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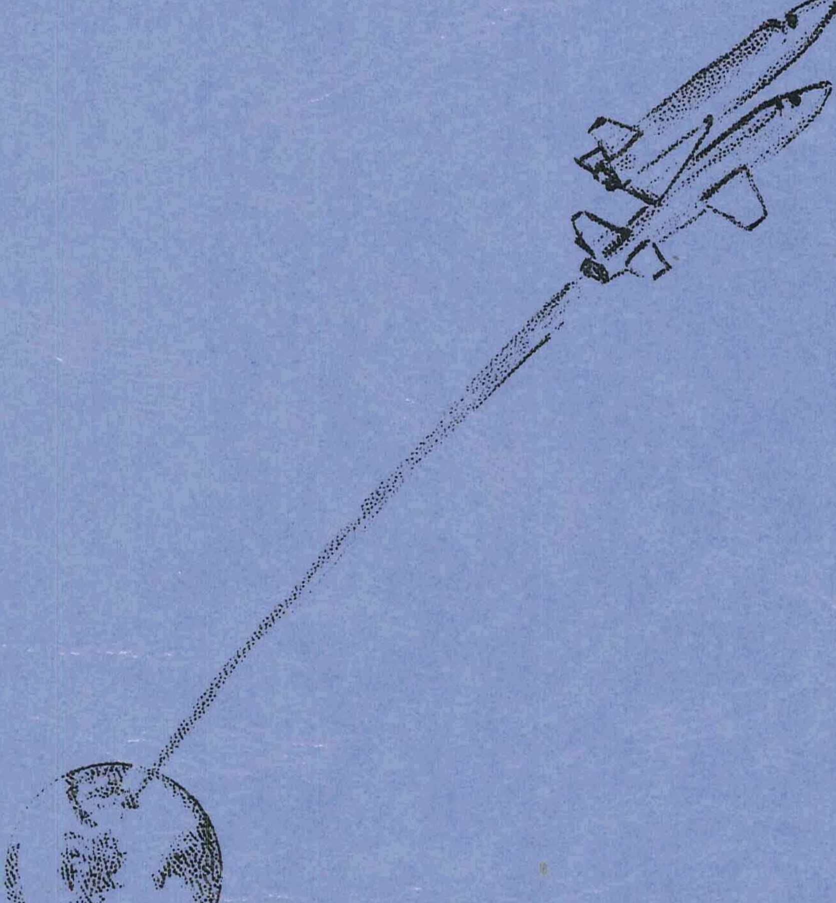
SPACE SHUTTLE DATA MANAGEMENT SYSTEM REQUIREMENTS ANALYSIS



FINAL REPORT

VOLUME I
STUDY SUMMARY

UNIVAC
DEFENSE SYSTEMS DIVISION
St. Paul, Minnesota

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UNIVAC

REPORT NO.
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SPACE SHUTTLE BOOSTER DATA MANAGEMENT SYSTEM (DMS) REQUIREMENTS ANALYSIS

Final Report

Volume I

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PREPARED FOR

George C. Marshall Space Flight Center
Final Report on
NAS8-30186, Mod 1

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ABSTRACT

The final report on the Space Shuttle Booster Data Management System (DMS) for NASA contract NAS8-30186 is in two volumes. The major items covered in Volume I are a discussion of the study results, a description of the booster mission, and a description of the functional requirements of the assumed avionics system. Volume II is devoted to subsystem interface description, subsystem computational requirements and a descript- of an analysis program generated and used during the study.

The reader should first read section 1.1 for a brief summary of the study and the major study conclusions. A summary of the detailed sizing data accumulated during the study is found in section 1.2. Further in-depth conclusions relating to various DMS configurations are found in section 1.3. Section 2 provides a detailed discussion of the Space Shuttle Booster mission and the function of the DMS. The reader interested in the detailed description of a particular subsystem is directed to the functional requirements in section 3, the interface requirements in section 4 and the computational requirements in section 5. The table of contents will direct the reader to the particular subsystem of interest.

