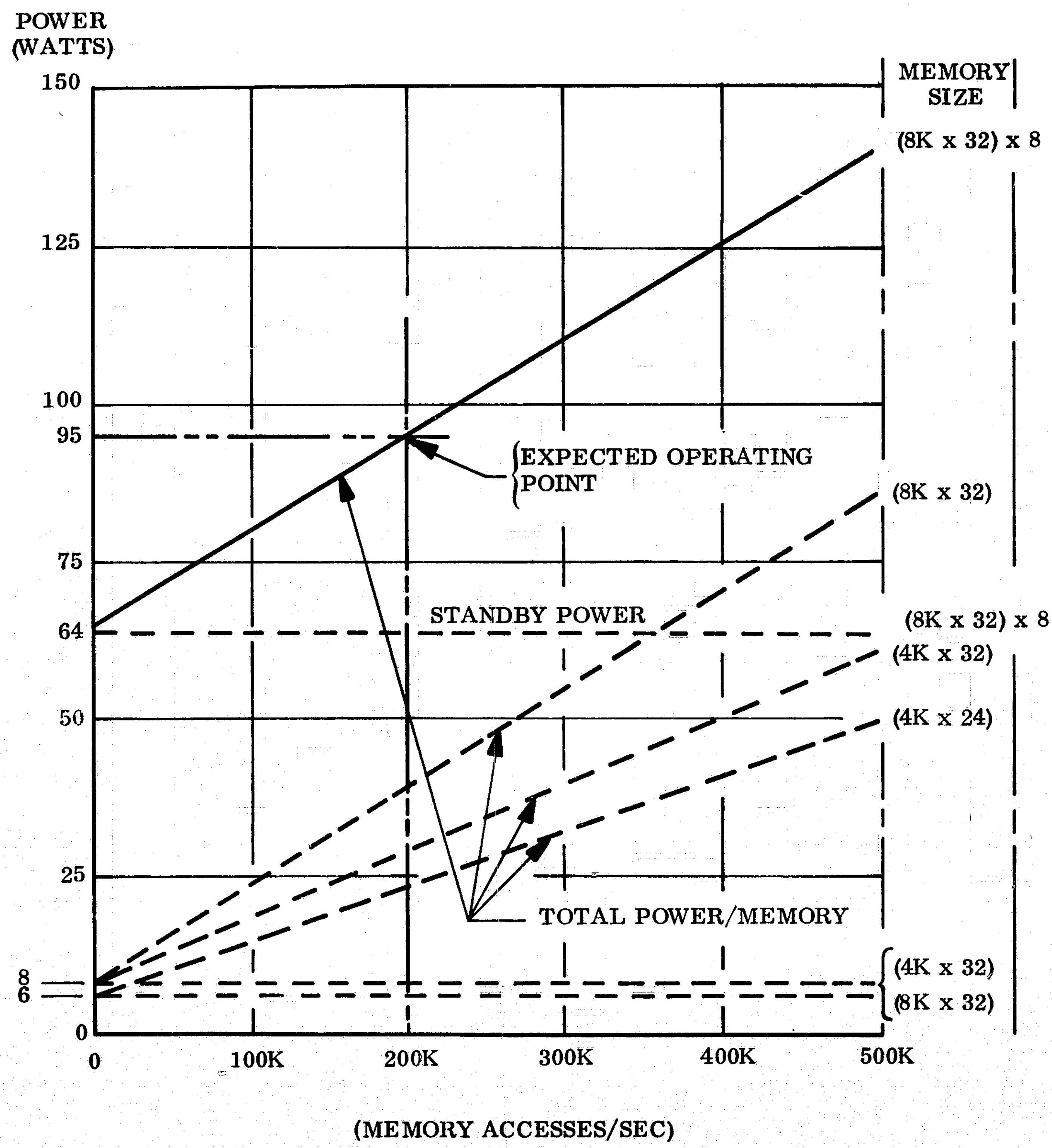


Fig. 4.8-9 POWER-SPEED CURVE FOR ACTIVE MEMORY

The tasks are relatively simple; however, the large data throughout requires high-speed operation, with corresponding high power required. As shown in Table 4.8-5, 45 watts is considered adequate.

4.8.3.5 Data Coordination System - Physical Characteristics . Table 4.8-5 summarized the DCS characteristics, all estimates being relatively conservative and well within 1972 state-of-the-art.

Table 4.8-5
DATA COORDINATION SYSTEM PHYSICAL CHARACTERISTICS

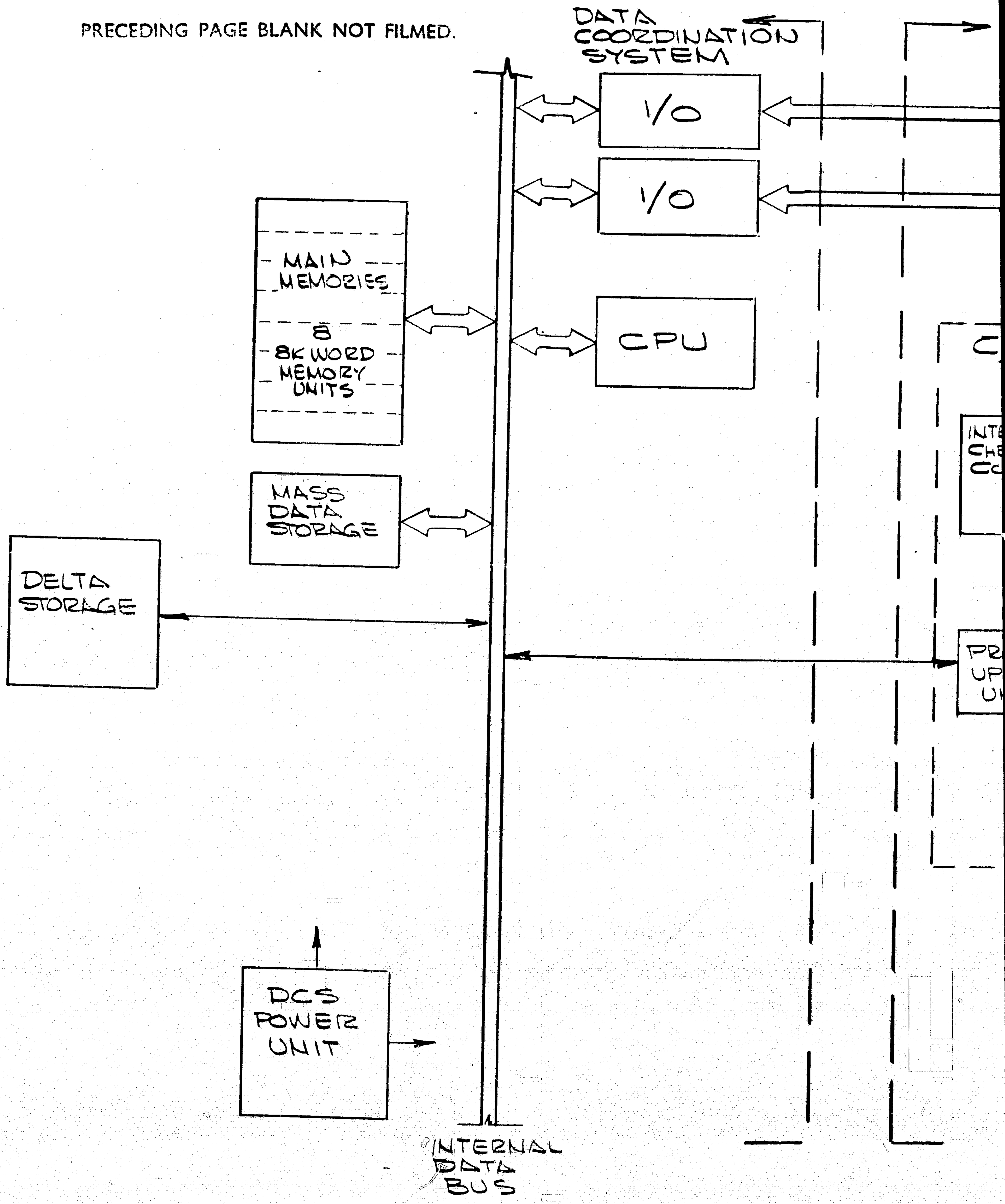
Unit	Weight/Unit (lb)	Power/Unit (watts)	Volume (in^3)
Memory	56	100	800
Central Processor	8	15	200
Input/Output	9	25	200
Power Unit	9	21	120
Hardware	20	--	80
Memory System Totals	102	161	1400

4.8.4 Complete Nonredundant Specifications

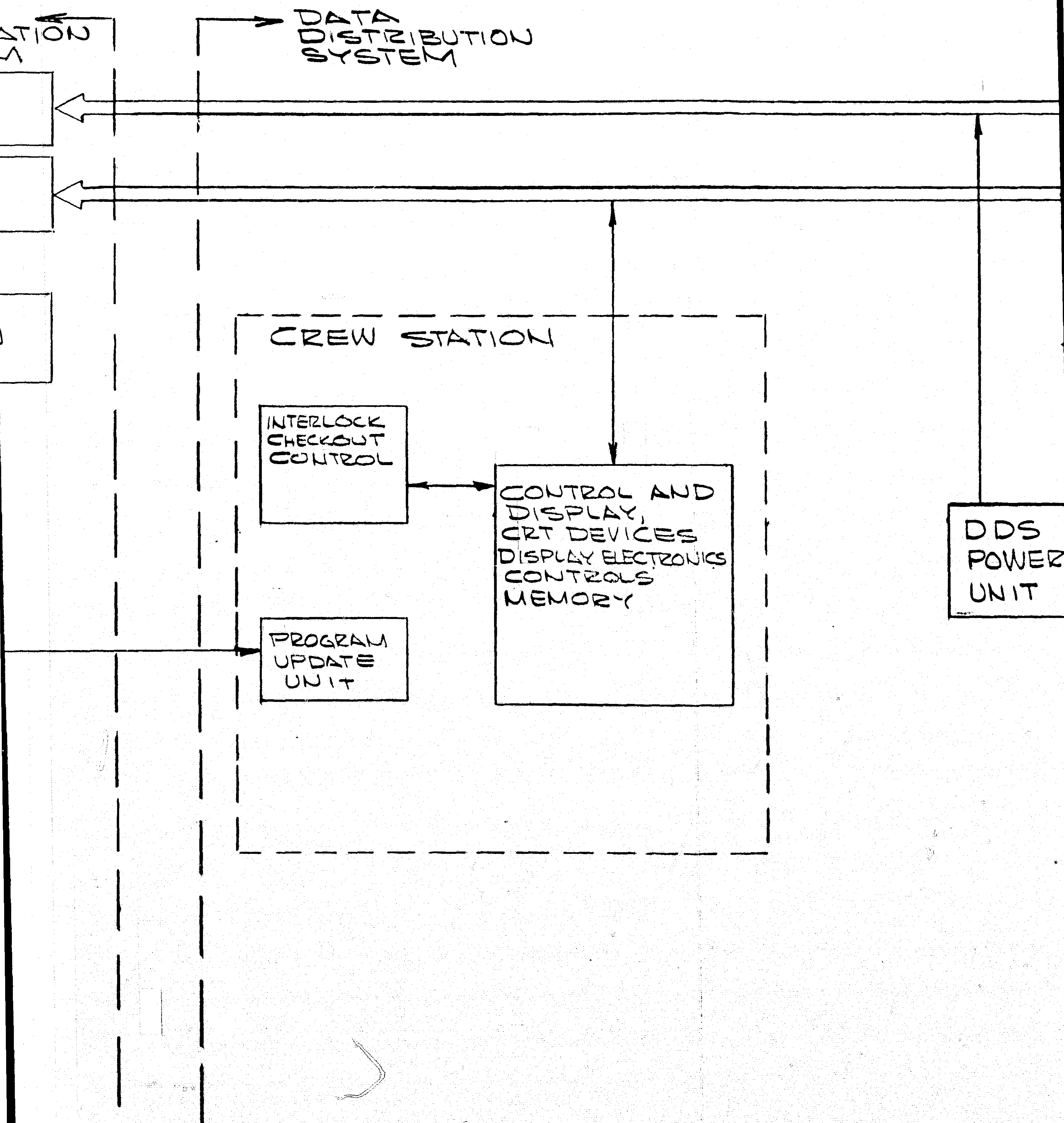
Figure 4.8-10 is a symbolic block diagram of the complete nonredundant system. The data configuration system requires a single CPU and a bank of 8 memory units, and an I/O section for each of the two DDS bus systems. The DCS modules are shown interconnected with a main internal bus structure. It is shown in this way for comparison purposes with the redundant configuration shown in the section 4.11.

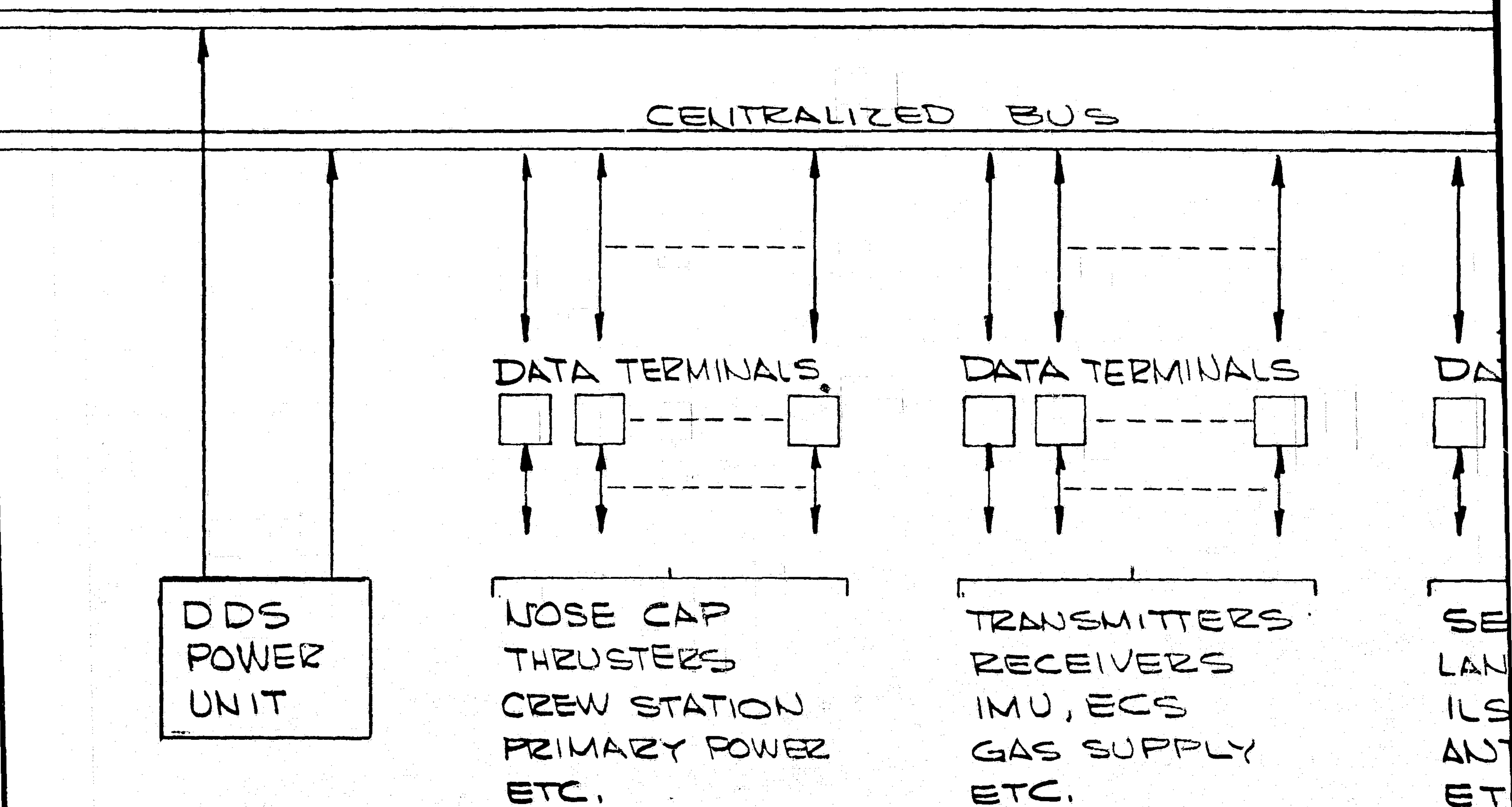
A summary of the electrical and physical characteristics of the complete system is presented in Table 4.8-6.

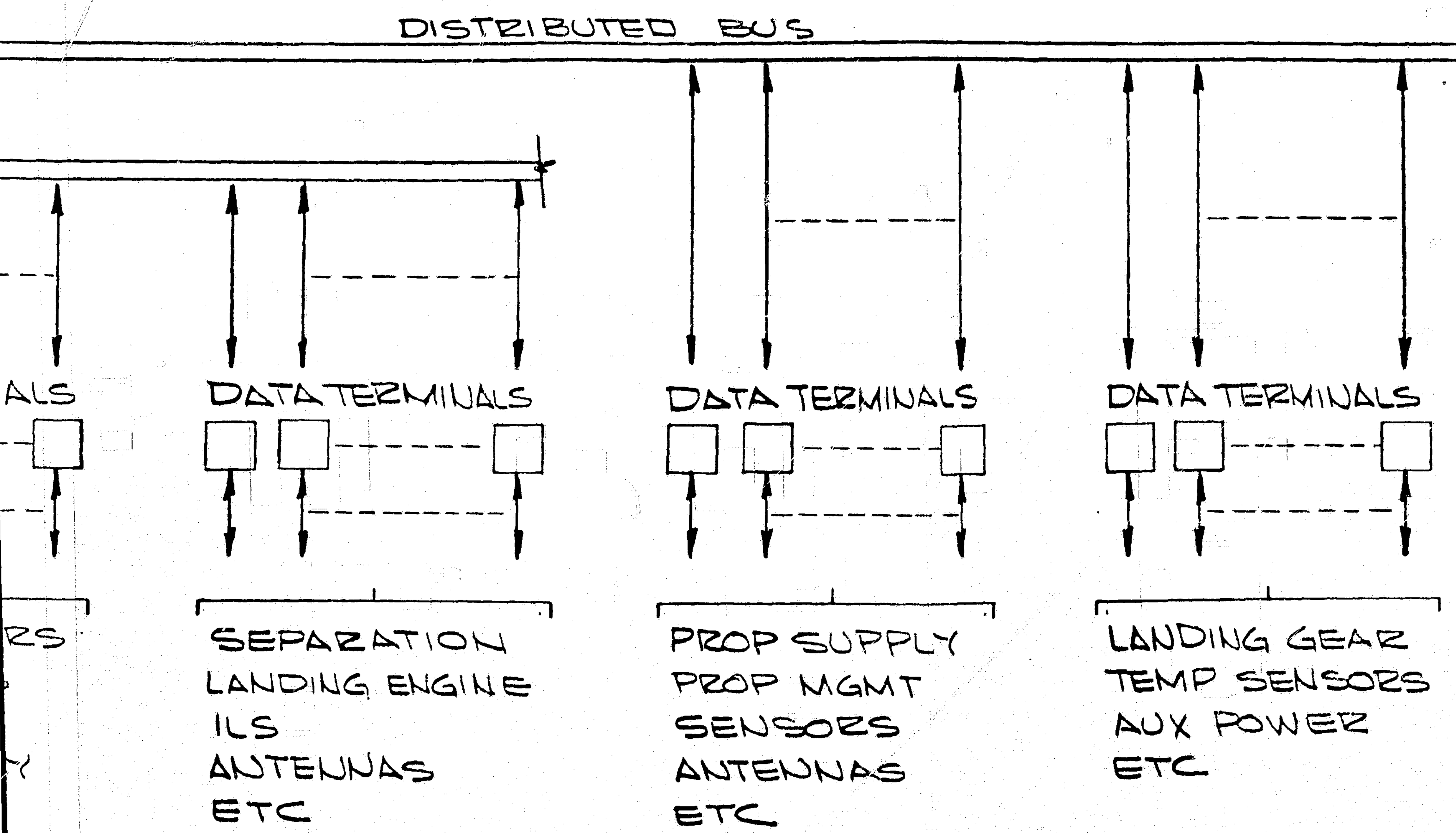
PRECEDING PAGE BLANK NOT FILMED.



FOLDOUT FRAME 1


ATION
DATA DISTRIBUTION SYSTEM
CREW STATION
INTERLOCK CHECKOUT CONTROL
CONTROL AND DISPLAY, CRT DEVICES DISPLAY ELECTRONICS CONTROLS MEMORY
PROGRAM UPDATE UNIT
DDS POWER UNIT


CENTRALIZED BUS
DATA TERMINALS
DATA TERMINALS
DDS POWER UNIT
NOSE CAP
THRUSTERS
CREW STATION
PRIMARY POWER
ETC.
TRANSMITTERS
RECEIVERS
IMU, ECS
GAS SUPPLY
ETC.

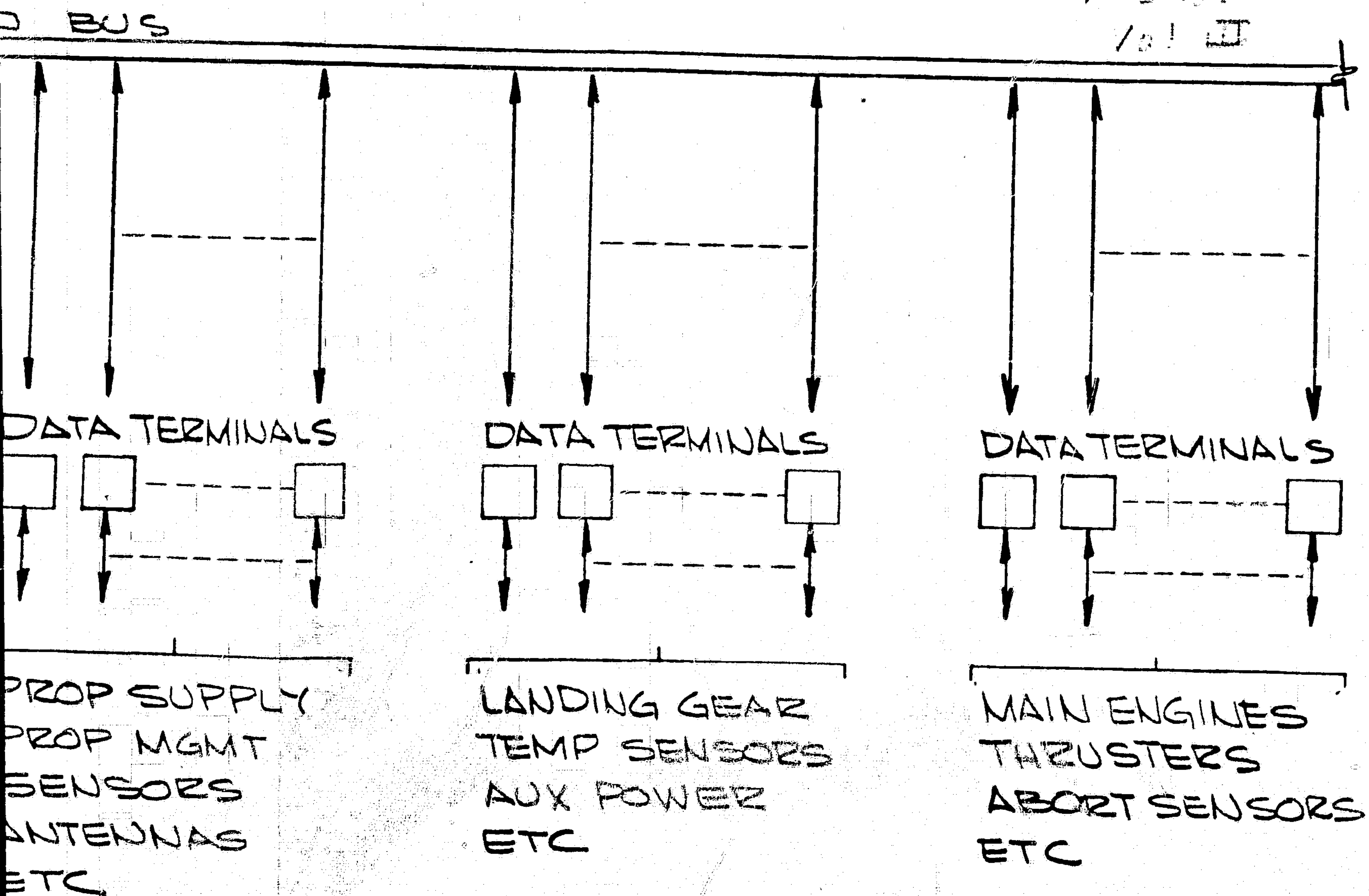


FOLDOUT F

INTEGRATED ELEC
CONFIGURATION

FIGURE

FOLDOUT FRAME 4



FOLDOUT FRAME 5

INTEGRATED ELECTRONICS SYSTEM
CONFIGURATION 3 - WITHOUT REDUNDANCY

FIGURE 4.8-10

PRECEDING PAGE BLANK NOT FILMED.

4.8.5 References

Ref. 1 "Independent Development Program, Integrated Electronic System Development," LMSC-A951928, May 15, 1969

Ref. 2 "UAC Modular Guidance System," Hamilton Standard, System Center, United Aircraft Corp., Aerospace Technology, March 25, 1968

Table 4.8-6

DDS AND DCS SUMMARY

Electrical Characteristics

<u>DCS</u>

Memory

Stack access time - 350 ns
Read-Restore time - 1 μs
Clear-Write time - 1 μs
Stack size - 8x8K words
Word size - 32 bits
Core size - 20 mil OD
Avg. access rate - 200K Hz
Avg. operating power - 30 watts
Avg. standby power - 64 watts

CPU

Type - 32-bit floating-point parallel processor
Add time - 1.5 μs (including memory access)
Multiply time - 7 μs (including memory access)

<u>DDS</u>

Buses

Type - Centralized (1000"), distributed (3000")
No. - 1 command and 1 data bus each
Data Rate (max) - 8K words/sec

Modules

Basic Types - 16-channel multiplexer (used for data retrieval and command decoding)
40 discrete channel expander
10-bit A/D converter
10-bit D/A converter
10-bit command protect module

Table 4.8-6 (Continued)

Physical Characteristics

	Weight/Unit (lb)	Power (watts)	Volume (in^3)
DCS	102	161	1400
Distributed Bus Modules	56	109	123
Distributed Cable Assy	43.3	-	450
Centralized Bus Modules	40.5	70	91
Centralized Cable Assy	15.2	-	150
Power Unit	10	27	120
Totals	277	367	2334

Note: DDS cable requirements determined for 1900" vehicle, converted to 1500" with 0.8 mult. factor.

4.9 SOFTWARE

A preliminary study has been conducted to identify the required computer software programs, to investigate the relationship between the level of vehicle subsystem integration, and to describe the total software task. The classes of onboard computational functions have been defined and estimates have been presented for the following software: malfunction detection and warning, operational support, interface control, computation, and executive.

Three organizations of the vehicle computation equipment are configured to correspond with the three IES alternatives discussed in the preceding sections. A basic summary of the number of words required for each of the three alternatives is shown in Table 4.9-1. These estimates are based on the assumption that there will be no integration of functions. Only those requirements that pertain directly to the particular alternatives are shown.

Table 4.9-1

ESTIMATED VEHICLE FUNCTION PROGRAM SIZE ESTIMATES
(In 1000's of 32-Bit Words)

Program	IES Alternative		
	1	2	3
Crew station	27	27	27
Malfunction detection and warning	16	16	16
Operations support	134	134	134
Interface control	*	2	2
Configuration control and sequencing	*	*	32
Structure/mechanical	*	*	6
Propulsion	*	*	4
Electrical power	*	*	4
Environmental control	*	*	2
Guidance/navigation	*	*	35
Vehicle control	*	*	19
Totals	177	179	281

*Mechanization of these functions is allocated to the appropriate subsystems.

4.9.1 Malfunction Detection and Warning

The malfunction detection programs will monitor signals from vehicle components, perform tests on these measurements, record selected test results, and inform the crew of defective components. Both malfunctions and engineering and trend data will be recorded.

The warning programs will monitor and test critical measurements, monitor the behavior of the vehicle to verify that the flight plan is being satisfied (to detect errors that may occur despite the proper operation of all vehicle functions), inform the crew of conditions that are or could become dangerous (such as failure or excessive loss of redundancy in a critical function or significant deviation from flight plan), and invoke execution of operations support programs to modify flight plans when an abort is necessary.

The MADAR system provides concepts and techniques that will be applicable to the Space Shuttle. Two varieties of onboard software are associated with MADAR and will also comprise most of the malfunction detection and warning software: test interpreter and supervisor, which monitors and tests according to encoded test programs and maintains smooth transition between test programs; and test programs that are the encoded versions of the logic programs written by engineers.

An estimated 1500 vehicle components with approximately 2000 test points are expected to be monitored by the malfunction detection program. The warning program will be concerned with about 10 percent as many test points.

4.9.2 Operations Support

The onboard operations support function includes the following: flight plan (calculations of perameters for the computations of the computing subsystems and preparation of instructions to the crew); reconfiguration of the Space Shuttle subsystems in the event of a component failure (turning off a malfunctioning component and switching in a backup); prelaunch checkout (minimum testing of vehicle components and subsystems to confirm flight readiness), and countdown, including fueling. The initial flight planning will be performed by an offboard system; the onboard system must be able to make in-flight adjustments to flight plans for mission abort.

4.9.3 Interface Control

The subsystem interface control and coordination function consists of routing intravehicle messages, reformatting messages, performing simple computations on messages, and generating trigger signals to subsystems. The last two aspects of the interface function are concerned with coordination of subsystems and may be performed by subsystems themselves.

4.9.4 Computation

The computation function consists of computing tasks specific to particular subsystems. There are seven "computing subsystems": crew station, structure/mechanical, propulsion, electrical power, environment control, guidance/navigation, and vehicle control.

Crew station (display and control) computation consists of formatting information for presentation to the crew, coordinating and maintaining displays, interpreting messages from the crew, and transmitting crew-provided information to appropriate subsystems.

The other subsystems monitor sensors attached to physical components (for control purposes, not for malfunction or danger detection), compute types and magnitudes of control stimuli required to maintain control of the subsystem and to satisfy the flight plan, and generate control signals to components. In Alternatives 1 and 2, many of the subsystem computations will be performed by specialized equipment; in Alternative 3, they will be performed by the central computation system.

4.9.5 Configuration Control and Sequencing

The configuration control and sequencing function will switch operational modes of subsystems and components and will monitor and control subsystems and components as necessary to ensure that proper timing relationships are preserved.

4.9.6 Executive

Executive software ordinarily refers to the set of programs of the following types: subtasks of many of the computer tasks; routines for accounting, failure detection and recovery, interprogram communication, program initiation, and schedule of execution of programs; and interrupt handler. The common subtasks are program components. Accounting activities are also built into the task programs. Reliability considerations have not been included in the study.

Interrupt handling involves acknowledgement of the high-priority communications from outside the computer and scheduling the execution of appropriate task programs. The amount of interrupt handling program not included in task programs or the scheduling program is on the order of 20 to 100 words. Interprogram communication may involve 50 to 200 words of instructions and an amount of message storage that depends on the messages, the queuing techniques, and the number of communicating programs. Interrupt handling and interprogram communication, including message buffers, will add approximately 10 percent to the sizes of the executive systems, but they should not exceed 1000 words nor be smaller that 300 words.

Program initiation is the process of locating the program to be executed next, loading it into the computer's high-speed memory (if it is not already there), notifying the program of the computer resources it may use, and transferring control of the computer to the program. These tasks are carried out under the direction of the scheduler or the resource allocation program. If the computer has sufficient high-speed memory to store all its task programs, the loading function of program initiation can be omitted. Otherwise, peripheral storage must be provided; this will increase the complexity of the scheduler and program characteristics.

The scheduler has responsibility for preventing conflicts among programs; for example, assuring that one program will not use for data storage a portion of memory in which another program is stored. Much of the scheduling of program execution will be done before flight by the offboard flight planning program. The executive software must be able to modify the predetermined schedules when such unpredictable situations as malfunction of a vehicle component or request from the crew occur.

The scheduler and program initiator require relatively large amounts of data storage in order to save information about the computer's programs and the schedules provided by the flight planning programs. These two components of the executive software, taken together, are sensitive to machine design and configuration. In the absence of fairly detailed specifications of the computers, high precision is not possible in size estimates for scheduling and program initiation.

Table 4.9-2 summarizes the relationships between executive software size and main memory and task program size. These relationships form the basis for the executive software estimates.

4.9.7 Alternative 1

The first candidate organization of vehicle computation uses separate computing machinery for each of the computing subsystems. In this configuration, the interface control function is performed by hardware associated with the data paths; there is no software involvement with interface control.

Many of the computational tasks may incorporate subtasks common to other tasks. These will include such processes as matrix arithmetic, input and output of data, formatting of messages, and transcendental function computations. These common subtasks will be programmed as subroutines; then, when two or more programs using the same subtask are executed by the same computer, only one copy of the subtask program need be stored in the computer's memory. The result is an economy of memory. On the basis of experience with large software projects, a 10 to 25 percent decrease in total program

Table 4.9-2
EXECUTIVE SOFTWARE

Component	Size*
Program Initiator	
Loader	400
Other	200
Scheduler	
With peripheral store (Program initiator requires loader)	600
Without peripheral store	400
Tables	
Computer system resources	1%
Schedules	2%
Program characteristics	
With peripheral store	5%
Without peripheral store	4%

Interrupt handling and interprogram communication (including tables):

10 percent of executive size (excluding these functions), but not less than 300 words or more than 1000 words.

---

*Program sizes in words; computer system resources table size in percent of main memory; other table sizes in percent of program words (excluding executive).

(Word length = 32 bits)

size may reasonably be expected as the programs are merged into a single computer.

In the first configuration, malfunction detection and warning plus operations support are combined in a single computer. A 10 percent size reduction is applied to this merger.

Table 4.9-3 presents estimates of program and memory sizes for each of the computers of Alternative 1.

4.9.8 Alternative 2

The second organization uses separate computing machines for most of the subsystem computations. However, the interface control and crew station computations are merged with operations support and malfunction detection and warning. Each subsystem now communicates only with the enhanced central subsystem; the number of intersubsystem data paths is reduced significantly. Moreover, as was discussed in the previous section, combining functions in a single computer produces additional program size savings.

The summary of the size estimates for the computers of Alternative 2 is shown in Table 4.9-3. In this table, a 10 percent size decrease is applied to the interface control and crew station estimates.

4.9.9 Alternative 3

The third organization of subsystem computing machinery is the configuration in which one control computer handles all computation for all subsystems. Integration of configuration control and sequencing the six subsystems into the central computer results in a 10 to 25 percent saving in those functions. A 10 percent saving is reflected in the estimates of Table 4.9-3.

Table 4.9-3

ESTIMATED NUMBERS OF WORDS REQUIRED*

Program	Alternative 1					Alternative 2					Alternative 3				
	Prog	Exec	Total Soft	Mem	Computer Reqd	Prog	Exec	Total Soft	Mem	Computer Reqd	Prog	Exec	Total Soft	Mem	Computer Reqd
Crew station	27	4	31	16	X	24					24				
Malfunction detection and warning	14					14					14				
Operations support	121					121					121				
	135	12	147	32	X										
Interface control	**2	1	3	4		2					2				
						161	14	175	48	X					
Configuration control and sequencing	32	4	36	16		**32	4	36	16		29				
Structure/ mechanical	6	1	7	8		6	1	7	8		5				
Propulsion	4	1	5	8		4	1	5	8		4				
Electrical power	4	1	5	8		4	1	5	8		4				
Environmental control	2	1	3	4		2	1	3	4		2				
Guidance/navigation	35	4	39	16		35	4	39	16		31				
Vehicle control	19	4	23	8		19	4	23	8		17				
	104	17	121	72		102	16	118	68		253	21	274	64	X
Totals	266	33	299	120		263	30	293	116		253	21	274	64	

*Word length, 32 bits — numbers in thousands of words.

**Mechanization of functions listed on this and lower lines is allocated to the appropriate subsystems. The estimates given would apply if each function were implemented in a dedicated programmable digital computer. Though this will not be the case for Alternatives 1 and 2, the estimates for these imaginary computers are presented in order to place the three alternatives in perspective.

4.9.10 Summary of Onboard Software

The size totals from Table 4.9-3 point out two advantages of increasing the degree of integration: the total amount of programming for onboard tasks decreases (this also results in a smaller memory requirement); and the ratio of required main memory to peripheral memory decreases. These two conclusions are valid, however, only if each of the functions listed is performed by a dedicated programmable digital computer. This will be true only for Alternative 3, as the majority of functions are assigned to the individual subsystems for Alternatives 1 and 2.

Table 4.9-4 summarizes the program and memory requirements for the three configurations, taking into account only those functions that will contribute to the actual software task. Alternative 3 involves a greater software effort than does either of the other configurations; however, this disadvantage is compensated by the significantly smaller amount of special-purpose hardware that will be required in this configuration.

Table 4.9-4

SUMMARY OF ON-BOARD SOFTWARE
(Size Estimates in 1000's of 32-Bit Words)

	Alternative		
	1	2	3
Program			
Task (subsystem)	162	161	253
Executive	16	14	21
Totals	178	175	274
Memory			
Main (high-speed)	48	48	64
Peripheral (low-speed)	130	127	210
Totals	178	175	274

4.9.11 Other Considerations

Table 4.9-4 demonstrates that, in the case of software memory requirements, the sum of the parts is never greater than the whole, and may be much less. Two aspects of integrating bear on this: merging programs obviates duplication of common subroutines, and merging computers decreases the number of executive systems. Moreover, the number of instructions in an executive program depends on the complexity of the system. Only the sizes of the executive's tables varies with task program sizes and numbers. Therefore, the size of the executive relative to task program size decreases with integration.

The impact of the requirement for high reliability of the subsystems computations on the software is not as easy to assess as is the impact of the basic functional requirements. Reliability constraints will dictate duplicate or triplicate storage of critical programs; however, probable impact of this requirement on the software may be alleviated by integration.

The difficulty in predicting the impact of reliability requirements, then, is related primarily to the modifications and additions to the executive software required to detect, diagnose, and overcome failures. This software will be closely tied to the organization and physical characteristics of the computing hardware and, therefore, may not be discussed specifically until the computing hardware is clearly defined.

4.9.12 Offboard Software

The onboard software contains no facilities for preparing programs or for nonoperational testing of programs. The onboard computing machinery will be designed and constructed to best serve the onboard computing requirements, whereas the program preparation and validation functions will require facilities distinct from those required onboard (for example, high-speed line printers). It is suggested that all program preparation and validation be performed by a ground-based, large-scale, general-purpose computer. This offboard supporting system will be selected for its efficiency of operation, convenience of use, and accessibility to programmers.

The major components of the supporting system software are as follows:

- Test program translator, which will translate programs written for malfunction detection and warning monitoring into a form suitable for interpretation by the onboard test program interpreter
- Test program environment simulator, which will simulate those aspects of the Space Shuttle of interest to the test programs and interpret test programs to permit debugging in the offboard computer
- Language translator, which will translate the task programming language into the language of the onboard computer (Alternative 3 is assumed here; if several different computers are used onboard, several different translators will be required.)
- Onboard processor simulator, which will simulate the onboard processor to permit offboard debugging of onboard programs (Use of different onboard processors will necessitate two or more of these simulators.)
- Program execution scheduler, which will translate onboard processing requirements generated by the offboard flight planner into detailed schedules of execution of onboard programs and allocations of resources to them

Estimates of the sizes for each of these five programs are presented in Table 4.9-5. The program all make use of well-known programming techniques. The development of the translators requires the definition of languages. The development of the simulators should, (and the estimates do) include design of facilities to aid programmers in debugging.

Table 4.9-5
OFFBOARD SOFTWARE

Program	Program Size (words)
Test program software	
Translator	15,000
Environment simulator	10,000
Programming language translator	15,000
Onboard processor simulator	10,000
Scheduler	15,000

In view of the high cost of programming, it is important that the programmer's task be made as easy as possible. For this reason, debugging aids are important parts of the simulators, and program analysis facilities are important in the translators. Moreover, the programming language used for writing the onboard programs should permit the greatest possible ease, conciseness, and clarity in the expression of computing processes. This will beneficially affect the costs of training programmers, writing programs, and preparing documentation.

These constraints on the programming language dictate that a high-level compiler language, strongly application-oriented, be used. Some programs, particularly those comprising the executive software and the test program interpreter, require that the programmer maintain control of the machine at a very intimate, hardware-oriented, level. Therefore, the flexibility provided by low-level assembly languages must be provided by the high-level compiler language translator.

The language-translator and onboard-processor-simulator programs provide another reason for favoring integration of onboard computing or, at least, using similar processors in all computing subsystems: one translator and one simulator are required for each type of onboard processor. Alternatively, this could be taken as an argument against use of a supporting system different from the onboard computers. In this case, though, the simulators are no longer needed; one language translator is still required for each onboard computer.

4.10 COMPARISON OF IES ALTERNATIVES

This section presents a comparison of the three alternative integrated electronics system configurations, which are described in sections 4.6, 4.7, and 4.8. The IES alternatives were intentionally implemented for a nonredundant set of avionics, in accordance with the study scope; this fact should be kept in mind while viewing the comparison, since a direct extrapolation of results to the case of a redundantly configured set of avionics may not always be possible.

4.10.1 Weight

The weight of avionics equipment allocated to each subsystem is presented in Table 4.10.1-1 for the three IES alternatives. The subsystems are grouped to permit subtotaling of avionics weights that would normally be allocated to avionics subsystems and those that would be allocated to nonavionic subsystems. Decreased cable weight accounts for most of the weight decrement from Alternative 1 to Alternative 2. Additional weight savings are possible in Alternative 3, primarily because of the elimination of subsystem computers. A more detailed breakout of weights for the data management subsystem is provided in Table 4.10.1-2. Note that the term "Data Management" is expanded to mean the central computer complex for Alternative 3.

4.10.2 Power

The comparison of power requirements in Table 4.10.1-1 and Table 4.10.1-2 is based on the power dissipation of each piece of equipment and does not reflect peak power consumption, average power consumption, or power duty cycle. The increase of power for Alternative 2 is assignable to data acquisition units, whereas the net decrease of power for Alternative 3 is due to elimination of subsystem computers outside of data management.

PRECEDING PAGE BLANK NOT FILMED.

TABLE 4.10.1-1

AVIONICS SUMMARY

Subsystem	Weight (lbs)									
	Alternative 1 Baseline Avionics				Alternative 2 (Increments)				Alternati (Incremen	
	Basic Equipmt.	Cabling	Instr.	Total	Basic Equipmt	Cabling	Instr	Total	Basic Equipmt.	Cabling
1. Structure/mechanical	157	100	160	417	0	-100	0	-100	-59	-100
2. Propulsion	40	400	200	640	0	-400	0	-400	-8	-400
3. Electrical power	565	3504	50	4119	0	0	0	0	0	0
4. Environmental control	0	20	50	70	0	-20	0	-20	0	-20
Subtotals	762	4024	460	5246	0	-520	0	-520	-67	-520
5. Guidance/Navigation	179	29	5	213	0	-29	0	-29	-35	-29
6. Vehicle control	71	200	50	321	0	-200	0	-200	-4	-200
7. Communications	67	50	10	127	0	-5	0	-5	-4	-5
8. Controls and displays	393	20	5	418	-60	-20	0	-80	-105	-20
9. Data management	375	228	5	608	-30	-208	0	-238	-58	-170
Subtotals	1085	527	75	1687	-90	-462	0	-552	-206	-424
	Basic weight							△		
Grand totals				6881				-1072		

- o No redundancies are included; totals are for a "single-thread," fully functional system.
- o Cabling and instrumentation weights are included in each of the five subsystems
- o Instrumentation power dissipation is included in each of the five subsystems.
- o 1972 SOTA is used for all the equipment in the five subsystems
- o Basic weight and basic power figures are for an avionics system designed to a "first" integration quantities are shown above (△).
- o The power values shown are only for purpose of comparing the alternatives of design integration. "on" simultaneously, as might be inferred from the tabulations.

FOLDOUT FRAME |

).1-1

SUMMARY

Alternative 3 (Increments)			
Basic quipmt.	Cabling	Instr.	Total
-59	-100	0	-159
-8	-400	0	-408
0	0	0	0
0	-20	0	-20
-67	-520	0	-587
-35	-29	0	-64
-4	-200	0	-204
-4	-5	0	-9
-105	-20	0	-125
-58	-170	0	-228
-206	-424	0	-630
			Δ
			-1217

Power (watts)								
Alternative 1 Baseline Avionics			Alternative 2 (Increments			Alternative 3 (Increments)		
Basic Equipmt.	Instr.	Total	Basic Equipmt.	Instr.	Total	Basic Equipmt.	Instr.	Total
260	150	410	0	0	0	-115	0	-115
380	180	560	0	0	0	-60	0	-60
615	40	655	0	0	0	0	0	0
0	40	40	0	0	0	0	0	0
1255	410	1665	0	0	0	-175	0	-175
485	4	489	0	0	0	-112	0	-112
368	40	408	0	0	0	-14	0	-14
473	8	481	0	0	0	0	0	0
752	4	756	-30	0	-30	-70	0	-70
443	4	447	+80	0	+80	+79	0	+79
2521	60	2581	+50	0	+50	-117	0	-117
Basic power					Δ			Δ
		4246			+50			-292

ystem.

st" integration alternative; for Alternatives 2 and 3, the incremental differences from the basic

n integration. They do not represent true power duty cycles nor would all of the packages ever be

PRECEDING PAGE BLANK NOT FILMED.

TABLE 4.10.1-2

DATA MANAGEMENT SUMMARY

Equipment	Weight (lb)								
	Alternative 1				Alternative 2				
	Basic Equip.	Cabling	Instr.	Total	Basic Equip.	Cabling	Instr.	Total	Basic Equip.
Delta Storage	53				0				0
Data Processor	22				0				+80
Control group interface	35				0				-35
Interlocking C/O control unit	16				0				0
Program update unit	8				0				0
Interface comparator	15x(3)				0				-45
Signal acquisition unit	5x(33)				-30				-58
Mass data storage	31				0				0
Subtotal	375				-30				-58
Cabling		228				-208			
Instrumentation			5				0		
Total				608				-238	

FOOTOUT FRAME 1

MMARY

					Power (watts)								
	Alternative 3				Alternative 1			Alternative 2			Alternative 3		
tal	Basic Equip.	Cabling	Instr.	Total	Basic Equip.	Instr.	Total	Basic Equip.	Instr.	Total	Basic Equip.	Instr.	Total
	0				30			0			0		
	+80				70			0			+91		
	-35				90			0			-90		
	0				65			0			0		
	0				20			0			0		
	-45				15x(3)			0			-45		
	-58				2.5x(33)			+80			+123		
	0				40			0			0		
	-58				443			+80			+79		
		-170											
			0			4			0			0	
238				-228			447			+80			+79

FOLDOUT FRAME 2

PRECEDING PAGE BLANK NOT FILMED.

4.10.3 Reliability

No quantitative comparison was made for reliability, but some observations may be stated. First, the decrease in cabling, connectors, and number of pin connections for Alternatives 2 and 3 will improve their reliability over that of Alternative 1. Second, for redundantly configured systems, Alternative 3 will have the advantage of being able to employ software to restructure the processor-memory grouping or the avionics equipment interconnections, making possible graceful degradation through assigned priority of functions and stored information.

4.10.4 Technical Risk

A scale of technical risk may be defined to extend from "use off-the-shelf hardware," to "modify existing designs," to "perform new design," and to "develop new technology." The technical risk for Alternative 1 is least, since the technique and the hardware for onboard checkout and fault isolation have been demonstrated. Provision of autonomous vehicle capability, however, falls into the category of new design. Alternative 2 is similar in concept to a system design proposed and about to be implemented for the S3A aircraft and its extensive avionics equipment. Some elements of the system have been demonstrated and, by 1972, experience with the complete system should exist. Alternative 3 will require the development of some complex software, in addition, the centralized control of all subsystems will require careful design to isolate a component catastrophic failure from the remainder of the system. Alternative 3 presents the greatest technical risk. Electronics component technology will be sufficient to support the system design of any of the three alternatives and, by itself, does not constitute a significant technical risk.

4.10.5 Sensitivity of Point Design

Sensitivity of a system design to modified requirements or to additional requirements is significant for a long-term program. Alternative 3 has a 100 percent margin of computer instruction rate capability (taking 1972 capability to mean a 1 microsecond cycle time) and will be limited only by software in its ability to accommodate new requirements. Also, spare memory capacity of a central computer complex may be allocated to any one of the subsystem functions as required. Both Alternatives 2 and 3 employ standard interfaces and multiplexed bus designs; also the present data rates are not high, so that the addition of equipment or the changed design of equipment can be accommodated. Alternative 1 is least flexible due to the extensive use of cabling, analog signal transmission, and nonstandard interfaces among subsystems and major components other than for test point access.

4.10.6 Data Acquisition/Distribution

Alternative 1 makes use of nonstandard interfaces for hard-wired interconnections, is most susceptible to EMI/noise based upon the number of lines and analog signal transmission, and has the lowest total data rate requirement. Alternatives 2 and 3 make use of clocked, digital-data transfer at about 200 K bits/second across standard interfaces. The technique of Alternative 2 requires only a two-wire bus, whereas Alternative 3 employs a 12-wire cable.

4.10.7 Data Processing

Alternative 3 requires an instruction rate of 419,000 instructions per second; the maximum rates for Alternatives 1 and 2 occur in the guidance/navigation/control computer and are 75,000 instructions/second. The configuration control function for Alternative 2 is distributed throughout (in subsystem controllers), and the rates are low. Storage requirements for each alternative are bulk, delta, program up-date, and main memory (total storage is 274,000 words). Only the main memory requirement varies as a function of the approach. On-line storage for Alternative 3 has been established at 64,000 words with Alternatives 1 and 2 about 6 percent higher (68,000 words). Alternative 2 requires 1,000 more words of storage (interface control) than Alternative 1. However, this quantity is included in the 32,000 words allocated for the data management function.

4.10.8 Subsystem Interface Definition

Subsystem interfaces are well-defined for both Alternatives 1 and 2. For the latter, a system interface specification will control all "box" suppliers, and all subsystems to use the standard interfaces will be defined in the Specification. For Alternative 3, a new group of interfaces must be defined wherever the integrated system extends into the normally defined subsystem areas. Problems of establishing revised organizations to work within the framework of revised subsystem interfaces must be resolved.

4.11 RELIABILITY/SAFETY REQUIREMENTS IMPACT

Rather than perform a complete reliability/safety failure modes analysis for each subsystem, reconfigure each subsystem to meet the reliability/safety requirements, and integrate the redundant subsystems, Only two examples were investigated to obtain an indication of the impact on weight and power of these requirements. The basis for this decision lies in the limited scope of the IES study. The two examples selected were the communications subsystem and the guidance, navigation, and control subsystem.

4.11.1 Communications Subsystem (Figs. 4.11-1 and 4.11-2)

The basic subsystem includes some functional redundancy; therefore, triple or quadruple redundancy is not required to satisfy the reliability/safety requirement. For example, Intercoms 1 and 2 are required by the basic system and would therefore be considered fail operational. To meet the requirement of fail operational-fail operational-fail safe, internal redundancy to intercoms 1 and 2 will be used as defined by the reliability model in Fig. 4.11.1-1. The weights summarized in Table 4.11-1, 4.11-2, and 4.11-3 are based on this approach and are established by taking percentages of the base weight. It is also assumed that the premod/demod is internally redundant.

Table 4.11-1

SUMMARY - COMMUNICATIONS SUBSYSTEM

RELIABILITY IMPACT OF ADDITIONAL WEIGHT AND POWER - COMMUNICATIONS SUBSYSTEM

Additional Equipment	Additional Weight (lb)	Additional Power (watts)
Intercoms 1 and 2	3.0	6.0
Premod/demod	4.0	3.0
Transmitters/receivers	22.0	0
Antennas	2.5	0
Total increment	31.5	9

IV

	Weight (lb)	Power (watts)
Baseline	67	473
Additional	31.5	9
Total	98.5	482

Table 4.11-2

RELIABILITY IMPACT - ADDITIONAL WEIGHT, POWER

COMMUNICATIONS - TRANSMIT FUNCTION

Additional Equipment	Additional Weight (lb)	Additional Power (watts)
Intercom 1	0.5	2
Transmit switch		
Amplifier		
Intercom 2	1.0	1
Premod/demod	2	1
Subcarrier mod		
Mixer		
Transmitter/receiver	10	(183)*
Ku transmitter		
Power amp		
Antennas	(Included in Table 4.11-1)	
Total increment	13.5	4

*Not considered additional, since only one is required at a time.

Table 4.11-3

RELIABILITY - ADDITIONAL WEIGHT AND POWER IMPACT

COMMUNICATIONS - RECEIVE FUNCTION

Additional Equipment - Receive	Additional Weight (lb)	Additional Power (Watts)
Intercom 1 critical subsections		
a. Amp 2	0.5	2
b. K-band receiver		
Intercom 2		
K-band receiver, switch	1.0	1
Premod/demod critical subsections		
Audio subcarrier discriminator	1	1
Data subcarrier discriminator	1	1
UHF transceiver	12	(250)*
UHF antenna	1.5	0
S-band switch (2)	1.0	0
Total increment	18	5

* Not considered additional, since only this or the baseline will be required at one time.

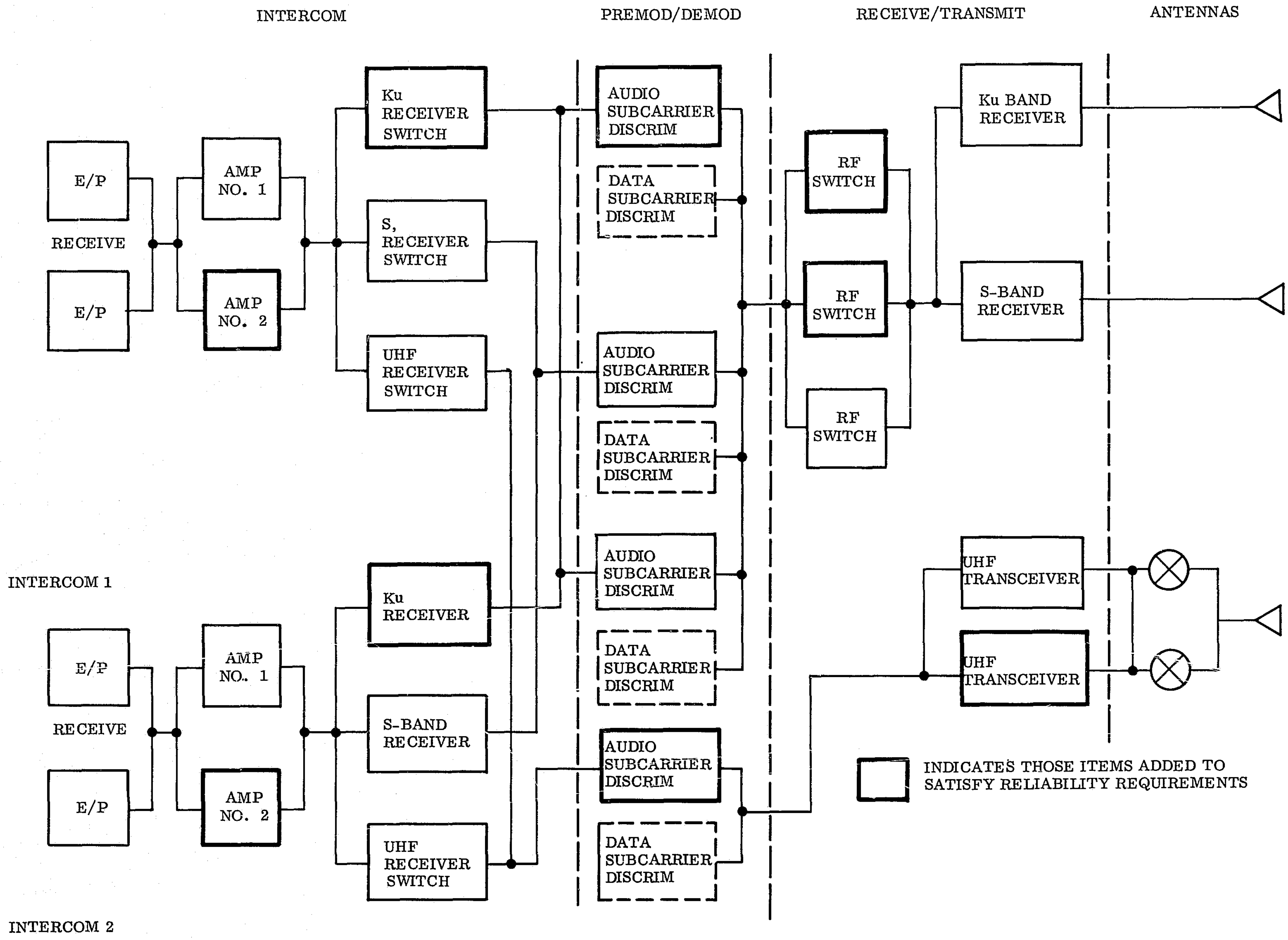


Fig. 4.11-1 Communications Subsystem Receive Function

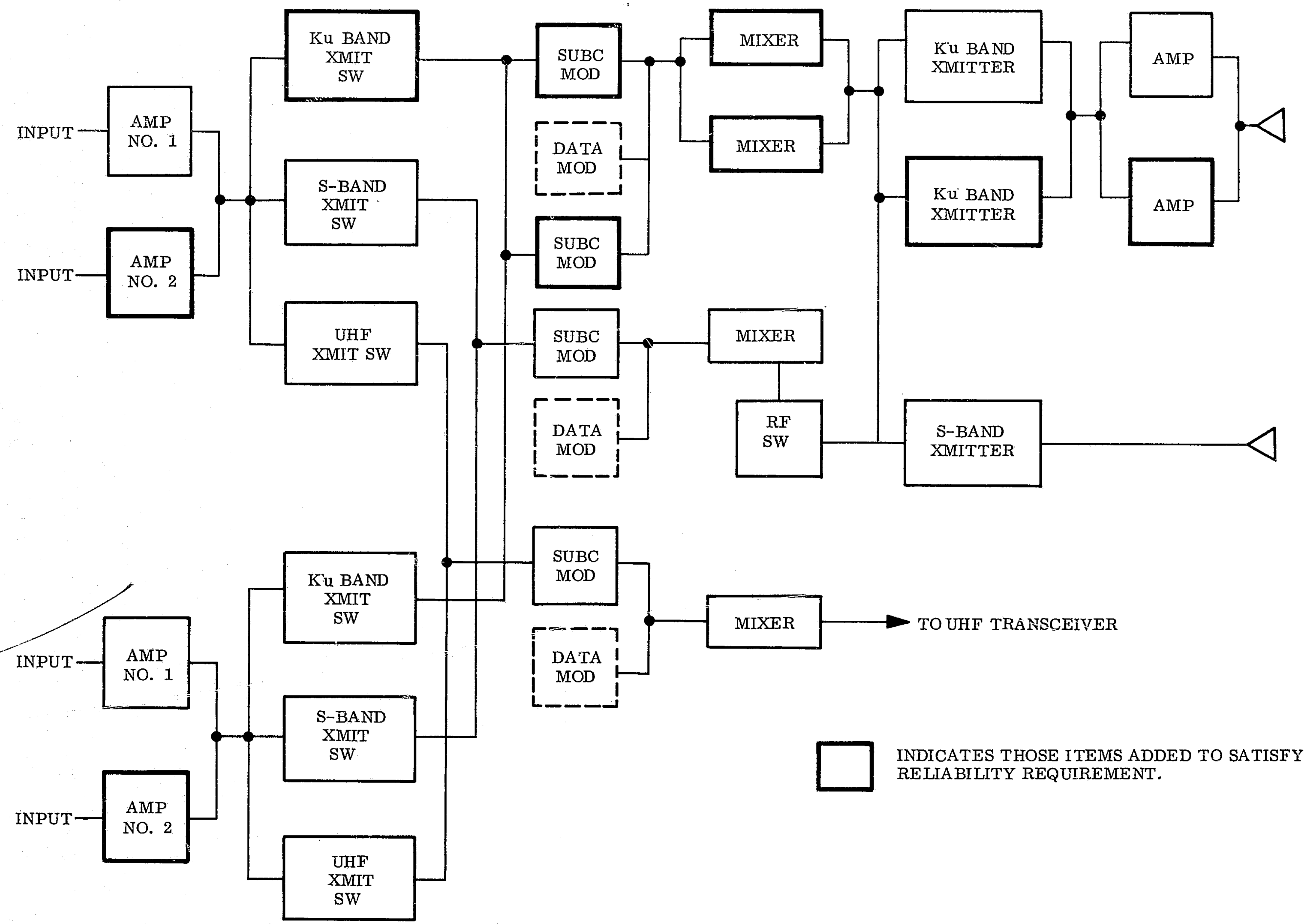


Fig. 4.11-2 Communications Subsystem Transmit Function

4.11.2 Guidance, Navigation, and Control Subsystem

Table 4.11-4 presents a summary of the additional weight and power required to satisfy the reliability criteria of fail operational-fail operational-fail safe for electronic equipment, and fail operational-fail safe for non-electronic equipment.

The following assumptions were made to arrive at the values tabulated:

- The reliability models are those shown in Figs. 4.11-3 through 4.11-8.
- To establish the power required, it was assumed that if more than two pieces of like equipment were required, two would operate active redundant and the remainder would be brought on as a failure occurred, except in the case of the horizon sensor and star sensor, where it was assumed that the primary unit could be shut down and the back-up brought on.
- The additional weights for equipment that would be required in the control display subsystem to satisfy the reliability criteria for the vehicle control and guidance/navigation subsystems were not included. It was assumed that these additional weights would be added into the control/display subsystem incremental weights.

Table 4.11-4 Reliability Impact on Weight and Power - Guidance, Navigation, and Control Subsystem

Additional Equipment	(1)	Boost Wt	Boost Pwr	Orbit Wt	Orbit Pwr	Rendezvous Docking Wt	Rendezvous Docking Pwr	Reentry Wt	Reentry Pwr	Approach and Landing Wt	Approach and Landing Pwr	Cruise and Self-Ferry Wt	Cruise and Self-Ferry Pwr
IMU (2,3)	2	76	125*										
Rate Gyro (2,3,4,5,6)	5	15	20*										
TVC Pitch Driver	5	7.5	125										
TVC Yaw Driver	5	7.5	125										
GNC Computer	3	118.2	126*										
Horizon Sensor	1			9.0	0**								
Star Sensor	1			9.5	0**								
Radar Altimeter	3			27.0	28								
RCS Driver	8			80	45								
Docking Sensor	1					26	32						
ACS Driver Left Elevon	2							4	20				
ACS Driver Right Elevon	2							4	20				
ACS Driver Left Rudder	2							4	20				
ACS Driver Right Rudder	2							4	20				
Temp Sensor Electronics	3							9	7.5				
Air Path Transducer	2									10	20		
Ground Approach Transponders	3									25.5	60		
Total Increments		224.2	521	125.5	73	26	32	25	87.5	35.5	80	0	0

	Weight (lb)	Power (watts)
Total Increment	436.2	739.5
Total Baseline	289.0	833.0
Total	725.2	1572.5

*Assumed that second unit operates active redundant and third brought on as needed.

**Assumed inactive redundant

(1) Number of additional items

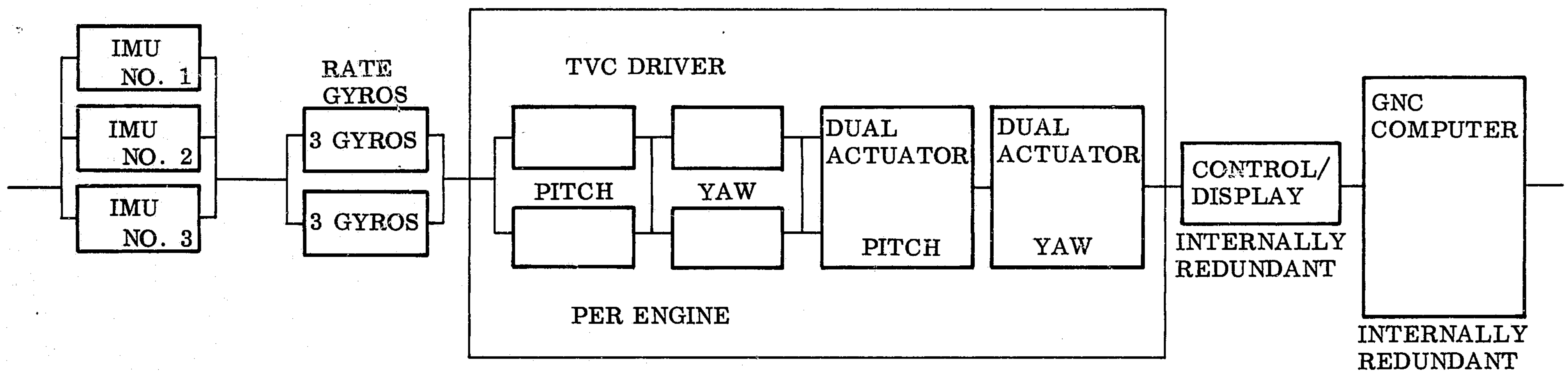


Fig. 4.11-3 GNC Reliability/Safety-Boost Phase

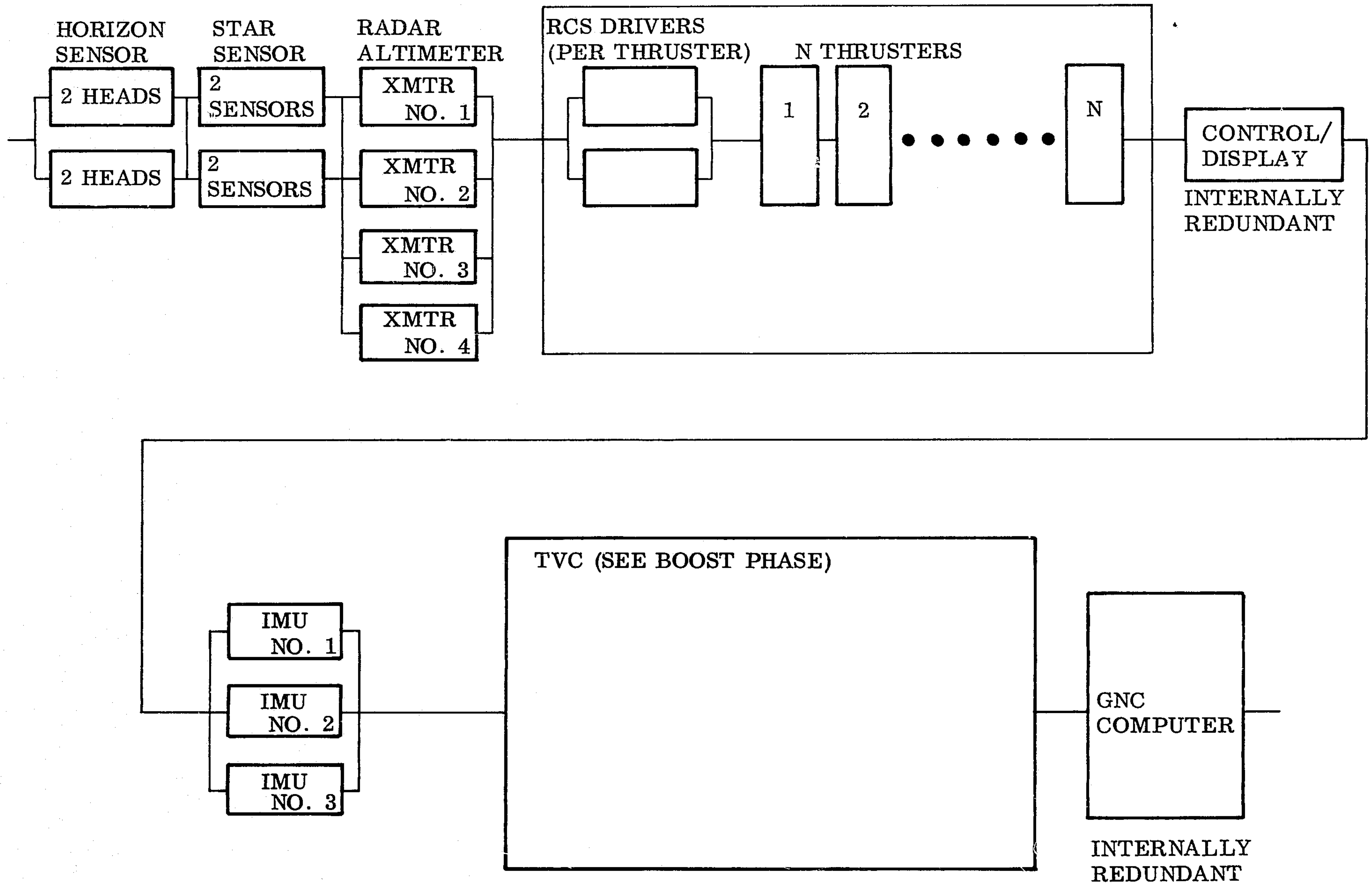


Fig. 4.11-4 GNC Reliability Safety-Orbit Phase

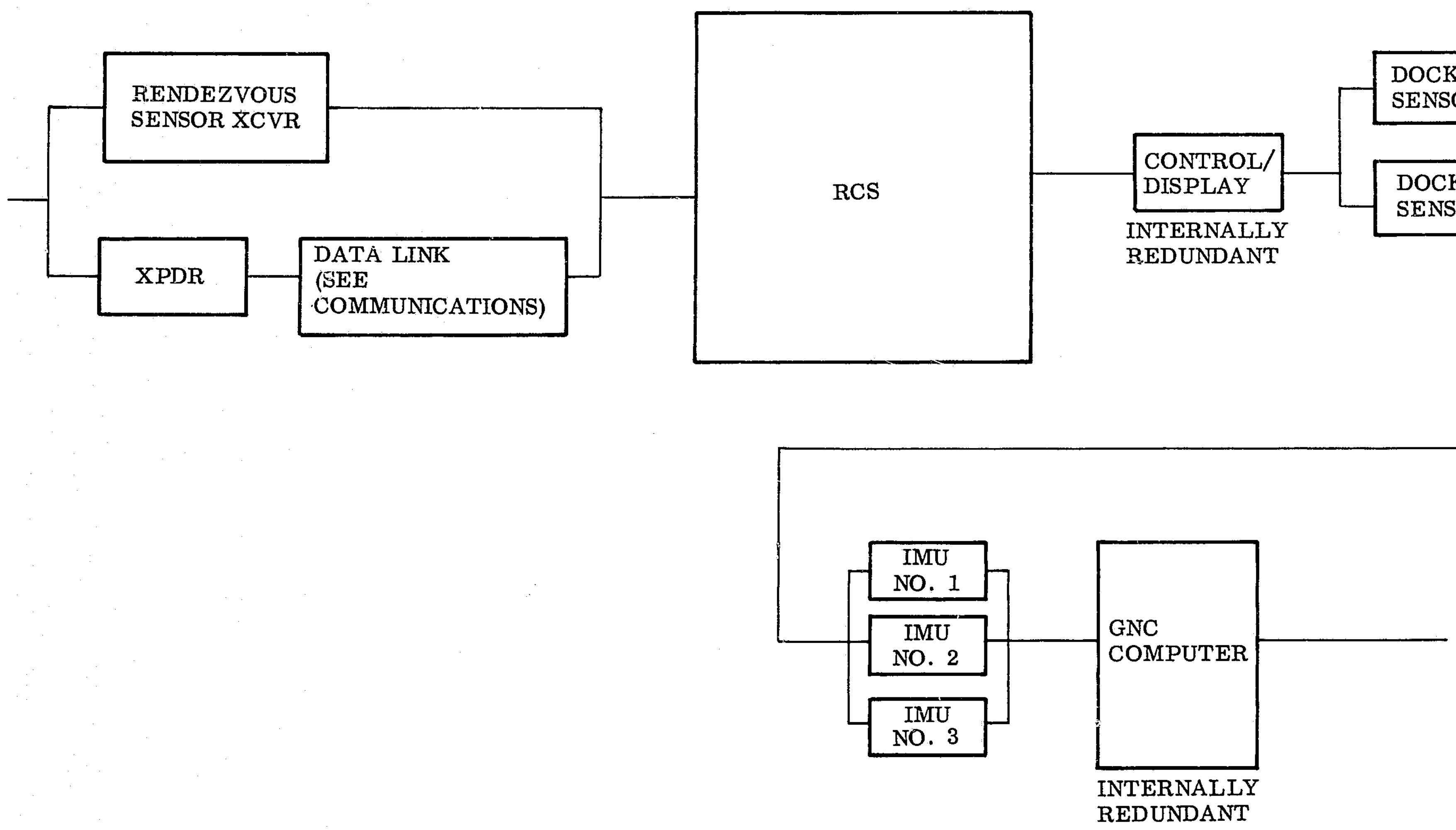


Fig. 4.11-5 GNC Reliability/Safety-Rendezvous and Docking Phase

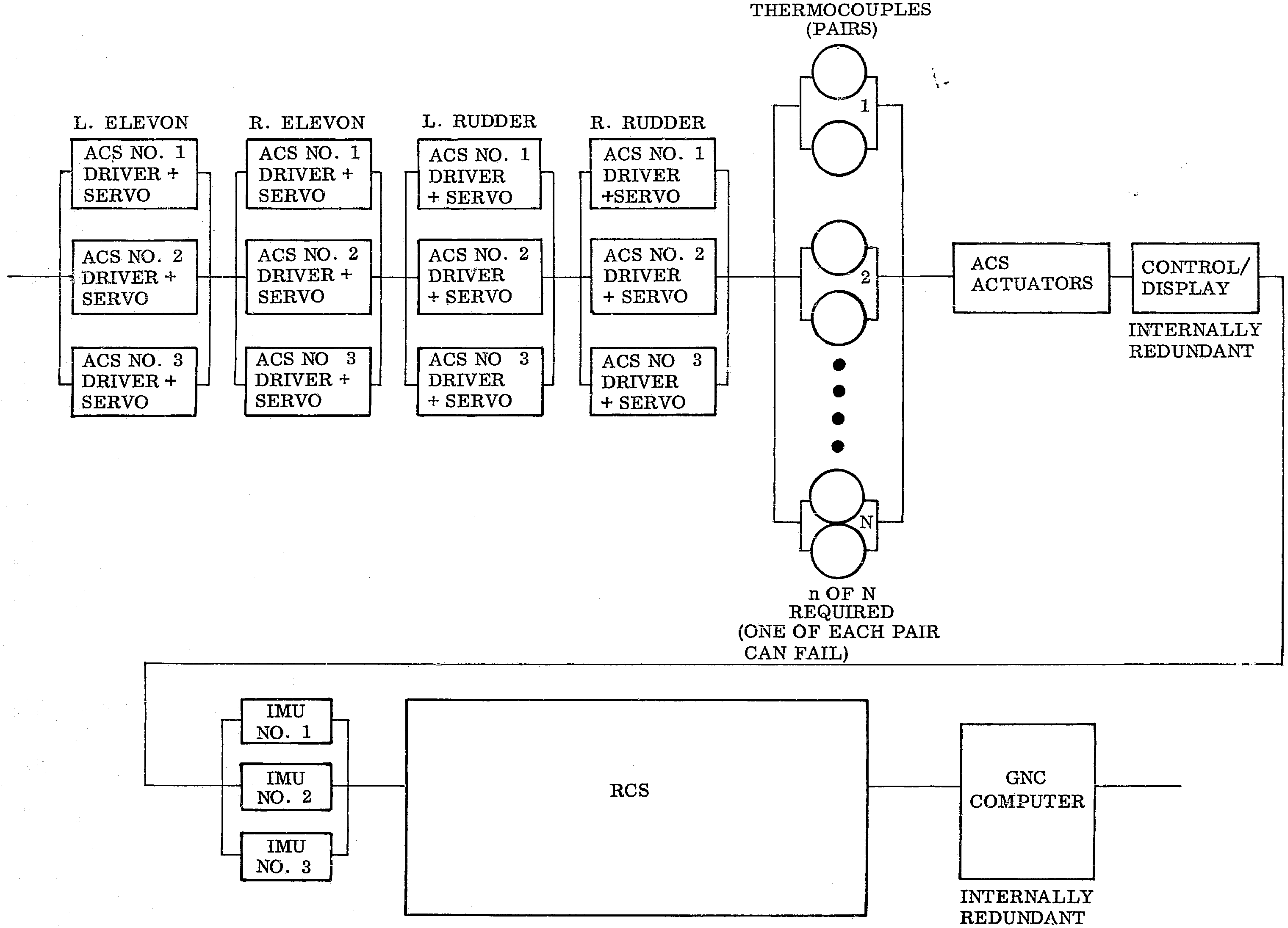


Fig. 4.11-6 GNC Reliability/Safety-Reentry Phase

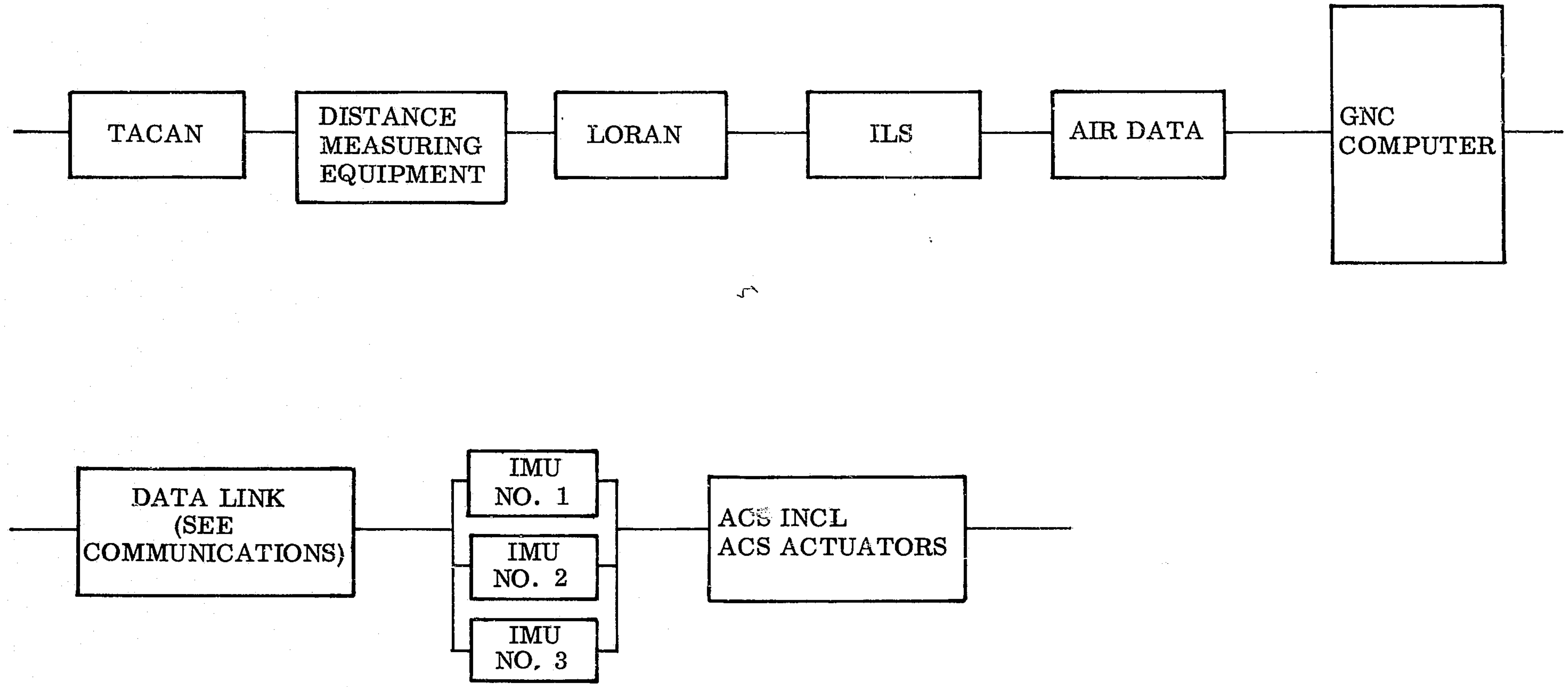


Fig. 4.11-7 GNC Reliability/Safety-Cruise and Self-Ferry Phase

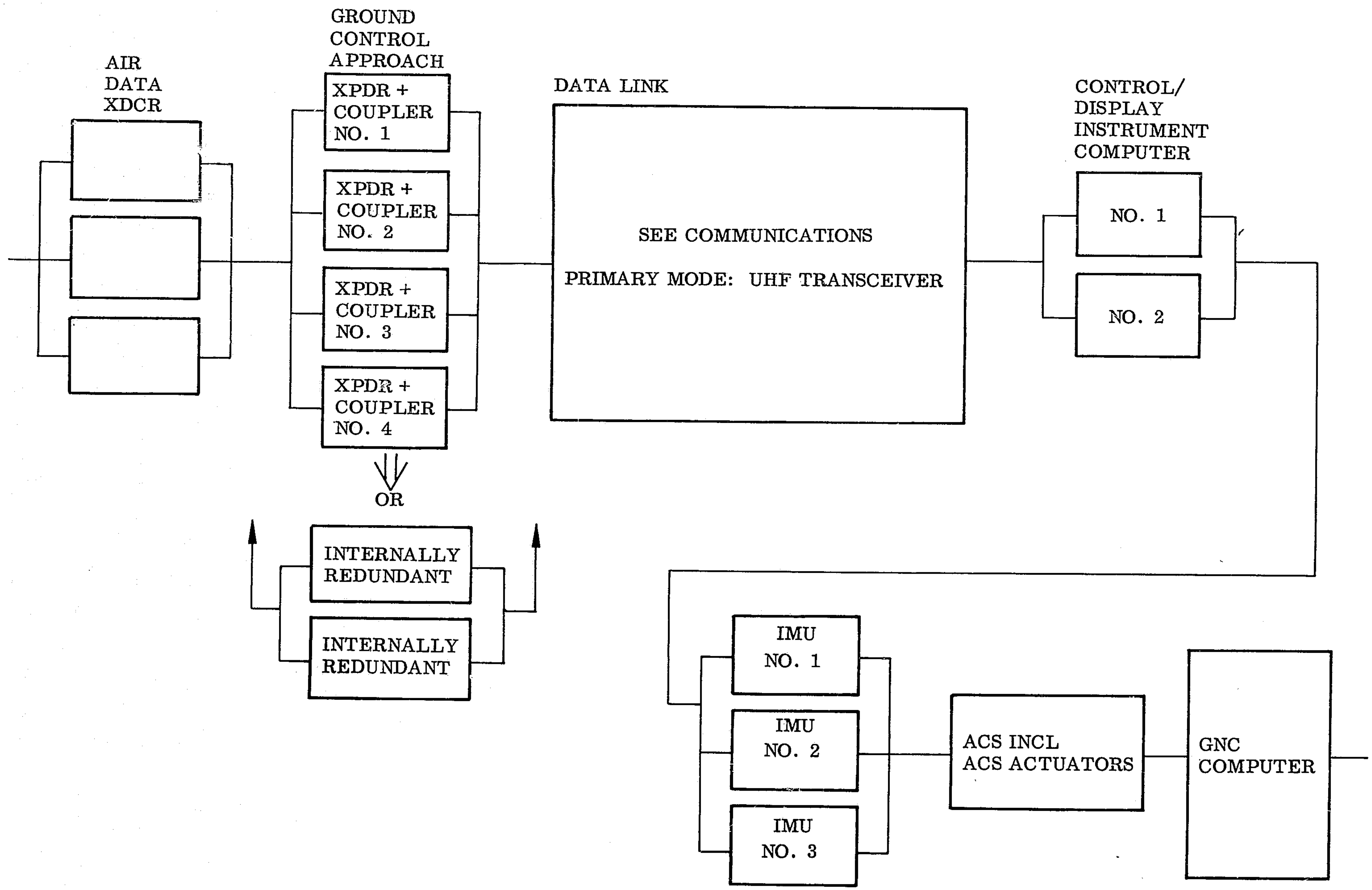


Fig. 4.11-8 GNC Reliability/Safety-Approach and Landing Phase

4.12 AVIONICS COMMONALITY

Though not part of the basic study requirements, differences in avionics between vehicle configurations and between orbiter and booster stages were to be flagged as time permitted. The Stage-and-a-Half vehicle configuration is included here since the study was initially oriented to the orbiter of that configuration.

4.12.1 Orbiter: Stage-and-a-Half, Two-Stage, Triamese

The orbiter avionics for the Stage-and-a-Half and Two-Stage vehicle configurations do not differ appreciably. This is quite reasonable since mission functions are the same and differences should be expected only where subsystems to be supported differ in their basic implementation. The Triamese orbiter avionics increment was not specifically tabulated but should be comparable to the Two-Stage orbiter avionics.

The principal avionics differences in going from the Stage-and-a-Half to the Two-Stage orbiter configuration occur because of changes in the propulsion subsystem: the number of main engines changes from five to two, the number of fuel tanks decreases from sixteen to nine, and the number of reaction control thrusters increases from fourteen 400-lb thrusters in three clusters to twelve 100-lb thrusters plus eighteen each throttleable thrusters. In addition, landing jet engines increase from two to four. Also, the requirement for wing deployment drops for the Two-Stage configuration.

4.12.1.1 Weight Increment. Summary of the weight increment of avionics equipment between One-and-a-Half-Stage and Two-Stage orbiter configurations are given in Table 4.12-1. The weight differences appear in equipment normally associated with subsystems other than data management and are not a function of the extent of integration of avionics.

Table 4.12-1

WEIGHT INCREMENT-ORBITER

Item	One-and-a-Half-Stage	Two-Stage	Increment for Two-Stage
1.0 Structure/mechanical subsystem wing deployment controller	10	0	-10
2.0 Propulsion			
Individual engine controller	5 ea @ 4	2 ea @ 4	-12
Landing engine controller	2 ea @ 6	4 ea @ 6	+12
4.0 Environmental control			0
5.0 Guidance/navigation			0
6.0 Vehicle control			
Primary engine throttle and gimbal drive electronics	5 ea @ 1.5	2 ea @ 1.5	-4.5
Reaction control valve and throttle driver	3 ea @ 10	3 ea @ 20	+30
7.0 Communications			0
8.0 Control/display			0
9.0 Data management (SAU)	(See Section 4.12.1.2)		-10
10.0 Instrumentation			
Structure/mechanical subsystem	160	150	-10
Propulsion subsystem	200	80	-120
Total increment	---	---	-124.5 lb

4.12.1.2 Data Handling Requirements. The number of data points for the Two-Stage orbiter decreases by approximately 100, as shown in Table 4.12-2. The number of signal acquisition units required decreases by two, for a weight decrement of 10 lb and a power decrement of 5 watts in the data acquisition portion of the data management subsystem. The impact on the data processing portion is insignificant.

4.12.2 Orbiter/Booster

No quantitative comparison of orbiter and booster avionics requirements was made. However, a cursory review indicates the following:

The electrical power subsystem for the booster will use primary batteries; the orbiter will use fuel cells and secondary batteries. Unmanned capability for the booster will increase the communications data link requirements. Guidance, navigation, and control requirements will decrease (no orbit, rendezvous and docking, and deorbit phases); and the star sensor, horizon sensor, rendezvous radar, and pulsed radar altimeter may be deleted, with a weight decrement of about 54 pounds. Also, the reduced number of mission phases for the booster will decrease the control/display requirement. It is not clear, however, that the panel will be significantly different; the difference may be primarily in the number of display formats and in the number of parameters displayed on a programmable CRT. Control and sequencing requirements for booster propulsion will increase because of the larger number of rocket engines on the booster. Additional data point access for an increased number of engines is easy to estimate.

The total number of test points required for the booster is expected to be on the order of 600 (based on subsystem level onboard checkout) versus 2000 for the orbiter single-thread system (based on unit level onboard checkout). Less in-flight monitoring for maintenance purposes is considered a reasonable approach for the booster, since the flight times are relatively short and the time on the ground for booster maintenance is considerably longer than for the orbiter. Weight savings on the booster through reduced onboard checkout and operations support capability should be thoroughly investigated and traded off with cost savings possible through orbiter/booster avionics commonality.

Table 4.12-2

DATA POINT INCREMENT

	One-and-a-Half-Stage	Two-Stage	Increment
Main Engines			
1.0 Structure/mechanical	50	20	-30
2.0 Propulsion	365	184	-181
5.0 Guidance/navigation	-	-	-
6.0 Vehicle control	93	40	-53
Reaction Control			
1.0 Structure/mechanical	22	33	11
2.0 Propulsion	130	192	62
5.0 Guidance/navigation	29	32	3
6.0 Vehicle control	28	96	68
Landing Engines			
1.0 Structure/mechanical	-	-	-
2.0 Propulsion	40	80	40
5.0 Guidance/navigation	-	-	-
6.0 Vehicle control	-	-	-
Wing Deployment			
1.0 Structure/mechanical	21	0	-21
Total increment			-101

Commonality in hardware design between the orbiter and booster elements is an important maintenance and checkout consideration. Commonality is probably more important to the autonomous onboard requirements than to the ground-based elements of the system.

Wherever common units can be installed in the flight vehicles, one-time development costs will exist for the following:

- Signal conditioning
- BITE design
- Subroutine definition
  - Checkout
  - Fault isolation
  - Configuration control
- Failure mode analysis
- Trend data base
- Limit definition
- Transducer development

The accumulative effect of a decision to maintain commonality between the orbiter and booster could potentially amount to 15 to 25 percent of the acquisition cost over completely differing designs. A continuing benefit over the life of the system is expected from reduced inventory and associated logistic problems. In view of airplane experiences of up to 10 to 1 ratios of operational life cost to acquisition cost, this latter consideration may well be the more important consideration.

4.13 CONCLUSIONS AND RECOMMENDATIONS FOR FURTHER STUDY

4.13.1 Conclusions

Weight Reduction. Significant weight savings are possible through use of multiplexed data buses and associated standard interfaces. A cable weight decrease of 982 lb was determined in going from Alternative 1 to Alternative 2. In addition, even more significant weight savings should be possible in the area of power distribution cabling by careful attention to the placement of power sources and switches and to the control of the distributed power. This latter point was not treated in this study. Weight reduction through elimination of distributed computers is evidenced by comparison of Alternatives 2 and 3.

Reliability. A redundantly configured system must be studied to assess the true impact of the reliability/safety requirements.

The total data management requirement for redundantly configured subsystems will increase appreciably but probably by not more than a factor of 2.5. Alternative 3 promises advantages in weight saving through use of software techniques to restructure the system on a priority basis to achieve graceful degradation. Software versus hardware tradeoffs were not performed in this study, so no conclusion may be reached as to possible weight savings.

Technical Risk. Electronics component technology of 1972 will be sufficient to support an Integrated Electronics System design. Development of complex software for executive control of a completely centralized system would be required. No significant technical risk exists for Alternative 2.

Flexibility. The use of standard multiplexed data bus and standard interface designs will permit the modification of individual pieces of equipment serviced by the bus without impacting the rest of the system. Also, the addition of more equipment to be serviced by the bus is possible; but, if this is anticipated during the initial design, additional storage for routines should be provided. These routines could also be developed initially and incorporated into the initial design. Additional flexibility for incorporating design parameter changes exists, primarily in Alternative 3 where software replaces some hard-wired mechanizations. Alternative 2 could also be modified for this capability. The use of BITE restricts the flexibility to incorporate changes of limits or other comparison bases.

Data Acquisition/Distribution. Interface and bus data transfer rates were not a problem for the avionics system studied. A redundant avionics system is not expected to cause a problem unless the interfaces are moved further into the black boxes to reduce the amount of signal processing electronics in the black boxes. Standard interface and standard multiplexed data bus designs are practical and are being demonstrated today. They should represent no significant problem in 1972. Grouping for local control by subsystems or at geographical locations in the vehicle reduces the need for intercommunication and consequently reduces data-rate handling requirements.

Self-Test and Warning. A total BITE concept embodying distributed built-in test equipment plus centralized processing to monitor and evaluate overall performance is recommended. Responsibility for failure and/or status reporting should be assigned to the subsystem or black box. The use of prime data validation by a central processor that has access to all vehicle system data will enhance the performance of integrity checks on individual equipment. Test point access requirements for abort warning are not additional to the basic test-point requirements, but higher sampling rates may be anticipated for some abort warning parameter test points.

Commonality. The Integrated Electronics System configuration is not impacted by the choice of orbiter vehicle configuration. Design requirements for a booster Integrated Electronics System will be less than for the orbiter but have not been evaluated in this study.

4.13.2 Recommendations for Further Study. A more extensive study should be performed for a redundantly configured set of vehicle subsystems. The scope of the integration problem studied should be expanded to include:

- Integration of multiple functions into one black box
- More detailed level of box definition and interface control requirements
- Extension of data interfaces as far as possible into each box where signal processing electronics may be integrated
- High data rates internal to actuator or effector closed loops
- Digitized voice channels
- Interface between orbiter and booster avionics and integrated electronics systems
- Interface between the orbiter Integrated Electronics System and the payload or cargo handling equipment
- Control of power distribution

Configuration control and sequencing requirements for each subsystem should be investigated in detail.

Software requirements for a centralized system should be established and a tradeoff study of software versus hardware should be performed.

The need for dedicated displays versus the capability of programmable displays should be investigated. The use of dedicated wires versus multiplexed buses for critical display and control parameters must be evaluated.

CONTENTS

Section 5
SPECIAL SUBSONIC FLIGHT OPERATIONS

5.1 APPROACH AND LANDING

5.1.1 Study Scope, Requirements, and Criteria

Under the approach and landing special emphasis task (8c), the critical aspects and problems of approach and landing of the Space Shuttle have been investigated. Flight dynamics of candidate vehicle configurations have been analyzed and preliminary landing patterns and design requirements established. Guidance and control requirements for all weather operation have been identified and control systems for approach and landing have been investigated. Results of control simulations (performed elsewhere) have been obtained and are applied to the LMSC vehicle.

The requirements and criteria applied in this study were derived from various customer and contractor sources. For the most part, they have not been identified as fixed and firm, but rather are "desired characteristics" and "present opinion." This approach of not fixing final requirements certainly seems appropriate at this early stage, and considerable effort in this task has been directed toward identifying approach and landing requirements. The list below specifies some of the more general requirements and criteria used in this study. Specific requirements are identified in the appropriate section.

- Horizontal landing vehicle, operating "like airliner"
- All-weather automatic landing system, plus capability for piloted landing
- Satisfactory vehicle flying and handling qualities
- Capability for one goaround
- Land on 10,000-foot runways

5.1.2 Flight Dynamics

Described in this section are the aerodynamic and flight performance aspects of approach and landing and tradeoffs that have led to the present vehicle design and operational concepts.

5.1.2.1 Approach and Landing Concepts. The desired operational requirements and flexibility of the Space Shuttle require that it possess many of the characteristics of airline and transport type aircraft. The following discussion presents the preliminary flight profile and restrictions from a 100,000-foot altitude (chosen as the end of the reentry phase) to the runway landing. Both flight mechanics and operational considerations are included.

The operational requirements on approach and landing include the following:

- Energy management for navigation and range control - The objective is to approach the landing field at the proper altitude, airspeed, and heading and to touch down on the runway at the proper location and velocities.
- Precision control throughout the landing phase - This is accomplished by altitude and airspeed control techniques comparable to those used in present aircraft instrument landings.
- Starting of jet engines for landing and go-around - The approach and landing profile must be designed so that a safe landing can be accomplished if the engines cannot be started.

Figures 5.1-1 and 5.1-2 illustrate the present concepts on approach and landing dynamics. These concepts are discussed in the following paragraphs, in which powered and unpowered landing profiles are also presented. Guidance requirements and guidance systems for approach and landing are discussed in section 5.1.3. For the entire approach and landing phase, automatic control is the primary mode, with pilot control possible at any time if desired.