A bibliography of the background literature used during the study including references in the report proper, is found at the end of Volume I.

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1.0 BOOSTER DATA MANAGEMENT SYSTEM (DMS) INVESTIGATION RESULTS

Univac has had the opportunity, under contract NAS8-30186 to review the DMS requirements for the Space Shuttle booster. The study began with a review of the space shuttle program, and the requirements placed upon the booster. From these mission requirements, it was possible to estimate the avionics systems that would be aboard. Next, the data management tasks required to service these avionics systems were listed. The logic, equations and interface needs were then accumulated. Finally a computer program ESCAPE (Evaluation of Shuttle Computational and Processing Events)(for the Univac 1108) was generated to compute total DMS loading as a function of the mission time-line. Subsequent modifications to the avionics systems configuration, or changes in precision, iteration rate etc., can easily be considered and evaluated using this technique.

This report documents the findings of the study. Volume I contains summary data, system/subsystem descriptions, and management level information. Volume II of the report contains the detailed interface, program logic and equation data. This section of the report, section 1, summarizes the results of the study. The conclusions presented in section 1.1 cover the Data Managment System (DMS) configurations that Univac considers primary candidates for the Space Shuttle Booster. The section also includes recommendations for further study. Section 1.2 describes the Space Shuttle Booster mission.

1.1 STUDY SUMMARY, CONCLUSIONS AND RECOMMENDATIONS

The objective of this study was to document a set of computational and data handling functions representative of the requirements of the Data Management System (DMS) for the Space Shuttle Booster Avionics System. The study developed a load model that was used to investigate a variety of computer configurations, resulting in immediate assessment of candidate computer configurations via graphical representation. A detailed assessment of candidate configurations is available via a computer program. The study results show that the major contribution to total computational load is imposed by the requirement for control of the main boost rocket engines. If all functions were to be lumped into a single computer, the computer would have to be capable of executing 1646 KIPS (including 75% excess capacity with an add time of .61 microseconds and a multiply of 4.86 microseconds). The computer storage requirements is 200,000 16 bit words*. The input/output data flow between the computer and all other systems is minimal; a data bus capable of handling a data rate of one million bits per second (including address data) is more than sufficient to handle the task. Use of a separate propulsion computer(s) reduces the DMS requirements to 608 KIPS (including 75% excess capacity with an add time of 1.64 microseconds and a multiply of 13.14 microseconds), with a negligible drop in memory requirements to 195,000 16 bit words.

Throughout this study, 16 bit instructions and 16 bit data words are assumed in the sizing counts. All 32 bit data words obviously require 2 16 bit words.

1.1.1 STUDY METHODOLODY

The above DMS requirements result from an extensive study of all avionics subsystems and represent an accumulation of the subsystem requirements. The subsystems were categorized into:

- Communications System
- Displays and Controls System
- Electrical Generation and Distribution System
- Flight Control System
- Navigation and Guidance System
- Propulsion System
- Vehicle Structures System

The functional requirements for each subsystem throughout the mission profile were developed. A search of all available space shuttle documentation was used as a basis for determining functional requirements and the detailed subsystem mechanization necessary to meet the functional requirements. Only the Space Shuttle booster was studied; many of the study conclusions are based upon assumptions which hold true only for the booster and quite different results may have been obtained if the study were directed toward the orbiter. The orbiter mission includes many phases not applicable to the booster e.g., the booster's navigation system has less stringent accuracy requirements, in that small velocity and position errors at thrust termination can be removed by the large delta velocity capability of the orbiter. A strapdown inertial measuring unit (rather than a platform system) was thus selected for the booster. Also the main rocket engine propulsion system on the booster will probably be different than that found on the orbiter, leading to a difference in the computational requirement. Many other less significant differences exist between the orbiter and booster, hence the study conclusions are not directly transferrable to the orbiter system.