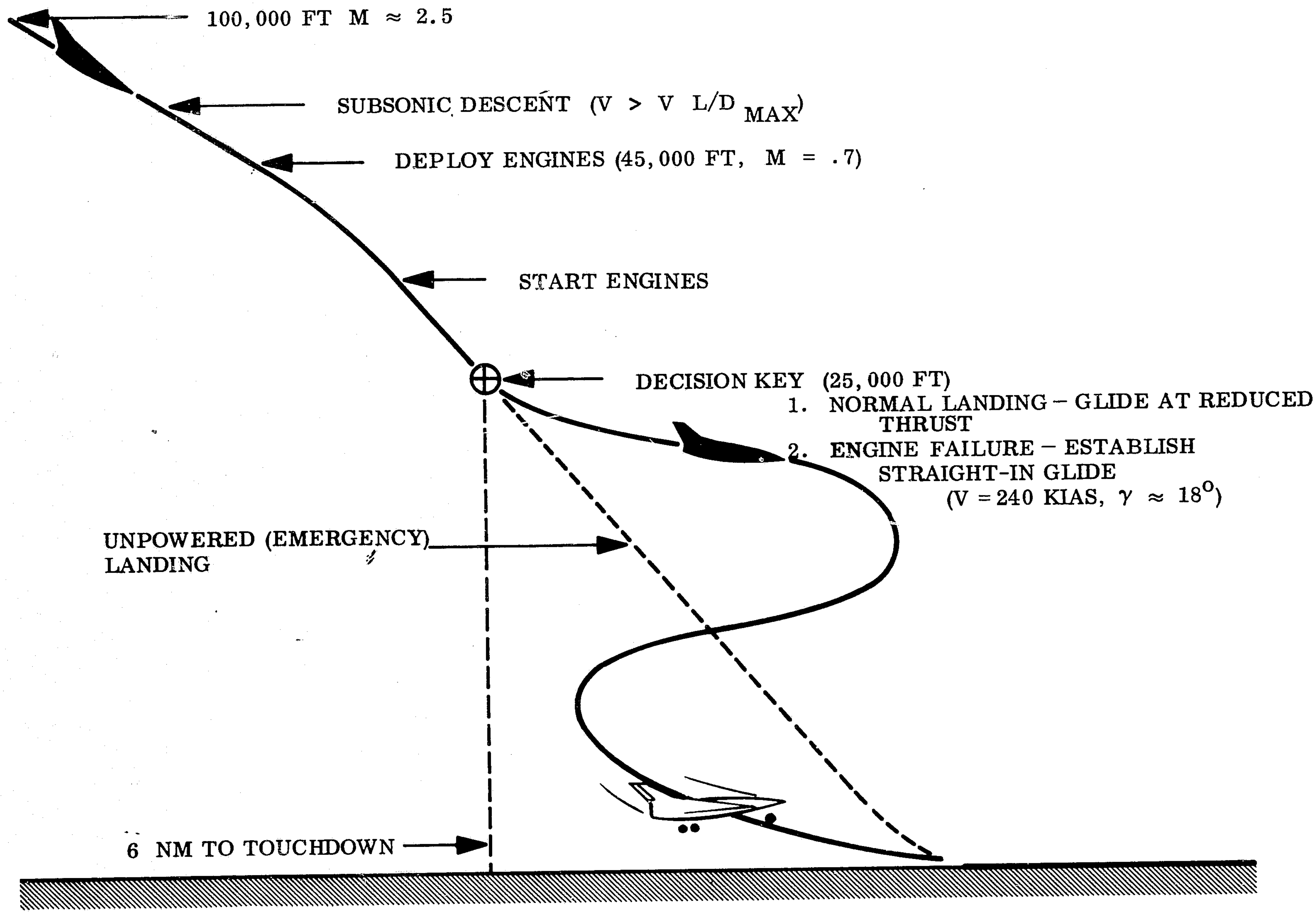


Fig. 5.1-1 TYPICAL ORBITER APPROACH

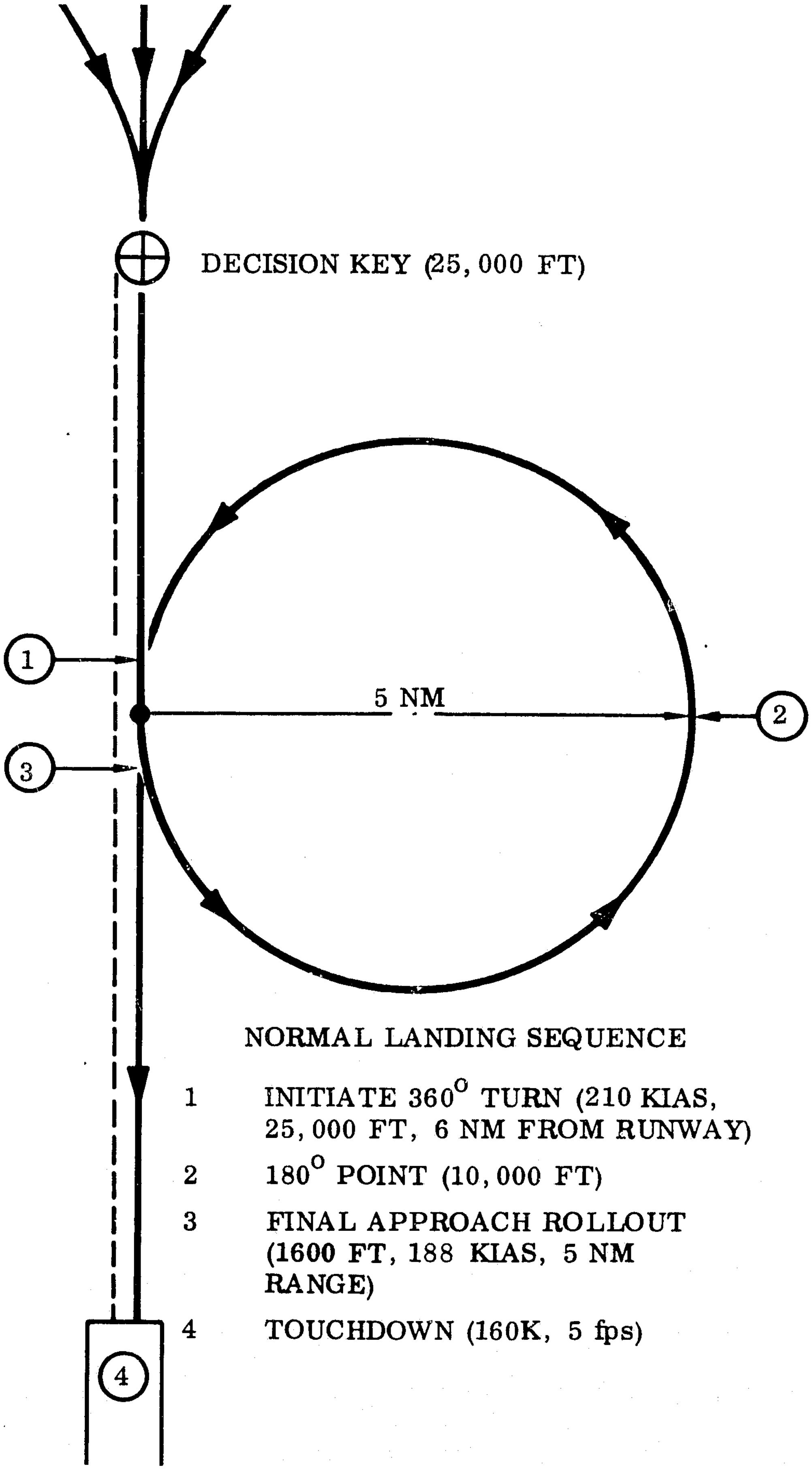


Fig. 5.1-2 LANDING GROUND TRACK

5.1.2.2 Vehicle Characteristics. Figures 5.1-3 through 5.1-6 show the performance characteristics of the orbiter and booster vehicles discussed in this section. It should be noted that the curve of flight path angle (γ) vs angle of attack (α) is basically L/D function and that the velocity vs α curve is the C_L function.

5.1.2.3 Descent. At the 100,000-foot altitude, the orbiter is established on an unpowered maximum L/D glide at approximately M = 2.5. The reentry and approach guidance will bring the vehicle to the proper heading at the landing decision key point. The guidance, through energy management techniques, will also control ground track so that decision key is reached at the proper altitude and airspeed. As the orbiter decelerates through M = 0.8 (approximately 60,000-foot altitude), glide angle is adjusted to establish a subsonic glide speed somewhat higher than that for maximum L/D. This operation, on the front side of the L/D curve, gives an inherent glide path stability for airspeed and range control and provides a significant energy management capability. For example, to lengthen range, the glide is shallowed and the velocity decreased; the glide is then at a higher L/D. This higher L/D glide, which is at a lower drag level, provides the additional energy necessary for the longer range. For spacecraft presently under study, velocity nominally would be held at M = 0.7 until a dynamic pressure (q) of 205 psf (250 KIAS) is reached, at which time q would be held constant.

Engine deployment (extension of jet engines into airstream) would be commanded at a 45,000-foot altitude, with a corresponding increase in glide angle to compensate for the increased drag. Engine start would be initiated immediately after deployment, and engines normally would be thrusting (at idle setting) by 35,000 feet.

Decision key, the next sequential event, is a critical point, since, if the engines have not started by this time (approximately 2 minutes after deployment), the spacecraft is committed to an unpowered landing. The rate of descent is quite high, and there is not time to attempt further starts if the

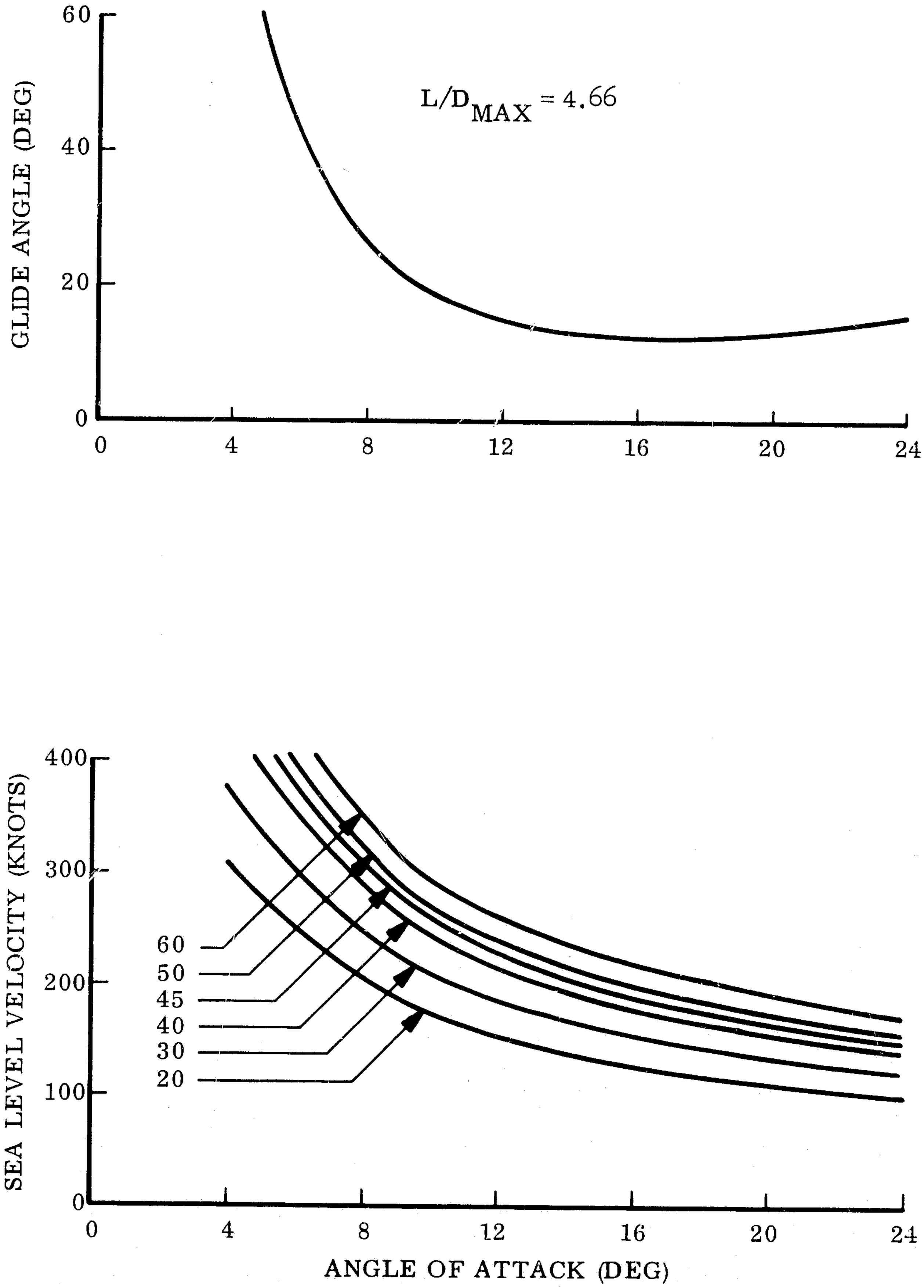


Fig. 5.1-3 ORBITER AERO CHARACTERISTICS (no thrust)

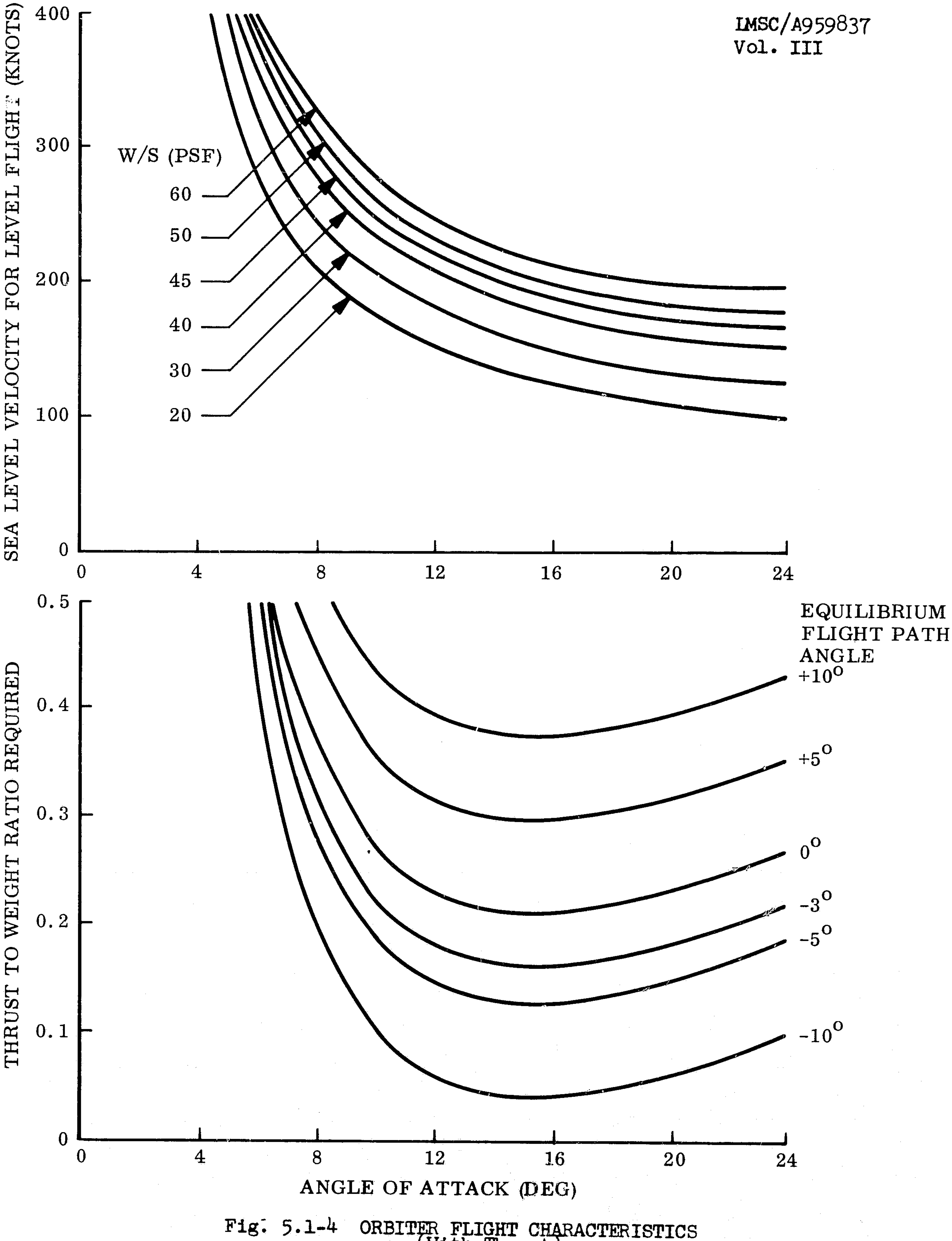


Fig. 5.1-4 ORBITER FLIGHT CHARACTERISTICS (With Thrust)

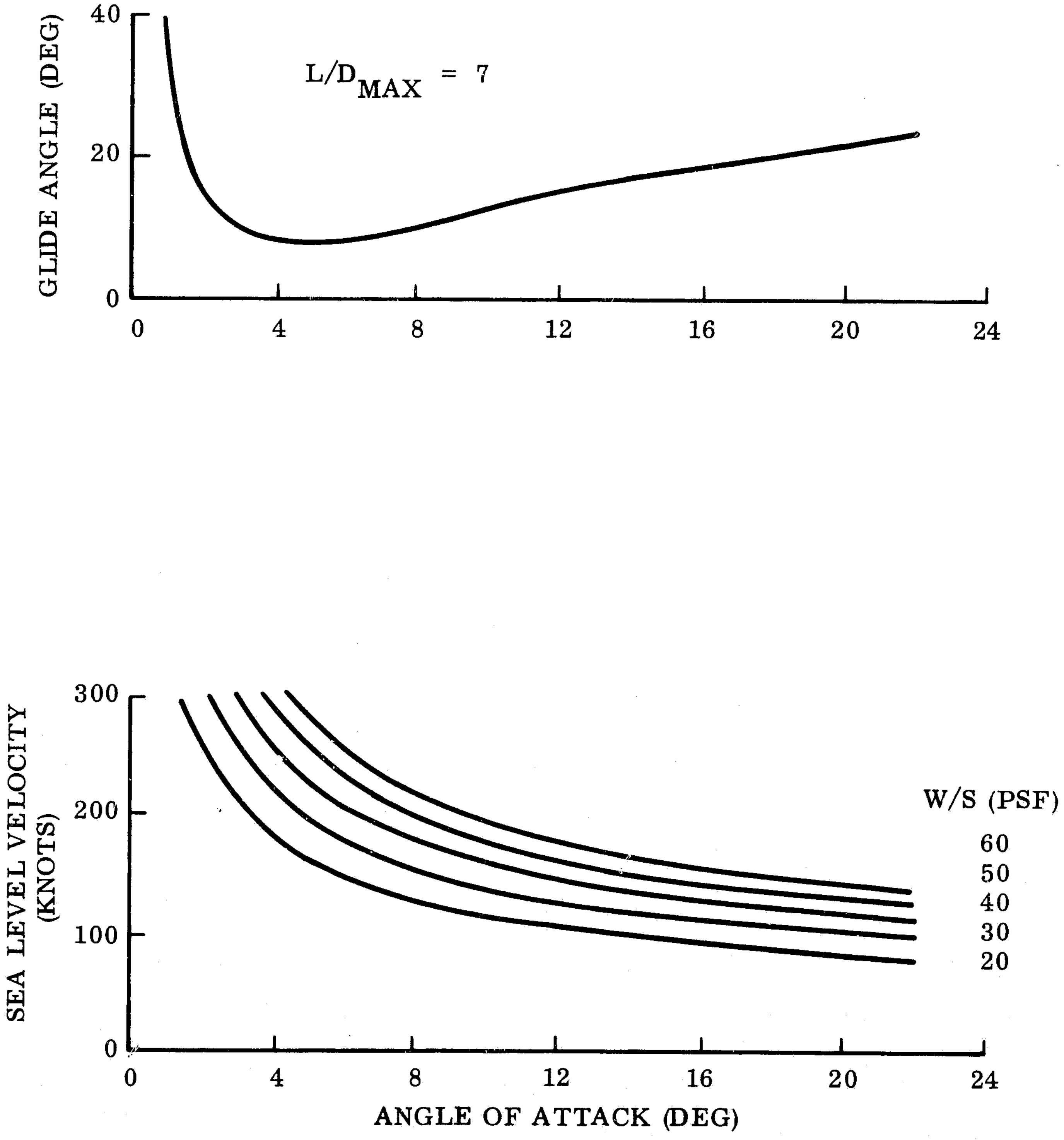


Fig. 5.1-5 BOOSTER AERO CHARACTERISTICS (No Thrust)

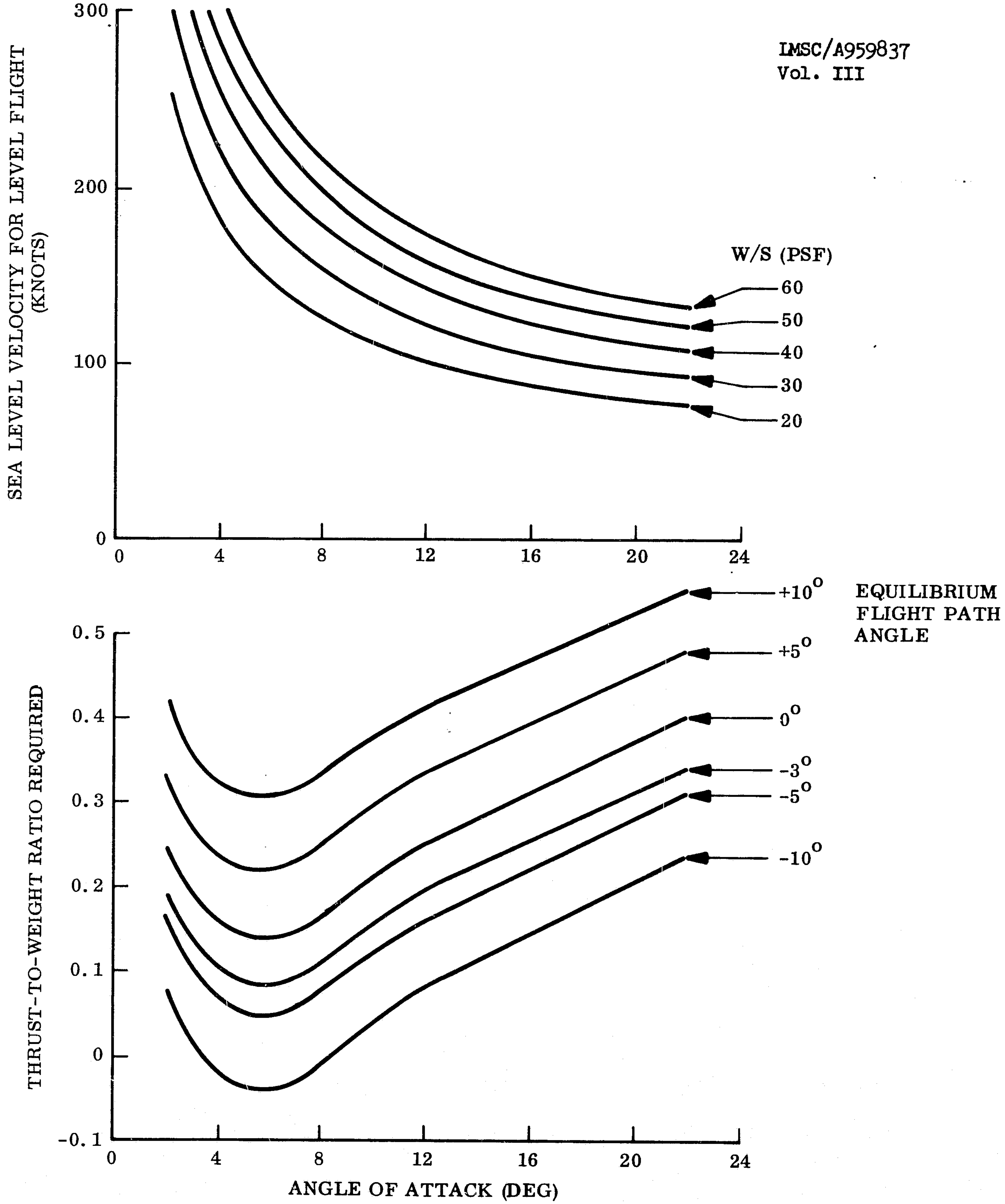


Fig. 5.1-6 BOOSTER FLIGHT CHARACTERISTICS (With thrust)

ability to perform an unpowered (emergency) landing is to be retained. Presently, decision key has been established at a 25,000-foot altitude. This is a conservative figure, and further analysis may show that this level can be lowered. Glide time from an engine start altitude of 40,000 feet down to 25,000 feet is on the order of 90 seconds, which is more than sufficient for several airstart sequences. The decision key must be at a proper ground range to perform the unpowered landing; the nominal range to touchdown for present spacecraft is 6 nm.

Figure 5.1-2 shows the spacecraft aligned with the runway at decision key; this requirement is not firm and other possibilities will be evaluated. For example, it may prove desirable to have the spacecraft aligned prior to decision key; or perhaps locating the decision key on a downwind or base leg (but still within unpowered glide range) would be preferable. With the computerized energy management and descent guidance system, it appears to be unlikely that a 360-degree gliding turn and high key point will be necessary. However, these features could be incorporated if desired.

5.1.2.4 Powered Landing. The nominal powered landing profile after decision key is shown in Fig. 5.1-1. Basically, this profile incorporates a 360-degree descending turn at moderate thrust levels, followed by a stabilized final approach path at a 3-degree angle and 5 nm in length. (The 3-degree and 5 nm figures are typical of those used in present aircraft instrument landings. Detailed analysis of this phase may result in modification of these values.) The final approach, landing flare, and touchdown follow standard large aircraft practice, wherein the specific constraints appropriate to large lifting-body spacecraft are observed.

At decision key, airspeed is reduced to approximately 210 KIAS by shallowing the glide. This airspeed will be chosen to enable modest radius turns while still providing sufficient load factor capability for turns and maneuvering. A moderate turn is initiated with the angle of attack near the L/D maximum point. Airspeed is maintained at 210 knots until near the end of the turn, at which time it is reduced to the final approach speed of 188 KIAS. Engine thrust is modulated throughout to provide an optimum altitude-time history and to meet the required conditions at rollout onto final approach. The point at which the 360-degree turn is concluded and the final approach is established is comparable to the present ILS outer marker. The aircraft is now configured for landing (landing gear extended, etc.) and established on the glideslope at the proper airspeed.

Figure 5.1-7 shows the final approach and touchdown profile. On final approach, the vehicle is steered and thrust is controlled so as to correct for position and velocity errors and to correct for winds and turbulence. A standard flare maneuver reduces vertical speed to that acceptable for touchdown. Thrust is reduced to idle just prior to touchdown, and touchdown is at 160K, with angle of attack slightly below the maximum usable C_L point. The derivation of approach and touchdown speeds is presented in 5.1.2.7, and rollout distances and runway length requirements are discussed in Section 5.1.2.8.

5.1.2.5 Unpowered Landing. The effect of failure of the jet engines to deploy or start is a significant failure mode. To perform a successful landing in the event of such failure requires the capability to land unpowered from the normal reentry and descent. To implement this capability, a decision key point has been defined. As described earlier, if the engines have not started at decision key, an unpowered landing will be performed. Automatic control is again the primary mode for flight control; however, the landing still can be performed by the pilot as is now done with X-15, HL-10, and other low L/D vehicles.

The unpowered landing approach follows closely the approaches successfully demonstrated on the X-15 and lifting bodies at NASA-FRC. An unpowered glide at 250 KIAS had been established prior to reaching decision key. After decision key, 240 knots airspeed will nominally be maintained down to the flare. This airspeed is on the "front side of the L/D curve," which provides a stable energy management regime as previously described.

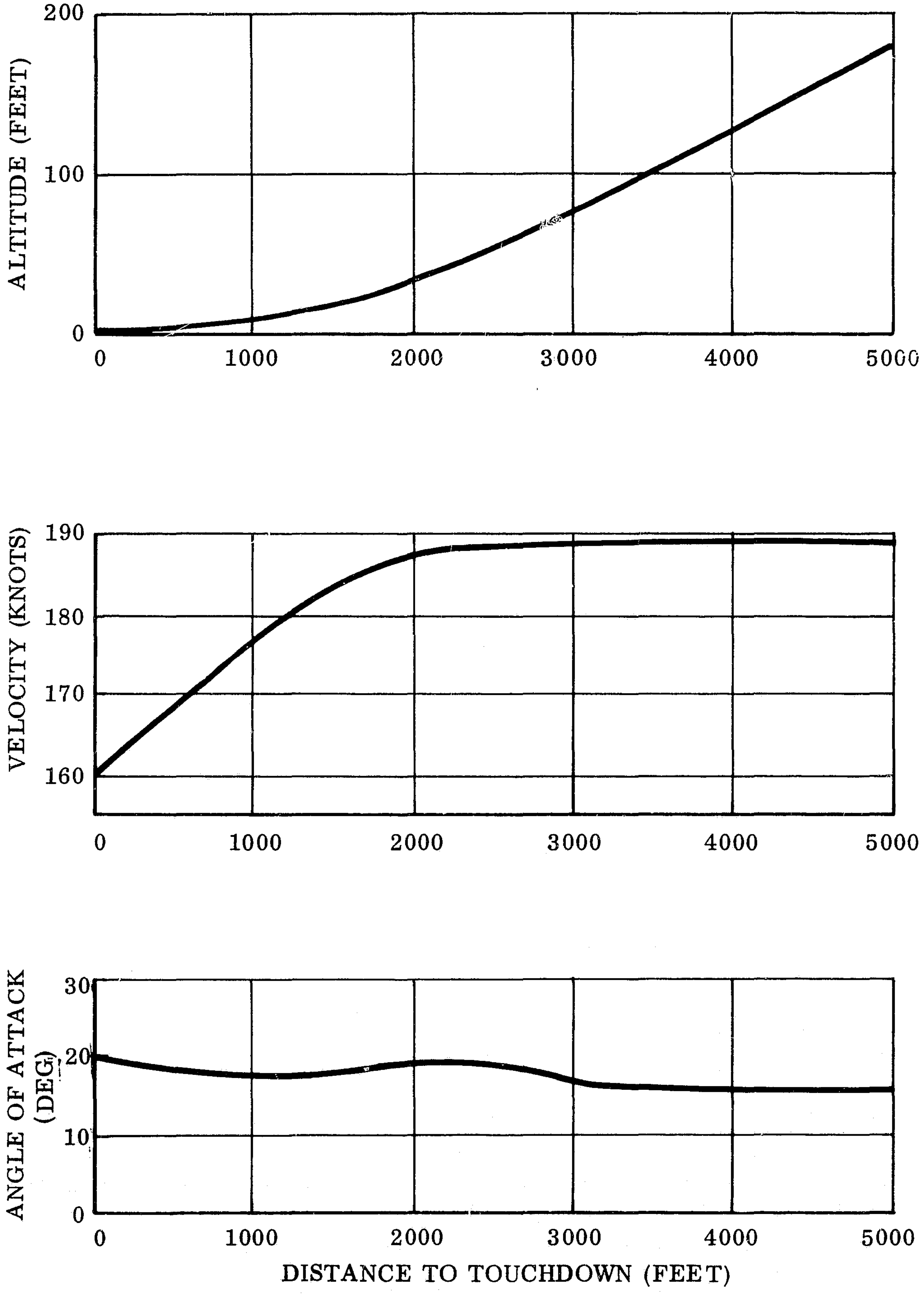


Fig. 5.1-7 ORBITER POWERED APPROACH PROFILES

Figure 5.1-8 shows a typical profile for spacecraft presently under study. The profile has been developed by computer simulation of the landing trajectory, integrated backwards in time from touchdown through float and flare to the approach glide. As established in Section 5.1.2.7, touchdown speed is slightly higher than for unpowered landing to provide an energy margin to avoid undershoot. The detail dynamics of unpowered landing have been discussed extensively in the open literature, primarily in connection with the X-15 and lifting body tests. The treatment here follows the standard approach with the addition of a slight flare just prior to touchdown to establish the touchdown rate of descent. Figures 5.1-9 and 5.1-10 show the effect on the profile of varying float time and flare g loading in the unpowered approach.

It is believed that a large lifting body, such as the Space Shuttle, can be landed successfully without power by using the technique described above.

Further investigations (including flight tests with low L/D aircraft) are recommended to increase confidence in the safety of unpowered landing.

5.1.2.6 Booster Landing. The preceding discussion is specifically oriented toward an orbiter vehicle returning from orbit. The booster (both Two-Stage and Triamese) will necessarily be approaching with engines already started and consequently will not require the unpowered landing approach pattern of the orbiter. Accordingly, the concepts of a decision key and rigid glide control are not required. It is anticipated that booster approach and landing will be similar to that in standard IFR aircraft practice. Approach requirements will be integrated with cruiseback requirements to develop an optimum booster return profile. Booster landing is similar to the spacecraft powered landing previously described, although the more favorable booster aerodynamics (lower wing loading, better stability, etc.) will probably ease control requirements.

Figure 5.1-11 shows typical booster landing profiles. These profiles are

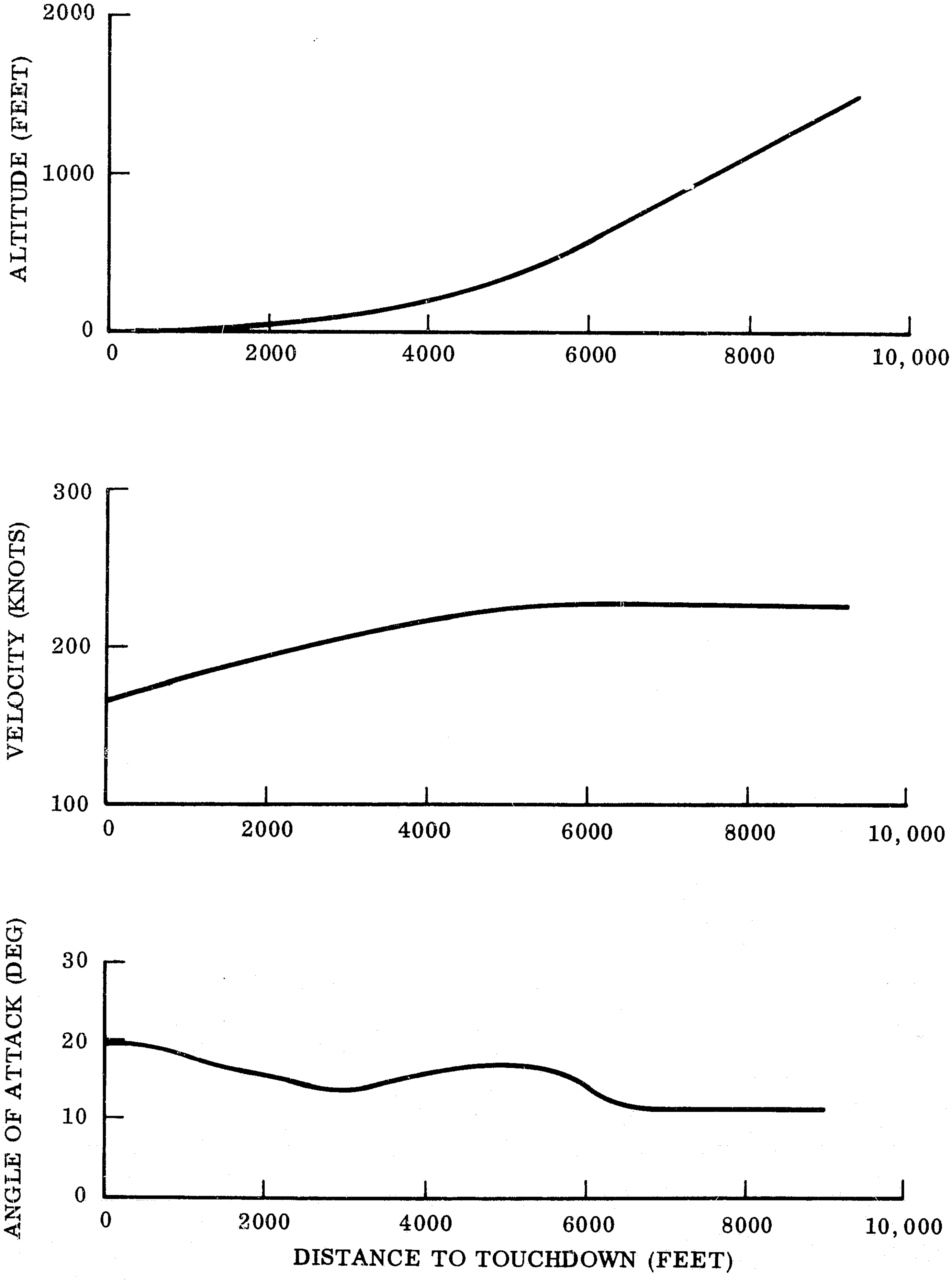


FIG. 5.1-8 TYPICAL ORBITER UNPOWERED APPROACH AND LANDING PROFILE

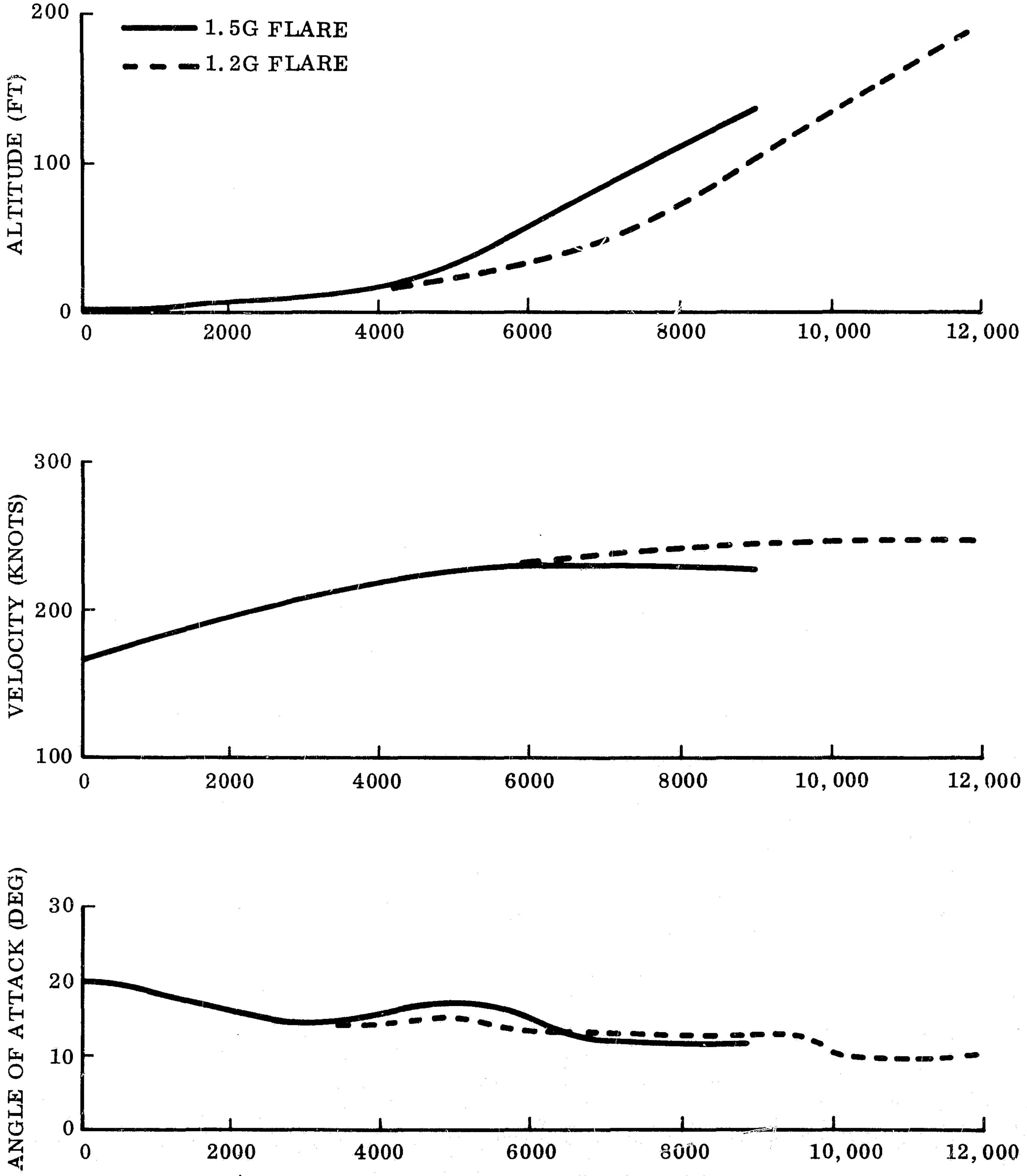


Fig. 5.1-9 EFFECT OF FLARE G LOADING ON UNPOWERED APPROACH

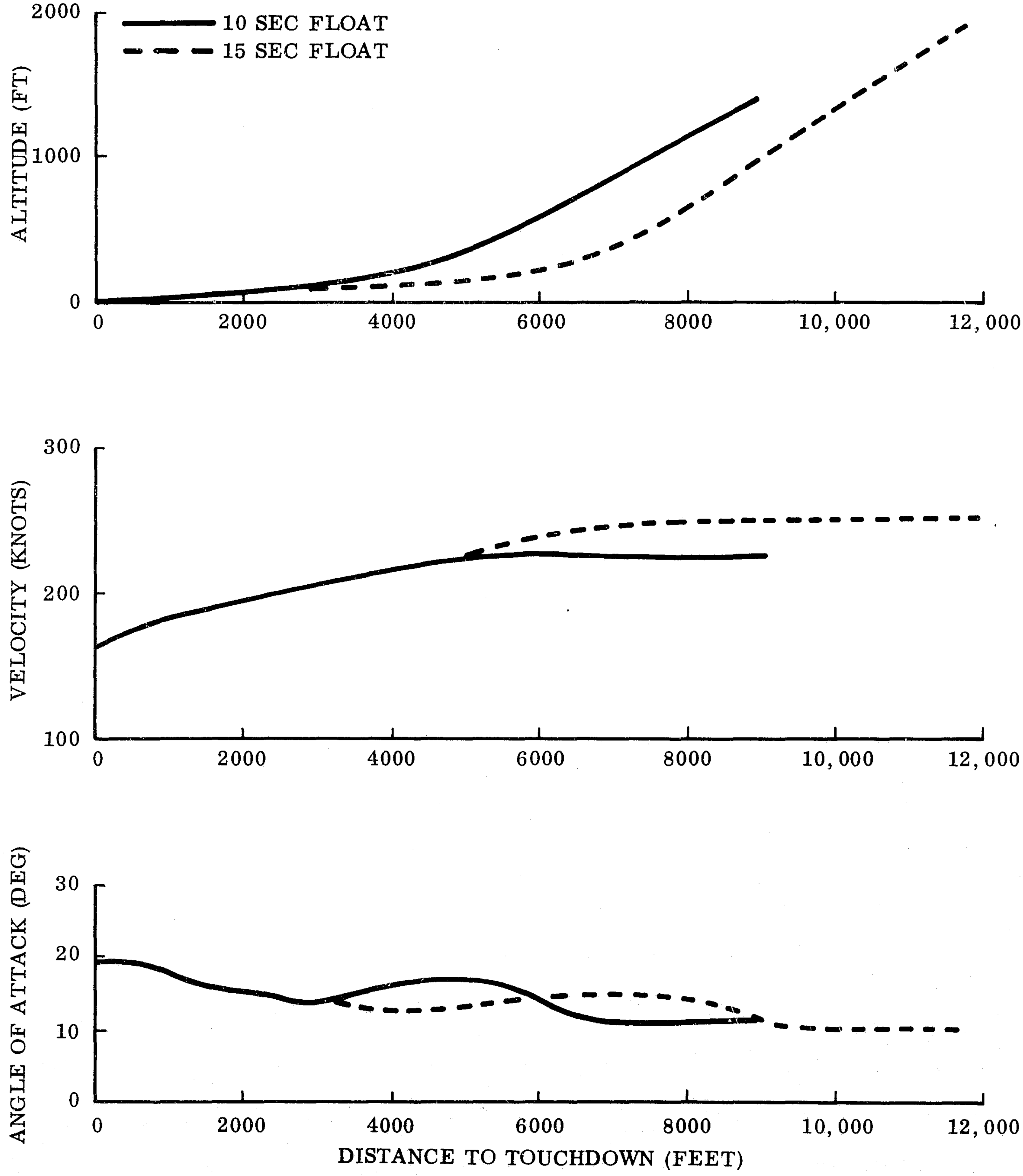


Fig. 5.1-10 EFFECT OF FLOAT TIME ON UNPOWERED APPROACH

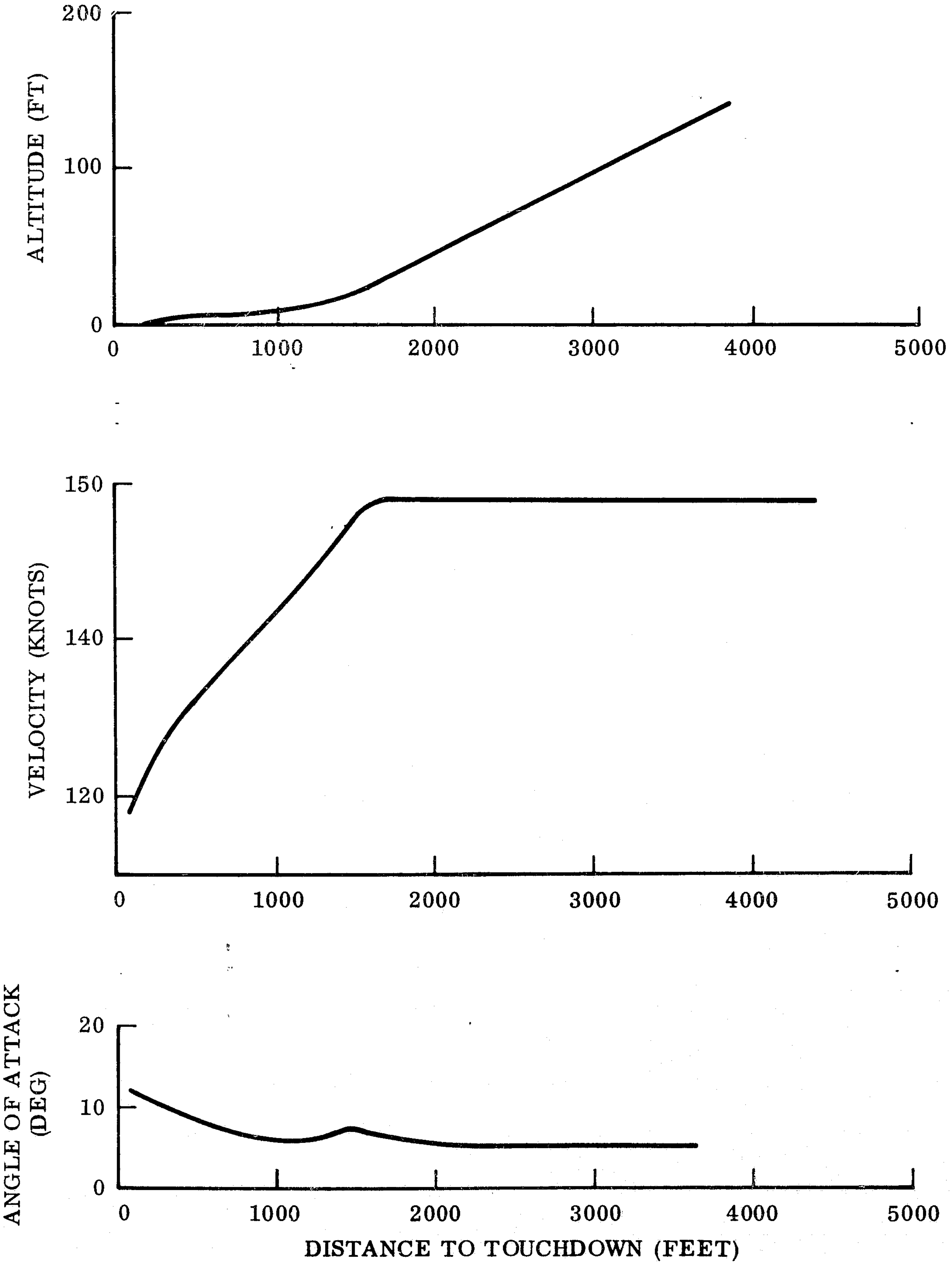


Fig. 5.1-11 BOOSTER POWERED APPROACH PROFILES

based upon Two-Stage booster aerodynamics. Aerodynamics for a Triamese booster were not available for this analysis; however, it is believed that they will be similar to the Two-Stage and will define essentially the same landing profiles.

5.1.2.7 Touchdown and Approach Speed Criteria. Touchdown and approach speeds used in the preceding paragraphs have been established according to published criteria for landing speeds. For touchdown, the criteria are those contained in the NASA Space Shuttle Task Group Report Volume II - Desired Systems Characteristics (June 12, 1969 revision). For approach, the criteria contained in Federal Aviation Regulation, Part 25, have been used. Touchdown and approach speeds for the present vehicles and the associated aerodynamic parameters are presented in Table 5.1-1.

Table 5.1-1
SUMMARY OF LANDING AND APPROACH SPEEDS

	Booster (2 Stage)	Orbiter
Wing Loading $\left(\frac{W}{S}\right)$	32 psf	45 psf (with 50,000-lb P/L)
Maximum usable α and C_L	14°/ .66	24°/ .613
Constraining Factor on α and C_L	Landing gear length	Lateral stability
Min V $\left[V_{Min} = \sqrt{\frac{2}{\rho o\ C_{LM}} \frac{W}{S}}\right]$	117 K	145 K
Powered Approach		
Nominal touchdown (NASA criteria) $\left[V_{TD} = 1.1\ V_{Min}\right]$	128 K	160 K
α at touchdown	13°	20°
Nominal approach (FAR criteria) $\left[V_{app} = 1.3\ V_{Min}\right]$	149 K	188 K
α on approach	9°	16°
Unpowered Approach		
Nominal touchdown (NASA criteria) $\left[V_{TD} = 1.15\ V_{Min}\right]$	--	167 K
α at touchdown	--	19°
Nominal approach	See Section 5.1.2.5	

The NASA touchdown speed criteria specifies the following:

$V_{TD} \leq$ 180 knots

$V_{TD} \geq$ 1.10 V min (powered landing)

$V_{TD} \geq$ 1.15 V min (unpowered landing)

V_{TD} = Nominal touchdown speed

V min = Minimum speed for flight

V min is defined by the max C_L that is available for use. Max C_L is determined by such factors as trim limits, stability and flying qualities limits, structural ground clearance limits, and other parameters that in general limit the maximum angle of attack. For the orbiter, α is limited to 24 degrees by the lateral stability parameter $C_{n\beta}$, which becomes negative past 24 degrees. For the boosters, α has been limited to 14 degrees by design of the landing gear. This enables sufficiently low-landing speeds so that no attempt has been made to increase the α limit. However, it will be possible to increase the usable C_L if desired in the future by lengthening the gear.

To minimize touchdown speeds and landing rollout distances, powered and unpowered touchdown speeds have been set equal to 110 percent and 115 percent of V_{min} for the powered and unpowered vehicles, respectively. It is believed that these factors will give an adequate safety margin to ensure necessary control response and stability. From Table 5.1-1, it can be seen that all touchdown speeds are below the 180-knot upper limit.

In summary, the vehicles presently under study (and without variable geometry wings) are capable of touchdown speeds that satisfy the recognized landing safety criteria. The effect of these speeds on runway length requirements is discussed in 5.1.2.8. All vehicles are capable of normal operation from 10,000-foot runways, even with braking effectiveness reduced by wet runways. Tradeoffs associated with variable geometry wings are discussed in 5.1.2.10.

The powered approach speeds of Table 5.1-1 are based upon the formula (specified in FAR 25)

$$V_{app} = 1.3 \ V_{min}$$

The factor 1.3 establishes an airspeed margin sufficient to provide satisfactory controllability and load factor capability for maneuvering and turbulence on the final approach. Approach speeds for unpowered landing are not determined from aerodynamic considerations, but are developed in Section 5.1.2.5 from energy requirements to complete the approach pullout.

5.1.2.8 Rollout Distances and Runway Lengths. The length of runway required for landing depends upon several factors, including the touchdown point, the touchdown speed, and the vehicle deceleration capability. A preliminary analysis has been performed on rollout distances and runway requirements; conservative conclusions indicate that both the orbiter and the booster can comfortably operate from 10,000-foot runways.

Figure 5.1-12 shows minimum rollout distance requirements for both vehicles as a function of touchdown speed and deceleration rate. These distances have been developed from landing roll simulations, with and without thrust reversal, by using the criteria as noted on the graphs. A friction coefficient of 0.6 has been chosen for maximum braking on a dry surface. For wet runways, a recognized rule of thumb for rollout distance = 1.67 x (rollout distance on dry runway) has been applied.

The rollout distance requirements for the orbiter are tabulated in Table 5.1-2 on the basis of the landing speeds developed as discussed in Section 5.1.2.7. Dry and wet runway distances are tabulated for a nominal speed touchdown and a touchdown 15 knots faster than nominal. As shown, a normal minimum rollout landing (powered, with thrust reversal, nominal velocity) on a wet runway requires only 4800 feet, while a worst-on-worst case (unpowered emergency landing, 15 knots above nominal touchdown, wet runway) requires 5800 feet.

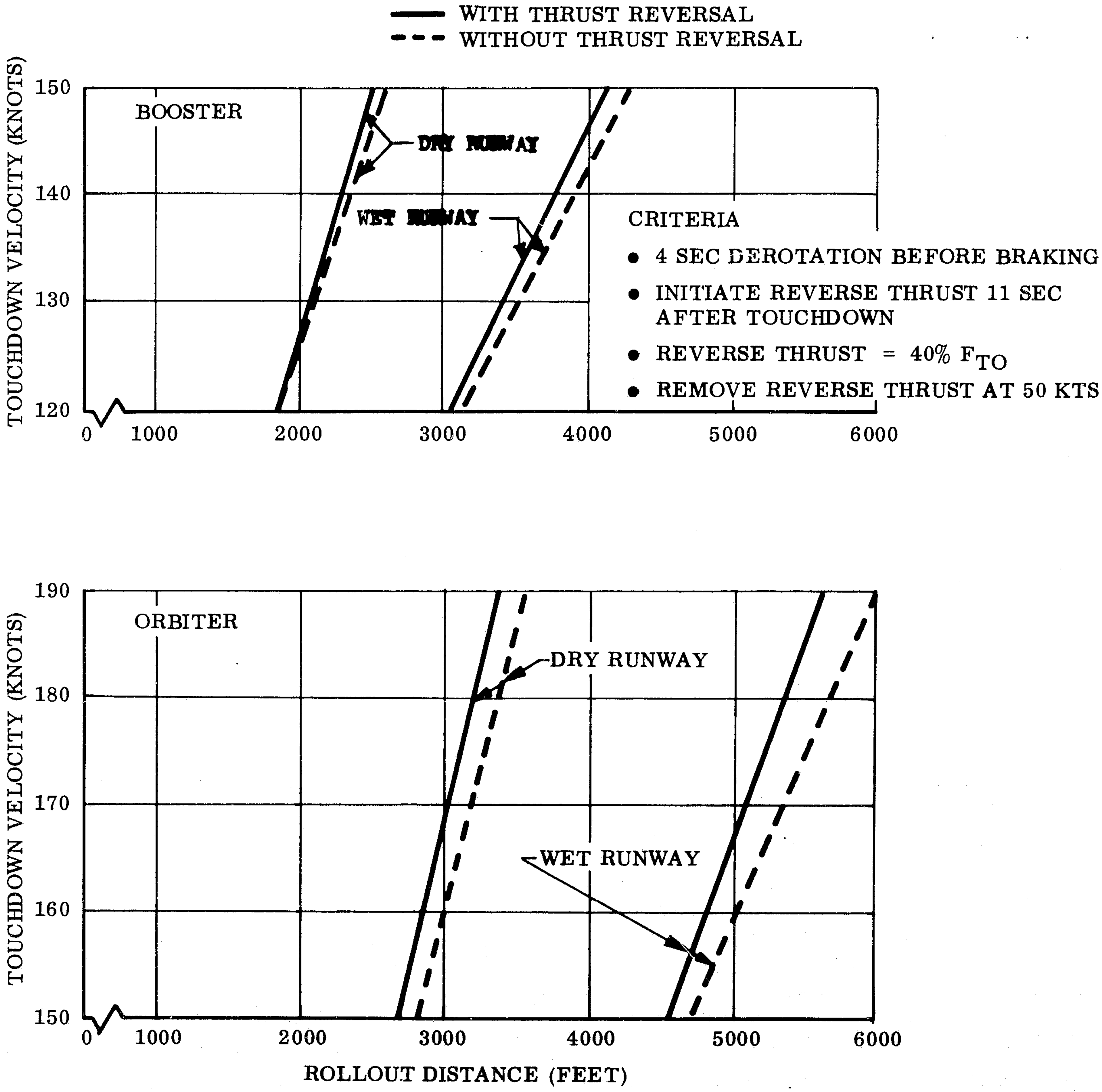


Fig. 5.1-12 MINIMUM ROLLOUT DISTANCES

TABLE 5.1-2

ORBITER VEHICLE ROLLOUT DISTANCES

	Powered Landing	Unpowered Landing
Nominal Touchdown Velocity		
Touchdown velocity (knots)	160	168
Rollout distance - dry runway (ft)	2900/3000	-/3200
Rollout distance - wet runway (ft)	4800/5050	-/5300
Touchdown 15 Knots Above Nominal		
Touchdown velocity (knots)	175	183
Rollout distance - dry runway (ft)	3200/3350	-/3500
Rollout distance - wet runway (ft)	5250/5550	-/5800

Note: X/X denotes with thrust reversal/without thrust reversal

A touchdown point 2000 feet from the end of the runway has been chosen as a worst case touchdown dispersion. This is believed to be conservative, since the FAA/USAF C-141 All-Weather Landing System program demonstrated touchdown dispersions within ± 500 feet 95 percent of the time; flight results from the unpowered X-15 and HL-10 landings at FRC-Edwards have been within a 1500 foot range. Adding this 2000-foot distance to the worst-case rollout results in a total runway length requirement of 7800 feet, well below the 10,000-foot criterion.

This conservative analysis, although preliminary, gives a high degree of confidence that the orbiter can be operated on a 10,000-foot runway. Further analysis performed to verify this conclusion may well show that the orbiter is capable of operation on 8,000-foot runways.

The orbiter runway requirement can be compared to that for the F-100C Century series jet fighter. The F-100C, which has a minimum touchdown speed of 155 knots, is regularly operated from 10,000-foot runways by operational pilots in all types of weather conditions.

For the booster, the lower touchdown speeds result in shorter rollouts, and thrust reversal is less effective in reducing rollout. Table 5.1-3 presents booster rollout distances for the various conditions, the worst-on-worst case here being 4,000 feet. Using the 2000-foot touchdown point results in a conservative runway requirement of 6000 feet.

TABLE 5.1-3
BOOSTER VEHICLE ROLLOUT DISTANCES
(Powered landing)

Nominal Touchdown Velocity	
Touchdown velocity (knots	128
Rollout distance - dry runway (ft)	2050/2100
Rollout distance - wet runway (ft)	3350/3450
Touchdown 15 Knots Above Nominal	
Touchdown velocity (knots)	143
Rollout distance - dry runway (ft)	2350/2450
Rollout distance - wet runway (ft)	3850/4000

X/X denotes with thrust reversal/without thrust reversal

5.1.2.9 Go-Around. A requirement exists for a capability to break off a landing approach prior to touchdown and return for another approach. In this section, performance and vehicle design requirements for go-around are presented and the need for go-around is discussed.

5.1.2.9.1 Go-Around Performance. Figure 5.1-13 shows a typical go-around pattern. Go-around is initiated by increasing thrust and establishing a climb to pattern altitude. The racetrack path brings the vehicle back to the

1 INCREASE THRUST AND INITIATE PULLUP

2 ESTABLISH MAX CLIMB AT MAX L/D

3 AT 1000 FT ALTITUDE, INITIATE 180° TURN AND INCREASE AIRSPEED TO 110% V L/D MAX

4 AT 2000 FT ALTITUDE, ESTABLISH LEVEL FLIGHT AT V = 110% V APPROACH

5 ROLL OUT OF TURN AT 180° POINT, ESTABLISH DOWNWIND LEG

6 AT 5 NM FROM TOUCHDOWN POINT INITIATE 180° TURN TO FINAL APPROACH

7 INTERCEPT GLIDE SLOPE AND RUNWAY BEARING WHEN APPROPRIATE

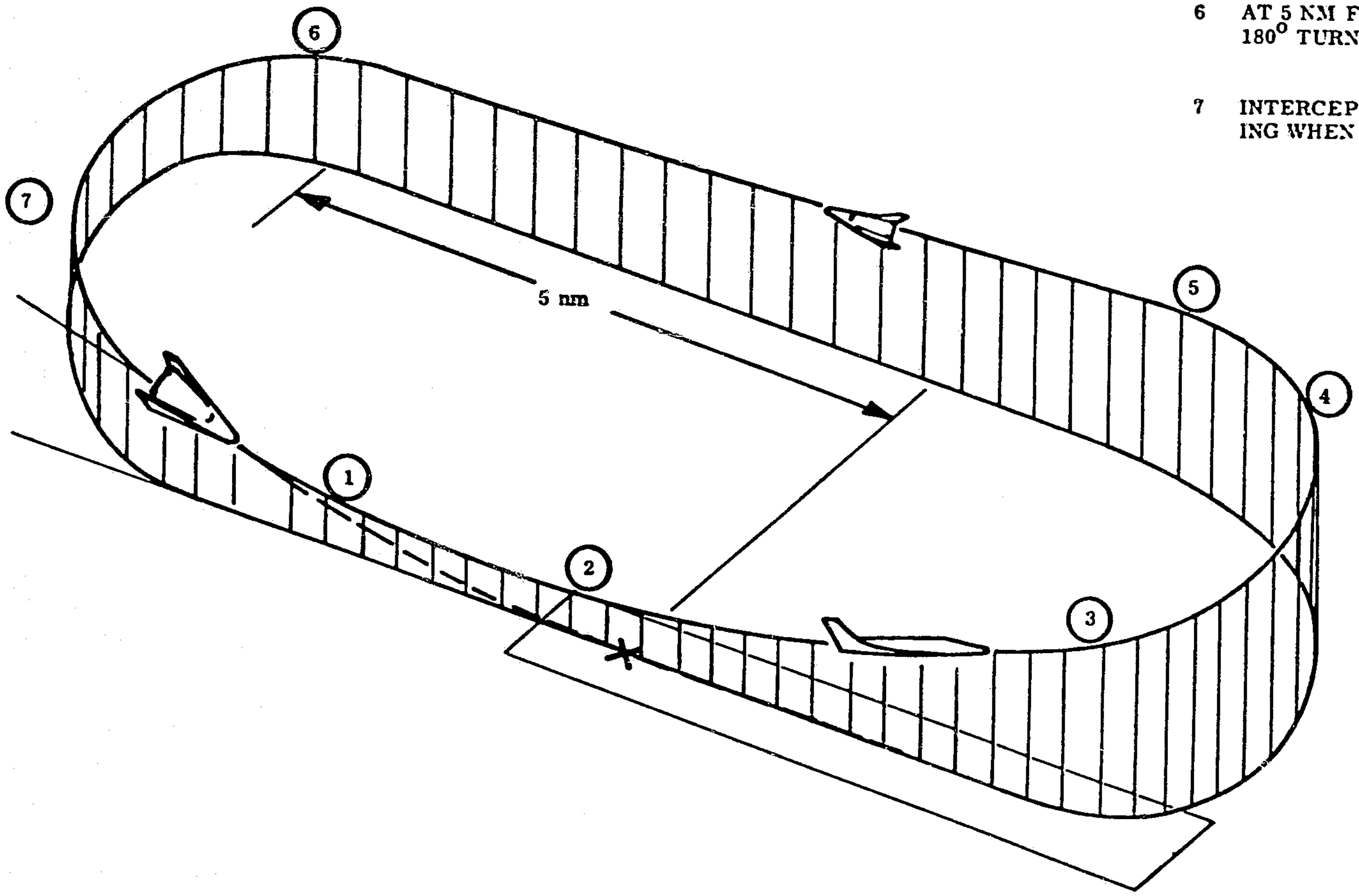


Fig. 5.1-13 Typical Go-Around Profile

final approach track.

Thrust must be sufficient to:

- Maintain level flight + .05 T/W margin at touchdown speed and configuration
- Maintain a 6-degree initial climb at maximum L/D in landing configuration
- Maintain 115 percent of V approach in level flight at 3000-foot altitude above runway
- Climb to 3000-foot altitude in 3 minutes

A performance analysis of a Two-Stage booster and orbiter configuration has yielded the following results:

Parameter	Orbiter	Booster
Vehicle landing weight (lb)	259,000	373,000
Wing loading (psf)	45	32
L/D max	4.66	7.0
Time required for go-around (min)	8.5	7.5
Fuel required (lb)	3500	3600
Max T/W ratio required	.30	.26
Duration of max thrust (min)	2.25	2.0
Downwind leg displacement from approach flight path (nm)	1.8	1.3

5.1.2.9.2 <u>Need for Go-Around</u>. It is appropriate at this time to examine the need for go-around and discuss the penalty that must be paid to provide a go-around capability. It is recommended that the tradeoffs in this area be examined further; for, if go-around can be eliminated from the orbiter, the launch system weight can be significantly reduced or, conversely, additional payload can be carried.

From an operational standpoint, the usual causes of aborting a landing approach and making a go-around prevail. One cause would be the inability to perform a safe landing because of misalignment with the runway or inability to reach the desired touchdown point. Another could be a vehicle hazard, either in the air or on the ground. It seems reasonable that the hazard problem can be effectively eliminated by proper traffic control and enforcement. The orbiter, at least, will require special air traffic control handling; for it is very unlikely that in descent and landing approach, it will be compatible with normal aircraft descent and approach control procedures.

The problem of misalignment with the runway is much more difficult, particularly for all-weather landing. There will always be a finite probability that a given landing approach will result in an unsafe landing or a crash. Ideally, it would be desirable to express this probability statistically as the number of go-arounds required per one million landings. However, it is extremely difficult to determine this number and certainly any quantification in this early stage of the program would be sheer speculation.

A capability for one (or more) go-arounds may reduce the probability of crash; but it can never eliminate it, since there is no way that a landing can be made 100 percent safe. In some instances, go-around may not help at all if the problem that caused the go-around (such as a malfunctioning vehicle system or extremely bad weather) still exists on the second landing attempt.

An acceptance of some finite probability of crash will be necessary, and the assessment of what probability is acceptable and whether go-around is required must be based on the penalty for go-around.

Tabulated below are the weights due to jet engine and fuel for go-around and for powered landing approach (L/D improvement) only on the orbiter:

	Go-Around	Powered Landing Approach
Selected jet engine	Turbo-fan	Lift fan
Thrust rating (takeoff static) (lb)	100,000	40,000
Installed engine weight (lb)	20,800	3,200
Fuel required (lb)	6,000	4,700
Total orbiter increment (lb)	26,800	7,900
Total launch weight increment (lb)	798,000	237,000

Certainly the penalty for having go-around capability is a significant percentage of the 50,000-pound payload. Eliminating go-around in the orbiter but retaining the powered approach would result in 18,900 pounds of additional payload capability. Future study must be conducted to evaluate this improvement versus the need for go-around. Furthermore, the complete elimination of jet engines should be considered, perhaps as an evolutionary change after the safety and suitability of unpowered landing has been demonstrated.

5.1.2.10 Variable-Geometry Wings. The application of variable-geometry, or deployable, wings to increase subsonic lift has long been a pertinent consideration in lifting body design. Wings offer the potential of landing speed reduction and associated easing of runway length and vehicle control requirements, with the penalty of increased vehicle weights. After thoroughly investigating the use of wings and after consideration of the tradeoffs, LMSC has established the present baseline as being without wings. The following paragraphs describe these tradeoffs and considerations.

Earlier in this landing study, a performance and weight analysis was performed on the then-current configuration to determine the effect of using wings on the orbiter. The following table summarizes the results of that analysis. Deployable wings on the boosters are not a relevant consideration, since the mild booster thermal environment permits design of adequately large fixed wings.

	With variable-geometry wings	Without wings
Max L/D	5.2	4.0
Touchdown speed (Powered approach)	155 K	185 K
Rollout distance (.25-g deceleration)	3738 ft	5089 ft
Total launch weight (booster plus orbiter)	3.65 M lb	3.15 M lb
Percentage weight (increase	16 percent	--

The choise of a wingless orbiter was based on this 16 percent weight penalty, and the considerations of deployment mechanism complexity and introduction of a critical failure mode (failure to deploy). Improved touchdown speeds for wingless bodies have been obtained by design iterations (Section 5.1.2.7), and runway rollout distances are within the capabilities of 10,000-foot runways (Section 5.1.2.8).

5.1.3 Guidance and Control Systems

This section presents the results of investigation into automatic landing systems and their application to the Space Shuttle approach and landing problems. A baseline system is described and its performance discussed. Landing aids and sensor are compared, the role of the pilot is discussed, and reliability aspects are introduced. The following discussion is addressed primarily to the problems of the returning orbiter. The booster is covered separately in Section 5.1.3.8.

5.1.3.1 Landing Systems and Requirements. The guidance and control system must be capable of performing both the powered and unpowered landings automatically or with a pilot at the controls.

The program requirement for all-weather landing capability necessitates a system capable of landing under FAA Category III conditions. (A distinction between Categories IIIa, IIIb, and IIIc need not be made at this point, since all three require instrument touchdown. Runway rollout and taxi control are the differences between the subcategories.) All present and projected landing systems have employed automatic control for touchdown under these low visibility conditions, and there is no evidence of any acceptable scheme in which pilot control is used for nonvisual landing. Accordingly, for Space Shuttle, fully automatic control has been selected as the primary mode for all landings. Repeated demonstrations of satisfactory automatic landing in good weather is the only way to build the pilot confidence necessary for acceptance of automatic landing under zero/zero conditions. The role of the pilot is discussed in further depth in 5.1.3.6.

The primary control task during the approach phase is to navigate the orbiter so that it will reach the decision key point within the required bounds of altitude, airspeed, and heading. (This navigation actually commences prior to reaching the 100,000-ft. altitude.) The navigation and energy management systems must have sufficient flexibility and accuracy to guide the vehicle throughout all possible reentry trajectories to within the yet to be defined window at decision key. After passing decision key, the emphasis shifts to precise spacecraft control (attitude, heading, glide angle, airspeed, thrust, etc.) required to perform a safe and precise landing, either with or without jet engine power.

5.1.3.2 Present and Projected Landing System Technology. The age of automatic landing is in its infancy at this time, and there are many programs in process that are aimed toward an operational automatic landing capability. In England, BOAC is landing airliners automatically, using an instrument landing system (ILS) coupled guidance approach. In the United States, several FAA and military projects are developing and evaluating automatic landing systems with the goal of all-weather operational capability. The previously slow pace has accelerated rapidly as both the airlines and the military have become aware of the benefits from an all-weather capability. The technology from these efforts is directly applicable to the Space Shuttle program.

The ILS is widely used for low-visibility approaches. Although alone it is not suitable for blind landing, ILS can be augmented (with radar altimeters, flare computers, and other devices) to enable automatic landings. The FAA and airlines have committed themselves to an all-weather landing evolutionary approach that builds from the present ILS. The military, primarily in connection with transport and cargo aircraft, is also proceeding in this direction.

Concurrently, other projects are developing all-weather capability without using ILS. The most advanced of these is the Navy's SPN-42 carrier-landing system, in which carrier-based radar tracking and data processing are employed to vector the aircraft to a hands-off carrier landing. SPN-42 is an automatic version of the military's ground-controlled approach system that has long been used for low-visibility approaches.

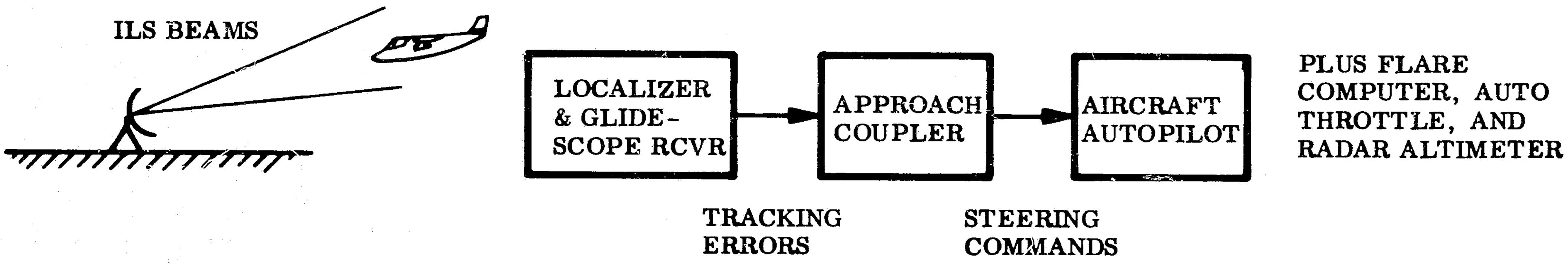
Many studies have been performed towards adapting these and other guidance techniques to automatic landing of lifting reentry vehicles. In the following paragraphs, details of all-weather landing system status are discussed.

5.1.3.2.1 ILS and Its Derivatives. The intersecting planes of the ILS glideslope and localizer beams define a "line in the sky", leading to the touchdown end of the runway. On the landing approach, deviations from this line are detected by the onboard ILS receivers; and for manual approaches these deviations are displayed to the pilot. For automatic approaches, the receivers are coupled to the aircraft autopilot; and deviations cause the issuance of appropriate steering commands. Since the glideslope beam becomes unreliable near the ground, and range or altitude is not available, a radar altimeter and flare computer are required for automatic touchdown.

Figure 5.1-14 shows the basic elements of an ILS.

Automatic ILS have been under development for some time, but are not operational (in the USA) at the present time. The furthest advance is represented by the joint FAA/USAF evaluation of the C-141 all-weather landing system. This program began with the development and certification of a Category II system and progressed to development and evaluation of a Category III system. The purpose of the Category III evaluation was to accumulate data and experience to be used in establishing criteria for Category III operations. Actual certification for Category III landings requires further development and evaluation of both ground and airborne equipment, as well as establishment of this operational criterion.

The new generation of airline and transport aircraft (C-5A, 747, L-1011, DC-10) are being designed for Category III ILS operation. The major technical problems are those associated with system reliability and failure modes and those associated with the reliability and accuracy of the ILS beam. (In the latter area, it has been found that ILS beams can have unacceptable bends and discontinuities and that they are susceptable to interference from overflying and taxiing aircraft. The FAA currently has several programs



- PRESENT USE – FAA CATEGORY II LANDINGS (100 FT, 1/4 MI)
- CATEGORY III REQUIRES IMPROVED TRANSMITTERS AND FURTHER REDUNDANCY
- MAJOR PROBLEMS – RELIABILITY AND OPERATIONAL REPEATABILITY
- APPLICABLE TO POWERED SPACE SHUTTLE LANDINGS
- HAS SIGNIFICANT DEFICIENCIES FOR UNPOWERED SPACE SHUTTLE

Fig. 5.1-14 Present ILS Based Landing Systems underway aimed at resolving or eliminating these problems). Use of the scanning beam system now undergoing FAA evaluation is discussed in Section 5.1.3.5.

There should be no difficulty in applying the airliner type Category III ILS to the landing of a powered Space Shuttle. Both the orbiter and booster vehicles are expected to need landing guidance and control of the same nature as large, high-performance aircraft; and much of the technology certainly is applicable. The major difficulties in the use of an ILS-based approach and landing system fall into two areas:

- ILS cannot guide the orbiter in its descent to the final approach path. Since ILS defines only a single beam extending from the approach end of the runway, it cannot provide the guidance necessary to engage and capture the final approach at the outer marker. Some other guidance will be necessary for navigation and energy management during the gliding descent and 360-degree turn prior to final approach.

- The unpowered, or deadstick, approach necessarily must follow a curved glideslope (Section 5.1.2.5), since the first flare occurs 1 to 2 miles from the touchdown point. Studies have been performed on an unpowered landing maneuver with two glideslope beams (one for the high speed approach and one for the post-pullout float) used and a radar altimter and path computer to guide during the two flares. These studies have not demonstrated the acceptability of this scheme, and ILS does not appear to be an attractive solution to unpowered orbiter landings.

The greatest advantage of ILS is that it is an established and operational system with capability (with improvements) to meet Category III requirements. If Category III ILS were installed at major airbases by 1975, these ground facilities would be available for use by the Space Shuttle. However, the need for additional facilities for descent navigation and energy management and the difficulty in performing an unpowered landing with ILS lead to consideration of guidance schemes that do not entail these problems.

In this context, a critical question is "at what landing fields must the orbiter be capable of making an automatic landing?" If the major landing sites will be the commercial airfields that are slated for Category III ILS, then a system that requires only the available FAA equipment would be very attractive. However, it seems reasonable to avoid these high density commercial fields -- because of the dangers posed by a Space Shuttle descending rapidly through congested airspace and the unavoidable disruption of normal air traffic -- and use the available military bases as primary fields.

The use of military bases also appears to be very attractive from an operational point of view. Figure 5.1-15 shows the wide distribution of military and low-density civil fields in the 48 contiguous states, which all have 10,000-foot runways. With this large selection, it hardly seems necessary to use high-density civil fields and to require a system to be compatible with projected FAA ground hardware, particularly since it appears that this hardware would require augmentation for Space Shuttle use.

5.1.3.2.2 Ground Vectored Control. The SPN-42 system used by the Navy for "hands off" landing on aircraft carriers is presently the only operational automatic landing system in the country. Figure 5.1-16 shows the elements of the SPN-42 system. Radar on the carrier tracks the aircraft and establishes its position in terms of azimuth, elevation, and slant range. The landing computer (on the carrier also) establishes a trajectory from a series of these position reports. This actual trajectory is compared to the reference or desired trajectory set into the computer, and position errors are determined. Aircraft steering commands are computed from these errors and transmitted to the aircraft via an RF data link. These commands enter into the autopilot and steer the aircraft through the normal control loop and hardware. An on-board automatic throttle system modulates engine thrust to maintain a preset airspeed.

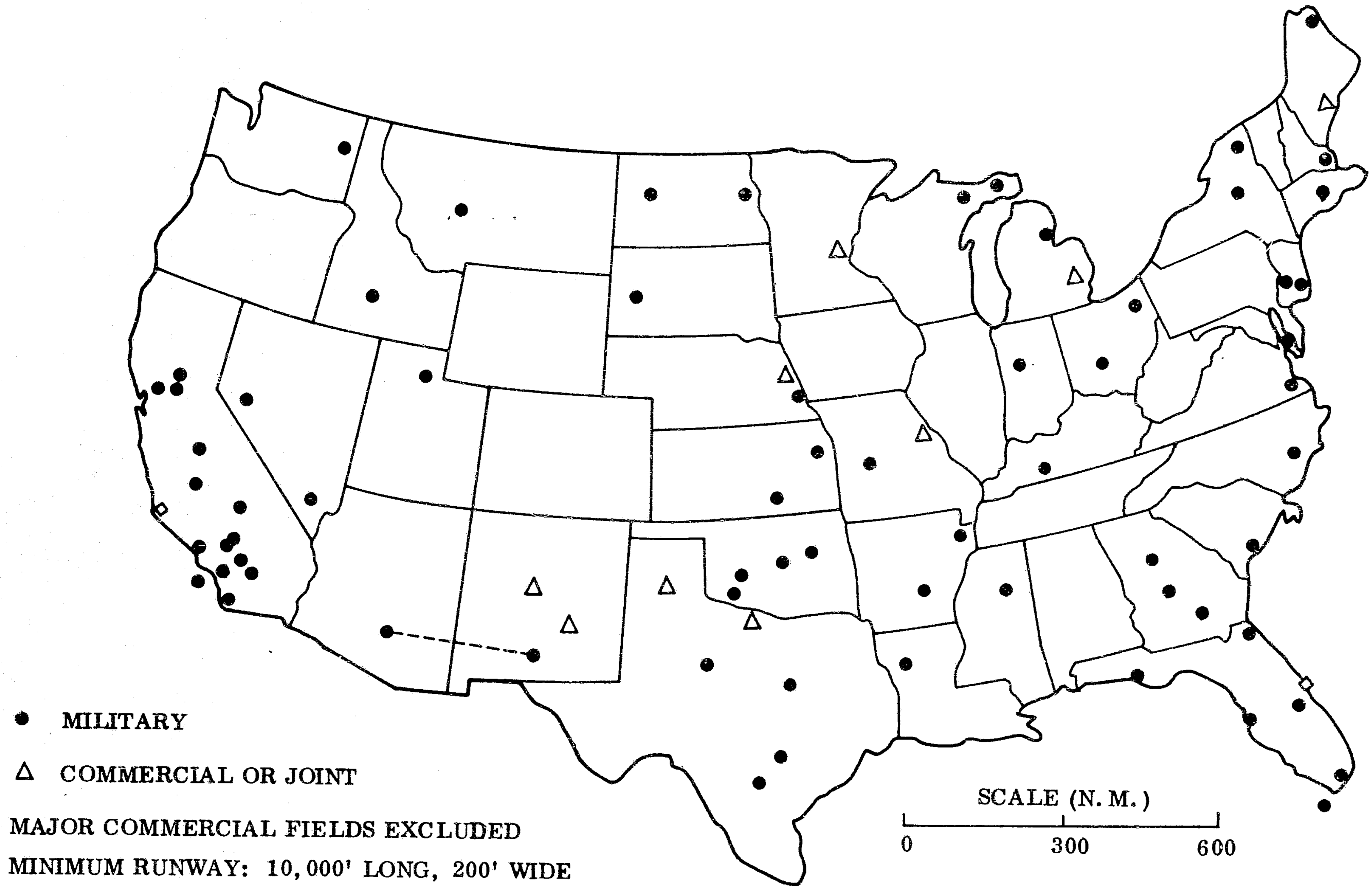
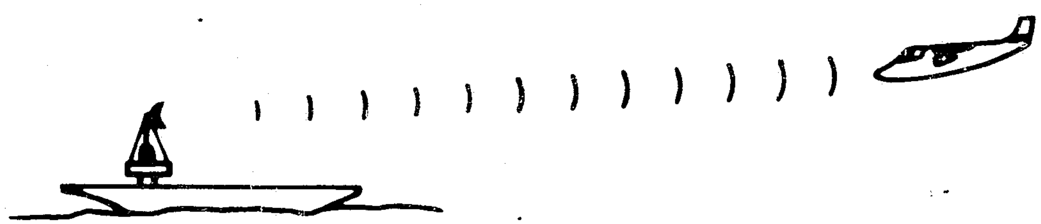
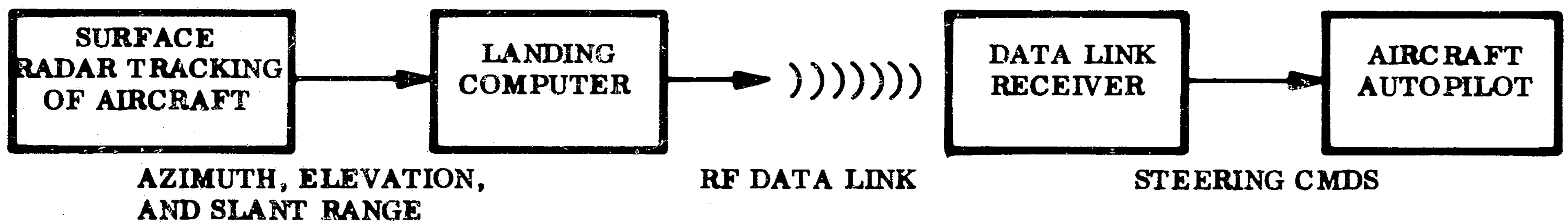


Fig. 5.1-15 Present Candidate Landing Sites



- OPERATIONAL ON NAVY CARRIERS
- AUTOMATIC GCA



- NEEDS SEPARATE RADAR SYSTEM FOR BACKUP
- APPLICABLE TO BOTH POWERED AND UNPOWERED SPACE SHUTTLE
- CAN INCORPORATE COMPUTATION FUNCTION INTO SPACECRAFT COMPUTER.

Fig. 5.1-16 Present Surface Radar Landing System (SPN-42)

This landing system concept is very attractive for the Space Shuttle. The major advantage over ILS-based concepts is that approach and landing paths are not constrained to the single line-in-the-sky of ILS. Consequently, the difficulties that ILS has with the descent to landing approach and with unpowered landings are not problems here. SPN-42 does, however, require considerable ground equipment.

5.1.3.2.3 On-Board Guidance with Update. An automatic landing system with the onboard digital computer and inertial measurement units, augmented by a ground-based update for earth-referenced position determination, was derived from the SPN-42 by moving the computation function from the ground (or carrier) to the digital computer in the vehicle. The existing inertial reference would also be utilized. In this discussion radar is treated as the means for updating; however, it is important to realize that radar is just one of several possible means of establishing the necessary ground reference. Other candidate updating systems are discussed in Section 5.1.3.5.

Figure 5.1-17 shows the elements of the onboard system. The control function is no longer performed in the same fashion as in the SPN-42. Rather, there is now an inertial guidance system, with the computer and the IMU used as the primary guidance elements. These elements are capable of performing all of the navigation and guidance computations required for approach and landing.

The purpose of the radar (or other ground reference) is solely to update the inertial reference in order to correct for drift and similar system errors. Conceptually, the orbiter has the capability to make a completely autonomous (without ground support) approach and landing. However, an IMU that can maintain linear accuracies within a few feet and velocities within a few feet per second (as is required for approach and touchdown) is not within the forseeable state of technology. Accordingly, the radar link is used to update the inertial reference so that it can guide to the accuracy required.

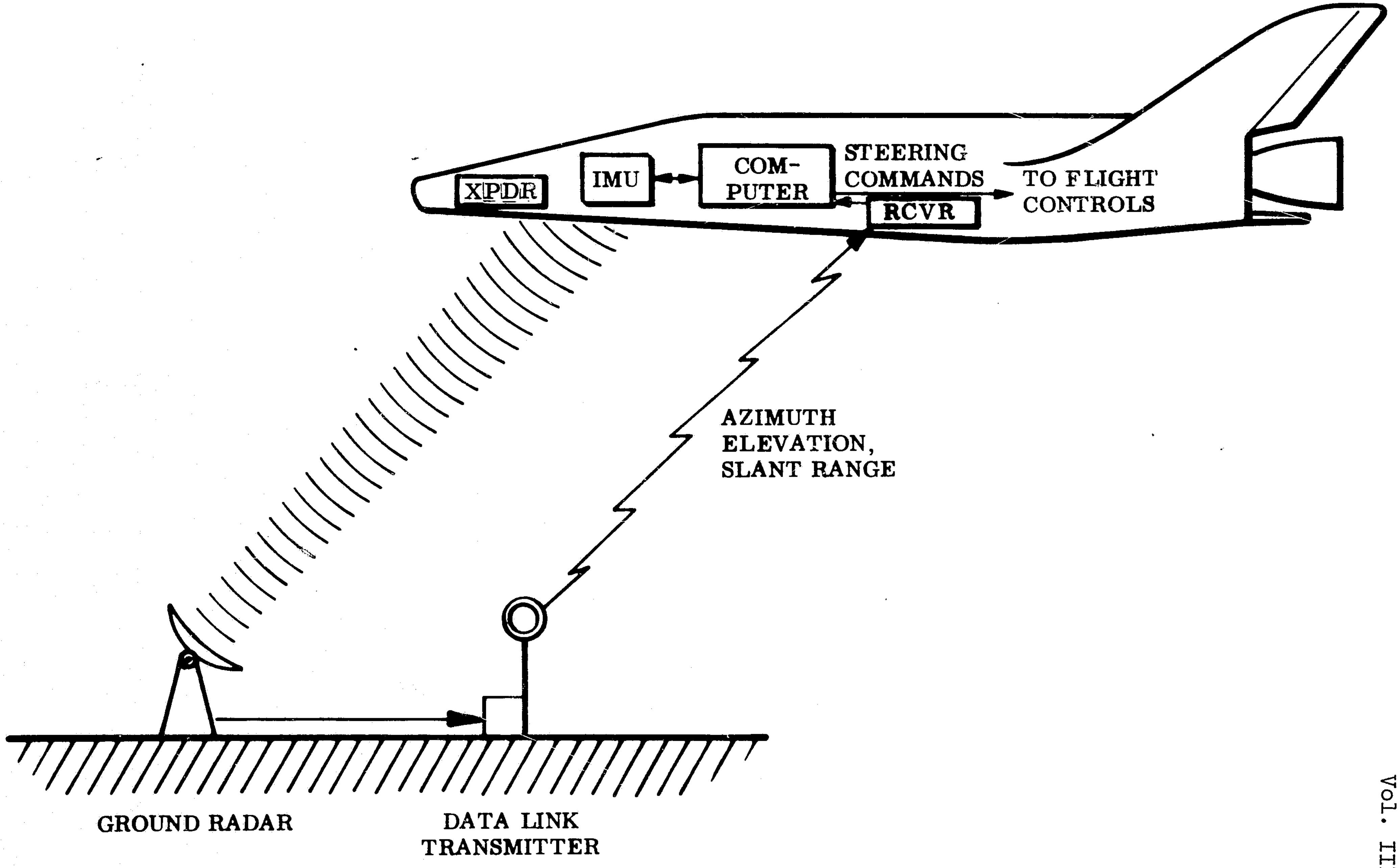

XPDR
IMU
COM-
PUTER
STEERING
COMMANDS
TO FLIGHT
CONTROLS
RCVR
AZIMUTH
ELEVATION,
SLANT RANGE
GROUND RADAR
DATA LINK
TRANSMITTER

Fig. 5.1-17

The navigation and guidance system described here - computer plus IMU control with updating - has numerous advantages over the beam-tracking (ILS type) and ground-radar controlled (SPN-42 type) concepts. These advantages include:

- Maximum vehicle autonomy
- Simple ground equipment state-of-the-art approach radar, landing radar, and data-link transmitter, for example
- Highly flexible - can program wide range of trajectories
- Guidance for both powered and unpowered vehicles
- Reentry guidance, energy management, and automatic landing accomplished by one onboard system

Analysis and simulation of the capabilities of this system have been performed by Bell Aerosystems Company, developers of the SPN-42. Section 5.1.3.4 contains a report on these studies.

5.1.3.3 <u>Operational Aspects</u>. The navigation and energy management control process that brings the orbiter within the required window at decision key starts at hypersonic reentry and continues down to the subsonic glide. Ground track, altitude, and airspeed control are accomplished by modulation of angle of attack, bank angle, and possibly speed brakes. Decision key is defined by nominal values of altitude, airspeed, and heading; these nominals have been tentatively established as 25,000 ft, 250 KIAS, and aligned with runway, respectively. Allowable tolerances on these values, plus the allowable ground track width, are defined to provide a sufficiently precise initial point for either a powered or unpowered landing.

Ground data for inertial reference update are provided during the descent. Studies to define update requirements have not yet been performed; however, it is anticipated that update will not be required until the orbiter has descended below a 100,000-foot altitude. If this is correct, the maximum update range required (from the landing field) will be on the order of 100 to 150 nm.

At decision key, the decision to perform either a powered or unpowered landing will be made, either automatically or by the pilot. The decision will be based upon successful (or unsuccessful) airstart of the jet engines. The normal mode will be a powered landing with the engines started. For powered landing, the computer program will generate steering and throttle commands to maintain proper altitude, airspeed, and groundtrack through the 360-degree turn, final approach, touchdown, and landing roll. Errors from the programmed nominal profile will be sensed and corrective commands generated. Update will be used throughout to provide the necessary inertial reference accuracies.

For an unpowered landing, the computer will command a straight-in glide approach. At approximately 1000-foot altitude, a flare is initiated to reduce the rate of descent to a suitable level for touchdown. This technique is similar to that successfully demonstrated in manual landings of the X-15 and the M2 and HL-10 lifting bodies at NASA-FRC. Simulations of automatic unpowered landings have successfully established control techniques for correction of initial condition errors and perturbations due to turbulence and shear winds. These simulations are discussed in Section 5.1.3.4.

5.1.3.4 Analysis and Simulation. This section describes the results of analysis and simulation of guidance for reentry vehicle terminal approach and landing. The material presented was supplied by Bell Aerosystems Company of Buffalo, New York, long a leader in the field of reentry vehicle guidance. The following is essentially a verbatim extraction from Bell report D6236-953002.

5.1.3.4.1 Requirements for Terminal Approach and Landing. Bell Aerosystems has investigated the guidance requirements for powered and unpowered approach and landing of a Space Shuttle. Of the two, the unpowered landing has the most demanding requirements and, therefore, dictates the guidance and navigational system configuration and determines accuracies that must be met. Powered landing can be considered in a similar light as the unpowered landing, for jet engine thrust effectively improves the L/D ratio (for 3-degree glidescope, L/D = 20). This improvement, in general, eases the control task.

The terminal guidance of unpowered, horizontal landing vehicles normally consists of (1) a terminal approach phase, during which the vehicle is aimed at an approach window; (2) a heading alignment phase, during which the vehicle's heading is lined up with that of the runway; (3) a final approach phase; (4) a flare; and (5) a final glide to touchdown. The guidance problems that are involved in these phases can best be described if they are considered in reverse order.

The primary guidance task during the final glide is one of making slight adjustments in the vehicle's touchdown velocity and position while maintaining the desired glide path angle at touchdown. Since the equilibrium flight conditions for an unpowered vehicle are all determined by the L/D ratio at which the vehicle is flying, a long term correction in any one of the touchdown conditions will normally compromise the others. Although studies have shown that a satisfactory compromise can be made; they have also shown that the size of the errors that can be corrected for are quite restricted. Consequently, initial conditions for final glide must be carefully controlled.

During the flare, the primary task of the guidance system is to brake the vehicle's rate of descent to a value that is compatible with the desired final glide path angle by the time the desired glide initiation altitude has been reached. Although it is possible to adjust the flare initiation altitude and normal acceleration during flare to correct for any errors that might occur in this altitude and altitude rate, very little can be done to correct for any velocity and position errors. If the vehicle is controlled so as to follow a fixed predetermined flare and glide profile, this severely restricts the errors that can be tolerated at flare initiation. This in turn restricts the size of the errors that can be corrected for during final approach, since it will be necessary to ensure that all phugoid flight path transients that are introduced by such correction are completely damped prior to reaching the flare initiation point.

These restrictions, arising from the concept of a predetermined, fixed-flare profile are somewhat artificial, since it is possible for the vehicle to correct for larger errors in final approach if the subsequent flare and glide phases are adjusted to account for the effects of this transient on the final touchdown conditions. Since the maneuvering capability of unpowered vehicles is inherently limited, it is important that the guidance system should not be further restrictive and should be capable of using all of this inherent maneuver capability to recover the vehicle from as large an area as possible. However, this does impose a requirement that the guidance system be capable of predicting the effects of corrective maneuvers on the expected touchdown conditions and of using this information to adjust the nominal flare and glide profiles as required to insure a successful landing.

Predictive guidance techniques investigated extensively at Bell have been found to enhance greatly the ability of a system to recover a vehicle. However, even with these techniques, it has been found that the errors that can be tolerated at the start of final approach are very small as compared to those that can exist at 100,000 feet; therefore, the guidance system must operate throughout the terminal approach phase so as to bring the vehicle into the final approach phase with a favorable position, heading, and total energy for straight-in final approach.

In the heading alignment phase, the primary task of the guidance system is to line up the vehicle's heading with that required for a straight-in final approach. Since this phase is normally quite short, even with predictive guidance techniques the errors that can be corrected in it are small compared to those that can exist at 100,000 feet. As a result, the guidance system must also bring the vehicle into the heading alignment phase with generally the right position and total energy for accomplishing the heading alignment turn. Both the position and energy required at the start of the heading alignment phase depend mainly on the magnitude of the turn required. As a result, the initiation point for this phase is, in terms of both position and total energy, a "floating" point, which must be adjusted by the guidance system according to the required turn. In addition since flight path transients have a significant effect on the short term turning capability of the vehicle in this phase, the guidance system should be capable of predicting the effects of these prior to the initiation of the turn and of adjusting the initiation point accordingly.

At this point, it can be seen that the guidance system must reduce all large errors before the heading alignment phase is reached. To do this, it must bank the vehicle to aim at the required heading alignment initiation point and modulate the vehicle's L/D as required to ensure that this point is reached with the proper total energy for heading alignment and the subsequent phases. Control of the vehicle's energy is a primary aspect in the terminal guidance of unpowered vehicles. In general, the range capability of a vehicle with a given L/D is somewhat directly proportional to its total kinetic and potential energy and, therefore, the nominal L/D required to dissipate a given amount of energy in a given range can be determined from this relationship. However, flight path transients also have a significant effect on the vehicle's range capability; and, if these effects are not accounted for in determining the required L/D, an unstable range/energy loop is likely to result. To prevent this, it is important that the guidance system be capable of predicting the effect of transient flight conditions on the range capabilities of the vehicle and of using this information to modulate the L/D of the vehicle to ensure that the vehicle will reach the desired point with the correct energy.

One of the primary requirements of a guidance system for an unpowered vehicle is that it be capable of predicting the effects of off-nominal flight conditions on the vehicle's maneuvering capability and of using this information to adjust the vehicle's flight path and the required flight conditions at the initiation of each phase as required to ensure a successful landing. It must also do this early in flight since the vehicle's capability to correct for errors decreases rapidly as the flight progresses. This, in turn, imposes requirements on the accuracy of the guidance and navigational systems.

For example, the total maneuvering capability of a vehicle with maximum subsonic L/D of 5.0 at a 30,000-foot altitude, Mach 0.7 final approach point can be described approximately by a circle with a radius of 5.65 nm. This is the area, or footprint, that the vehicle can successfully land in. From it, the range errors that can be tolerated in guiding the vehicle to this point can be determined directly, and the errors that can be tolerated in other parameters can be determined by converting them to equivalent range errors. Normally, it is not desirable to plan on using more than 50 percent, or 2.83 nm, of this capability for correcting for all 1σ errors. If this is done, correcting for a single 3σ error will requre using only about 63 percent of this total capability.

If the errors in the guidance system itself were zero, all of this 2.83-nm capability could be used for correcting for errors in the navigational information, vehicle characteristics, and for disturbances. In practice, if 20 percent of this capability is allocated for correcting 1σ guidance errors, 98 percent of it will still be available for correcting all other errors. This 20 percent represents a maneuvering capability of about 0.23 nm.

To avoid exceeding this capability, the guidance system must be inherently accurate at the 30,000-foot level to within the following 1σ tolerances:

Range	0.23 nm
Cross range	0.23 nm
Altitude	0.08 nm
Velocity	22 ft/sec
Heading	1 degree

If the remaining 98 percent of the 2.83-nm capability that is available for correcting 1σ errors is equally divided in correcting for errors in the navigational data, for errors in vehicle characteristics, and for disturbances, the capability that is available for correcting for 1σ navigational errors is about 1.58 nm. To prevent exceeding this capability, the maximum errors that can be tolerated at 30,000 feet are as follows:

Range	0.64 nm
Cross range	0.64 nm
Altitude	0.22 nm
Velocity	61 ft/sec
Heading	2.8 degrees

From this example, it can be seen that the accuracies required are dependent on the vehicle maneuvering capability and how much of this capability is allotted for correcting for each type of error. Since the maneuvering capability of an unpowered vehicle decreases to near zero as the touchdown point is approached, this imposes a requirement for increasingly accurate guidance and navigational information as the flight progresses. In addition, these requirements apply regardless of the fact that a powered landing is planned, since the vehicle must retain its unpowered landing capability until engine start has been satisfactorily completed. Updating of navigation data accuracy will be necessary to enable the guidance system to maintain the vehicle on a flightpath that stays within acceptable limits of the vehicle's rapidly decreasing maneuver capability as the flight progresses.

5.1.3.4.2 <u>Description of the Guidance System</u>. Bell has undertaken studies on guidance systems based on:

- Repeated fast time integration of the vehicle's equations of motion in an environment described by stored data

- Empirical functions of height, speed, vehicle weight, and climb angle, stored after initial (i.e., preflight) computation of trajectory data by repeated integration of the vehicle's equations of motion for various nominal and off-nominal conditions.

Of the two types, the first has been found to meet the required guidance accuracies for unpowered flight. This type of guidance system is one that employs a predictor (which is a model of the vehicle in differential equation form) to predict the maneuvering capability of the vehicle during the flight.

Prior to decision key, the predictor is used to predict the unpowered maneuvering capability of the vehicle in the form of a ground area attainable (GAA). The range and cross range errors between the position of the final approach point in this predicted GAA and the nominal point in it are then used in guidance laws that control the vehicle's energy and heading in such a manner that the vehicle is aimed at this point and passes through it with the proper energy and heading for final approach.