The DMS computational load varies as a function of mission phase.
The major mission phases used during the study were:

- Prelaunch Checkout
- Boost
- Coast
- Reentry
- Cruise
- Landing

with subphases defined as required. The functional requirements were expanded into detailed computational requirements in the form of flow diagrams, equations, logic, and mission phase requirements. Program execution estimates were generated based upon the detailed computational requirements. Each program was assigned a start and completion time with respect to the mission time line; each program was represented as a constant computational load during the scheduled period. Computational load was determined by counting the number of instructions executed in a normal (as opposed to abnormal or failed) pass through the program. Instructions were segregated into two categories: short instruction times (add, substract and housekeeping instructions) and long instructions times (multiplication, divide and long shift operations). The required instruction executions in each category were multiplied by the required program iteration rates to obtain the instruction executions per second in each category. The instruction executions per second in each category were summed at each mission time point for each operative program, resulting in what is termed the "raw computational speed" estimates. A Univac 1108 computer program was developed which generates a time history of speed, memory and input/output requirements as a function of program module speed, memory and input/output specifications (in conjunction with DMS configuration specifications).

These estimates must be increased to arrive at the final requirements. Many studies have been performed to determine the ratio between raw estimates and the computer capacity necessary for a minimum cost system. Cedric O'Donnell, Alexander Williman and Barry Boehm (references 27 and 28) discussed the results of two such studies. In the determination of the computations required by each subsystem, one tends to overlook a function rather than to include unnecessary functions. This has been long recognized when specifying computer capacity for avionics system and other computer applications. Historically, a 15 to 20% increase above raw data estimates has been used, however, this factor has always proved to be grossly inadequate. This inadequacy is reflected in overruns in software costs and schedules.

There are a number of reasons why a 15 to 20% excess is inadequate. Initial program pressures usually require a higher priority for the generation of hardware designs and their subsequent release to production. Production of detailed software specifications receives secondary priority, which delays the initation of software development and causes many software changes to occur after initial software development is completed. Inadequacies in subsystem design can often be remedied with software changes. Software development is thus faced with a shortened schedule and continuous modifications. A reduced schedule can only be met by increasing the number of programmers. Increased efficiency is obtainable by using programming aids such as compilers, where efficiency in program execution speed and computer memory is sacrificed for program efficiency. Furthermore, modification of a program usually results in less efficient coding than would be generated had the modification been included in the original software specification. Barry Boehm (reference 28) justifies a 50 to 100% excess capacity over raw estimates in achieving minimum cost hardware plus software development. Univac recommends a 75% excess capacity.

1.1.2 COMPUTATIONAL SPEED REQUIREMENTS SUMMARY

Figure 1-1 is a summary of the computer speed requirements for the major flight phases. Multiply and add instructions have been combined using an 8 to 1 ratio. All values are given in units of thousands of instructions per second (KIPS). The data presented in the figure is based upon the raw computational speed estimates and does not include the requirements of 6.6 KIPS per computer. Only the major mission phases are shown in the figure; within each phase, the subphase was chosen that resulted in the largest total requirements for the mission phase. The ground checkout phase is not included since it does not contribute to the establishment of computer speed requirements; checkout functions can be distributed over a period of time.

Subsystem	Boost	Coast	Reentry	Cruise	Landing
Communications	17.2	17.2	9.4	8.5	0.6
Display and Controls	43.2	43.2	43.2	43.2	43.2
Electrical Generation	2.5	2.5	2.5	2.5	2.5
Flight Control	45.6	29.5	38.6	34.7	34.7
Navigation & Guidance	172.2	203.6	242.7	225.9	230.5
Propulsion	651.7	4.8	3.7	16.4	15.4
Vehicle Structures	1.6	0.9	0.9	0.9	1.1
Total	934.0	301.7	341.0	332.1	329.0