The operation of the system in the final approach and landing phase is essentially the same for powered and unpowered flight. In powered flight, the predictor is used to predict the flight conditions that will result at the end of each flight phase if the 360-degree spiral turn (from decision key to final approach) and the powered final approach and landing were flown at the nominal power setting and with the nominal attitude commands. The errors between the predicted and desired flight conditions are then used in guidance loops, which modulate the vehicle's thrust level and attitude as required to ensure that the desired flight conditions at the end of each phase and at touchdown will be achieved. For an unpowered final approach and landing, the operation is the same, except that the errors are predicted for an unpowered nominal flight and guidance loops, which modulate the vehicles' attitude, only are used. These guidance loops are not greatly different, since the control of a powered vehicle in many instances is very similar to the control of an unpowered vehicle of higher L/D.

This system has been designed to be mechanized on an airborne digital computer with a computational speed compatible with those available on such computers as the Honeywell ALERT or the IBM CP-2 and with a memory (for the guidance function alone) of at about 6000-24 bit words or the equivalent. The system can receive its required navigational inputs from a navigational system, a radar, or a combination of both. For a reentry vehicle, it is recommended that an airborne navigational system be used with ground based radar updating to achieve the required navigational accuracy. The system has been designed to operate either in a completely automatic mode, in which case attitude steering commands are sent directly to an autopilot, or in a manual mode, in which the attitude steering command errors are displayed on the flight director needles of a three-axis attitude indicator and the pilot closes the control loop to null these errors. Displays other than conventional displays should include a situation display that shows the position of the desired landing site relative to the predicted landing point if the nominal commands were flown all the way to touchdown.

This predictive type of guidance system, which has been investigated extensively by Bell, has been found to overcome all major problems associated with the terminal guidance of horizontal landing reentry vehicles. It was first applied at Bell to the reentry guidance of lifting reentry vehicles. The preliminary design of a predictive energy management system for reentry vehicles was done from 1960 to 1962 under Air Force Contract AF33(616)-7463. From 1962 to 1964, Bell optimized the design of this predictive energy management system and conducted extensive simulation studies on it, including piloted realtime studies, under Air Force Contract AF33(657)-8330. Under a supplement to this contract, Bell also adapted this predictive energy management system design for terminal guidance of the X-15 from the end of boost to a high key point at the start of the landing approach pattern. From this design, specifications for the equipment requirements were prepared. These included specifications for the airborne computer, navigations information, displays, pilot controls, and interfacing with the adaptive flight control system in the X-15. This was done as a part of an Air Force program for flight testing an advanced reentry navigation and guidance system in the X-15.

In addition to designing the predictive EMS for X-15 terminal guidance, Bell developed an X-15 EMS simulation program for the SDS D30 digital computer at NASA-FRC from 1964 to 1965 under NASA Contract NAS4-915. This program was used in conjunction with the X-15 simulation at NASA-FRC for evaluation and pilot training purposes. From 1965 to 1967, under Contracts AF33(615)-2519 and NAS4-1002, Bell also assisted NASA-FRC in the evaluation of the system, with the programming and checkout of the system on the ALERT airborne digital computer through to the ground based checkout of the airborne system. Although the program was discontinued before actual flight tests were begun, the work done on it was sufficient to indicate that the predictive guidance concept is a good approach to the problem of terminal guidance of horizontal landing reentry vehicles and that a system employing such a concept can be mechanized with current state-of-the-art airborne equipment.

Since 1967, Bell has been engaged in studies on adapting the predictive energy management system concepts that were developed in previous programs to the final approach and landing of reentry vehicles. It was found that inherent flexibility of the predictor approach make it easy to adapt this technique to the approach and landing phase and to vehicle designs with widely varying aerodynamic characteristics. In doing so, it was shown that problems such as those associated with controlling the effect of off-nominal and transient flight conditions on longitudinal control during final approach on the required flare indication altitude and on flare control can be easily overcome. Based on these studies, a preliminary design of a predictive automatic landing system was developed and simulated.

Flight path guidance employing automatic pilot and auto throttle control laws for powered final approach and landing were developed for a wide variety of Navy and Air Force aircraft and are in daily use in the Bell-built SPN-10 and SPN-42 automatic landing system now operational on Navy carriers.

5.1.3.4.3 Advantages of the Predictive Guidance Technique. The primary advantages of the predictive approach to terminal guidance are its versatility, accuracy, and flexibility. Since it is a predictive system, which successfully predicts the vehicle's maneuvering capabilities on the basis of the actual flight conditions, it is not restricted to any predetermined flight conditions or profile. It can predict the vehicle's capability to maneuver and land from any initial flight condition. If these flight conditions are off-nominal, it can predict the effects of these on the planned nominal trajectory and determine whether corrective action is required. It can also predict the effects of flight transients and disturbances on the required flare initiation altitude for both powered and unpowered landings. It can also predict when such a landing would push the vehicle to the limits of its maneuvering capability or constraints and signal that a go-around is necessary.

With its predictive feature, this technique has an inherent capability to provide very accurate guidance. Studies have shown that under ideal conditions, two things must be satisfied to achieve this. The predictor model must be a reasonably accurate representation of the real vehicle, and the errors in the navigational information must at all times be small compared to the total maneuvering capability of the vehicle. Since the maneuvering capability of the vehicle decreases very rapidly through flare to touchdown, this imposes a requirement for increasingly accurate navigational information. However, this requirement is not unique to this technique, since no guidance system can be more accurate than the information supplied to it.

The basic guidance law is to maneuver the vehicle rapidly so that the landing site is centered in its maneuvering capability (i.e., to the center of its recovery funnel). This ensures that the vehicle will have the greatest margin to correct for errors in any direction. The ability of the predictor to predict flights with the same constraints imposed on them that are imposed on the real vehicle also ensures that maneuvering the vehicle anywhere in the predicted maneuvering capability will not violate any of these constraints. The predictor also has the capability of predicting future flight criticalities. For example, it can predict whether some control action now will cause some vehicle constraint to be exceeded at a future point.

It can be seen that the predictor technique is very flexible and can be readily adapted to the study of a wide variety of guidance problems. The predictor can be easily adapted from one vehicle to another simply by changing the aerodynamic characteristics supplied to it and modifying the constraint control loops to be compatible with the constraints of the vehicle being considered. This makes it very useful, both as a study tool and as an airborne guidance technique that can be applied to a wide variety of vehicles.

5.1.3.4.4 Simulation and Results. A digital simulation program for this predictive automatic landing system has been developed by Bell on an IBM 7090 digital computer and used to simulate the system. As shown in Fig. 5.1-18, this program contains a simulation of a reentry vehicle and of the proposed automatic landing system. These are tied together in a manner that permits controlled flights to be simulated from the point of terminal acquisition touchdown.

In operation, the equations of motion for the reentry vehicle are initialized with the desired initial conditions and then integrated. At a frequency of 10 times per second, the simulated vehicle flight conditions are transferred to the simulated automatic landing system. The vehicle position and velocity data are then used in the predictor, maneuver command, and constraint control equations to generate the vehicle attitude commands. These commands are then transmitted back to the reentry vehicle simulation and used during the next 1/10-second cycle. This procedure is repeated to obtain a controlled simulation of the flight to touchdown.

Since the guidance requirements for unpowered flight are most demanding, this simulator has been used mainly to evaluate the operation of the system on unpowered flights. The results shown here are for an unpowered vehicle with a maximum subsonic L/D of 3.0, which is a more difficult problem than expected for the higher L/D Space Shuttle. These include results for straight-in runs, runs where various lateral turning maneuvers are required prior to final approach, runs with off-position at the end of boost, runs with off-nominal flight conditions (out of equilibrium) at the end of boost, and runs with off-nominal flight conditions at the point of flare initiation.

The operation of the proposed system on a nominal flight for a straight-in approach is shown in Figs. 5.1-19 through 5.1-22. The initial vehicle flight conditions on this flight were those for equilibrium* flight at

*Equilibrium flight conditions are defined as those for the nominal angle-of-attack and bank angle that result in the vertical forces being balanced and the rate of change of the vertical forces being zero.

Mach ≈ 2. The initial vehicle position was selected so that the range and cross-range errors were initially zero.

It can be seen from the plot of predicted nominal range and range in Fig. 5.1-20 that the system keeps the flight very near on-nominal all the way to touchdown. It can also be seen from the plot of the flight conditions in Fig. 5.1-19 that the resulting trajectory is quite smooth. The increase in the glide path angle, shown in this figure at an altitude of about 5000 feet, is due to the programmed increase in the nominal L/D as the vehicle enters the heading alignment and final approach phase.

The angle-of-attack and bank-angle commands are shown in Fig. 5.1-21. Although the commands are quite smooth, they do vary significantly. This is because the nominal angle of attack that is required to obtain the nominal L/D varies with Mach number; the system range control loop has a relatively high gain, which is continuously modulating the commands to hold the vehicle exactly on nominal; and the system damping loop is further modulating the commands to damp out any flight path transients that are induced by the range control loop.

An expanded view of the final approach and flare and glide phase of flight is shown in Fig. 5.1-22. As shown, the required flare initiation point occurs almost at the predicted nominal altitude of 800 feet. The plot of the predicted flare altitude required shows that the flare occurs almost as the planned nominal flare and that little modulation of the net vertical acceleration from the planned value of g is required. The glide-path angle during the glide part of flight is also very close to the planned nominal value of -2.0 degrees. The plot of the angle-of-attack command shows that the command never exceeds the maximum trimmable value for this vehicle of 20 degrees and that the angle of attack at touchdown is not higher than the maximum allowable value for this vehicle of 15 degrees.

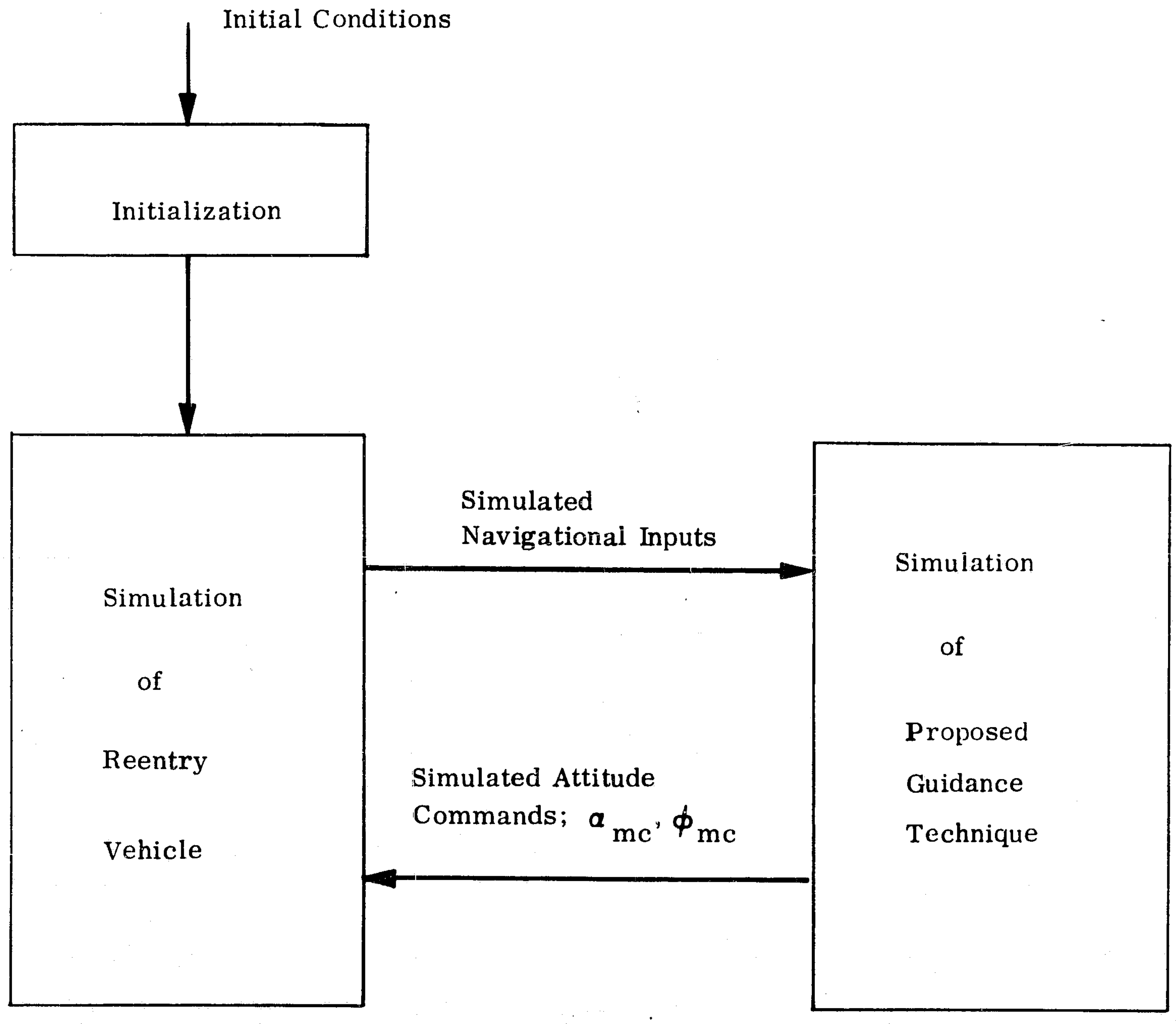


Figure 5.1-18. Block Diagram of System Simulation

Initial Cond: $V = 2000$ FPS $R = 30.5$ n.mi.
$h = 59000$ ft $R_c = 0.0$ n.mi.
$\gamma = -10.1^\circ$
$\xi = 0.0^\circ$

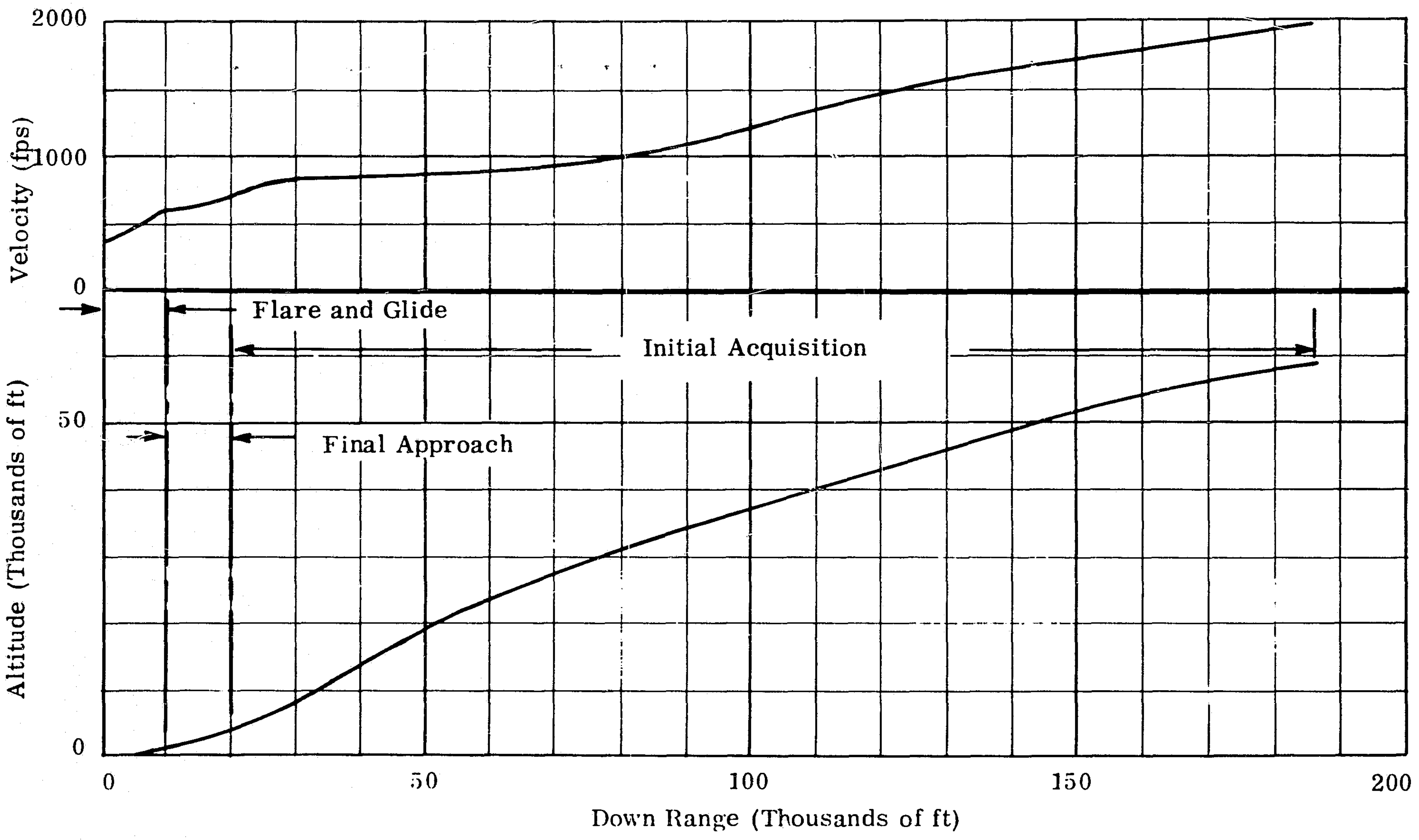


Figure 5.1-19. Nominal Flight (Flight Conditions)

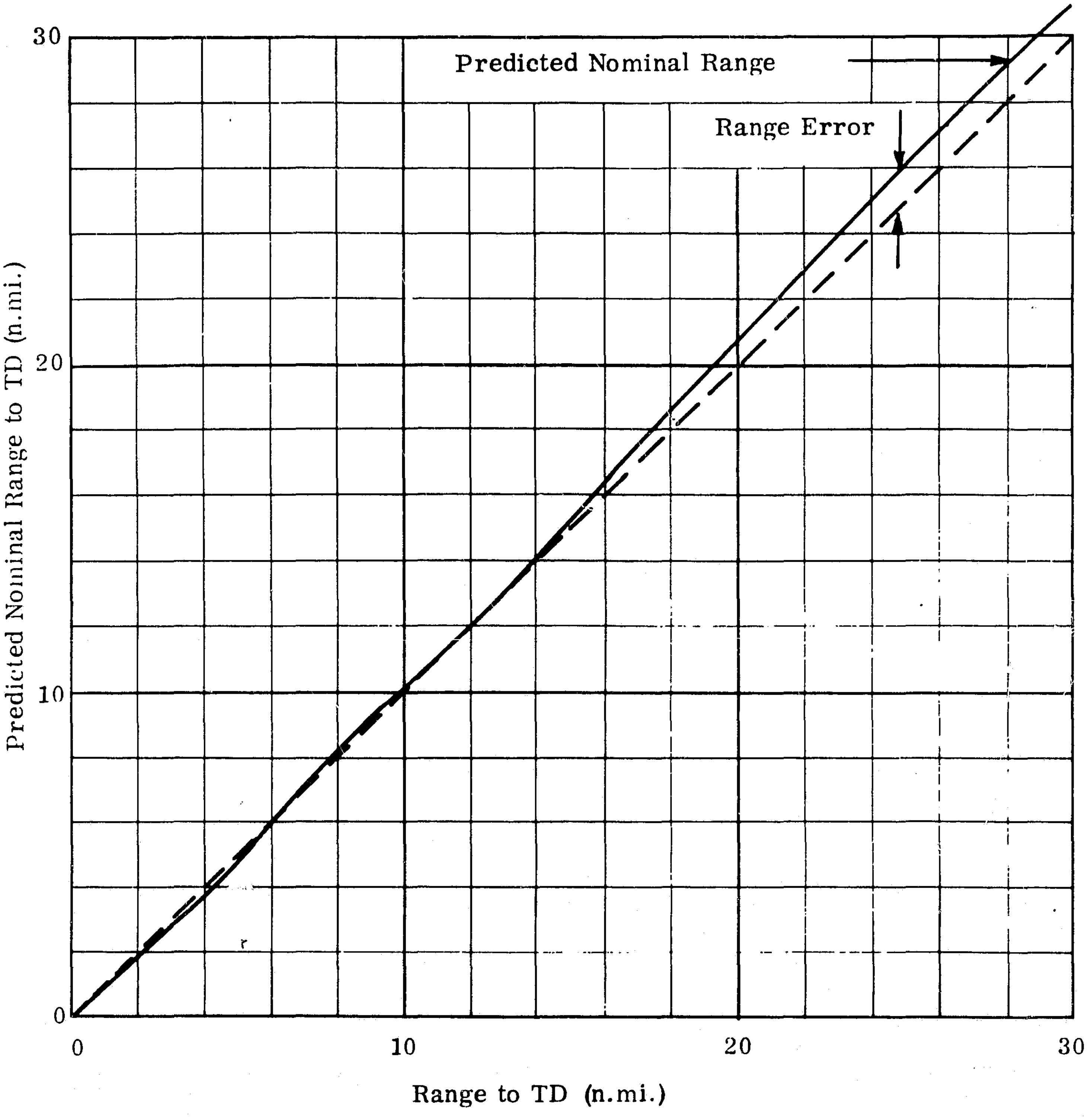


Figure 5.1-20. Nominal Flight (Range Data)

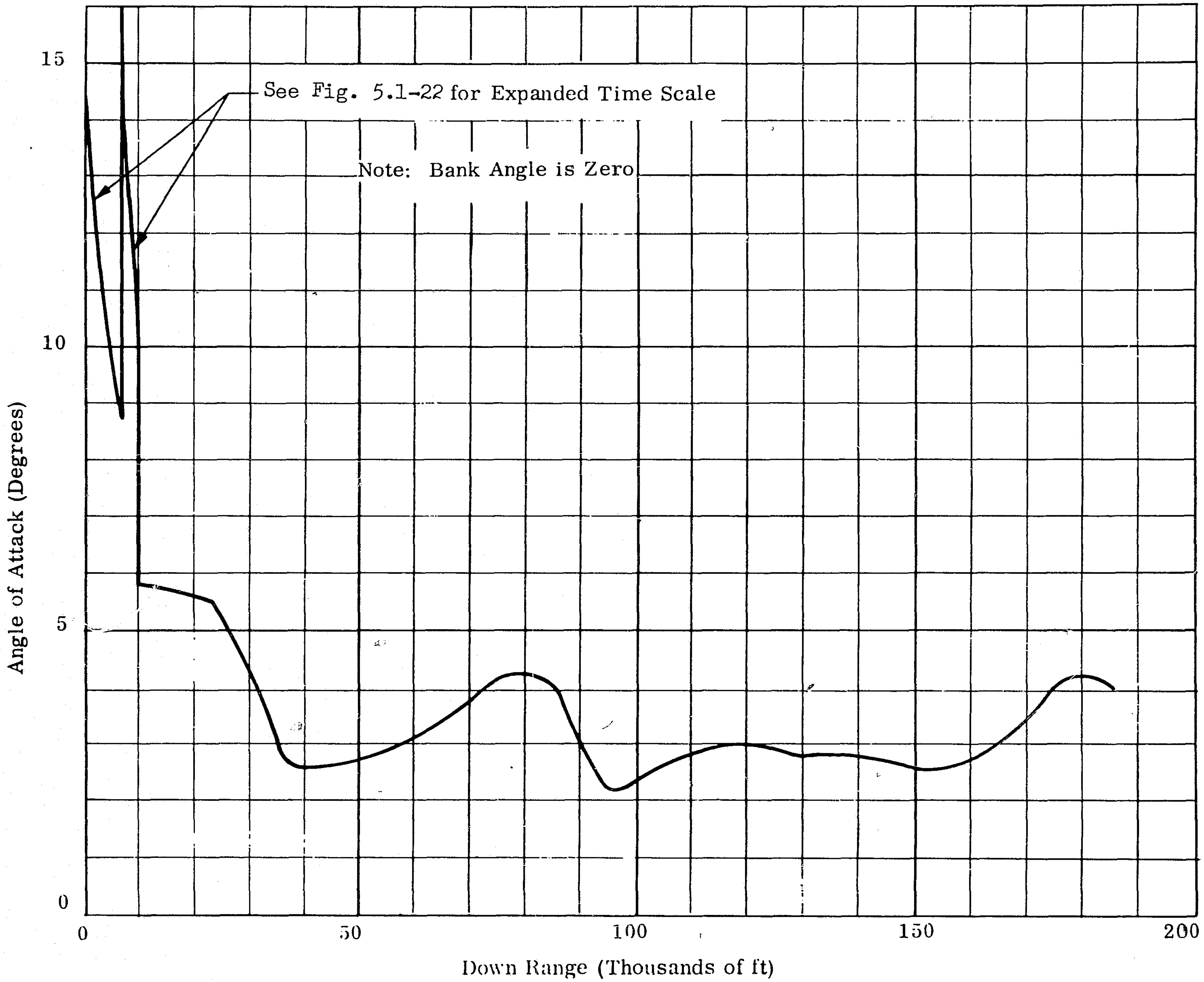


Figure 5.1-21. Nominal Flight (Commands)

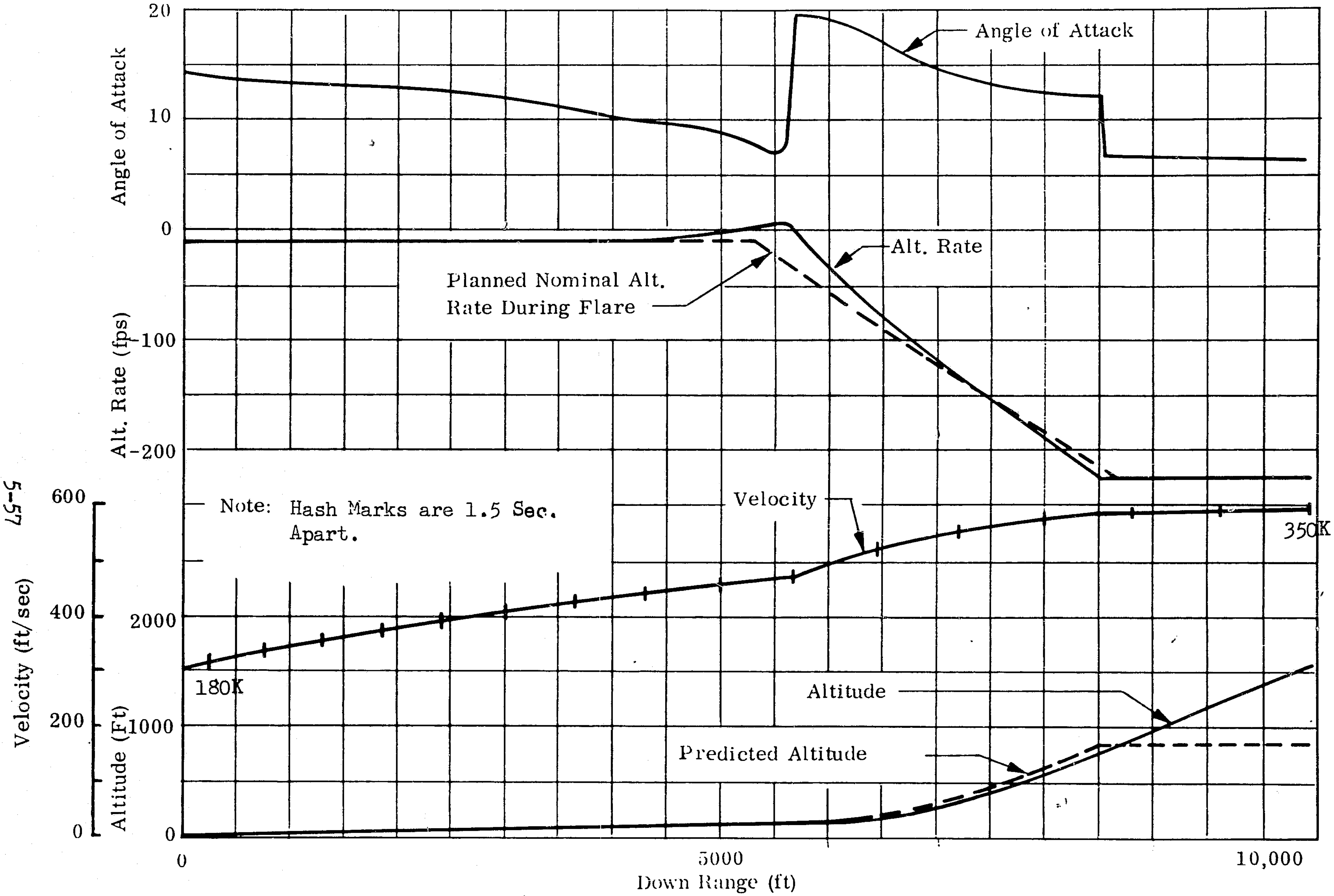


Figure 5.1-22. Nominal Flight (Expanded View of Flare and Glide)

LMSC/A959837

Four flights were made to evaluate the operation of the system when the vehicle position was initially off-nominal. On these flights all of the initial vehicle conditions were identical to those for the nominal flight except for the initial vehicle position. This was moved down range to obtain initial range errors of ± 5 nm and laterally to obtain initially cross-range errors of ± 5 nm. These errors are about 25 percent of the maximum that can be tolerated at these flight conditions.

The operation of the system in the presence of range errors is shown in Fig. 5.1-23. It can be seen from the figure that the system corrects for these errors rapidly and brings the trajectory back on to the planned nominal and that the error that requires range shortening is of a magnitude that can be corrected simply by pitch control, without an S-turn maneuver.

The operation of the system under the presence of cross-range errors is shown in Fig. 5.1-24, which shows that the system rapidly corrects for these errors by turning toward the desired touchdown point and also that the heading offset that results from this turning is corrected for by the system during the heading alignment and final approach phase.

One flight was made to evaluate the operation of the system in the presence of off-equilibrium flight conditions. On this flight, all of the initial conditions were identical to those for the nominal flight except that the flight path angle was 20.1 degrees greater than that required for equilibrium flight at Mach 2. Since transient flight conditions have a significant effect on the range traveled on a nominal L/D flight from the same energy condition, these off-equilibrium flight conditions resulted in the vehicle's range position being far off-nominal, even though its total energy and range position were nominal for equilibrium flight conditions.

The results of this run are shown in Figs. 5.1-25 and 5.1-26. The plot of predicted nominal range and range in Fig. 5.1-26 shows the ability of the system to predict the effect of the off-equilibrium flight conditions on the range that will be traveled if the nominal L/D is flown. This enables the system to start correcting for this error immediately. This is a desirable feature in this system, because without it the system would not be able to detect any range error until the integrated effect of the off-nominal rate of energy dissipation due to the transient became significant. The plot of the flight conditions in Fig. 5.1-25 shows that the system corrects for this error and brings the flight back to nominal. However, it can also be seen that the flight is well into the final approach phase before this happens. Had the system not accounted for the increment in nominal range due to the flight path transient and had to wait until it resulted in an error in the nominal range versus energy profile before it could start to correct for it, the system probably could not have brought the flight back onto nominal before the required flare initiation point. It can also be seen that the damping loop results in the flight path transient being well damped.

Two runs were made to evaluate the operation of the system on flights in which dogleg maneuvers are required prior to touchdown. The first of these was for a 90-degree turn and the second for a 135-degree turn. In each case, the vehicle heading and position was changed so that the range errors were initially zero. All other flight conditions were identical to those for the nominal flight.

The ground tracks for these two runs are shown in Fig. 5.1-27, which shows that the system guides the vehicle so that it ties into a circle at the beginning of the heading alignment and final approach phase. It also shows that during the heading alignment phase, the system guides the vehicle around the circumference of this circle and that the radius of the turn is about 4.25 nm, as planned. The system also controls the energy that the vehicle has at the start of the heading alignment and final approach phase and the vehicle energy during the heading alignment turn so that the vehicle has sufficient energy for a nominal final approach of about 2 nm, as planned.

Initial Cond: $V = 2000$ ft/sec

$h = 59{,}000$ ft

$\gamma = -10.1^\circ$

$\xi = 0.0^\circ$

$R_c = 0.0$ n.mi.

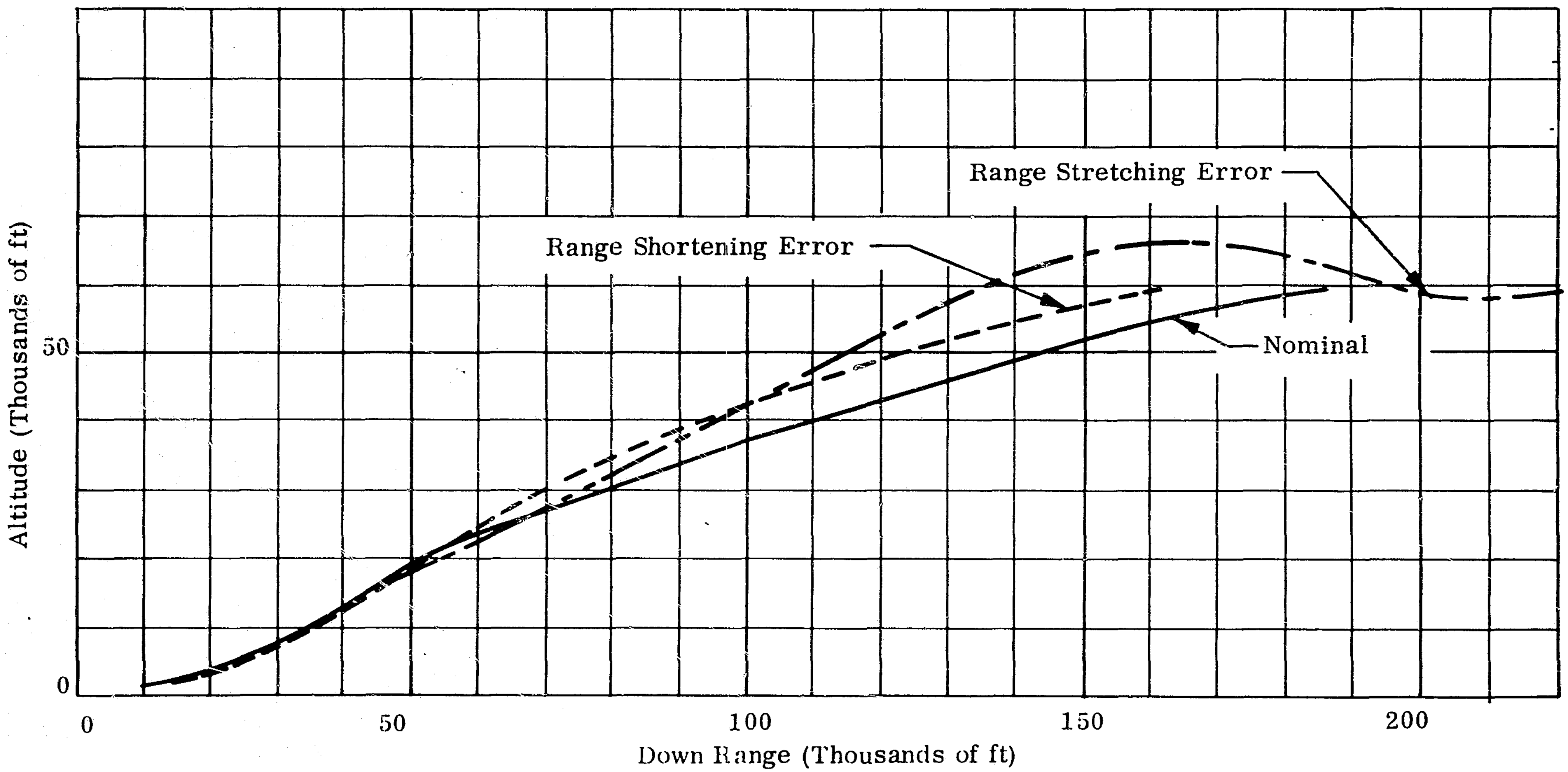


Figure 5.1-23. Influence of Range Errors for Straight-In Approach

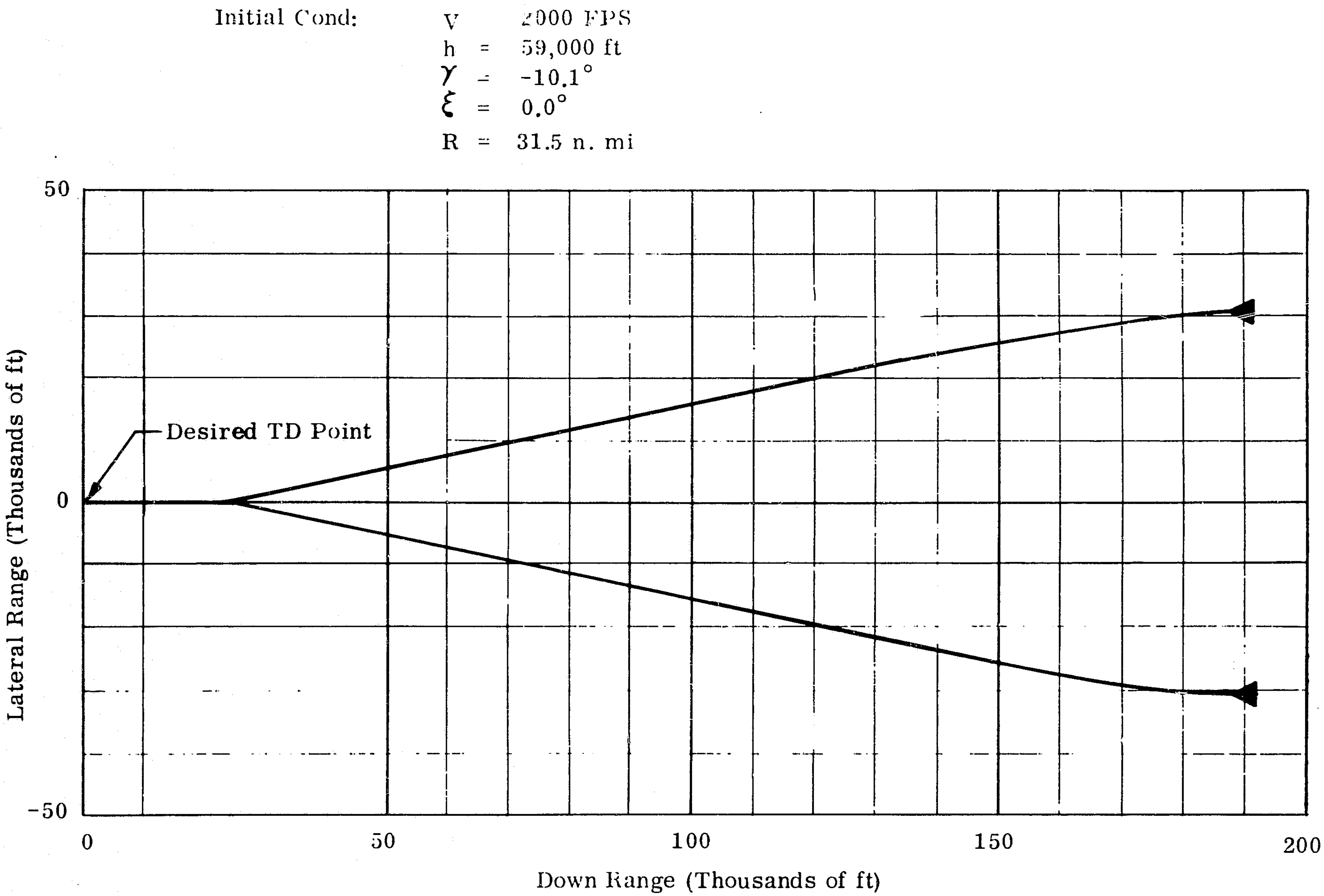


Figure 5.1-24. Influence of Lateral Errors for Straight-In Approach

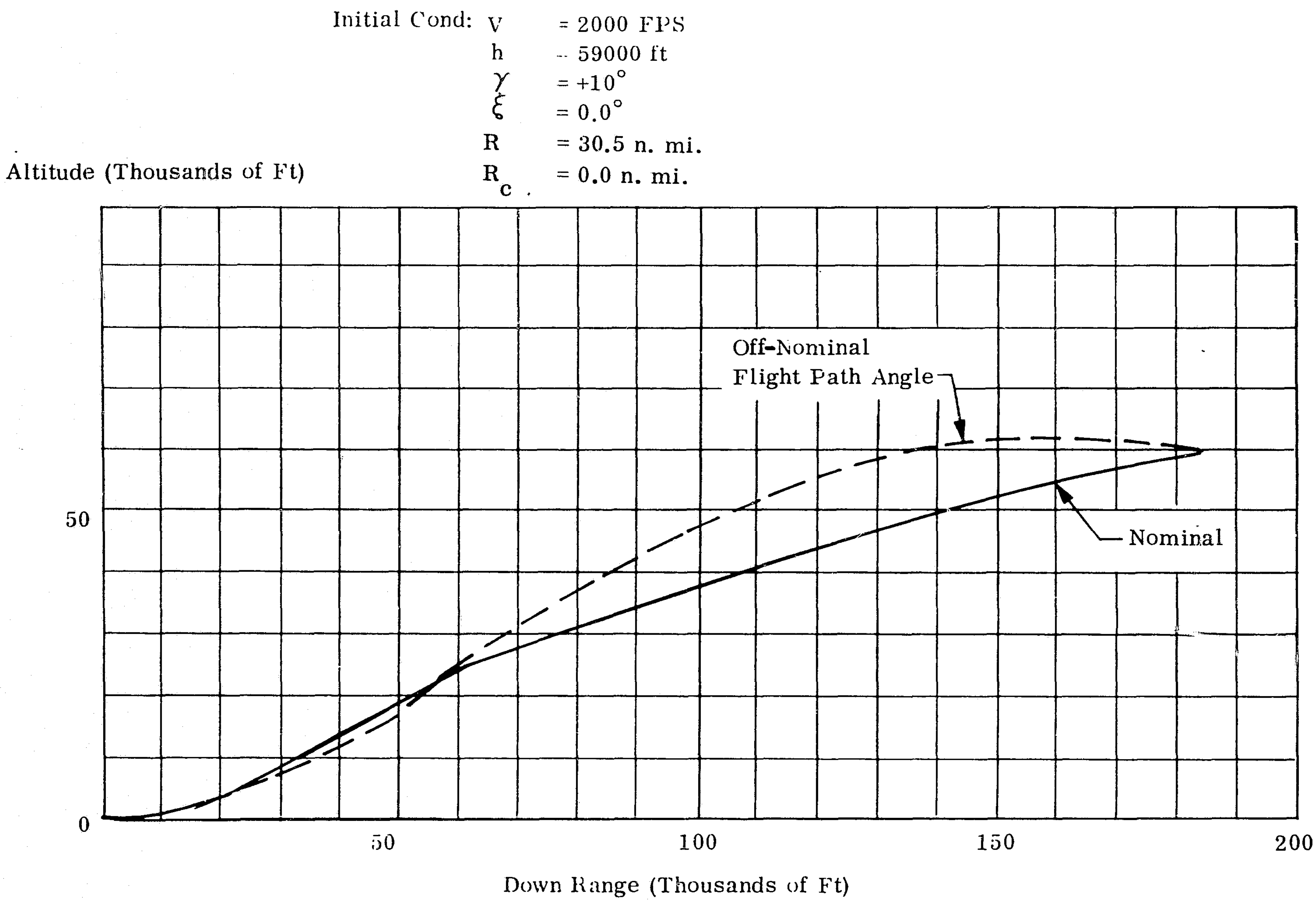


Figure 5.1-25. Influence of Off-Nominal Flight Conditions

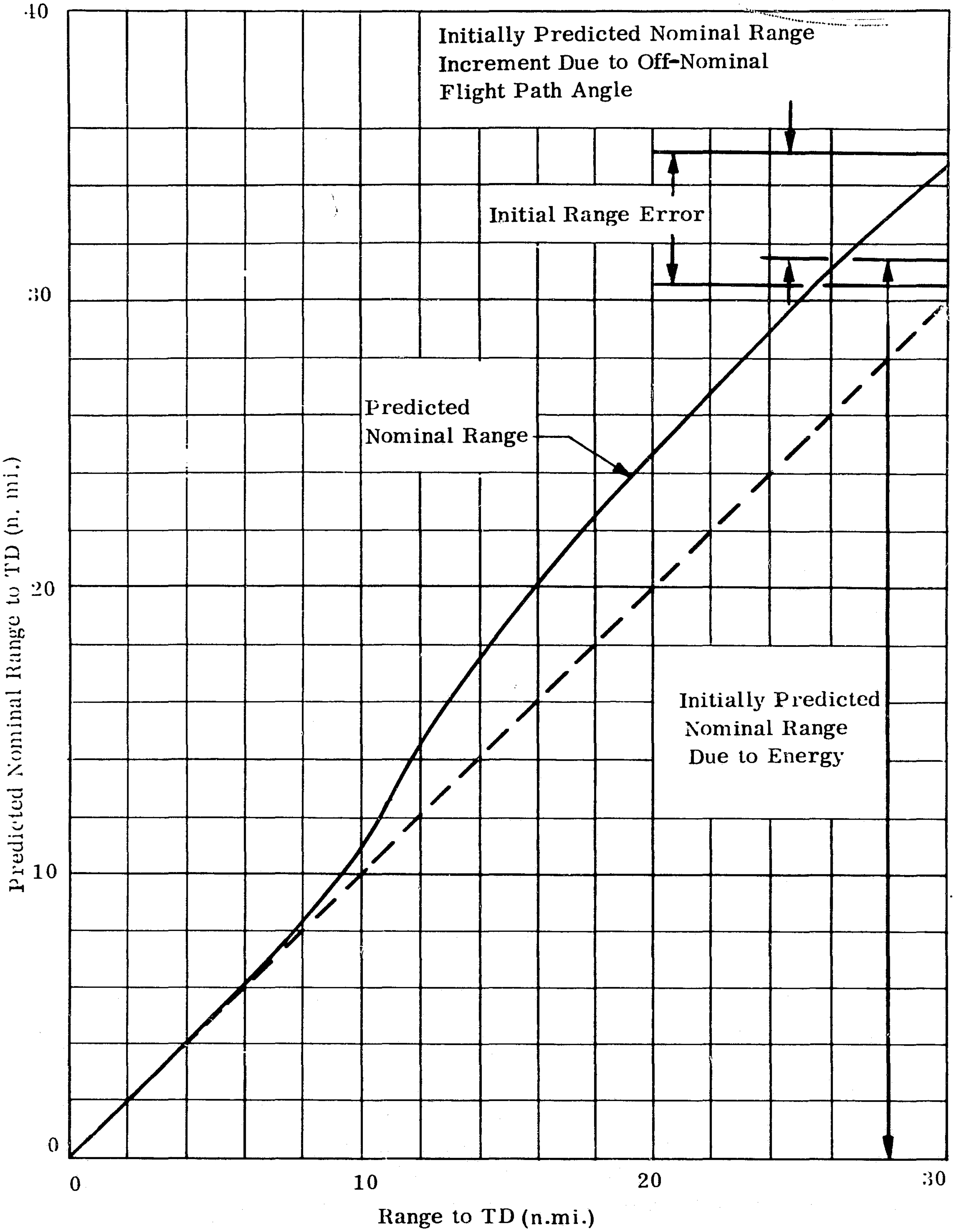


Figure 5.1-26. Influence of Off-Nominal Flight Conditions (Including Range Error)

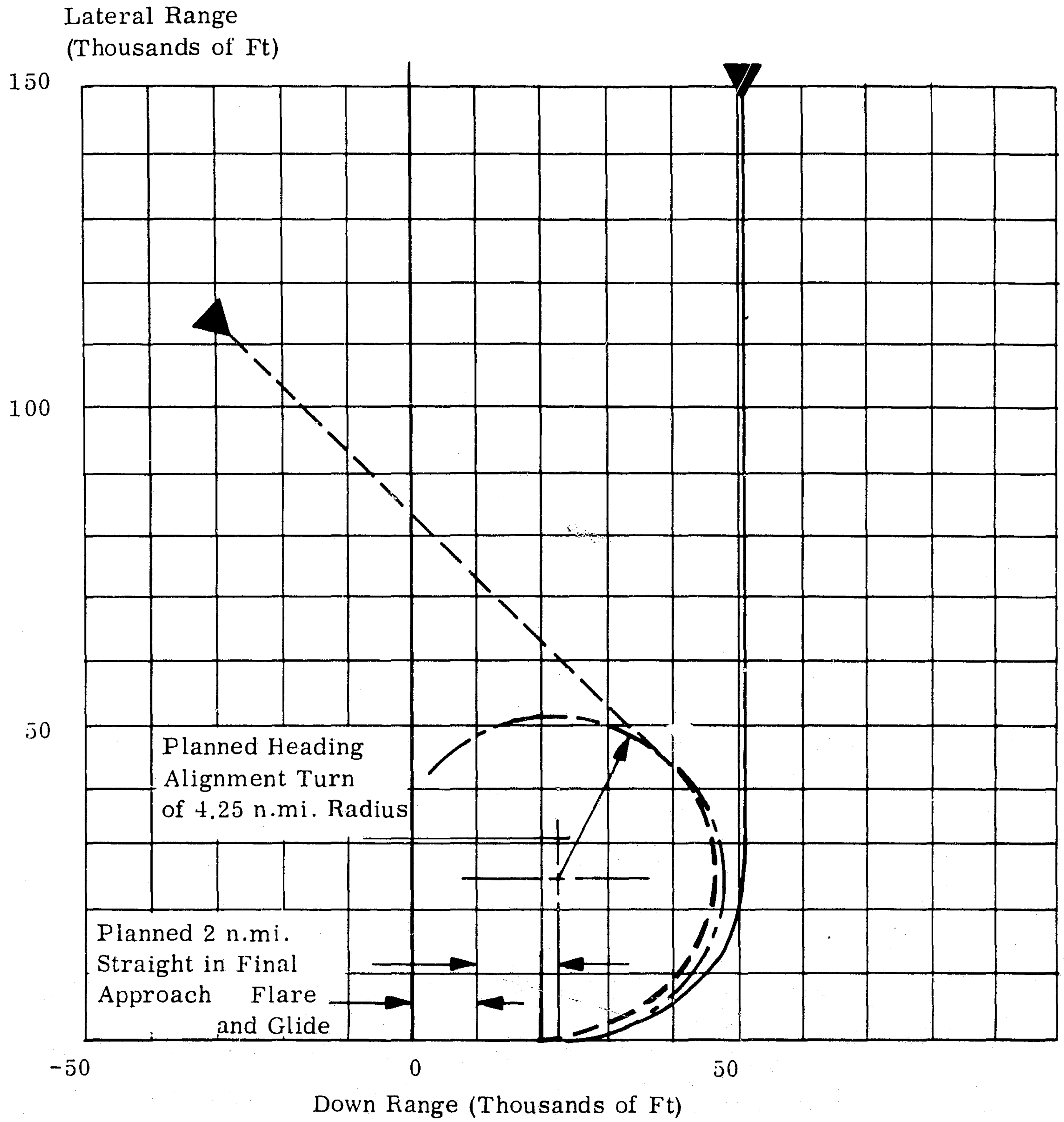


Figure 5.1-27 Influence of Dogleg Maneuvers

These flights demonstrate the ability of the system to control the vehicle through planned dogleg maneuvers or maneuvers that are unplanned but which may be required if wind shifts or other factors that change the planned landing direction are programmed in after the flight is started.

Although it is expected that the vehicle flight conditions will normally be nominal by the time the flare initiation point is reached, in some cases it may not be possible to ensure this. This could happen if the initial range errors are so large that the flight path transient required to correct them is not completed by the time the flare initiation point is reached or when far off-nominal atmospheric conditions are encountered. To evaluate the ability of the system to predict the required flare initiation altitude and to adapt the flare in the presence of these off-nominal flight conditions, two flights were made. On the first of these, the initial vehicle glide path angle was made shallower than nominal as the flare initiation point was approached; and on the second, it was made steeper than nominal.

The trajectories for these runs are shown in Fig. 5.1-28, which shows that the system automatically adapts the flare initiation altitude as required to compensate for the off-nominal flight conditions. Furthermore, it shows that the system controls the vehicle to a new range point at flare initiation to compensate for the effect that the off-nominal flight conditions have on the range traveled during the flare and glide.

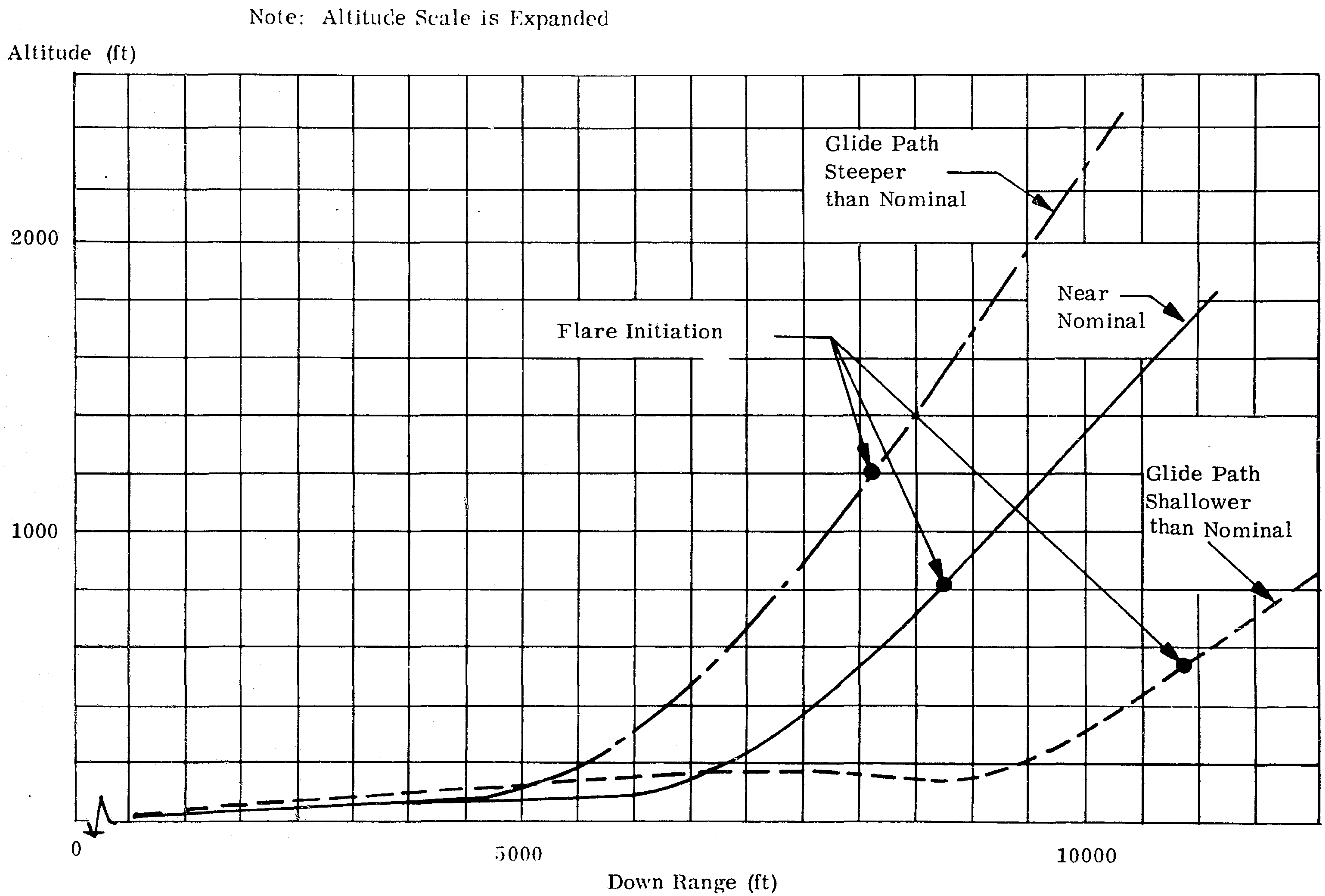


Figure 5.1-28 Influence of Off-Nominal Flight Conditions at Flare

5.1.3.5 Updating Sensors. In the preceding discussion, conventional radar is treated as the source of position data for inertial reference updating. In this section, some of the radar requirements and other sensors that may be applicable to the updating task are discussed.

The need for long-range (100-150 nm) tracking and also a high-precision sensor for the final approach and touchdown may well necessitate two separate radars. For approach and touchdown, a unit much like present precision approach radars should be satisfactory; and for the longer ranges, there are several choices that should satisfy requirements. The radar technology necessary for the update task appears to be well within the state-of-the-art.

One alternative to ground radar for updating is to use the scanning beam, which is an outgrowth of the ILS. Rather than defining a single line in the sky, the reassuring beam provides instantaneous azimuth and elevation on board the aircraft through use of a pair of coded, sweeping beams. Together with a conventional DME, a scanning beam can provide the spatial positioning required for inertial reference update.

The scanning beam system is presently being evaluated by the FAA. Results have been encouraging, and the FAA is considering using it in place of ILS for Category III landing systems. In its present form, the scanning beam has only a 10 degree elevation angle and ± 5 degrees azimuth range capability. This would not be satisfactory for Space Shuttle, where high elevation and all-azimuth capability would be required. In concept, the larger angles could be achieved; the Army is building a unit with 60 degree scan capability.

The scanning beam certainly is an attractive candidate for the Space Shuttle landing system. Of particular interest in further tests will be the determination as to whether a scanning beam system that might be adopted by the FAA would have the flexibility and range required for the Space Shuttle orbiter. As an ILS replacement, a scanning beam need not have wide-angle and long-range capability. If it is to be used by Space Shuttle, perhaps now is the time to make FAA aware of the requirements.

Another means of updating that warrants investigation is the use of conventional navigation aids, such as VOR, Tacan, DME, Loran, and satellites, to provide the long-range, relatively low-precision descent and approach navigation data. For the high-precision final approach and touchdown the ILS, scanning beam, or precision radar could be used. This hybrid approach has the advantage of perhaps the maximum use of existing and available equipment, therefore requiring the least special ground support. Various studies and evaluations are now underway on the updating of aircraft inertial navigators from ground stations, and much of this should be applicable to the Space Shuttle.

On-board radars for position update have been investigated, but there does not appear to be a system (either present or projected) capable of meeting Space Shuttle requirements. The accuracy of state-of-the-art and experimental on-board radars is not sufficient for automatic landing. The U.S. Air Force is experimenting with on-board radar for tactical navigation and low-visibility approaches, but the expected accuracy is far from meeting Space Shuttle requirements. Future developments in on-board radars will be followed nevertheless, since they provide the maximum vehicle autonomy and independence from ground installations.

5.1.3.6 <u>Role of Pilot.</u> The role of the pilot in approach and landing is discussed from two aspects -- automatic versus manual control and pilot displays.

<u>Automatic versus Manual Control.</u> The requirement for all-weather landing establishes the need for capability to touch down on the runway without visual runway reference. All present and projected landing systems have employed automatic control under these low-visibility conditions, and there is no evidence of any acceptable system entailing pilot control for nonvisual landing. Therefore, an automatic control system, without a pilot in the loop, appears to be required for Space Shuttle. For pilot acceptance of automatic landing, repeated demonstration of satisfactory landings in good weather will be necessary. This is the only way to build confidence in the performance and reliability of the landing system. Accordingly, fully automatic control has been selected as the primary mode for all landings, good weather or bad.

Furthermore, it is recommended that control for the descent and approach phase be automatic. In a manual mode, computer-generated navigation and energy management data will normally be used for guidance, so it appears preferable for these data to be fed directly to the automatic flight control system.

The pilot's role during approach and landing is primarily that of a systems monitor and decision maker. Considerable study will be necessary to establish the extent that he should participate. Certainly, he will always have the capability to control the vehicle manually (except in the final phases of a zero visibility landing, where, if automatic control has failed, a go-around is required). Advantage should be taken of the pilot's unique judgment and decision making capabilities for such aspects as failure analysis and alternate field selection. In later phases of this program, the man/machine relationships must be investigated in considerable depth. Present recommendations on pilot versus automatic control roles are summarized below:

Pilot Function	Automatic Function
Supervise and manage entire approach and landing process.	Control vehicle at all times (except during some emergencies).
Perform decision-making function in event of abnormal situations.	Switch to redundant circuits in event of malfunctions (fail operational).
Initiate mode changes, for example, go-around	

Pilot Display Requirements. The pilot must have sufficient flight and vehicle data displayed to him to provide the information necessary to monitor flight progress, exercise judgment in his decision-making role, and control the vehicle when required. Because of the mass of data involved and the criticality of some flight phases (such as zero/zero landing with a single pilot), the display problem is considerably more difficult than with present aircraft. Certainly, a sophisticated data system will be needed for the Space Shuttle.

As the pilot role becomes defined in the later phases of the program, it will become necessary to examine the display problem in further detail. Display requirements necessarily depend upon the specific pilot tasks and monitoring required. These items will be investigated in detail.

5.1.3.7 Reliability and Redundancy. Assurance of a safe landing under zero/zero conditions in event of hardware failures requires a highly reliable and redundant landing system. The term "fail operational" is used to describe a system that will remain correctly and safely operational following a single failure. This concept, presently being applied to automatic landing systems on the new generation airliners, must be applied to the Space Shuttle to provide adequate system integrity for zero/zero landing.

A fail-operational system requires independently redundant channels with appropriate monitoring and switching elements to assure system operation in event of failure. The simplest concept is a system with two identical operational channels and a third monitoring channel to "vote" in event of disagreement between the operational channels and to switch out the incorrect channel.

In practice, there are many schemes for implementing a fail-operational system; so detail investigation is needed to select the optimum route. With any route, careful design and analysis will be required in mechanization of landing system elements to assure reaching reliability goals without creating a monstrous (and inherently unreliable) monitoring and switching system. This latter aspect is one of the most difficult and critical that will be facing the subsystem designers.

5.1.3.8 Booster Aspects. The booster vehicle has somewhat different requirements for approach and landing; therefore, booster tradeoffs and hardware considerations are somewhat different from those pertaining to the orbiter.

The booster will return to the launch base or alternate field at subsonic speeds with the jet engines thrusting. The navigation problem is much like that with present aircraft navigation; certainly it is much simpler than the reentry and energy management problem of the orbiter. Also, the booster does not require the unpowered landing capability; therefore, the control problems are very similar to those with aircraft.

It appears that booster return and landing could be accomplished according to present aircraft practices. For example, return navigation could be by VOR or other radio aids and zero/zero landing by ILS, as described in Section 5.1.3.2. This is not to say that the booster should not be operated with the same techniques and systems as the orbiter, but rather that the booster does not have (for return and landing) the same requirements. For ascent and the 180 degree turn after separation, the booster will require some sort of inertial control; consequently, it may be preferable to use this equipment for return and landing. A definite consideration is to use the same guidance and control system as the orbiter's to reduce development costs. In this event, approach and landing would be accomplished in the same manner as for the orbiter.

It appears to be feasible to fly the booster unmanned. Because of the high degree of automation and redundancy required to provide all-weather landing capability, the necessary pilot functions are of a decision-making nature. All normal flight operations and redundant channel switching in event of component failures would be accomplished on board. A ground-based pilot would have such functions as commanding go-around. Possibly he could fly and land the booster in drone fashion, although the need for this capability remains to be established.

The question of unmanned booster operation is really one of design execution rather than concept. Vehicle systems must have the flexibility and reliability to meet whatever performance and safety standards are established for unmanned operations. The guidance and control system developed in this report for the orbiter would be particularly attractive. The high level of vehicle autonomy and flexibility of the on-board computer should enable a straightforward unmanned operating mode. It is recommended that if unmanned booster operation is desired, specific operational criteria be established to enable development of system design requirements.

5.1.4 Summary

The major conclusions in this study are summarized below, and the critical technical issues warranting further study are presented.

5.1.4.1 Major Conclusions.

- Power-on landing of Space Shuttle is possible. Velocities, sink rates, attitudes, etc., are comparable to those of present high-performance aircraft.
- Based on X-15, HL-10, and M2-F2 flight test results, unpowered approach appears to be possible for emergency (or perhaps primary) landing. Feasibility must be confirmed.
- Vehicles appear to be compatible with operation on 10,000-ft runways.
- Penalty to provide go-around capability is high weight of jet engines and fuel.
- Variable-geometry wings not necessary. Landing speeds are acceptable without wings.
- All-weather landing appears to be feasible, but requires fully automatic control.
- Onboard guidance with updating is attractive landing system.
- Role of pilot is not clear; it includes system monitoring at a minimum.
- Landing system reliability will be major problem.
- Unmanned booster operation appears to be feasible.

5.1.4.2 Critical Technology Areas.

- Feasibility of unpowered landing must be established. Flight tests with low L/D with variable stability aircraft (e.g., CAL T-33) to simulate Space Shuttle flying characteristics are suggested.
- Candidate landing fields for a returning orbiter should be identified more specifically. (Choice of an automatic landing system depends in part on the type of fields that will be used.)
- Criteria for vehicle handling qualities must be established.
- Further analysis of runway length requirements is warranted. Identification of potential landing sites should be included.
- The need for powered landing and go-around elimination should be examined in the light of resultant major system weight benefits.
- Further guidance simulation with Space Shuttle configuration and trajectories should be performed, and guidance laws and landing system hardware requirements should be defined.

- Integration of approach and landing guidance with reentry guidance should be studied.
- Roles of the pilot and automatic system should be defined further.
- Required piloting aids -- displays, controls, stability augmentation, etc. -- should be established.
- Reliability requirements should be defined and system solutions -- redundancies, monitors, switching logic, etc., -- formulated.
- Further investigation of RF sensors for landing system updating is warranted.

5.2 SELF-FERRY

This section is concerned with the capability of the orbiter and booster vehicles for self-ferry; i.e., to fly as an aircraft from one airfield to another. Ferry performance has been evaluated on the basis of a range of jet engine thrusts and fuel capacities. The restrictions on range and the problems of increasing range are discussed. Possible landing fields for ferry and considerations for in-flight refueling are presented. Also, a discussion is included of the tradeoff between accepting a low ferry range and increasing vehicle launch weight to increase the inherent range capability.

5.2.1 Ferry Performance

A parametric study has been made to establish the ferry capabilities of the orbiter and ferry vehicles. The ferry operation is defined as a vehicle taking off over a 50-foot obstacle, climbing to altitude, cruising at altitude, descending and landing. The parameters affecting this operation are as follows:

- <u>Jet Engine System</u>. The parameters in the jet engine system selected for these vehicles are thrust and specific fuel consumption and their variations with Mach number and altitude.

- <u>Operation</u>.
  Vehicle Takeoff: In this phase, the method of takeoff is essential to determining how the vehicle will become airborne. The parameters affecting takeoff are vehicle weight, attitude of the aircraft in order to effectively use the vehicle's aerodynamics, and the surface conditions of the airport runway in rolling friction of the wheels on the runway.
  Climb to Altitude: The parameters influencing the phase are velocity or Mach number, and specific fuel consumption. The controlling factor here is establishing the most efficient mode to attain the desired cruise altitude.

Cruise: Since the cruise phase dominates the ferry range, it is essential that the fuel consumption during this phase be held to a minimum. The parameters affecting this phase are altitude and velocity.

Descent: The mode of descent to accomplish a successful landing of the vehicle is influenced by such parameters as descent velocity and vehicle altitude.

Landing: (This phase of the ferry mission is described in section 5.1.)

5.2.1.1 Study Approaches

- Jet Engine System. The jet engine system used in the study is representative of a Pratt & Whitney rubberized turbofan engine (bypass ratio of 5) assumed operating at maximum power for takeoff, and at maximum continuous power for climb.

- Operation.

  Vehicle Takeoff: In this phase the approach was to assume that the vehicle uses four jet engines for takeoff at maximum takeoff thrust and clears a 50-foot obstacle. In this aspect the approach assumed that the vehicle maintains zero angle of attack until takeoff, where it is instantaneously rotated to takeoff attitude; i.e., for the orbiter α = 20 deg and for the booster α = 15 deg.

  Climb: In this phase it was assumed that the vehicle climbed at an average velocity corresponding to that attained at an altitude midway between sea level and cruise.

  Cruise: In this portion of the study the ranges to be covered were maximized by Breguet's formulation of the range equation.

  Vehicle Descent: In this phase it was assumed that the distance covered in descent is identical to that covered in climb to altitude and that the fuel expended was half that used during climb.

The performance profile for a typical ferry mission for the orbiter and booster vehicles is shown in Fig. 5.2-1. Goaround capability has not been

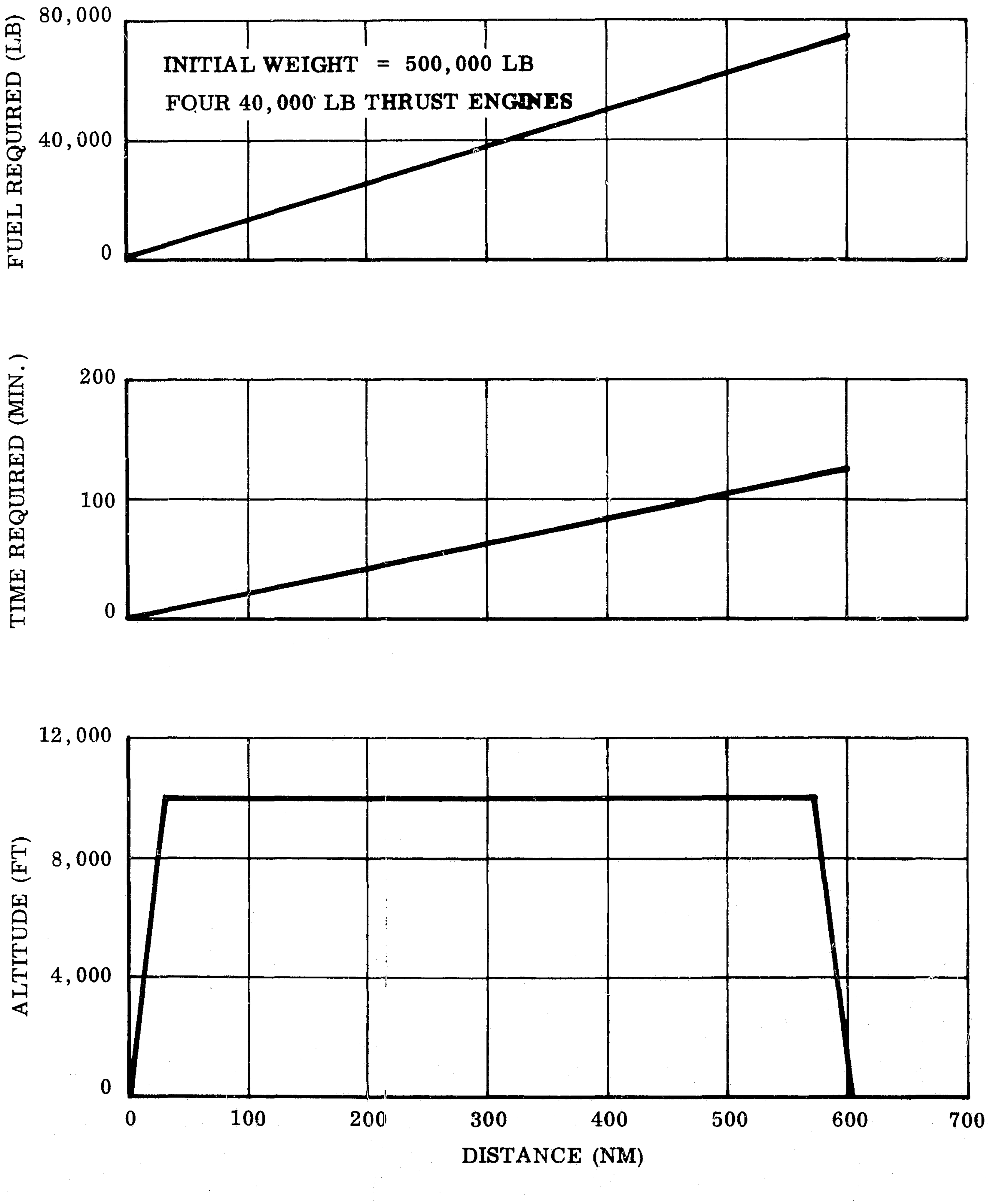


Fig. 5.2-1 Typical Profile for Ferrying Booster Vehicle

included in this study. The ferry performance presented here used the aerodynamics for the two vehicles shown in Fig. 5.2-2.

5.2.1.2 Ferry Range Parametrics

The carpet plots (Figs. 5.2-3 and 5.2-4) show the fuel required for four 40,000-lb thrust engines to power the orbiter and booster 200, 400, and 600 nm at cruise altitudes of sea level, 5,000 ft., and 10,000 ft. In this study, the orbiter was assumed to have initial takeoff weights ranging from 250,000 to 400,000 lb; for the booster the takeoff weights range from 300,000 to 500,000 lb. From Fig. 5.2-3, it can be seen that the orbiter having a takeoff weight of 390,000 lb would have difficulty in cruising at 10,000 ft, since this vehicle is already at its service ceiling (rate of climb is 100 ft per min).

From these figures, it can be seen that the effect of altitude is one of reduced fuel requirement. However, this advantage may be of minor significance when evaluated over the range to be traversed.

Other engine thrust levels have been analyzed; findings are as follows:

- The significance of engine size is on rate of climb, service ceiling,and takeoff distance.
- Engine size did not affect the conclusions drawn from the carpet plots presented as far as range and fuel weight requirements.

5.2.1.3 Takeoff Performance

Takeoff performance has been investigated parametrically for the orbiter and booster vehicles from the standpoint of effects of vehicle weight and jet engine size. The results are depicted in Fig. 5.2-5 for the orbiter and Fig. 5.2-6 for the booster. This study assumed sea-level conditions and standard day operation. It is interesting to note here that at takeoff the orbiter, with an angle of attack of 20 deg, uses a greater lift force. However, the booster must takeoff at a lower angle of attack (15 deg in this

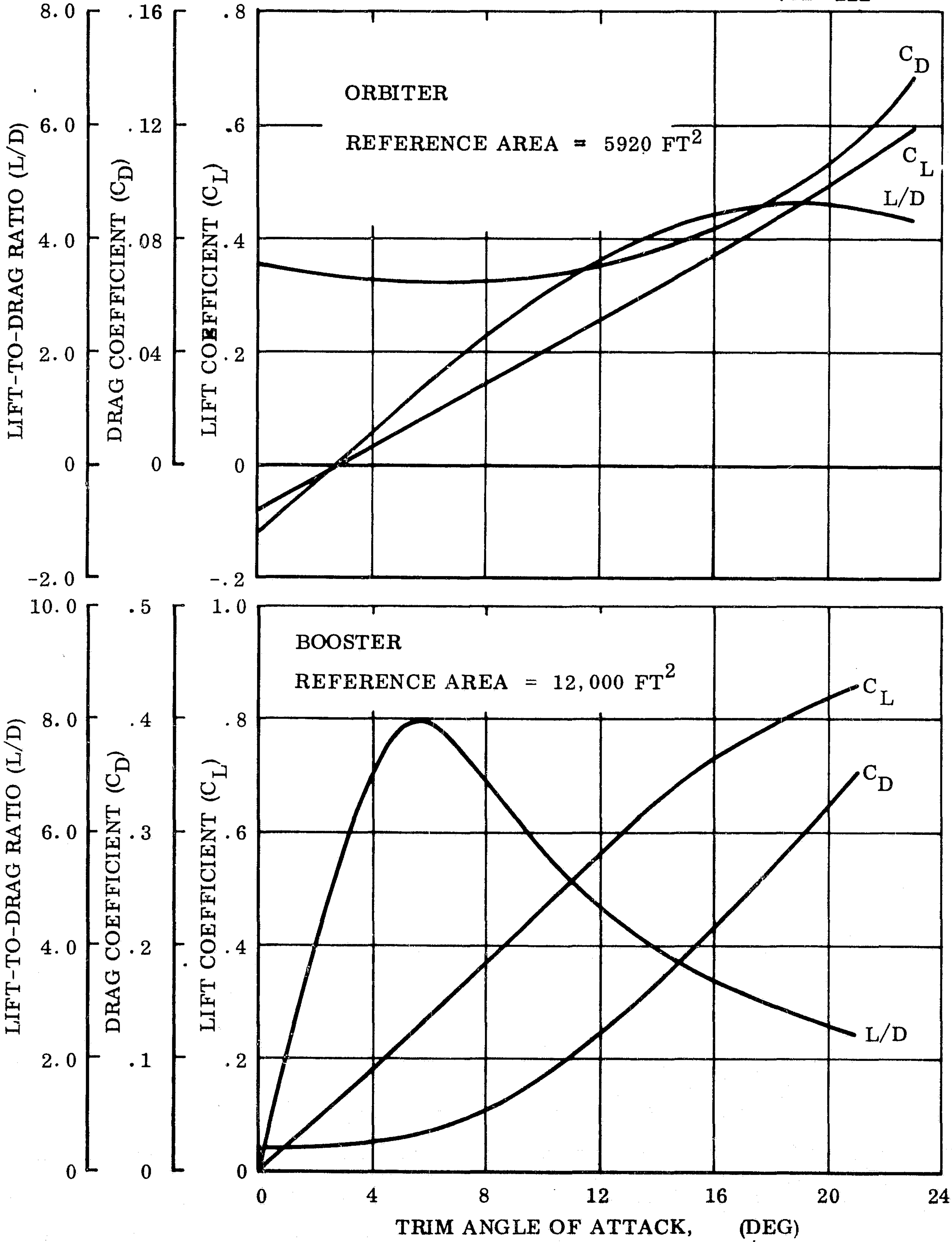


Fig. 5.2-2 Orbiter and Booster Subsonic Aerodynamic Characteristics

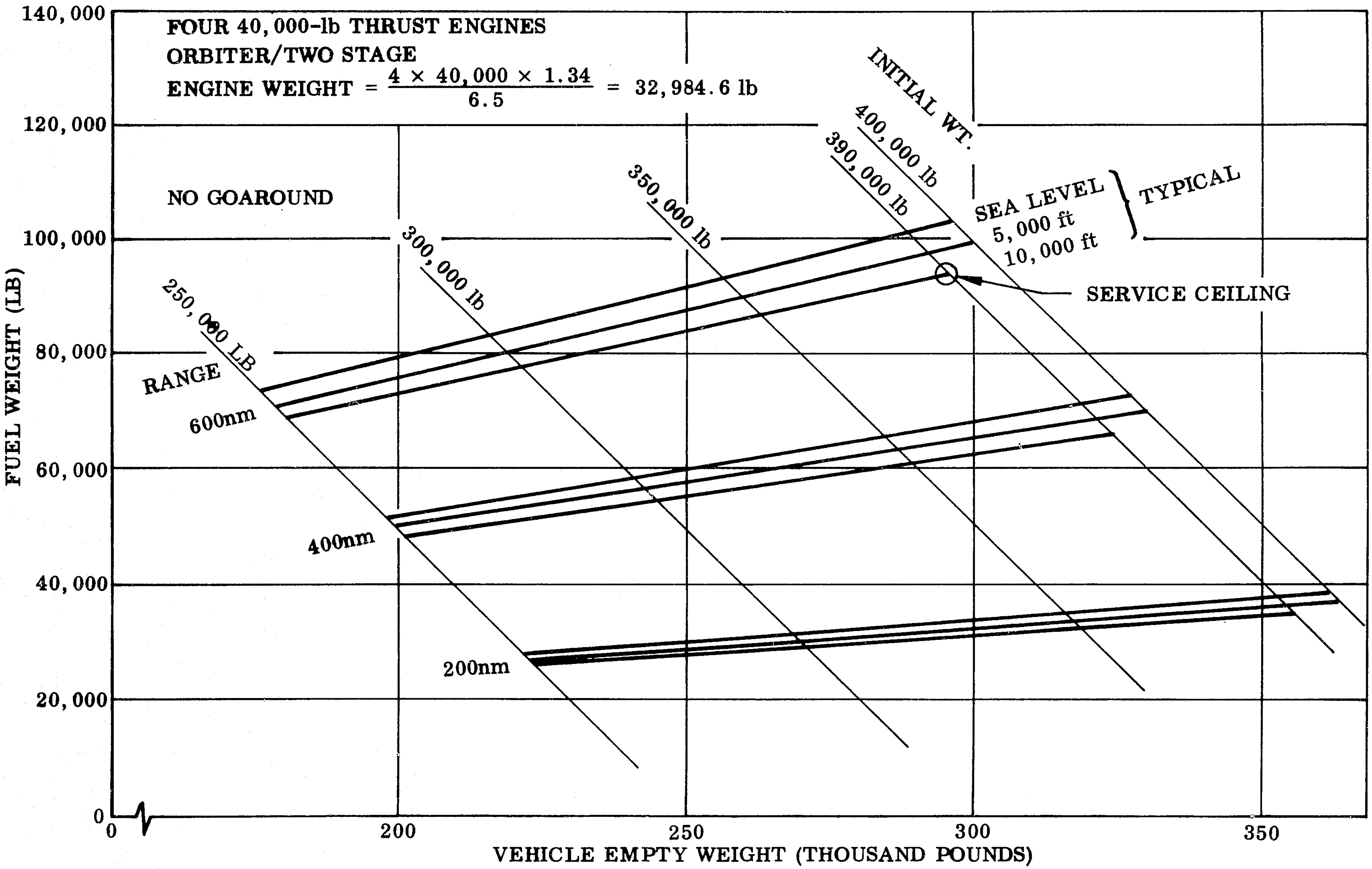


Fig. 5.2-3 Fuel Required for Ferry Mission of Orbiter at Various Altitudes and Initial Weights - Orbiter/Two-Stage

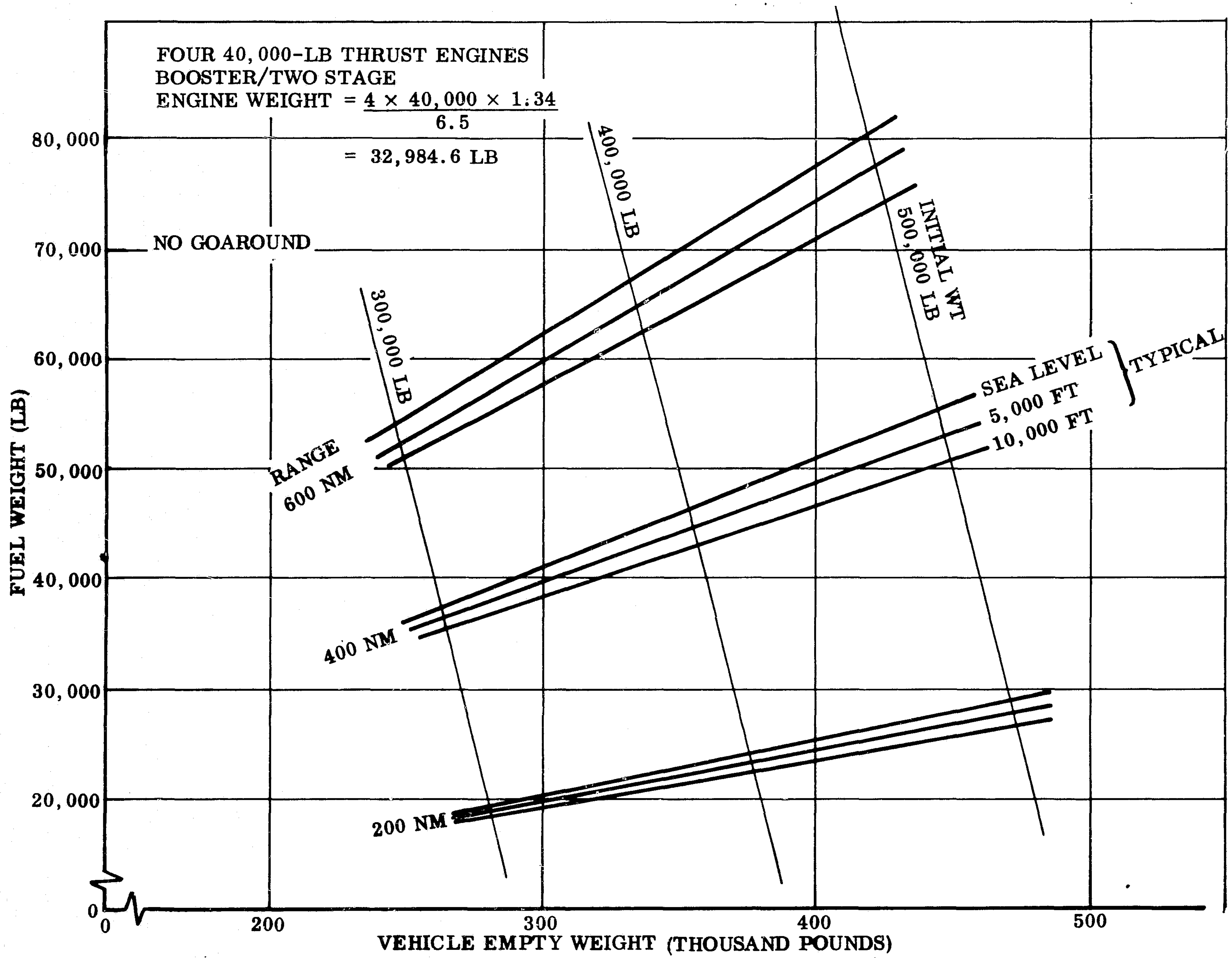


Fig. 5.2-4 Fuel Required for Ferry Mission of Booster at Various Altitudes and Initial Weights - Booster/Two-Stage

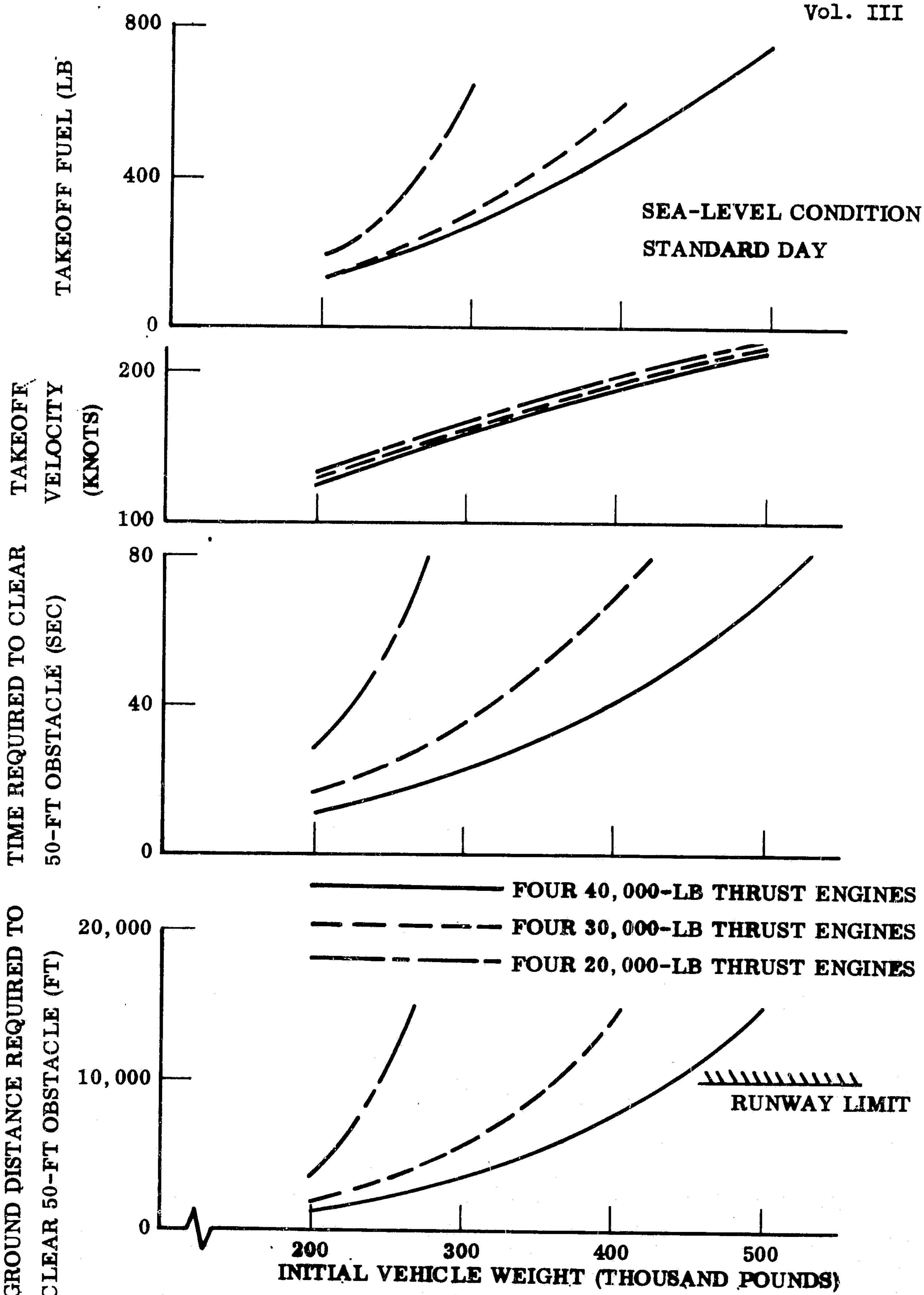


Fig. 5.2-5 Takeoff Performance for Orbiter Vehicle

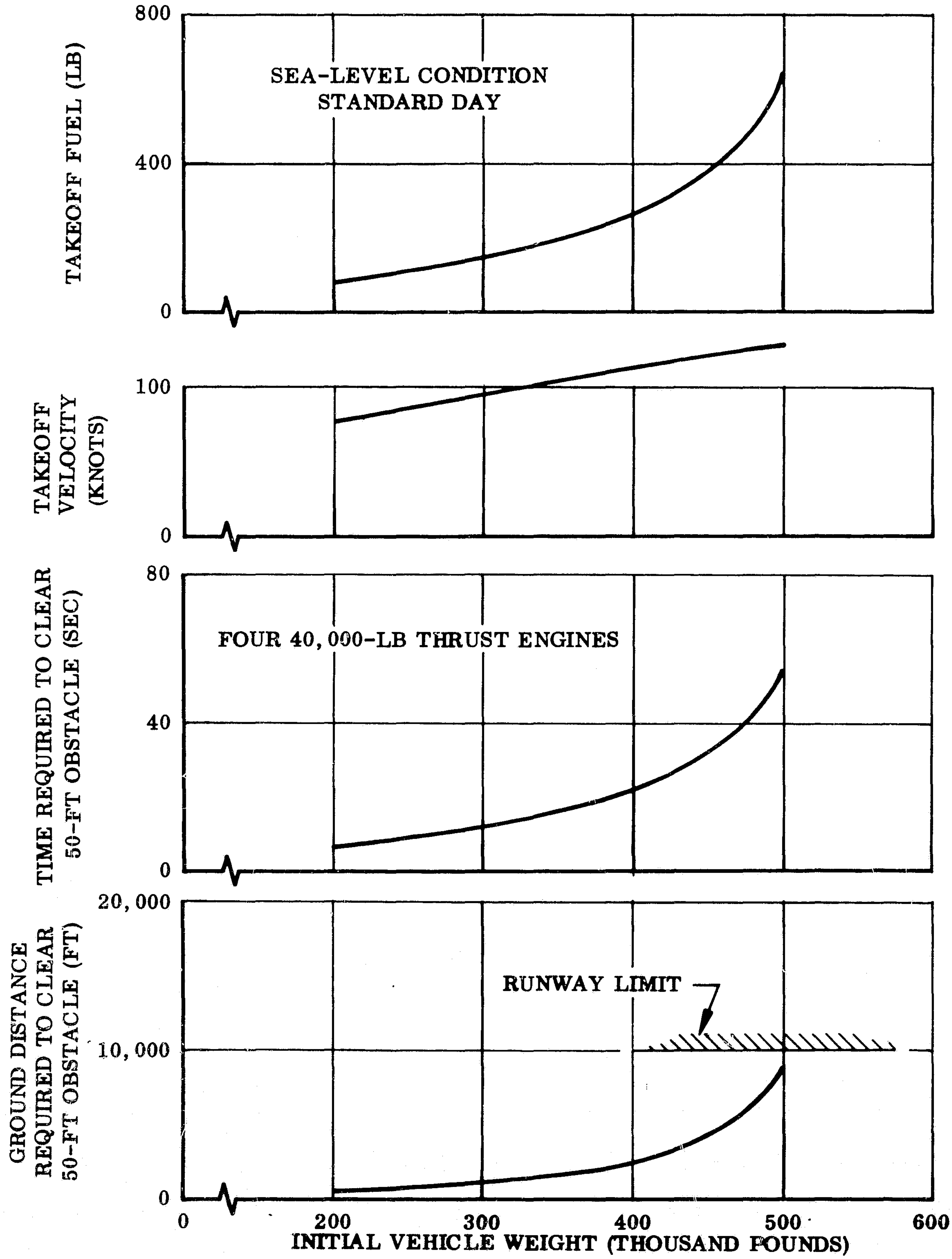


Fig. 5.2-6 Takeoff Performance for Booster Vehicle

study) to use the lower drag coefficient without drastically reducing the vehicles lift. At higher angles of attack, the booster has difficulty taking off with four 40,000-lb thrust engines, and jet assist is necessary.

This study is continuing to include the effects of hot and cold day and altitude performance as well as vehicle malfunctions.

The other consequential aspect of the jet engine is its influence on cruise altitude. This effect is shown in Fig. 5.2-7, which shows the influence of vehicle weight and jet engine thrust on the orbiter's service and absolute ceilings. With the engines presently considered for the nominal launch system (four 25,000-lb thrust, sized for goaround), Fig. 5.2-7 indicates that the vehicle would be limited to a 10,000-ft ferry cruise altitude.

5.2.1.4 Performance Summary

Figures 5.2-8 and 5.2-9, for the orbiter and booster respectively, relate the ferry mode to the vehicles making up the nominal launch systems. From these figures it can be seen that:

- For the 50,000 lb of fuel for the ferry (replacing the payload with fuel), the orbiter cruising at 10,000 ft has a range of approximately 350 nm, and cruising at 5,000 ft its range decreases to 333 nm. For the same amount of fuel, the booster has a range of 368 nm if it cruises at 10,000 ft, and if cruising at 5,000 ft its range decreases to 350 nm.
- Range is not significantly dependent upon engine thrust. If the cruise altitude is below the service ceiling for the vehicle/engine combination, it can cruise at that altitude.
- Both nominal vehicles using four 40,000-lb thrust engines have the capability of taking off in less than 3,000 ft of runway.
- For both nominal vehicles, fuel weight has little effect on landing distance, because most of the fuel weight has been used, and the vehicle is near its empty weight.