Figure 1-1 Total Booster Raw Computational Speed Reqmts. (KIPS)

Figure 1-1 shows that the largest computational speed requirements occur during boost. The propulsion system is the main contributer to the computational speed requirements during boost. During boost, the propulsion system controls the main booster rocket engines (12 engines were assumed during the study). Associated with each rocket engine is a complex control system for which the DMS determines the commands to the several proportional and on/off valves based upon desired thrust and sensed engine operation characteristics. The combined complexity and high iteration rate required by the engine control system results in the high computational speed requirements. It is highly probable that the main rocket engine control will not be included in the DMS computers. If the requirement to control the main rocket engines is removed from the DMS, the boost propulsion contribution drops from 651.7 to 2.0 in Figure 1-1 resulting in a total boost requirement of 284.3 KIPS. Figure 1-2 tabularizes the speed requirements of the major mission phases under the assumption that the propulsion system main rocket engine control is done elsewhere.

Mission Phase	Raw Speed Requirements	Minimum Capacity (50% Excess)	Desired Capacity (75% Excess)
Boost	290.9	436.4	509.1
Coast	308.3	462.4	539.5
Reentry	347.6	521.4	608.3
Cruise	338.7	508.0	592.7
Landing	335.6	503.4	587.3

Figure 1-2 Booster Computational Speed Requirements with Propulsion Control Removed (KIPS)

The executive requirements (assuming a single computer) have been included in these figures. The first column represents the raw data speed requirements and the last column, the recommended computational speed, which is determined by increasing the first column entries by 75%. The middle column represents the minimum recommended system which is 50% greater than column 1. These numbers again assumed an 8 to 1 ratio between multiply to add execution times. The maximum speed requirements occur during reentry with a desired computational speed of 608.3 KIPS. This represents a computer having a 1.64 microsecond add time and a 13.14 microsecond multiply time. The minimum speed requirements of 521.4 KIPS represents a computer having a 1.92 microsecond add time and a 15.32 microsecond multiply time.

Special attention was given to the computational load imposed by the generation of mathematical functions generally performed by subroutines, in particular sine and cosine , arcsine and arcosine, arctangent, exponential, logarithm, square root, and matrix multiply. For example, Figure 1-3 gives the raw data estimates for the subroutine functions during reentry and the percentage of the total computations being performed. A total of 32.8% of the computation time was spent in evaluating the subroutine functions. Nearly half of that time was spent in evaluating matrix multiplies. A significant reduction of the computational load on the main processor would occur if special hardware were provided to generate these subroutine functions.

Function	KIPS	%
Matrix Multiply	55.9	16.1%
Arc Tangent	26.1	7.5%
Sine and Cosine	13.8	4.0%
Arcsine and Arcosine	1.6	0.5%
Square Root	9.7	2.5%
Exponential	5.0	1.4%
Logarithm	1.8	0.5%

Figure 1-3 Subroutine Computational Requirements (Reentry)

The numerical integration of the strapdown inertial sensor data is the next major candidate (after the propulsion system main rocket engine control) for exclusion from the central DMS computers. Utilization of a separate strapdown computer would provide navigation outputs compatible with a platform system, facilitating switching between dissimilar redundant inertial navigation mechanizations, thus providing increased system redundancy and reliability. Strapdown computations could be performed by a DDA or a microprogrammed general purpose computer. Removal of the high iteration rate strapdown transformations and integrations from the DMS reduces the flight program requirements by 124.8 KIPS. Figure 1-4 shows the computational speed requirements of the DMS for each phase of the flight mission with the strapdown integration removed. Reentry remains the mission phase with the maximum computational speed requirements. The desired computational capacity of 389.9 KIPS represents a computer having a 2.56 microsecond add time and a 20.52 microsecond multiply time. The minimum capacity speed requirement of

334.2 KIPS during reentry represents a computer having a 3.00 microsecond add time and a 23.96 microsecond multiply time. Mechanizing the strapdown system external to the DMS also changes the central computer memory requirements and the computational time devoted to software mathematical subroutine evaluation. The raw data memory estimates of the central computer are reduced by 2.0 K words. The matrix multiply subroutine function is removed from the central computer reducing the total reentry software subroutine computational requirements to a raw data estimate of 58.0 KIPS which is 26.0% of the total computational requirements, as compared to 113.9 KIPS is shown in Figure 1-3.