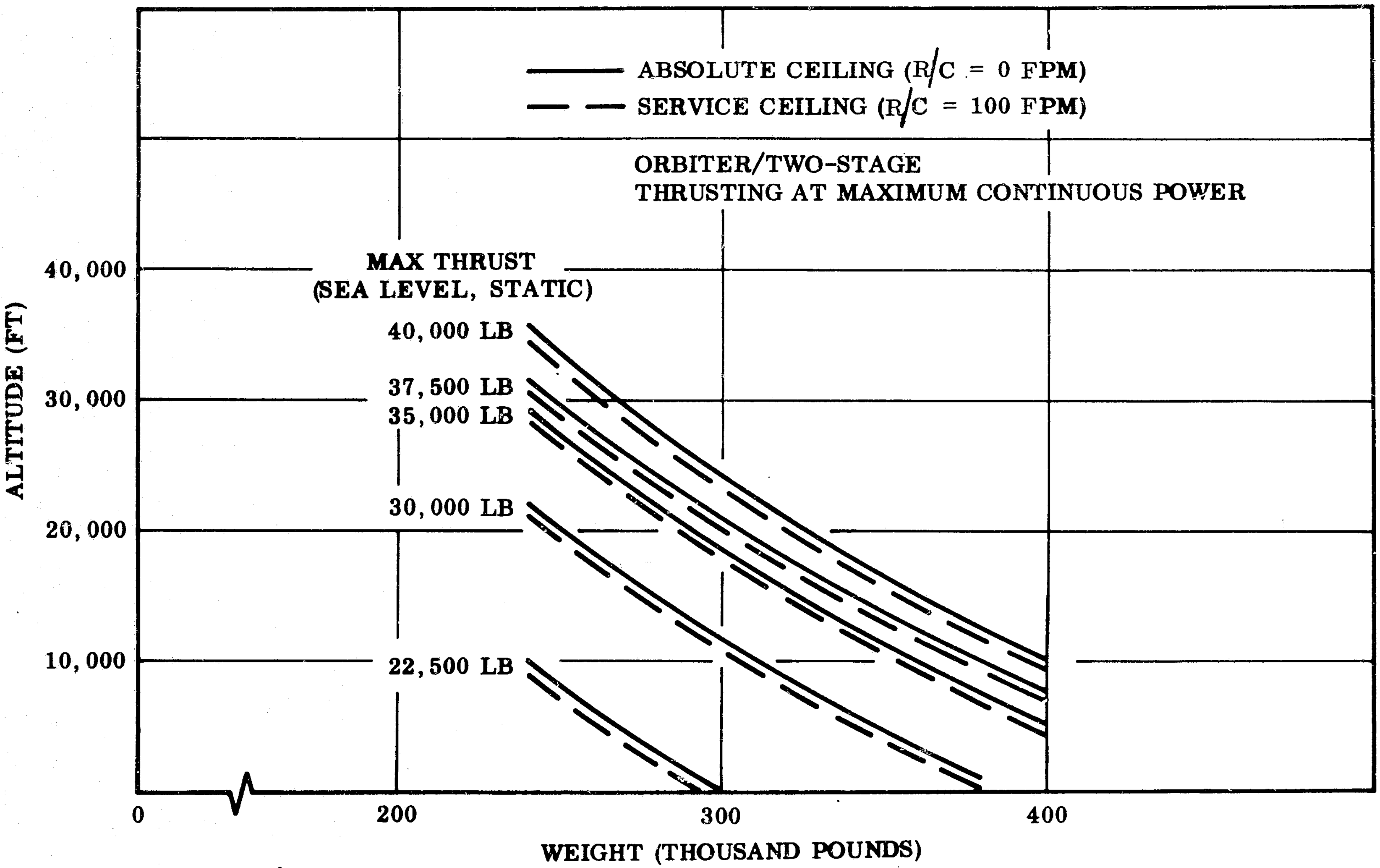


Fig. 5.2-7 Absolute Service Ceiling - Altitudes vs Vehicle Weight

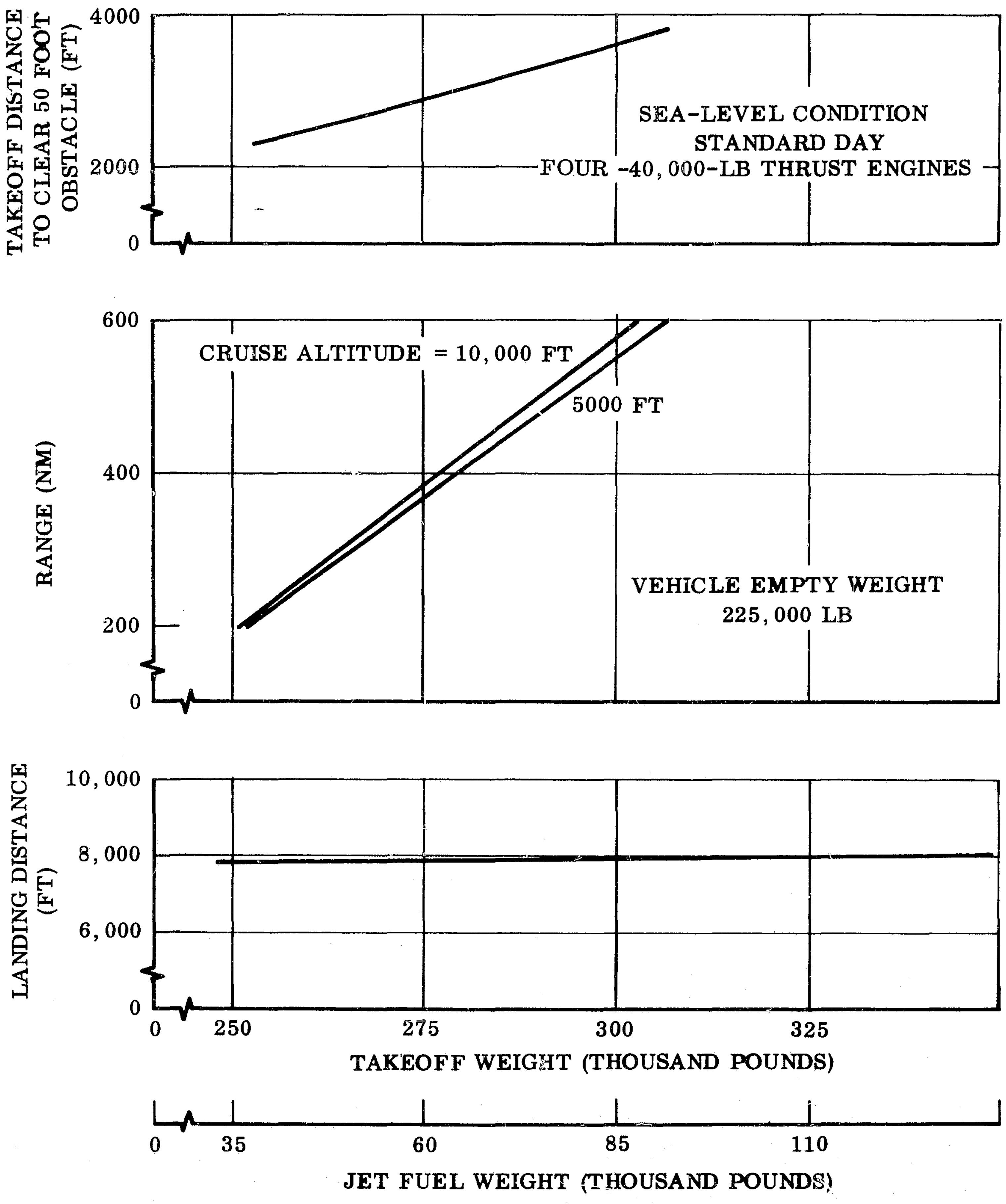


Fig. 5.2-8 Ferry Performance for Orbiter Vehicle

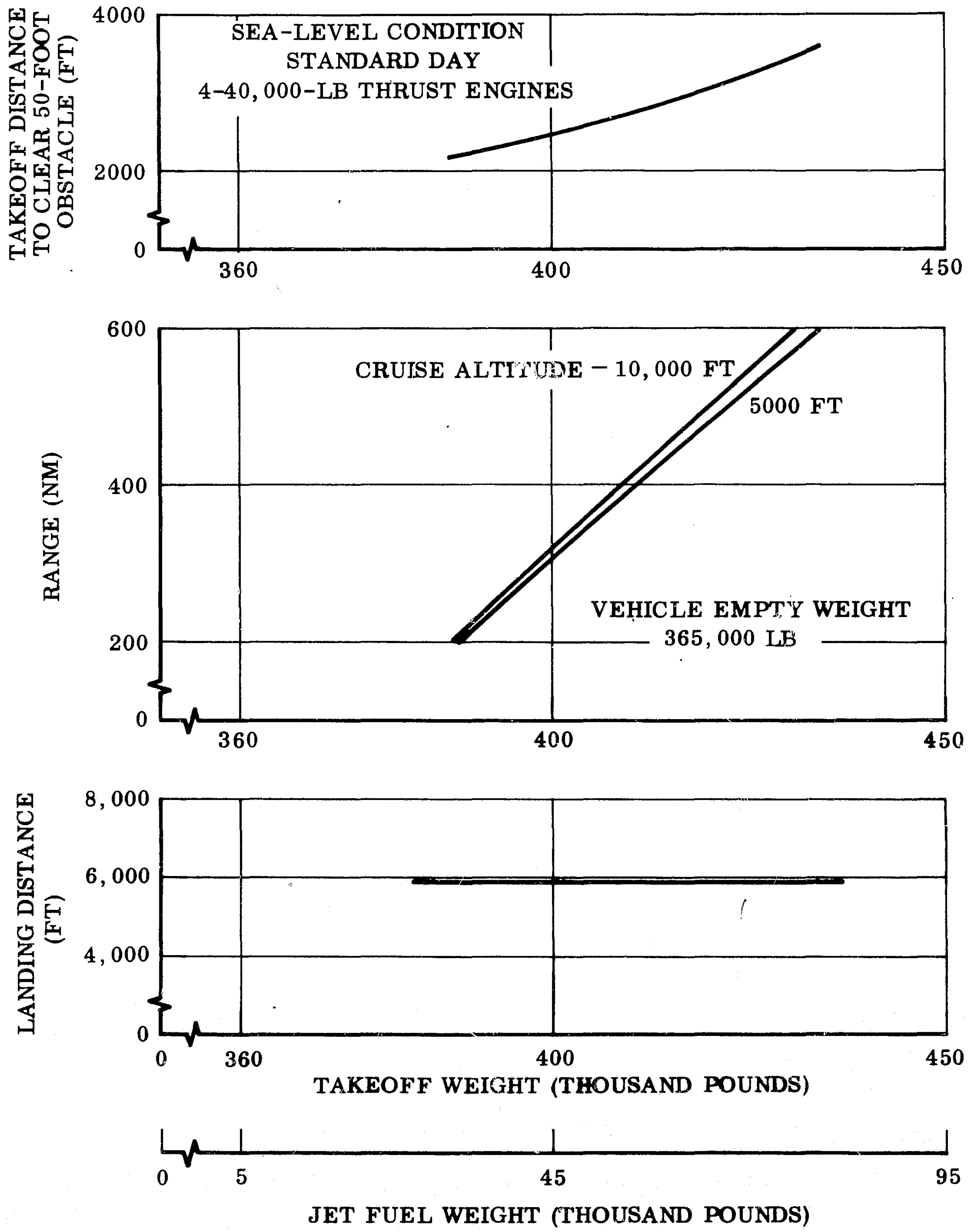


Fig. 5.2-9 Ferry Performance for Booster Vehicle

5.2.2 Operational Aspects

The operational requirements and considerations for self-ferry heavily influence the need for ferry range. Self-ferry is required for such missions as return to launch base from an alternate landing site, and ferry from manufacturing plant to launch base.

Figure 5.2-10 shows airfields in the 48 contiguous states that are candidates for Space Shuttle use. All have a minimum runway length of 10,000 feet and a minimum width of 200 ft. Excluded are some high density civilian airfields and all inactive fields. It is seen that for any ferry mission (including a West Coast to East Coast trip), a minimum ferry range of 300 to 400 nm would be required. The most critical leg appears to be between Davis-Monthan AFB, Arizona and Holloman AFB, New Mexico. This is the most attractive route across the Continental Divide, but still is a 300 nm leg over relatively high terrain with few emergency landing fields.

Outside the 48 states the situation is much different. To return from overseas bases could require ranges in excess of 2000 nm (for example, from Hawaii to the West Coast); this, of course, would require inflight refueling.

5.2.3 In-Flight Refueling

In-flight refueling is attractive for extension of ferry range without incurring the vehicle weight penalties involved in increasing jet fuel capacity and engine thrust. It appears that the "flying boom" refueling technique used on Strategic Air Command B-52 bombers will be applicable to Space Shuttle vehicles. Both the orbiter and booster are expected to have stable handling characteristics during cruise and should be adequate for the stationkeeping task required for refueling. Data on the capabilities of the KC-135 tanker were not available for this study, but it is anticipated that the data will be compatible with the orbiter. The booster's low-cruise airspeed might possibly be a problem for a KC-135, but data were not available to confirm this.

The major problems with in-flight refueling may well be those associated with the low service ceiling of the orbiter and booster. Results of early analysis (section 5.2.2) indicate that both may be limited to near 10,000-ft altitude,

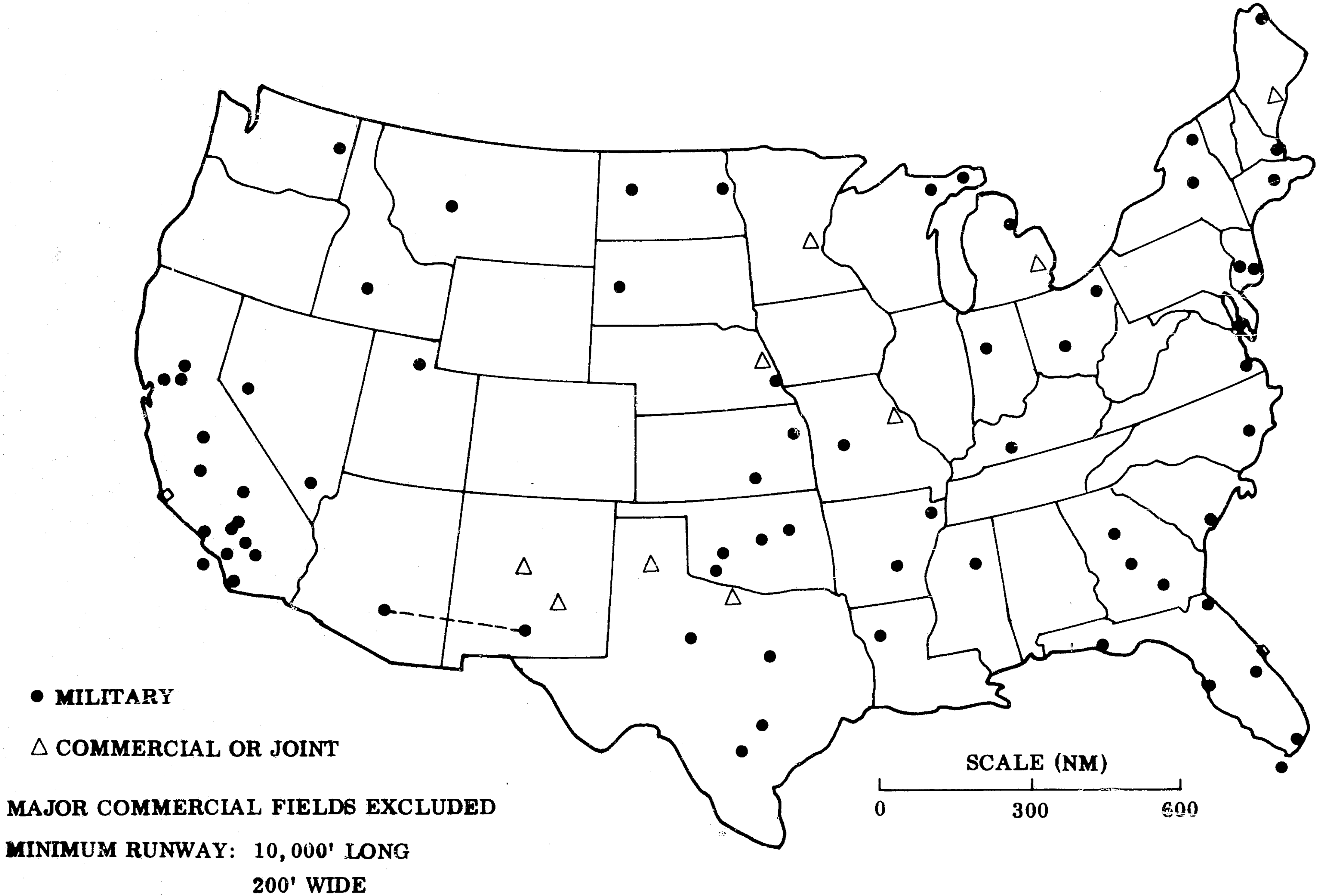


Fig. 5.2-10 Candidate Airfields for Ferry

whereas refueling is normally done above 20,000 ft. At 10,000 ft, problems arise because of weather (higher probability of cloudiness), turbulence (ground heating effect), and terrain clearance over mountain areas.

Nevertheless, as discussed in section 5.2.6, in-flight refueling offers the ability to increase ferry range without increasing vehicle weight. It should be investigated in depth if additional range is desired.

5.2.4 Impact of Ferry Capability on Vehicle Design

To reduce system launch weight, fuel tanks for ferry cruise have not been designed into the basic vehicles. For ferry, special fuel tanks will be added.

For the orbiter, it is envisioned that the payload will be removed and a fuel tank installed in its place. The 15 ft by 60 ft payload bay has a volume of 10,600 cu ft. Since 50,000 lb of jet fuel would require a volume of only 1000 cu ft, sufficient volume exists for any feasible fuel load. The limits would be based on such factors as takeoff distances and structural loads. Takeoff distances are discussed in section 5.2.2. The structural load capability would certainly exist for 50,000 lb of fuel; the structural impact of loads beyond this value will have to be assessed. It is expected that landing gear loads during taxi and takeoff will become critical.

For the booster, there is no payload to be removed, so the fuel and tank is an increment over the basic vehicle. Presently, it is envisioned that the fuel will be carried in an externally mounted tank added for ferry, but this concept is not fixed. The effect of the added weight on structural loads will be assessed also.

5.2.5 Conclusions

In section 5.2.2, with 50,000 lb fuel for ferry, it was concluded that ferry range for both the booster and orbiter is limited to 300-400 nm. The question now arises as to whether this rather low range should be accepted, or whether the vehicles should be designed for greater range.

5.2.5.1 Implications of 300-400 nm Ferry Range

With this limited range capability, there is a choice, for longer trips, between using in-flight refueling or landing frequently to refuel. The acceptability of this range limit in many respects depends upon the type of ferry missions that will be necessary. If there will be frequent long-distance missions, the operational requirements and problems associated with frequent landings and or in-flight refuelings may be unacceptable. On the other hand, if long missions are infrequent, then most likely the operational inconveniences on those rare occasions would be tolerable.

Further operational analysis is necessary to answer this question, but in any event it does appear that the 300-400 nm range provided by 50,000 lb ferry fuel could be acceptable.

5.2.5.2 Implications of Increasing Ferry Range

From Figs. 5.2-8 and 5.2-9 it can be seen that to increase ferry range to, say, 600 nm requires a fuel load of 90,000 lb for the orbiter and 80,000 lb for the booster. In general, it can be stated that to carry these fuel loads will require modifications to the nominal vehicles described in Volume I of this report.

For the orbiter, with the present four 25,000 lb jet engines, Fig. 5.2-5 indicates that a standard, day takeoff with 50,000 lb of fuel requires 6500 ft of runway. With 90,000 lb of fuel, approximately 10,000 ft would be required, clearly an unacceptable takeoff run. Additional thrust for takeoff would be necessary with 90,000 lb of fuel, requiring either larger jet engines or thrust augmentation such as after burners or JATO. The increased takeoff weight (45,000 to 60,000 lb) will increase structural design requirements, probably including design loads on the landing gear.

The inevitable increase in orbiter empty weight will be reflected in an increased dry weight in orbit. To impose this penalty on the overall booster-vehicle system appears undesirable at this point unless increased ferry range is a highly desirable characteristic. More analysis is necessary to support a requirement for longer range and the associated vehicle penalties.

Appendix B
SUMMARY OF ELECTRONICS COMPONENT TECHNOLOGY (1972)

The evaluation of an integrated electronics system (IES) requires a projection of component types and characteristics that will influence the development of this concept. Areas that have been reviewed are signal acquisition, computational elements and data storage and distribution.

MISSION DEFINITION

The requirement is for a multimission space vehicle used in earth orbit - specifically, a cargo transfer spacecraft. Environmental considerations are mechanical (acceleration, vibration, and shock), physical (temperature, pressure, and radiation), and mission time. The first grouping has a relatively mild requirement, resulting from a profile definition for "average" personnel transfer. Thermal considerations, including active cooling if required, are the most important in the next group. Pressure is not a problem area, and radiation levels that are safe for man are light doses for electronic piece-parts (500, 10^4, and 10^7 RADS for man, MOS, and bipolar respectively). The mission time requirement is considered to be the most difficult criteria to satisfy.

DISCUSSION

Although projection based on published literature and interviews with various users and suppliers is highly subjective, current thinking can still be used to relate the development schedule to the 3-5 year gestation period (concept to flight) for electronic systems. The consensus is that LSI (defined as 100 gates per chip) will be available with established reliability/confidence levels for most common digital functions. These will be constructed by the use of both bipolar and P-MOS processes. Complementary MOS (C-MOS) circuits will be developed in lesser quantities.

Hybrid circuits will be catalog items for applications requiring a variety of components, e.g., an oscillator requiring a crystal, resistors, capacitors, and semiconductors.

It is believed that an integrated system designed and constructed today would have performance characteristics similar to those of the 1972 unit. Anticipated gains are improved reliability (fewer parts), lower power consumption (MOS parts), a slight speed increase (reduced external connections as well as piece-part improvements), and modest weight and volume improvements.

Expanding upon the areas reviewed, signal acquisition implies conditioning (amplification/filtering), signal selection (multiplexing), and formatting (A/D conversion). Hybrid active filters, consisting of IC amplifier, film resistors, and chip capacitors, will be used more extensively for signal conditioners. Reliability data will be available to support these selections, even though for limited application; however, no technology funding is anticipated.

MOS multiplexers will double in capacity (16 to 32 lines), but switching times without overlap will remain in the 1.5 to 2.0 microsecond range. This requires a large chip (130 x 130 mils) and a 50-pin package. Monolithic J-FET multiplexers recently introduced are undesirable because of the shorting type failure induced by power loss.

Signal formatting (A/D conversion) will be done at the remote locations to reduce noise effects on data transmission. Historically six to eight bits have been used for missile system data links, because noise and errors as well as sensor resolution made finer measurements meaningless. Onboard data processing will permit recovery of this information; hence the projected system will be 10 to 12 bits.

Preliminary estimates affecting data rates are 2000 test points, 1 sample/second (average) and 30 bits (data, address, parity, and control). This requires a serial transfer rate of only 120 kHz for an interrogate/response operating mode. If all interface data transfer is managed, the data rate will increase by a factor of 30 (sample rate for computation functions estimated at 10 samples/second, average, and triple redundant units). This results in a total system data rate of about 3.6 MHz without margin for growth. (The S3A ASW aircraft will use a single bus transfer at 6 MHz.) MOS logic requires a single chip with a maximum conversion of 400-500 kHz per bit, while bipolar needs four chips with a 3-4 MHz per bit capability. The number of buses used in a design will affect component selection (bus data rate $= \frac{\text{system data rate}}{\text{number of buses}}$).

Submultiplexers will provide a digital-to-digital capability to reduce the distribution bus requirements. They will probably use bipolar elements for a high-speed capability.

Two other components used in the formatter (A/D converter) are the comparator amplifier and the ladder network. The amplifier will be a hybrid unit to satisfy the characteristics of high-slew rates, high-input impedance, and low drift (matched components). The ladder network will probably be thick film resistors (14 bits resolution currently produced), although thin films are also available if design analysis indicates a need for greater stability.

The selection of preferred logic elements will be based on various items. Reliability considerations indicate that the largest available arrays are desirable in implementing the design. Table 1 outlines projected growth for the various logic families. Discretionary wiring has circumvented the yield problem by cell selection with unique interconnections. The technique, developed for a high-speed, phased-array radar is an expensive process, and the reliability for a "one-of-a-kind" design has not been firmly established. Approach feasibility has been demonstrated, and questions on its competitive position (wrt cost/reliability) should be ascertained by 1972. The average density for fixed interconnection bipolar devices will grow from 35 (current MSI average) to about 100 gates/chip, with a few devices at the 200-gate level. No significant technology changes are expected; gradual improvements in process controls and personnel will permit fabrication of devices with reduced geometry and surface leakage. The yield problem is less severe for MOS. The devices have a 20:1 size advantage over bipolar transistors and simpler processing (38 versus 135 steps). Low thresholds (1-2 volts) will be the more common MOS device. C-MOS will not have a broad product line and will not be a contender in the 1972 time frame. The most dense arrays will be in MOS technology with discretionary wiring as an alternate if current uncertainties are resolved.

The second area is speed-power considerations. Figure 1 shows the cost for achieving high speeds. Multiprocessors degrade reliability by increasing parts count, as demonstrated by the parameters given in Table 2,(design study of a 24-bit arithmetic unit using various logic families). Equivalent performance can be approximated by using three times the low-power devices (i.e., 5.4 watts and 336 parts) and four times the MOS (7.2 watts and 68 parts). On the basis of these numbers, MOS-LSI is the logical selection for a multiprocessor computer.

The speed/power characteristics for each logic family (Fig. 2) show MOS devices to be limited to about 5 MHz, with a projected growth to 10 MHz by 1972. The latter should be adequate for the computer design. However, MOS is relatively inefficient when used for combinatorial logic other than sequential arrays. Bipolar logic with multilayer metalization (three levels) is much more flexible as well as faster. The projection is that bipolar devices will be used for control logic and interface elements.

The current trend in packaging technology is the use of hybrid circuits. The reliability gained by reducing interconnections is partially offset by the lengthy high-temperature operation necessary to mount multiple chips on the substrate. The major advantages are the application of semiconductor process controls to packaging and the effective increase in device density. Eight to 10 discrete integrated circuits per package are being routinely produced.

Table 1

ESTIMATE OF SEMICONDUCTOR PARAMETERS (1972)

Logic Family	Bipolar				P-MOS				C-MOS		
	Fixed Interconnect		Discretionary Wiring		Static		Dynamic				
Parameter	69	72	69	72	69	72	69	72	69	72	Units
Gates/chip	73	200	500	1000	100	200	426	1000	18	400	each
Operating speed	5-15	5-20	5-15	5-20	2	5	5	10	2	-	MHz
Power (static)	10-50	1.5-50	10-50	1.5-50	.5-1.5	.1-.5	$5X10^{-6}$	$1X10^{-6}$	10^{-5}	$5X10^{-6}$	mw
Power (dynamic, 1 MHz)	10-50	1.5-50	10-50	1.5-50	.5-1.5	.1-.5	10^{-1}	$5X10^{-2}$	5	5	mw

Note: Current estimates for the ultimate CKT/CHIP are 400 (fixed bipolar), 10,000 (discretionary wiring), and 50,000 (MOS). Bipolar devices can be operated in excess of 100 MHz; P-MOS is limited to 10-15 MHz.

Table 2

ARITHMETIC UNIT SUMMARY

Function	TTL	TTL (Low Pwr)	MOS	Units
Add time	0.54	1.4	2	μsec
Multiply time	5.46	14.0	24	μ sec
Power	9.5	1.6	1.8	watts
Parts	112	112	17	each

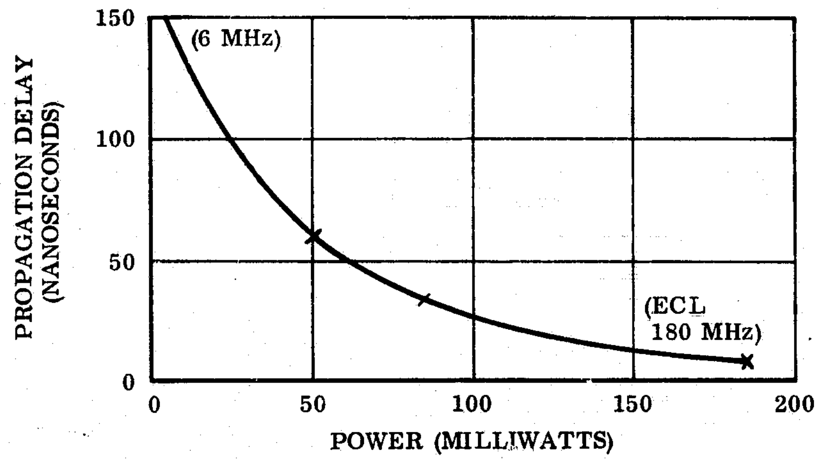


Fig. 1 Speed vs Power for TTL Flip-Flop

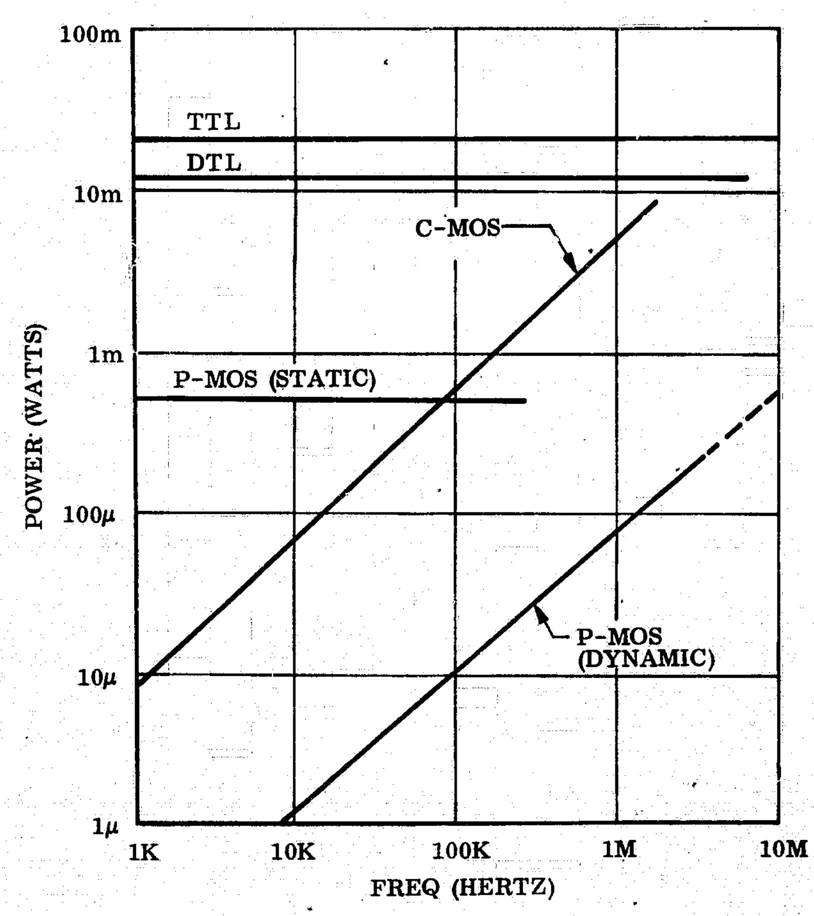


Fig. 2 Speed and Power Characteristics for Logic Families

The memory functions are read-only-memories (code conversions, look-up tables, etc.), scratch-pad memories (short-term data storage), main memory (program and data storage), and mass memory (offline storage). Read-only-memory types are core "rope," bipolar arrays, MOS arrays, and silicon-on-sapphire (experimental). Direct compatibility with the processing logic makes the bipolar and MOS prime candidates for this application. Bipolar densities will increase from 256 to 1024 bits by late 1970, with a 50 nanosecond access time. MOS devices will grow from 2048 bits to 4096 bits, with approximately 200 nanosecond access times.

The projected characteristics for both the scratch pad and main memory functions are given on Table 3. Scratch pad memory must match the logic speed of the system; in this case, the MOS processing logic prevails. Desirable characteristics of the main memory are NDRO, nonvolatile, low-drive currents ($<$200 ma) and short-cycle times. Plated wire and planar thin film types, both requiring close process control, satisfy these criteria.

Many new types of offline data storage have been presented or are in development stages. Beam-scanning approaches, optical and electron, are in the laboratory and cannot be considered. Thermoplastic recording has no known user, and dielectric recording developed under NASA Goddard is nearing flight test. This is primarily a video system; data retrieval has not been incorporated in the equipment. Magnetic recording techniques are the most advanced and probably the safest to use.

Data distribution buses for transfer rates will be transformer-coupled balanced lines with pulsed serial data. Hybrid driver circuits will replace discrete designs for the longer lines. Monolithic receivers will be essentially the same. Preliminary design analysis on the S3A systems (operating at 6 MHz) has indicated that this approach is adequate. Alternate designs, such as FM carrier or fibre-optics, are potential candidates to improve data fidelity. Although the principles involved are well established, neither of these techniques has been demonstrated.

Table 3

ESTIMATE OF CHARACTERISTICS FOR MEMORY ELEMENTS (1972)

Type	MOS	Bipolar	Core	Plated Wire	Thin Film	Monolithic Ferrites
Read-out	NDRO	NDRO	DRO	NDRO/DRO	DRO/NDRO	DRO
Full-cycle time	0.2	0.05-0.1	0.5	0.2	0.2	0.5
Array size	256 X 1	128 X 1	16 k X 128	4 k X 128	1024 X 64	256 X 100
Volatility	Yes	Yes	No	No	No	No
Read current	-	-	400	200	150-200	400
Write current	-	-	400	200	150-200	100
Bit current	-	-	400	30	$\pm$ 25	35
Sense voltage	-	-	20	2.5	$\pm$0.5-1.5	4-10
Standby-power (μw)	200	1000	0	0	0	0
Packing density (bpsi)	15 k	7.2 k	4.5 k	1.4 k	3.2 k	10 k
Organization	LS	LS	2-1/2D	LS	LS	LS
Batch process	Yes	Yes	No	Semi	Yes	Yes
Cost/bit (cents)	1-4	4-5	1-2	2-4	2-3	-

LMSC is currently developing, under an independent development program, a data terminal concept; several units are scheduled for fabrication. In brief, the data bus is tapped as required by small modules that contain decoding logic (address and data) and the necessary signal conditioning for that function. Flat cables will be used as the bus media, with operation at 100 kbps. The approach reduces direct wiring, ensures a compatible interface for each unit, and virtually eliminates junction boxes. This technique, when demonstrated, is directly applicable to the IES concept.

CONCLUSIONS

An integrated electronics system (1972) has been hypothesized and components selected to satisfy the need. Details and rationale are given in Table 4. In consideration of the development schedule, only anticipated off-the-shelf units have been selected for this "final" design. Mechanization changes to accommodate this available component concept will not significantly affect performance parameters.

Table 4

INTEGRATED ELECTRONICS SYSTEM (1972)

Function	Component	Range	Rationale
Signal Acquisition			
Conditioning	Hybrid active filters	dc 50-kHz	Specialized applications (wrt gain and dynamics) result in continuing current technology
Selection	MOS-LSI	32 lines	Expected growth on 150X150-mil chip and 44-pin package
Formatting	Success approximates A/D converter	10-12 bits	Analog signal with noise requires 6-8 bits. Presumes data recovery by onboard computer processing
	Ladder network		Current thick film technology is 14 bits
	Bipolar logic/switches		Approaching 4-MHz bit rate in data conversion
	Hybrid comparator amplifier	± 5 volts	Custom design for high-speed (100V/μsec), low drift operation
Control logic	Bipolar-LSI	100 gates	High-speed (> 10 MHz) and greater flexibility (made possible with multilayer metalization) necessary for data control
Processing logic	MOS-LSI	200 gates	Adequate speed (2-5MHz) for multiprocessor operations (giving inherent redundancy), low-power consumption, and improved reliability

Function	Component	Range	Rationale
Memory			
Read-only memory (ROM)	MOS-LSI	4096 bits	Replace command/computation functions with look-up tables
Scratch pad memory (RAM)	MOS-LSI	256 bits	200 nanosec cycle times and output directly compatible with process logic
Main memory	Plated wire	-	Combination of NDRO, and nonvolatile operations with average power and cycle times (0.3 μsec) and fair development status
Mass storage	Magnetic tape		Development status inadequate on other techniques as compared to magnetic tape
Data Transmission			
Transformer coupled balanced line	Hybrid drivers	1-10 MHz	Insufficient experience with RF or laser links

Appendix C
REQUIREMENTS DEFINITION EXAMPLE (PROPULSION)

Each of the nine subsystems was reviewed and analyzed for the control, monitoring, and testing requirements. The interface, test points, and power were then estimated. The analysis was made by operational phase so that the loading on the various electronic system configurations could be determined. Descriptions of the subsystems were extracted from the related studies, and routines were estimated. A portion of the propulsion subsystem requirements analysis has been extracted to serve as an example. The engine ignition and operational power has not been determined.

The requirements presented are best estimates at this time and will be continually changed and refined. Sufficient information has been made available, however, to conduct the study with a high degree of confidence for fulfilling the finalized system needs. A contingency allowance of 20 percent across the board was included in the study to permit fairly extensive requirement changes.

The propulsion subsystem provides the impulse for launching, controlling, operating, reentering, and landing the ILRV. Portions of the subsystem are required for each of the operational modes, i.e., launch, injection, orbit transfer, orbital maneuvers, reentry, and landing.

The propulsion subsystem is comprised of three major units: primary propulsion, reaction control, and landing-aid propulsion.

The primary propulsion system consists of three (or five) high-pressure, two-position, bell-nozzle Pratt & Whitney rocket engines and the propellant supply, which includes retro-propellant, valves, disconnects, vents, manifolds, and interconnecting plumbing. Also included are the engine control circuits, the propellant management system, the monitoring and sensing transducers, the checkout and fault-isolation circuits, and the displays. In addition to these are the safety monitors and controls such as the leak detectors.

The reaction control system consists of three clusters of thrusters (14 thrust chambers), propellants, pressurant, valves, disconnects, vents, bursts discs, manifolds, and interconnecting plumbing. Also included are control circuits, propellant management system, monitoring and sensing transducers, checkout and fault isolation circuits, displays, and safety monitors.

The landing-aid propulsion system consists of two turbojet engines, propellant supply, valves, disconnect, manifold, and interconnecting plumbing. Also included are engine control circuits, propellant control circuit, monitoring and sensing transducers, checkout and fault isolation circuits, engine-positioning controls, and displays. A safety control with monitors is also included.

The sequence of events, which follows, illustrates the propellant loading, the primary engine ignition sequence, and the primary propellant utilization and management from prelaunch to injection.

Phase Event	Condition/Requirement	Control/Monitoring
Preflight	All vehicle and GSE components in "go" conditions	"Go" conditions displayed for propellant loading
Propellant Loading		
Initial condition	All valves closed. Tanks cleaned. Area control - safe	Monitor valve position. Onboard detectors energized. Safety interlock system - "on"
Load JP-5 for Turbojets	Meter and control fill from GSE. Disconnect fill lines	Monitor capacitive probes - calibration. Verify valve positions and valve fill sequence. Visual inspection for leaks, spills, etc.
Load RCS Load GHe (2)	Check pressure with GSE and meter with GSE	Calibrate and monitor pressure transducers. Maintain temperature monitoring. Verify valve positions. Check relief valve
Load N_2O_4(2)	Check pressure with GSE and meter with GSE	Calibrate and monitor pressure transducers. Maintain temperature monitoring. Verify valve positions meter during fill check for leaks
Load $N_2H_2(CH_3)$ (2)	Check pressure with GSE and meter fill with GSE	Calibrate and monitor pressure transducers. Maintain temperature monitoring. Verify valve positions meter during fill check for leaks
Primary Propellant		
Load tanks	All tanks cleaned and filled with gas. All valves closed	Valve positions verified. Tank temperature and pressure monitored. Condition of equipment and status of system displayed

Phase Event	Condition/Requirement	Control/Monitoring
He purge	Purge system activated	Onboard He purge system valve, flow, temperature, and pressure monitored and controlled
	Leak detection system activated	Monitor and display safety system. Initiate cryogenic propellant loading sequence
Childown GH_2	All fill lines, storage tanks, venting, and safety system activated	Temperature, pressure, liquid level, flow meters, etc., monitored and displayed. Vents monitored
GOX	Same as GH_2	Same as GH_2
LH_2)) Fill LOX)	Same as childown	Follow loading program. General sequence: fill LH_2 tanks then fill LOX tanks. Maintain vehicle balance. Fill at maximum rate
Hold and top-off	Maintain purge system operation. Maintain line temperatures. Maintain pressure and venting. Maintain liquid levels	Continue status and condition monitoring. Continue display: Temperature Pressure Valve Position Propellant Distribution Flow meters Vapor/liquid Leak detectors Liquid level Electrical, voltage current
	Engine valves open. Engine lines and pumps chilled and filled with LOX and LH_2	
CARGO DELIVERY		
Launch		
Preignition	Isolate tanks. Pressurize tanks with GSE He. Disconnect fill lines.	Close valves. Control pressure. Continue leak detection.

Phase Event	Condition/Requirement	Control/Monitoring
Ignition	Engine ignition sequence. Combustion chamber valves open. Pumps star. Engine to 10% thrust in 2 secs; engines start at 200-250 msec intervals	Initiate start by command signal to engine control. Monitor and display sequence. Engine ignition sequence
Launch to Injection		
Primary propulsion		
Propellant utilization	Control LOX/LH_2 ratio	Continuous monitoring, comparing of LOX and LH_2 flow, level in tanks.
Propellant management	Control spacecraft trim	Balance spacecraft CM by controlling flow from each main tank monitor, control all tanks. Maintain tank pressures

A more detailed description of the control and monitoring requirements was then prepared. The initial conditions were stated; the number of valves, commands, and sensors identified; and an estimate of the rates stated. The primary propellant loading routine, engine starts, run, and shutdown routines, and the reaction control system propellant loading follow.

PRIMARY PROPULSION -
CRYOGENIC PROPELLANT LOADING ROUTINE

Initial Conditions	All valves closed	
	All tanks and lines purges	
	Tanks, engines, and manifolding at ambient temperature	
1. Instrumentation	Activate monitoring system and propellant management	
	Monitor valve positions	40 valves @ 1 sps each valve
	Monitor liquid level	9 sensors @ 1 sps each
	Monitor tank temp press	61 sensors @ 1 sps each
2. Safety System (External to tanks)	Activate Leak detectors	8 sensors @ 10 sps each
	Activate He purge lines	4 cmos @ 1 per sec
3. Liquid Hydrogen Fill	Open fill line, tanks, vent valves	10 CMDS @ 1 per sec
	Monitor valve positions	40 valves @ 1 sps/valve
	Monitor tank pressures	16 sensors @ 10 sps each
	Vent AT - psig (cmo/automatic)	8 CMD discrete
	Monitor temperature	45 sensors @ 5 sps each
	Increase fill rate at ^{o}K	4 CMD discrete
	Monitor level in each tank, CP	9 sensors @ 5 sps each
	Monitor level in each tank, OPS	35 sensors @ 1 sps each
	Close valves as tanks fill	8 CMD discretes
	Close main fill line valve	1 CMD discrete
4. Liquid Hydrogen Hold	Monitor tank pressure	16 sensors @ 1 sps each
	Monitor tank temperature	45 sensors @ 1 sps each
	Vent at ___psig or ___^{o}K	8 CMDS discrete

PRIMARY PROPULSION
CRYOGENIC PROPELLANT LOADING ROUTINE (Cont'd)

5.	Liquid Oxygen Fill	Open fill line, tanks, vent valves	10 CMDS @ 1 per sec
		Monitor valve positions	40 valves @ 1 sps/valve
		Vent at ___psig (CMD/automatic)	8 CMDS discrete
		Monitor temperature	45 sensors @ 5 sps each
		Increase fill rate at ___ °K	4 CMDS discrete
		Monitor level in each tank, CP	9 sensors @ 5 sps each
		Monitor level in each tank, OPS	35 sensors @ 1 sps each
		Close valves as tanks fill	8 CMDS discrete
		Close main fill line valve	1 CMD discrete
6.	Liquid Oxygen Hold	Monitor tank pressure	16 sensors @ 1 sps each
		Monitor tank temperature	45 sensors @ 1 sps each
		Vent at ___psig or ___ °K	8 CMDS discrete
7.	LH_2 Top Off	Open fill line valve	1 CMD discrete
		Open each tank fill valve	8 CMDS discrete
8.	LOX Top Off	Open fill line valve	1 CMD discrete
		Open fill line valve	8 CMDS discrete

LMSC-A959837
Vol. III

PRIMARY PROPULSION
PRIMARY ENGINE STARTUP, RUN, AND SHUTDOWN ROUTINES

Initial Conditions	Propellant loaded, topped off, and in hold condition Engine propellant manifold, engine cool down valves Engine bypass valves, and engines at ambient temp Engine nozzles set to initial burn position Propellant instrumentation, and valve monitoring Operating	
1. Instrumentation	Set fuel/oxidizer ratio	5 CMDS discrete
	Set throttle positions	5 CMDS discrete
	Monitor engine pressures	25 sensors @ 5 sps each
	Monitor engine temperatures	20 sensors @ 5 sps each
	Monitor engine position controls	25 sensors @ 5 sps each
	Monitor LH_2 & LOX pump rpm	10 sensors @ 5 sps each
	Monitor LH_2 & LOX flows	10 sensors @ 5 sps each
	Monitor ignition voltage	5 sensors @ 5 sps each
2. Cooldown Routine	Sequence manifold valves from temperature sensors	30 valves @ 1 per sec max
3. Launch disconnect	Propellant line disconnects	3 CMDS discrete
4. Start Engines	Sequence start (one eng. every 250 ms): ignition signal, LOX flow, LH_2 flow, flow rates, etc. Balance pump speeds	5 CMDS 1 per sec
5. Propellant Supply (Boost Phase)	LOX Supply valving	10 CMDS discrete, 1/sec max
	LH_2 supply valving	10 CMDS discrete, 1/sec max

PRIMARY PROPULSION
PRIMARY ENGINE STARTUP, RUN, AND SHUTDOWN ROUTINES (Cont'd)

6. Run Engines	Nozzle positions	5 CMDS discrete, 10sec/CMD
	Throttle positions	5 CMDS Cont. @ 0.1 sec/CMD
	Fuel/oxidizer ratio, course	5 CMDS 1 per second
	Fuel/oxidizer ratio, fine	5 analog 200 ms/signal
	Engine control - operating limits	90 sensors 10 sps
7. Propellant Supply (Boost)	LH_2 level, temp, press	6 sensors 5 sps each
	LOX level, temp, pres	26 sensors 5 sps each
(Injection)	LH_2 level, temp, press	54 sensors 1 sps each
	LOX level, temp, press	54 sensors 1 sps each
	Valves - tank balance	30 CMDS 1 CMD every 5 sec
8. Engine Shutdown	Throttle back	5 CMDS discrete
	LOX Supply off	10 valves 2 CMDS/sec
	Ignition off	5 CMDS 1 per sec
	Cooldown - LH_2 flow	10 sensors 5 sps
	LH_2 supply off	10 valves 2 CMDS/sec

LOCKHEED MISSILES & SPACE COMPANY

LMSC-A959837
Vol. III

REACTION CONTROL SYSTEM PROPELLANT LOADING ROUTINE

Initial Conditions	All tank, line, and thruster valves closed	
	Tanks, lines, etc. purged and ready for loading	
	Tanks, lines, thrusters at ambient temperature	
1. Instrumentation	Activate monitoring system	1 CMD
	Monitor valve positions, voltage	27 valves @ 1 sps
	Monitor liquid level, CP	4 sensors @ 1 sps each
	Monitor tank and line temp press	38 sensors @ 1 sps each
2. Load GHe (5-10 min)	Monitor tank temp and press	10 sensors @ 10 sps each
	Open gas fill line valve	1 CMD discrete
	Open tank No. 1 valve	1 CMD discrete
	Fill to 4500 psia, close valve	1 CMD discrete
	Open tank No. 2 valve	1 CMD discrete
	Fill to 4500 psia, close valve	1 CMD discrete
	Close gas fill line valve	1 CMD discrete
	Monitor tank temp press	10 sensors @ 0.1 sps
3. Load N_2O_4	Monitor tanks, temp pressure, level	12 sensors @ 10 sps each
	Open N_2O_4 fill line valves, 2	2 CMDS discrete
	Open N_2O_4 tank No. 1 valves, 1	1 CMD discrete
	Fill tank No. 1, close valve 1	1 CMD discrete
	Open N_2O_2 tank, No. 2 valve, 1	1 CMD discrete
	Fill tank No. 2, close valve, 1	1 CMDS discrete
	Close fill line valves, stop monitoring	3 CMDS discrete

LMSC-A959837
Vol. III

Along with the sequence of events, functional descriptions of the subsystem rate and routine estimates and a tabulation of the required equipment and instrumentation were made on an operational phasing basis. An example of this tabulation follows.

PRECEDING PAGE BLANK NOT FILMED.

INTEGRATED ELECTRONICS,

		PHASE				
		I	II	III	IV	
		Pre-flight	Launch-Injection	Orbit Insertion-Rendezvous	Docking-Transfer	Or S
2.1	Main engines	C/O	All	1-5	None	Nor
2.1	Operational Control					
	Start/stop-run	Routines	Prog 1	Prog 2		
2.1	Instrumentation					
	Press, LH_2 pump	Indicator	5 SPS*	1-5 SPS		
	LOX pump	Indicator	5 SPS	1-5 SPS		
	Main Burn Lox Inject.		5 SPS	1-5 SPS		
	Preburner chamb		5 SPS	1-5 SPS		
	Preburner LH_2 Inject		5 SPS	1-5 SPS		
	Temp, Preburn Chamber	Indicator	5 SPS	1-5 SPS		
	Main burn Chamb Skin	Indicator	5 SPS	1-5 SPS		
	Heat exchange discharge		5 SPS	1-5 SPS		
	Nozzle coolant dis.		5 SPS	1-5 SPS		
	Position, power lever		5 SPS	1 SPS		
	P.U. input		5 SPS	1 SPS		
	Preburn LH_2 valve		5 SPS	1 SPS		
	Main Burn LOX valve		5 SPS	1 SPS		
	Preburn LOX valve		5 SPS	1 SPS		
	RPM, LH_2 Pump		5 SPS	5 SPS		
	LOX pump		5 SPS	5 SPS		
	Flow, LOX		5 SPS	5 SPS		
	LH_2		5 SPS	5 SPS		
	Voltage, supply	Indicator				
	Helium pressure-GSE	Indicator				
2.3	Disconnect, DCV 1	Load	Discrete			
	DCV 2		5 SPS	Event		
	DCV 3	Vent	Discrete			
2.3	Cooldown valve, CDV 1	Discrete	1 SPS	1 SPS	1 SPS	.01
	CDV 2	Discrete	1 SPS	1 SPS	5 SPS	.01
	CDV 3	Discrete	1 SPS	1 SPS	5 SPS	.01
2.3	Bypass valve BV 1	Discrete	1 SPS	1 SPS	.01 SPS	.01
	BV 2	Discrete	1 SPS	1 SPS	.01 SPS	.01
	BV 3	Discrete	1 SPS	1 SPS	.01 SPS	.01
	BV 4	Discrete	1 SPS	1 SPS	.01 SPS	.01
	BV 5	Discrete	1 SPS	1 SPS	.01 SPS	.01
	BV 6	Discrete	1 SPS	1 SPS	.01 SPS	.01
	BV 7	Discrete	1 SPS	1 SPS	.01 SPS	.01

FOLDOUT FRAME

INTEGRATED ELECTRONICS, PROPULSION SUBSYSTEM

PHASE							
II	III	IV	V	VI	VII	VIII	
Launch-Injection	Orbit Insertion-Rendezvous	Docking-Transfer	Orbital Stay	Deorbit-Entry	Landing	Post-flight	
All	1-5	None	None	1-5		As req'd	
						Routines	Go-no g
Prog 1							Start e
	Prog 2						Adjust
				Prog 3			
							Reqd pe
5 SPS*	1-5 SPS			1-5			Conditi
5 SPS	1-5 SPS			1-5			Conditi
5 SPS	1-5 SPS			1-5			Conditi
5 SPS	1-5 SPS			1-5			Conditi
5 SPS	1-5 SPS			1-5			Conditi
5 SPS	1-5 SPS			1-5			Conditi
5 SPS	1-5 SPS			1-5			Conditi
5 SPS	1-5 SPS			1-5			Conditi
5 SPS	1-5 SPS			1-5			Conditi
5 SPS	1 SPS			1			Primary
5 SPS	1 SPS			1			Primary
5 SPS	1 SPS			1			Primary
5 SPS	1 SPS			1			Primary
5 SPS	1 SPS			1			Primary
5 SPS	5 SPS			5			Primary
5 SPS	5 SPS			5			Primary
5 SPS	5 SPS			5			Supplied
5 SPS	5 SPS			5			Supplied
							Engine i
							Initial
Discrete						Cooling	Connect/
5 SPS	Event						Connect/
Discrete						Vent	Connect/
1 SPS	1 SPS	1 SPS	.01 SPS	1 SPS	.01 SPS	Discrete	Open/clo
1 SPS	1 SPS	5 SPS	.01 SPS	1 SPS	.01 SPS	Discrete	Open/clo
1 SPS	1 SPS	5 SPS	.01 SPS	1 SPS	.01 SPS	Discrete	Open/clo
1 SPS	1 SPS	.01 SPS	.01 SPS	1 SPS	.01 SPS	Discrete	Open/clo
1 SPS	1 SPS	.01 SPS	.01 SPS	1 SPS	.01 SPS	Discrete	Open/clo
1 SPS	1 SPS	.01 SPS	.01 SPS	1 SPS	.01 SPS	Discrete	Open/clo
1 SPS	1 SPS	.01 SPS	.01 SPS	1 SPS	.01 SPS	Discrete	Open/clo
1 SPS	1 SPS	.01 SPS	.01 SPS	1 SPS	.01 SPS	Discrete	Open/clo
1 SPS	1 SPS	.01 SPS	.01 SPS	1 SPS	.01 SPS	Discrete	Open/clo
1 SPS	1 SPS	.01 SPS	.01 SPS	1 SPS	.01 SPS	Discrete	Open/clo

PROPULSION SUBSYSTEM

V rbital Stay	VI Deorbit-Entry	VII Landing	VIII Post-flight	Remarks
one	1-5		As req'd	
			Routines	Go-no go
				Start engines - 1 second increments - on cmd
				Adjust f/o ratio and thrust levels - on cmb
	Prog 3			
				Reqd per engine during engine burn
	1-5			Condition monitor 0-5 vdc output
	1-5			Condition monitor 0-5 vdc output
	1-5			Condition monitor 0-5 vdc output
	1-5			Condition monitor 0-5 vdc output
	1-5			Condition monitor 0-5 vdc output
	1-5			Condition monitor temp probe
	1-5			Condition monitor thermocouple
	1-5			Condition monitor temp probe
	1-5			Condition monitor temp probe
	1			Primary measurement lin or L pot 0-5V
	1			Primary measurement lin or L pot 0-5V
	1			Primary measurement lin or L pot 0-5V
	1			Primary measurement lin or L pot 0-5V
	1			Primary measurement lin or L pot 0-5V
	5			Primary measurement freq to DC
	5			Primary measurement freq to DC
	5			Supplied with engine 0-5 vdc
	5			Supplied with engine 0-5 vdc
				Engine ignition - volt monitor
				Initial pressurization only - ground supply
			Cooling	Connect/disconnect, LH_2 Leak Det, DT
				Connect/disconnect, LH_2 Leak Det, SC
			Vent	Connect/disconnect, LH_2 Leak Det, SC
1 SPS	1 SPS	.01 SPS	Discrete	Open/close
1 SPS	1 SPS	.01 SPS	Discrete	Open/close
1 SPS	1 SPS	.01 SPS	Discrete	Open/close
1 SPS	1 SPS	.01 SPS	Discrete	Open/close
1 SPS	1 SPS	.01 SPS	Discrete	Open/close
1 SPS	1 SPS	.01 SPS	Discrete	Open/close
1 SPS	1 SPS	.01 SPS	Discrete	Open/close
1 SPS	1 SPS	.01 SPS	Discrete	Open/close
1 SPS	1 SPS	.01 SPS	Discrete	Open/close
1 SPS	1 SPS	.01 SPS	Discrete	Open/close

FOLDOUT FRAME 2

FOLDOUT FRAME 3

INTEGRATED ELECTRONICS, PRO

			PHASE				
			I	II	III Orbit	IV	
			Preflight	Launch-Injection	Insertion-Rendezvous	Docking-Transfer	Or S
2.3	Engine Main Valve	EMV 1	Discrete	1 SPS	1 SPS	.01 SPS	
		EMV 2	Discrete	1 SPS	1 SPS	.01 SPS	
		EMV 3	Discrete	1 SPS	1 SPS	.01 SPS	
2.3	Propellant main Valve	PMV 1	Discrete	1 SPS	1 SPS	.01 SPS	
		PMV 2	Discrete	1 SPS	1 SPS	.01 SPS	
		PMV 3	Discrete	1 SPS	1 SPS	.01 SPS	
		PMV 4	Discrete	1 SPS	1 SPS	.01 SPS	
2.3	Vent valve	VV 1	Discrete	1 SPS	0.1 SPS	.01 SPS	
		VV 2	Discrete	1 SPS	0.1 SPS	.01 SPS	
		VV 3	Discrete	1 SPS	0.1 SPS	.01 SPS	
		VV 4	Discrete	1 SPS	0.1 SPS	.01 SPS	
		VV 5	Discrete	1 SPS	0.1 SPS	.01 SPS	
2.3	Check Valve,	CV 1	Discrete				
		CV 2	Discrete				
		CV 3	Discrete				
2.3	Spacecraft LH_2 Tank 1						
	Liquid level,	CP 1	5 SPS load	1 SPS	5 SPS burn	.01 SPS	On
	Optical Pt.,	OPS 11	5 SPS load	1 SPS	5 SPS burn	.01 SPS	On
		OPS 12	5 SPS load	1 SPS	5 SPS burn	.01 SPS	On
		OPS 13	5 SPS load	1 SPS	5 SPS burn	.01 SPS	On
		OPS 14	5 SPS load	1 SPS	5 SPS burn	.01 SPS	On
	Temp Sensor,	T 11	5 SPS load	1 SPS	5 SPS burn	.01 SPS	On
		T 12	5 SPS load	1 SPS	5 SPS burn	.01 SPS	On
		T 13	5 SPS load	1 SPS	5 SPS burn	.01 SPS	On
		T 14	5 SPS load	1 SPS	5 SPS burn	.01 SPS	On
		T 15	5 SPS load	1 SPS	5 SPS burn	.01 SPS	On
	Pressure	P 5	5 SPS load	1 SPS	5 SPS	0.1 SPS	0.
		P 6	5 SPS load	1 SPS	5 SPS	0.1 SPS	0.
2.3	SC LH_2 Tank 2						
	Liquid Level	CP 4	5 SPS load	1 SPS	5 SPS burn	.01 SPS	.0
	Optical Pt,	OPS 15	5 SPS load	1 SPS	5 SPS burn	.01 SPS	.0
		OPS 16	5 SPS load	1 SPS	5 SPS burn	.01 SPS	.0
		OPS 17	5 SPS load	1 SPS	5 SPS burn	.01 SPS	.0
		OPS 18	5 SPS load	1 SPS	5 SPS burn	.01 SPS	.0
	Temperature,	T 16	5 SPS load	1 SPS	5 SPS burn	.01 SPS	.0
		T 17	5 SPS load	1 SPS	5 SPS burn	.01 SPS	.0
		T 18	5 SPS load	1 SPS	5 SPS burn	.01 SPS	.0
		T 19	5 SPS load	1 SPS	5 SPS burn	.01 SPS	.0
		T 20	5 SPS load	1 SPS	5 SPS burn	.01 SPS	.0
	Pressure,	P 7	5 SPS load	5 SPS	5 SPS	0.1 SPS	0.
		P 8	5 SPS load	5 SPS	4 SPS	0.1 SPS	0.

LMED.

INTEGRATED ELECTRONICS, PROPULSION SUBSYSTEM (CONT'd)

	PHASE							
	I	II	III	IV	V	VI	VII	V
	Preflight	Launch-Injection	Orbit Insertion-Rendezvous	Docking-Transfer	Orbital Stay	Deorbit-Entry	Landing	Post
	Discrete	1 SPS	1 SPS	.01 SPS				Disc
	Discrete	1 SPS	1 SPS	.01 SPS				Disc
	Discrete	1 SPS	1 SPS	.01 SPS				Disc
MV 1	Discrete	1 SPS	1 SPS	.01 SPS		1 SPS		Disc
MV 2	Discrete	1 SPS	1 SPS	.01 SPS		1 SPS		Disc
MV 3	Discrete	1 SPS	1 SPS	.01 SPS		1 SPS		Disc
MV 4	Discrete	1 SPS	1 SPS	.01 SPS		1 SPS		Disc
	Discrete	1 SPS	0.1 SPS	.01 SPS		1 SPS		Disc
	Discrete	1 SPS	0.1 SPS	.01 SPS		1 SPS		Disc
	Discrete	1 SPS	0.1 SPS	.01 SPS		1 SPS		Disc
	Discrete	1 SPS	0.1 SPS	.01 SPS		1 SPS		Disc
	Discrete	1 SPS	0.1 SPS	.01 SPS		1 SPS		Disc
	Discrete							
	Discrete							
	Discrete							
	5 SPS load	1 SPS	5 SPS burn	.01 SPS	On cmd	5 SPS burn	0.1 SPS	On c
	5 SPS load	1 SPS	5 SPS burn	.01 SPS	On cmd	5 SPS burn	0.1 SPS	On c
	5 SPS load	1 SPS	5 SPS burn	.01 SPS	On cmd	5 SPS burn	0.1 SPS	On c
	5 SPS load	1 SPS	5 SPS burn	.01 SPS	On cmd	5 SPS burn	0.1 SPS	On c
	5 SPS load	1 SPS	5 SPS burn	.01 SPS	On cmd	5 SPS burn	0.1 SPS	On c
	5 SPS load	1 SPS	5 SPS burn	.01 SPS	On cmd	5 SPS burn		On c
	5 SPS load	1 SPS	5 SPS burn	.01 SPS	On cmd	5 SPS burn		On c
	5 SPS load	1 SPS	5 SPS burn	.01 SPS	On cmd	5 SPS burn		On c
	5 SPS load	1 SPS	5 SPS burn	.01 SPS	On cmd	5 SPS burn		On c
	5 SPS load	1 SPS	5 SPS burn	.01 SPS	On cmd	5 SPS burn		On c
	5 SPS load	1 SPS	5 SPS	0.1 SPS	0.01 SPS	5 SPS burn	5 SPS	On c
	5 SPS load	1 SPS	5 SPS	0.1 SPS	0.01 SPS	5 SPS burn	5 SPS	On c
	5 SPS load	1 SPS	5 SPS burn	.01 SPS	.01 SPS	5 SPS burn		On c
	5 SPS load	1 SPS	5 SPS burn	.01 SPS	.01 SPS	5 SPS burn		On c
	5 SPS load	1 SPS	5 SPS burn	.01 SPS	.01 SPS	5 SPS burn		On c
	5 SPS load	1 SPS	5 SPS burn	.01 SPS	.01 SPS	5 SPS burn		On c
	5 SPS load	1 SPS	5 SPS burn	.01 SPS	.01 SPS	5 SPS burn		On c
	5 SPS load	1 SPS	5 SPS burn	.01 SPS	.01 SPS	5 SPS burn		On c
	5 SPS load	1 SPS	5 SPS burn	.01 SPS	.01 SPS	5 SPS burn		On c
	5 SPS load	1 SPS	5 SPS burn	.01 SPS	.01 SPS	5 SPS burn		On c
	5 SPS load	1 SPS	5 SPS burn	.01 SPS	.01 SPS	5 SPS burn		On c
	5 SPS load	1 SPS	5 SPS burn	.01 SPS	.01 SPS	5 SPS burn		On c
	5 SPS load	5 SPS	5 SPS	0.1 SPS	0.1 SPS	5 SPS burn	5 SPS	On c
	5 SPS load	5 SPS	4 SPS	0.1 SPS	0.1 SPS	5 SPS burn	5 SPS	On c

ULSION SUBSYSTEM (CONT'd)

V	VI	VII	VIII	
ital ay	Deorbit-Entry	Landing	Post-flight	Remarks
			Discrete	Open/close
			Discrete	Open/close
			Discrete	Open/close
	1 SPS		Discrete	Open/close
	1 SPS		Discrete	Open/close
	1 SPS		Discrete	Open/close
	1 SPS		Discrete	Open/close
	1 SPS		Discrete	Open/close
	1 SPS		Discrete	Open/close
	1 SPS		Discrete	Open/close
	1 SPS		Discrete	Open/close
	1 SPS		Discrete	Open/close
				Test by flow/volume measurement
				Test by flow/volume measurement
				Test by flow/volume measurement
cmd	5 SPS burn	0.1 SPS	On cmd	Primary propellant measurement
cmd	5 SPS burn	0.1 SPS	On cmd	Secondary propellant measurement 0.3 vdc
cmd	5 SPS burn	0.1 SPS	On cmd	Secondary propellant measurement 0.6 vdc
cmd	5 SPS burn	0.1 SPS	On cmd	Secondary propellant measurement 1.2 vdc
cmd	5 SPS burn	0.1 SPS	On cmd	Secondary propellant measurement 2.9 vdc
cmd	5 SPS burn		On cmd	Third-order-level determination
cmd	5 SPS burn		On cmd	Third-order-level determination
cmd	5 SPS burn		On cmd	Third-order-level determination
cmd	5 SPS burn		On cmd	Third-order-level determination
cmd	5 SPS burn		On cmd	Third-order-level determination
1 SPS	5 SPS burn	5 SPS	On cmd	Primary tank safety eventing
1 SPS	5 SPS burn	5 SPS	On cmd	Secondary tank safety eventing
SPS	5 SPS burn		On cmd	Primary propellant measurement
SPS	5 SPS burn		On cmd	Secondary propellant level measurement
SPS	5 SPS burn		On cmd	Secondary propellant level measurement
SPS	5 SPS burn		On cmd	Secondary propellant level measurement
SPS	5 SPS burn		On cmd	Secondary propellant level measurement
SPS	5 SPS burn		On cmd	Third-order-level and safety determination
SPS	5 SPS burn		On cmd	Third-order-level and safety determination
SPS	5 SPS burn		On cmd	Third-order-level and safety determination
SPS	5 SPS burn		On cmd	Third-order-level and safety determination
SPS	5 SPS burn		On cmd	Third-order-level and safety determination
SPS	5 SPS burn	5 SPS	On cmd	Primary tank safety and venting
SPS	5 SPS burn	5 SPS	On cmd	Secondary tank safety and venting

ME 2

FOLDOUT FRAME 3

Block diagrams, schematic drawings, see (Fig. C-1), and power profiles were also prepared. The following is a summary of the electrical power required for the propulsion subsystem

PRIMARY PROPULSION - ELECTRICAL POWER REQUIREMENTS

Primary Propellant Supply

Valving. Approximately 90 valves are required for loading manifolding, separation, and controlling the propellants. Most of the valves are electrically controlled pilot type requiring from 2 to 5 amperes at 28 volts DC. The valves are two-position type, i.e., either closed or fully open. The operation of the valving is slow and, thus, can be sequenced so that at one time no more than two valves require power. The operation of a single valve is not required more frequently than once every 10 seconds. Fast action and regulation are achieved by pressure from the propellent either in the liquid or gas state.

Propellant Management, Liquid Level, and Flow Sensing. The control and monitoring of the primary propellant is on a low-demand basis because of the large value of propellant. The 90 optical point, 90 temperature, 40 pressure, 20 liquid level capacitance, 16 leak, and 14 liquid/vapor sensors plus the 90 valve position and 4 flow sensors comprise the instrumentation. The operational requirements are such that it is not necessary to operate all of the sensors at the same time. These may be turned on and turned off in groups according to the mission phase. During the prelaunch and operational phases, the groups can be sequenced for checkout.

Total power required without sequency is approximately 360 watts.

The propellant management function requires about 25 watts.

Main Engine. The individual engine controllers require about 20 watts each, or a total of 100 watts. The engine ignition and operational power has not been determined.

REACTION CONTROL SYSTEM

Propellant Supply. Approximately 23 valves are required. Operation of the valves is similar to those of the main propulsion system in that only one is required to operate at a time. The total electrical load is 2 to 5 amperes at 28 volts DC.

Thrusters. Fifty-six valves are required, and operating in groups of 4 per thruster. The operation of as many as six valves could be required at the same time. For each cluster, then, the maximum power would be about 12 amperes at 28 volts DC. The thrusters would not be required during main engine burn except for a single main engine burn condition for a maximum of 5 minutes.

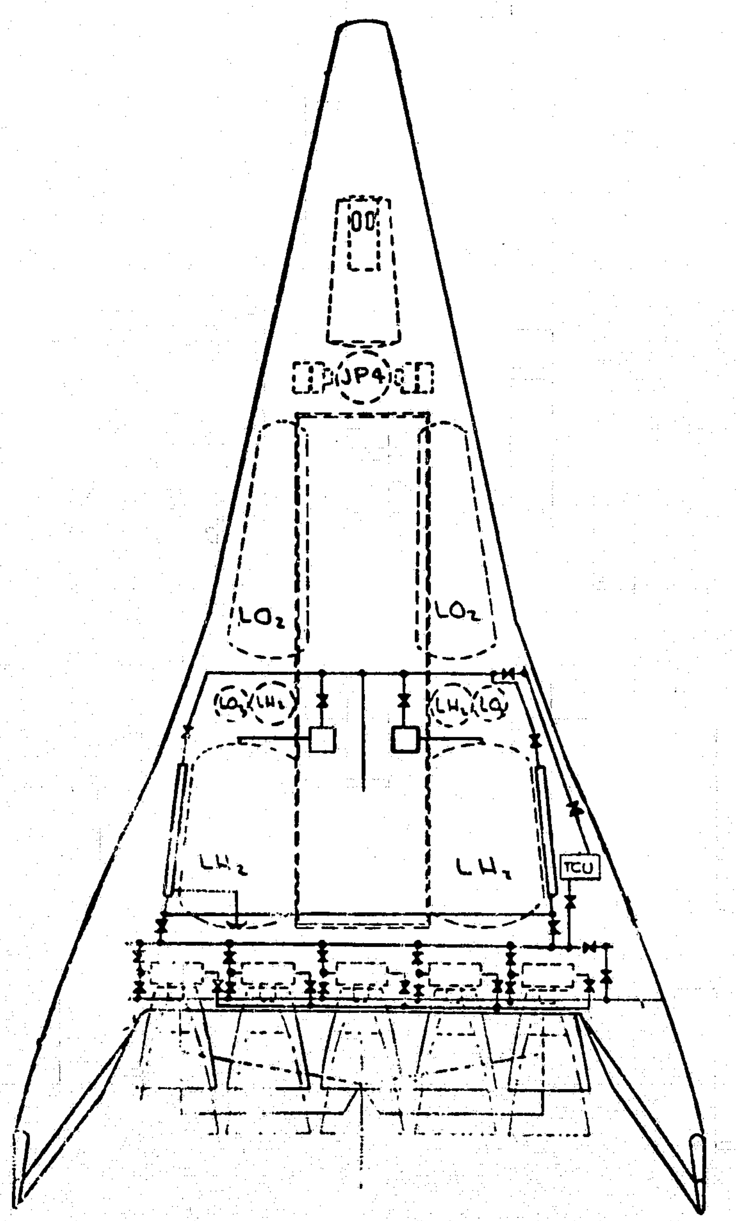
Propellant Management. The 48 transducers and associated electronics require about 20 watts.

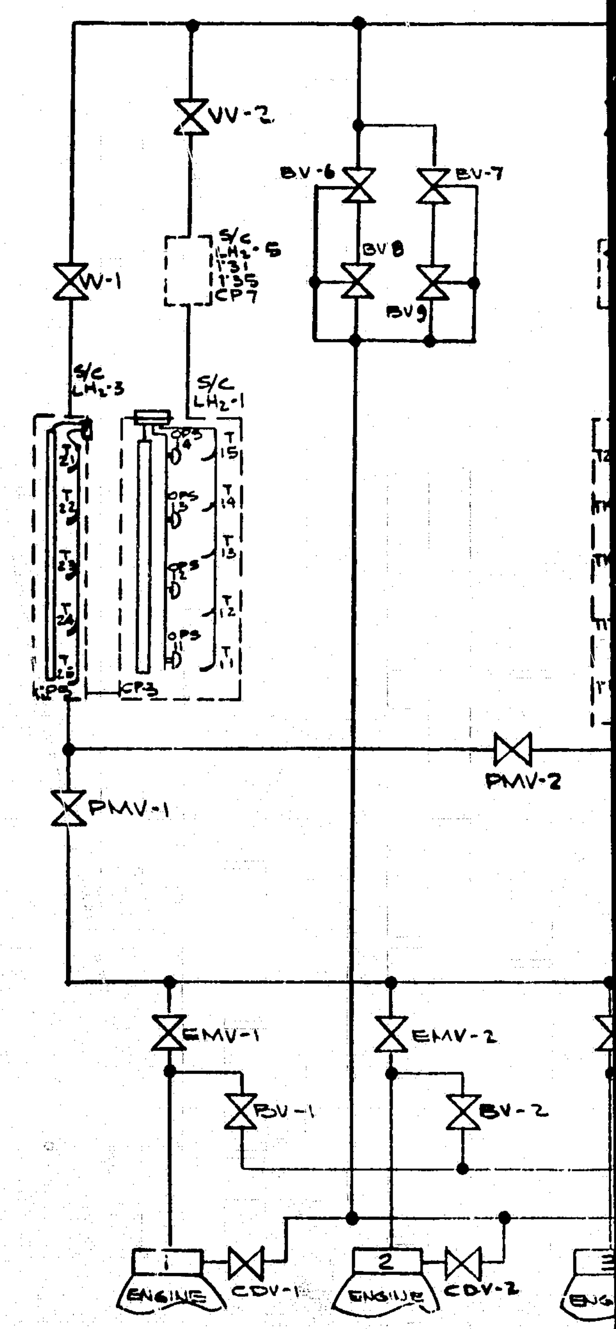
LANDING AID ENGINE

Propellant Management. About 8 valves are required for propellant management, operating one at a time. About 3 amperes at 28 volts DC is required. Monitoring requires about 20 watts additional.

Engine Controllers. Controllers using 20 watts each are required.

The requirements presented are best estimates at this time and will be continually changed and refined. Sufficient information has been made available, however, to conduct the study with a high degree of confidence for fulfilling the finalized system needs. A contingency allowance of 20 percent across the board was included in the study to permit fairly extensive requirement changes.

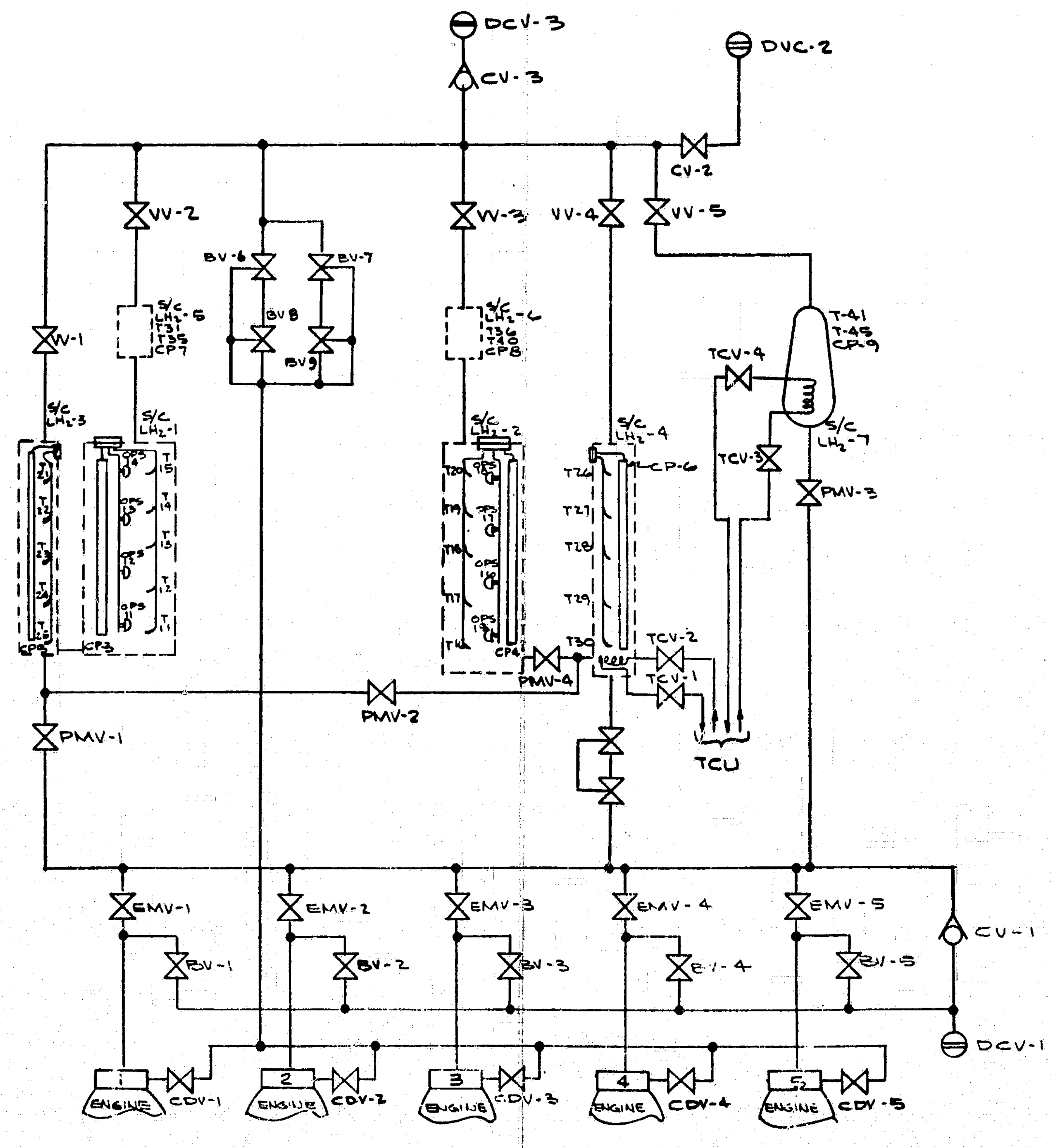




TEMPERATURE SENSOR
PRESSURE SENSOR
OPTICAL POINT SENSOR
DISCONNECT
CHECK VALVE
VALVE
(CP) CAPACITIVE PROBE
(EMV-) ENGINE MANIFOLD VALVE (NO.)
(BV-) BYPASS VALVE -(NO.)
(CDV-)

(DCV-) DISCONNE
(CV-) CHECK VAL
(TCV-)
(PMV-)
(T-) TEMPERATU
(OPS-) OPTICAL P

FOLDOUT FRAME



TEMPERATURE SENSOR
PRESSURE SENSOR
OPTICAL POINT SENSOR
DISCONNECT
CHECK VALVE
VALVE
) CAPACITIVE PROBE
)
) ENGINE MANIFOLD VALVE (No.)
) BYPASS VALVE - (NO.)
)

(DCV-) DISCONNECT VALVE - (No.)
(CV-) CHECK VALVE - (NO.)
(TCV-)
(PMV-)
(T-) TEMPERATURE SENSOR - (No.)
(OPS-) OPTICAL POINT SENSOR - (NO.)

Appendix D

APPLICATION OF BITE TO ONBOARD CHECKOUT

Onboard checkout (OBC) provides in-flight monitoring for safety and maintenance and permits minimal requirements to be placed on ground support equipment. The test objective is to detect and isolate failures in line replaceable units (LRU). The test function may be distributed, centralized, or a combination. Following are some factors to be considered in partitioning the OBC system.

TESTING PHILOSOPHY

The classical test technique for electronic systems is to provide a controlled excitation and monitor the signal flow through the circuitry. For testing an online unit, this approach interferes with system operation. An alternate technique is given in Fig. 1. The sequence of operations would be to test the function, end-to-end, and if satisfactory, proceed to the next function. The module tests would be exercised in the event of a failure. The diagram implies that the tester is, essentially, a parallel channel performing the same function. This technique, used in scientific computers, avoids errors and requires diagnostic routines only when a failure is detected. Obviously, the approach cannot be directly applied to an online, realtime system, because the tester failure rate would be somewhat higher (as a result of a switching logic increment) than in the primary channel. It is of interest to note that triple redundancy with majority voting logic is an extension of this scheme and could be considered a form of OBC.

A different criteria must be adopted for OBC operating on a noninterference basis with minimal test equipment complexity; specifically, the test goal must be gross failure detection and not critical evaluation of performance parameters. The primary reasons for eliminating performance evaluation are considerations of the instrumentation accuracy required and the difficulty of obtaining and implementing adequate mathematical models. Experience gained on

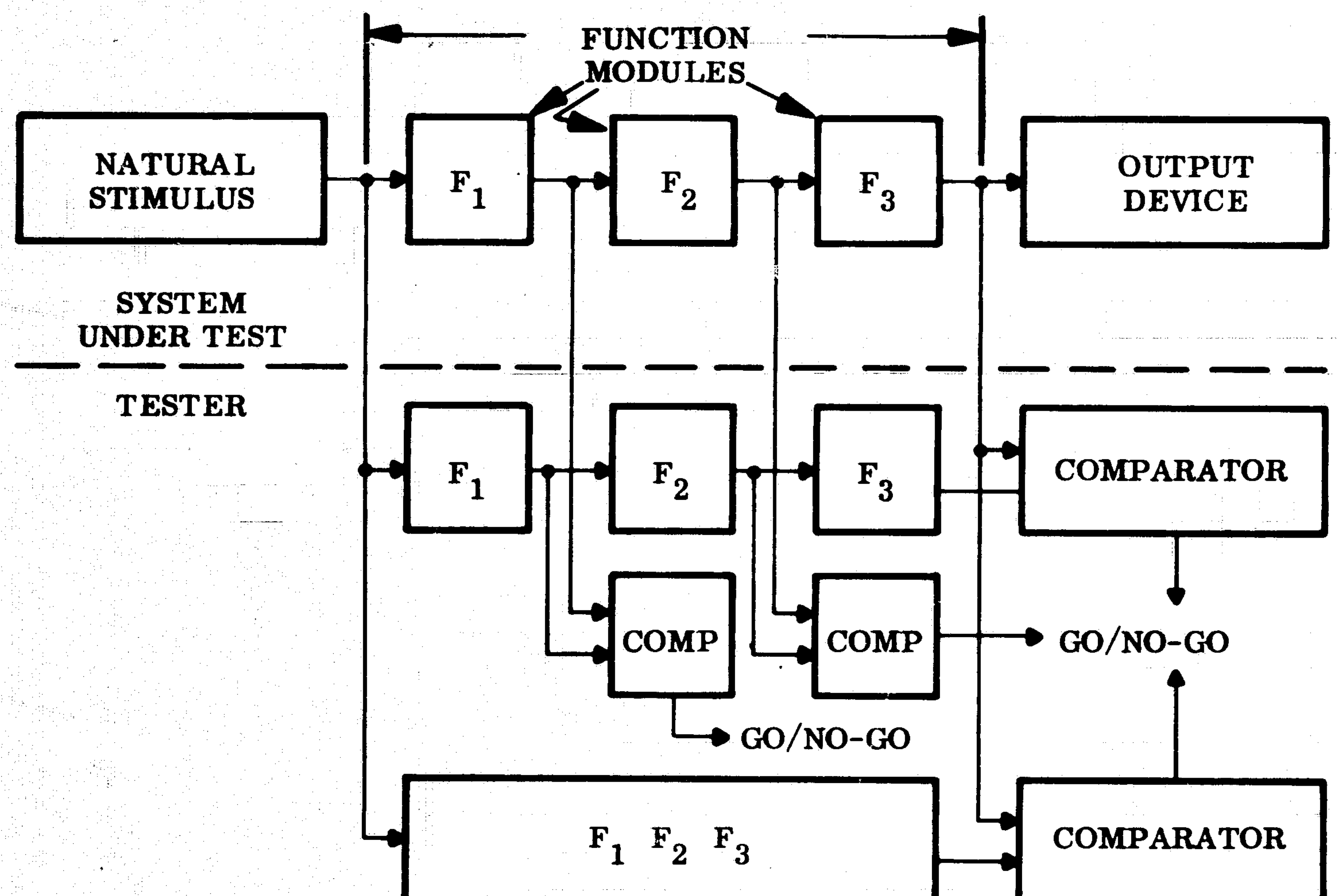


FIG. 1 SIMPLIFIED TEST DIAGRAM

MADAR (Malfunction, Analysis, Detection, and Recording, a centralized system used on the C5A aircraft) has shown OBC to be effective when a qualitative rather than quantitative testing is conducted. This has resulted in a major simplification of the test equipment, which is constrained to relatively static monitoring instead of the more precise, dynamic measurements made by a dedicated tester on an isolated unit. The basic measurement used in MADAR is voltage amplitude. Signal conditioners are used for converting data format; typical examples of conditioning units are peak detectors and frequency-to-voltage converters as well as common line-drive amplifiers. The acceptance limits are pre-set at levels representing a reasonable operating range. Logical decisions are also a part of the basic program.

SPECIAL CASE - DIGITAL COMPUTER

The current trend in system implementation is to use a general-purpose machine for all onboard computations. Secondary effects such as unit temperature and power (supply voltage and current) are easily monitored but programming OBC for a computer logic check is impractical; hence BITE is used. Self-test is achieved through parity checks and by programming a sample problem for comparison with a stored solution. The typical program exercises the I/O for a complete end-to-end computer check as well as individual tests on internal functional blocks. The test outputs are a go/no-go signal and a coded word identifying the faulty component. Transient errors, resulting in loss of data,are detected by bus parity checks. The computer program generally includes a routine for handling parity errors; however, storage of a cumulative count of errors is not included. This cumulative count would be of interest in establishing data error rates, a good indication of noise margins.

DISTRIBUTED VS CENTRALIZED

BITE (Distributed Built-In Test Equipment), confined to a LRU, cannot monitor overall performance. Some form of centralized processing (MADAR type) is desirable to evaluate system capabilities. Hence, the central processor has been retained throughout this study; its implied function is an all-station command to test the integrity of the system cabling, coupled with elementary logic routines to relieve the crew of routine mental tasks and to assess the impact of detected failure patterns.

Results from a preliminary system sizing indicate that 2000 test points will be needed for the OBC function in the reference system. A typical signal distribution for a centralized system is shown in Fig. 2. The functions contained within each unit are defined in Fig. 3a, and two variations of BITE Systems are shown in Figs. 3b and 3c (remote comparison and remote comparison/decision capability respectively). Results of the trade study summary (Table 1) indicate that the greatest system advantage can be obtained with a total BITE concept. The partial BITE, delegation of the comparison function only, is the most inferior of the approaches reviewed.

The characteristics of each technique are also given in Table 1. The MADAR-type system requires analog data transmission (differential) and A-D conversion; hence reliability is reduced and cabling weights are high. The partial

Table 1

OBC TRADEOFF SUMMARY, OBC SYSTEM

Parameter	Weighting Factor	MADAR Type Score	MADAR Type Weighted	Partial Bite Score	Partial Bite Weighted	Total Bite Score	Total Bite Weighted
Power	10	3	30	1	10	10	100
Weight	10	1	10	1	10	10	100
Cost	2	8	16	2	4	10	20
Reliability	8	5	40	8	64	10	80
Data rate capability	5	8	40	10	50	10	50
Trending	4	10	40	0	0	0	0
System test (end-to-end)	8	10	80	0	0	0	0
Total			256		138		350

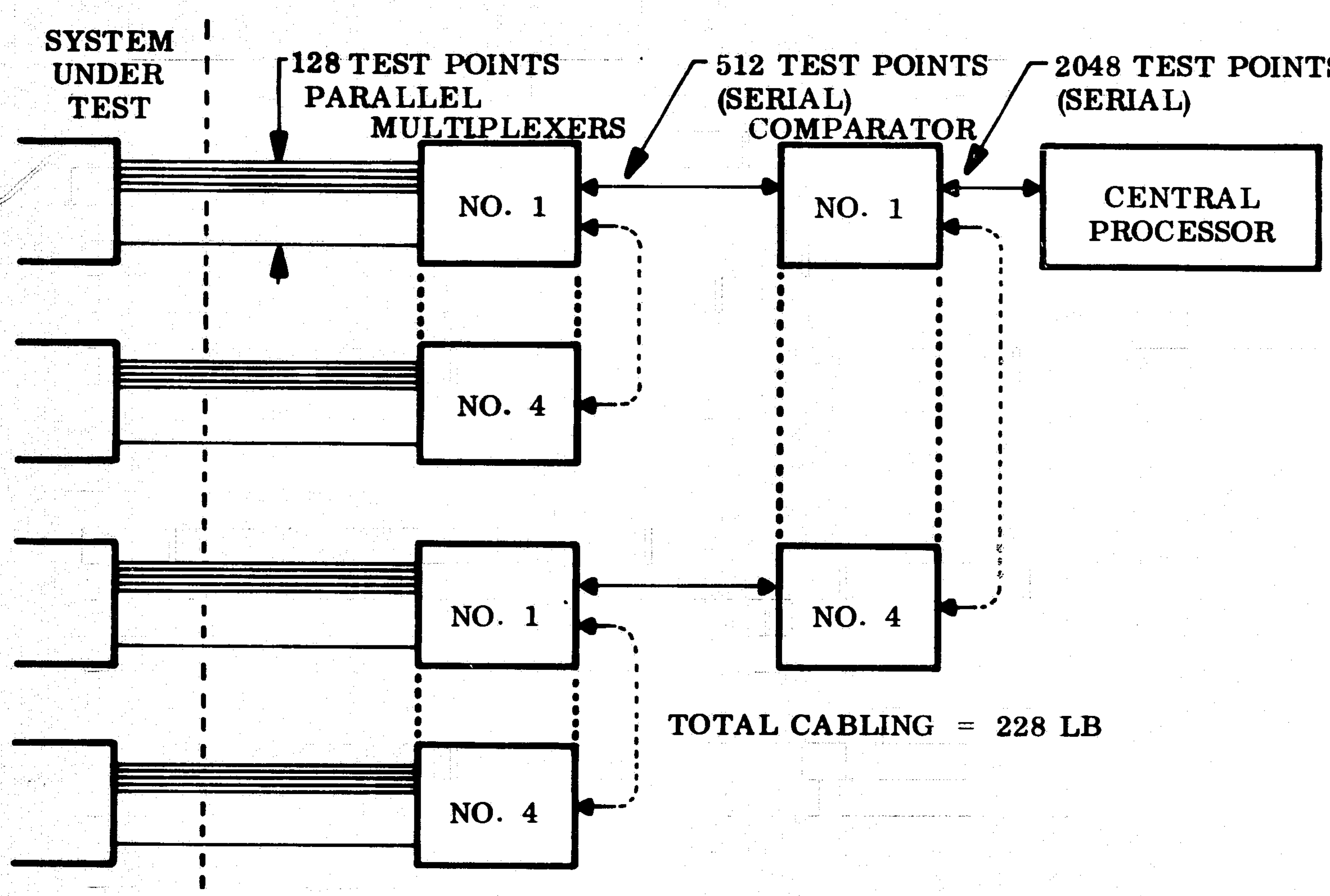


FIG. 2 OBC (MADAR TYPE) SIGNAL DISTRIBUTION

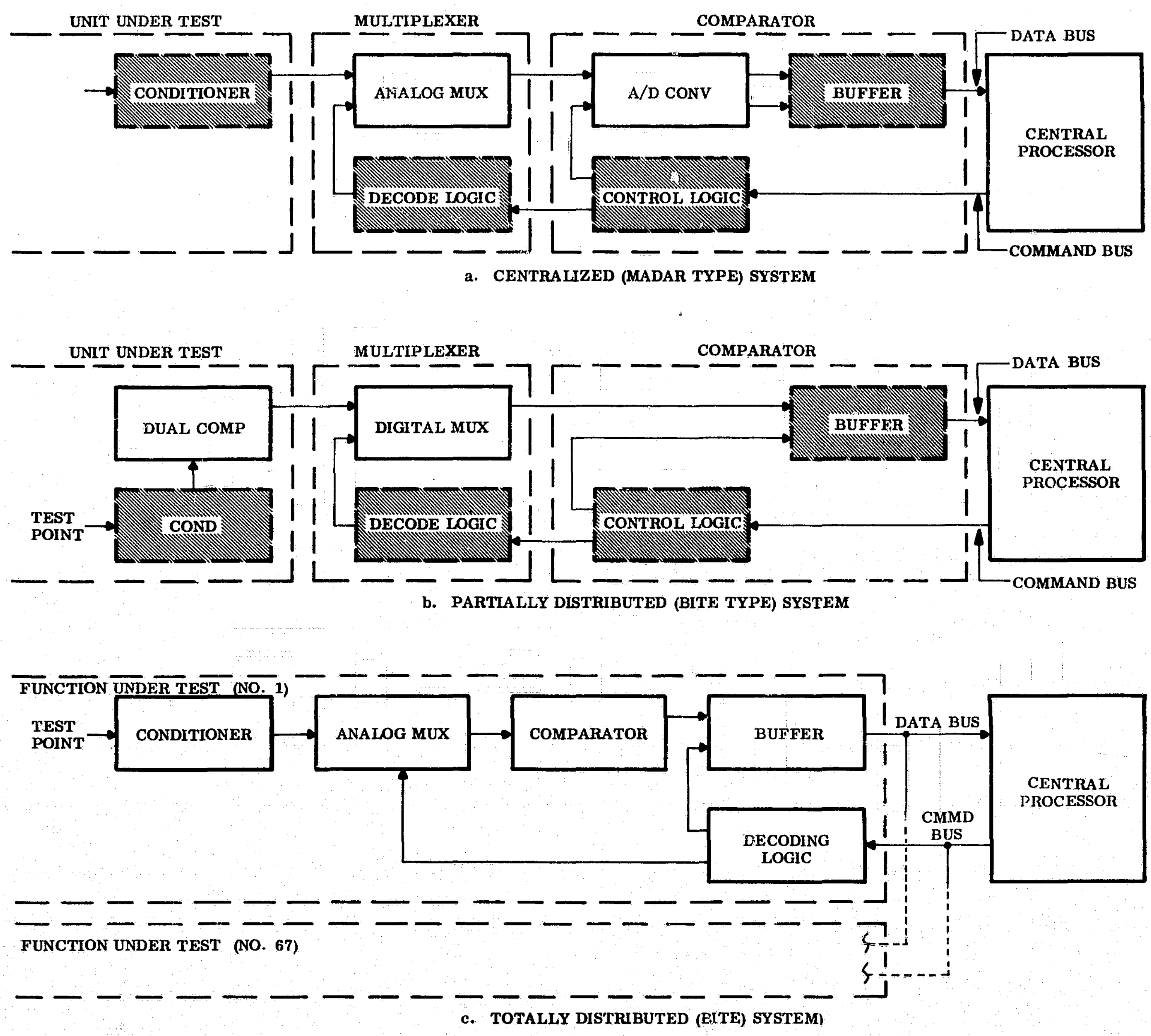


FIG. 3 BASIC OBC UNIT FUNCTIONS

BITE approach requires single-line digital data, but the high cost of non-multiplexed comparators more than offset any possible gain. The total BITE system is based upon a need for only 67 comparators. A summary of the test point/function distribution (Fig. 4) was used to ascertain this value. It was assumed that simplifications in the requirements on the decoding logic would be equal to the increased number of units required. This is admittedly optimistic (804 bits versus 640 bits for the system); however, it is more than compensated by the high probability that the signal conditioners could also be multiplexed in a hard design of this configuration. Both BITE techniques do not have trending or end-to-end test capability. Degradation detection by trend data analysis could still be provided by the central processor for a limited number of critical-wear sensitive components (hybrid approach). End-to-end testing needs are assumed to be satisfied by the sum of all parts (including a cable integrity check) being tested.

The incremental values used in the comparison are given in Tables 2 and 3. Linear multipliers were derived from current data sheets, price lists, production units, etc. This approach neglects the overhead base for each function, but the trends are still significant. This is most apparent in the total BITE presentation where the cabling (on a per-wire basis) and the central processor (synchronizing/recording functions only) have negligible effect on the total system.

The centralized system is limited to a maximum data rate capability of 4 MHz; this rate is based on a 1 microsecond/bit conversion rate in each of the A to D units. The BITE systems are essentially digital; they could be operated at higher rates (theoretically approaching 10 MHz). The system requirements will probably be closer to 500 kHz; hence any of these are more than adequate.

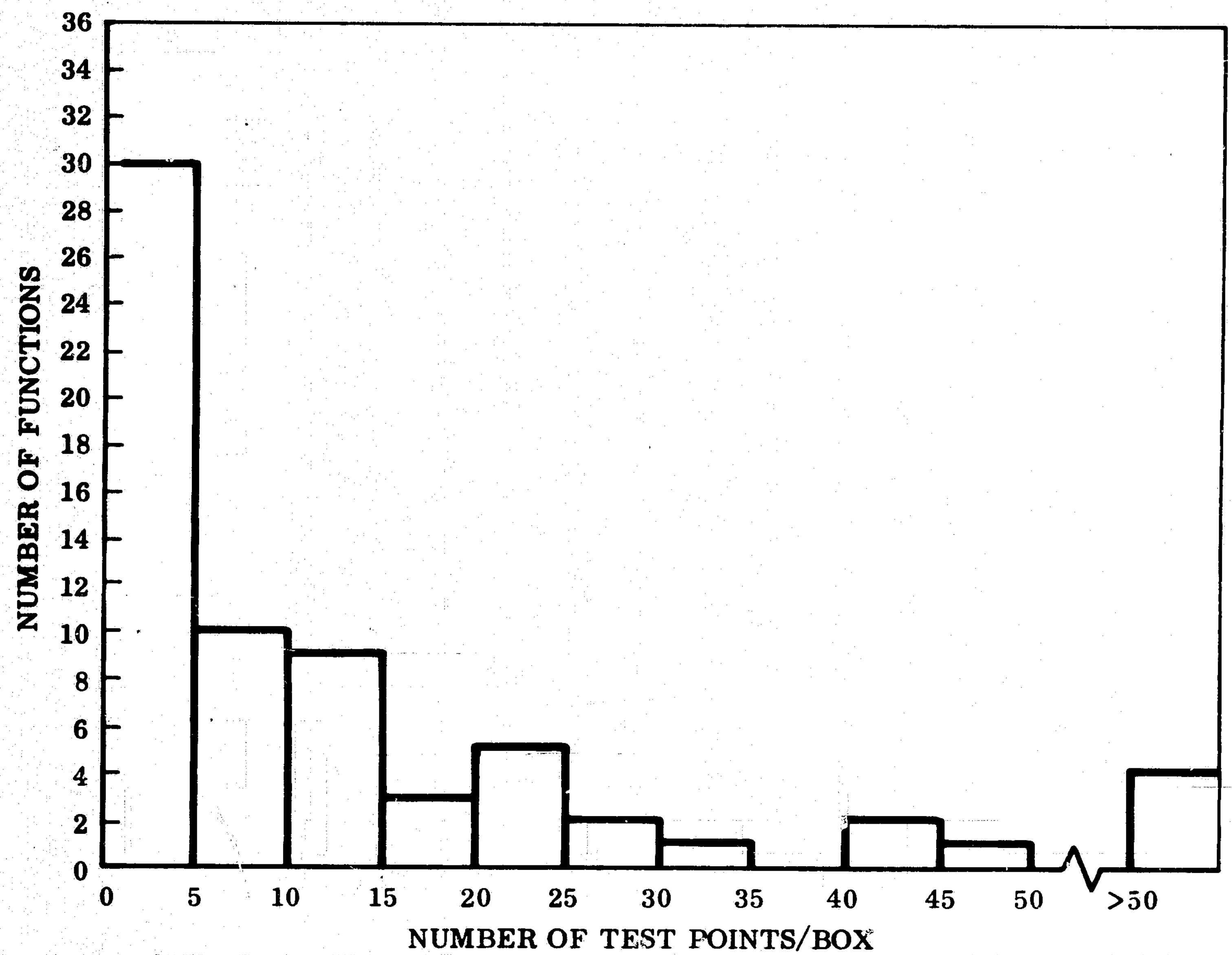


FIG. 4 TEST POINT DISTRIBUTION

Table 2

SYSTEM PARAMETER INCREMENTAL ESTIMATES

Parameter / Function	Power (watts)	Weight (lb)	Cost (dollars)	Single Channel Reliability (%/1000 hr)
Centralized (MADAR) System				
Analog multiplexer	21.8	1.81	17,200	0.1360
A/D converter	8.8	0.66	2,600	0.1160
Central processor	51.2	16.0	7,680	0.2560
Cable	0	130.0	-	0.0032
Total	81.8	158.47	27,480	0.5112
Partial Distribution (Remote Comparators)				
Comparator	300	66.0	58,000	0.0146
Digital multiplexer	39	1.92	6,850	0.0970
Central processor	12.8	4.0	1,920	0.0640
Cabling	0	65.0	-	0.0016
Total	351.8	136.92	66,770	0.1806
Total Distribution				
Analog multiplexer	10.9	0.91	8,600	0.0680
Comparator	10.1	0.22	1,943	0.014
Central processor	0.16	0.05	25	0.0008
Cabling	0	2.2	-	0.0016
Total	21.16	3.38	10,568	0.0844

Table 3

BACKUP DATA

Linear Multipliers

Weight -	0.0066 lb/electronic part	
	0.002 lb/digital word (32 bits)	
Cost -	0.03 dollar/bit memory	
	3.20 dollar/electronic part (fab, assy, inspection, test)	
	All other parts current list prices	
Power -	0.2 milliwatt/bit of memory	
	0.150 milliwatt/comparator	
	0.135 milliwatt/8 channel digital mux	
	0.080 milliwatt/8 channel (diff) analog mux	
Reliability -	1×10^{-6}	%/1000 hr/bit of memory
	0.0001	%/1000 hr/solder joint
	0.0008	%/1000 hr/cable connector
	0.0002	%/1000 hr/resistor
	0.004	%/1000 hr/IC

CONCLUSION

It is important that the OBC system approach be defined and incorporated as a part of the initial electronics design. The results of this study indicate a total BITE concept to be the most economical technique. The design should apparently converge the system to a single line as rapidly as possible. In favor of total BITE, the original equipment designer, theoretically, can (1) select the most critical parameters and (2) incorporate some dynamic measurements (reduced constraint on local tests). The level of centralized processing required for reporting status and performing integrity tests on cabling and BITE functions should not significantly change the total system. However, one major design decision to be made is the application of data management to all interfaces. The test data could be interleaved on the now-required signal transmission network. This may result in a "free-ride" for OBC with respect to distribution parameters (e.g., cable weights, A-D converters, and a large share of the multiplexing). The point ratings (Table 1) for centralized (MADAR) and total BITE systems would then become approximately the same. For this case, the selection of a preferred approach would be based on reevaluation of several parameters such as higher data rates (e.g., control operations require greater bandwidth than test functions) and definition of multiplexing responsibility (signal or test function).