Mission Phase	Raw Speed Reqmts.	Minimum Capacity (50% Excess)	Desired Capacity 75% Excess)
Boost	116.1	249.2	290.7
Coast	183.5	275.2	321.1
Reentry	222.8	334.2	389.9
Cruise	213.9	320.8	374.3
Landing	210.8	316.2	368.9

Figure 1-4 Booster Computational Speed Requirements with Propulsion Control and Strapdown Integration Removed (KIPS)

1.1.3 COMPUTATIONAL STORAGE REQUIREMENTS SUMMARY

There are several factors which contribute to the determination of the required memory size. The computer memory must be capable of holding both the instructions and data required of the mission programs. Generally, there are two types of data: constant and variable. Constant data are such items as control system gains and difference equation coefficients which do not change during the duration of the flight. Vehicle velocity and attitude is representative of variable data which does change during the mission. In accumulating memory sizing requirements, constant and variable data were segregated since some computers utilized a different memory technology to mechanize constant and variable data storage requirements. Constant data (along with the program instructions) can be mechanized using read only memory while variable data requires read/write memory. It is most probable that a single read/write memory technology will be used throughout, since the DMS is to be used both during flight for mission and subsystem control, and on the ground for diagnostic checkout. Word length is an important factor when specifying memory size. Word length can also be an important contributor to software development cost. The scaling problem is an item often referred to when discussing the high cost of software. A much discussed solution to the scaling problem is the implementation of floating point hardware. This solution is costly because of the additional logic required in the arithmetic section and sometimes results in the reduction of computational speed due to the more complex operations required by floating point operations. Every computational task performed has an accuracy requirement associated with it. If fixed point computations are used, the most severe scaling problems occur when the computer word length barely meets the program accuracy requirements. With minimum word length available, every data item must be scaled at the highest possible scaling that can be achieved without generating overflows. All calculations including minor intermediate steps must be scrutinized carefully to insure that the maximum accuracy is maintained without obtaining overflows. Excess word length beyond minimum accuracy requirements alleviates much of the scaling problem. The computational accuracy requirements fall into two categories. Most computations, except for navigation and guidance, can be performed using a minimum word length of 12-15 bits. The navigation and guidance computations require a minimum accuracy of 24 bits. The recommended word length is 16 bits for all functions except navigation and guidance which requires 32 bit precision.

The total size of the DMS memory is dependent upon the size, reliability, redundancy, and speed of the mass memory system. As the flight progresses, some programs are phased out and new programs required. If the mass memory has high reliability (including the data flow from the mass memory to the DMS computer memory) and is quadrupley redundant, new programs can be read from the mass memory to the DMS memory as they are needed. The new programs would be stored in locations previously occupied by phased out programs. The transfer of a program from mass memory to DMS memory must occur with sufficient speed to insure that the new program is completely loaded when needed and without slowing the computational speed of the DMS computer below acceptable limits. Figure 1-5 is the tabularized memory requirements for each major subsystem during each major flight mission mode.

Subsystem	Boost	Coast	Reentry	Cruise	Landing	All
Communications	17.5	17.5	15.7	6.7	2.0	22.2
Displays & Control	19.7	19.3	19.1	16.6	15.3	64.8
Electrical Genr.	.7	.7	.7	.7	.7	1.0
Flight Control	2.7	1.6	1.9	1.8	1.8	5.3
Navigation & Guid.	4.7	4.5	5.9	6.3	6.5	10.8
Propulsion	3.1	3.1	2.6	2.6	2.6	3.6
Vehicle Structures	.2	.1	.1	.2	.4	2.3

Figure 1-5 Total Booster Memory Requirements (K 16 bits words) Excluding Engine Control, Executive and Subroutines

The data is expressed in units of thousands of 16 bit words of memory. The numbers represent an accumulation of both fixed and variable storage. Thirty-two bit constants and variables required by the navigation and guidance programs have been counted as two words. The entries do not contain the memory required for the executive program and the software subroutines. If more than one computer is used in mechanizing the DMS, an executive program and the math subroutines package will be required in each computer. A very simple executive program has been assumed for the space shuttle booster application. Complex executive programs are required when the computer system usage is highly variable and unknown, however, the computational tasks of the DMS are well known prior to flight, allowing a simple executive to be used. The memory and computational loading of the executive is dependent upon the number of programs to be executed during flight and upon the number and iteration rate of each program being executed at any time. These variable memory and computational requirements have been counted with the requirements for each individual program. Configuration control and management, as well as malfunction detection and analysis have been included with these subsystem program counts. The remaining executive memory requirements plus storage requirements for the math subroutine package is 1460 words. The last column in Figure 1-5 is the memory requirements if all flight programs plus final prelaunch checkout is resident in the computers throughout flight. Maximum memory requirements occur during boost if an adequate mass memory is available. Figure 1-6 is a tabularized summary of the memory requirements showing the raw data estimates, the minimum requirements and the desired requirements.

Mission Phase	Raw Memory Reqmts.	Minimum Reqmts. (50% Excess)	Desired Capacity (75% Excess)
Boost	50.1	75.2	87.7
Coast	48.3	72.4	84.5
Reentry	47.5	71.2	83.1
Cruise	36.4	54.6	63.7
Landing	30.8	46.2	53.9
All	111.5	167.2	195.1

Figure 1-6 DMS Memory Requirements Summary (K 16 bit words)

The desired capacity with an adequate mass memory is 87.7 K words, and without an adequate mass memory is 195.1 K words. The data in Figure 1-6 was generated assuming all computational requirements except for the main rocket engine control are implemented in a single computer.

1.1.4 RELIABILITY REQUIREMENTS SUMMARY

The Fail Operational Fail Operational Fail Safe (FOFOFS) requirement demands a minimum of quadruple functional redundancy on critical systems. The avionics computer will be critical for mission operational success. A highly integrated system can be expected to result from an emphasis on cost and weight reduction and high degree of integration will most likely demand an operating DMS computer to ensure mission safety. Some basic assumptions must be made concerning the nature of failures and the required response to failures to determine the level of redundancy required to achieve a FOFOFS computer system. One of the primary redundancy candidates utilizes synchronized redundant computers with majority voting on the computer outputs to determine the correct system output. If it is assumed that identical simultaneous failures can occur , seven computers are required in a majority voting system that will survive three failures. Seven computers are required in order to have four correctly operating computers in majority over three failed computers. If simultaneous failures are ignored, then a failed computer can be removed from the system, reducing the number of redundant computers required to five. With five redundant computers and failed computers removed from the system after each failure, then the voting is four against one at the first failure, three against one at the second failure, and two against one at the third failure. If failures are random, the probability of three identical simultaneous failures occuring is negligible. If failures are causative, identical simultaneous failures are more probable. However, if a high probability of three identical simultaneous failures does exist, the system design is totally inadequate for the space shuttle application. Thus it can be assumed that five computers in a majority voting mechanization are sufficient to meet the FOFOFS requirement.

Four computers constitute the theoretically minimum number required to meet the FOFOFS redundancy requirement. With four computers, one remains after three failures. With four computers, majority voting can be used to resolve the first two failures since the vote would be three against one after the first failure or two against one after the second failure. A vote of one against one after the third failure is not sufficient to resolve the conflict. With four computers, built-in test equipment (BITE) is required to determine the third failure. Majority voting logic is very simple, so it is a primary candidate for resolving the first two failures. The third failure can be resolved by using either BITE or a fifth computer. A trade-off between BITE and a fifth computer must be based upon total system cost, weight, power, reliability etc. The complexity of BITE is dependent upon the portion of the computer being tested. BITE for a memory system consists of adding parity to the memory. The simplicity of mechanizing memory parity is evident by the fact that most large military and commercial computers include memory parity. BITE for a complex processor and input/output section is much more difficult to mechanize and may approach the size and complexity of a second processor and input/output section. By adding majority voting between the computer memories and processors and input/output sections, it is possible to achieve a system meeting the redundancy requirements having four memories with parity BITE and five redundant processors and input/output sections.

1.1.5 FURTHER STUDY RECOMMENDATIONS

The DMS requirements estimate are based on the assumptions and system philosophies that were developed early in the study. The basic assumptions used in the study reflect Univac's interpretation of NASA and industry thinking at the initiation of the study. Since the initiation of the study, many of the initial assumptions have been seriously reviewed by NASA and some of the assumptions are likely to change. In order to obtain meaningful results within the manpower and schedule limitations of the study, boundaries were placed upon the areas investigated. Further study recommendations fall into two categories; those resulting from a change in basic assumptions, and those which broaden the areas of investigation. Major recommendations for further study are:

a. Update Total Study

Many of the major and minor assumptions made during the study are based upon the results of Phase A study reports. The Phase B studies are presently nearing completion. It is recommended that this study be updated by incorporating Phase B study results.

b. Reduced System Sophistication

A highly sophisticated automated system with FOFOFS redundancy under extreme operating conditions, has been studied. A change in basic assumptions, modifying the load placed upon the crew would modify the DMS requirements. It is recommended that a less sophisticated design be studied.

c. Expand to Include Orbiter

The present study was restricted to the study of the booster. The orbiter has more stringent navigation accuracy requirements, additional mission phases including orbiting, rendezvous and on orbit experiments, a more critical reentry environment and may have unpowered landing (no go-around capability). These additional and changed mission requirements should result in quite different DMS requirements. It is recommended that the study be expanded to include the orbiter.

d. Expand Computational Summaries

The present study categorized instructions into two groups according to their relative execution times. These categories allowed detailed sizing studies based upon a computer with instructions falling into the assumed categorization. By expanding the number of categories and collecting additional trade off data, more detailed questions on computer architecture could be answered. Typical of the questions which would be answered are :

- What are the memory and speed requirements of a computer with and without index registers?
- What are the memory and speed requirements of a computer having a register file organization?
- What is the computational speed impact imposed by floating point arithmetic?

e. Dynamic Simulation

Each scheduled program was assumed to place a constant computational load upon the computer. A dynamic simulation would simulate the actual program execution, determining in detail peak computational loading points during the mission. A dynamic simulation can also be used to determine the influence of variations in computer capabilities upon total computational performance.

f. Low Cost DMS Study

Minor emphasis has been placed on economics in this study. It is recommended that the study be reviewed from an economic viewpoint, and recommendations be made regarding total development costs. Such costs would include available hardware, available software, hardware vs. software trade studies, software generation checkout and verification costs, new hardware development costs etc. Such a study could outline the positive and negative factors relative to a minimum cost system.

1.2 SPACE SHUTTLE BOOSTER MISSION DESCRIPTION

This section examines some of the major considerations which affect the design and functional use of the booster's Data Management System. These include:

- Vehicle configuration
- Space Shuttle Vehicle (SSV) mission
- Operational modes
- SSV desired characteristics
- System assumptions on results of tradeoff studies

1.2.1 VEHICLE CONFIGURATION

The SSV is designed to provide a cost effective transportation system for servicing earth orbital missions. The SSV is composed of a booster and an orbiter, both of which are reusable with minimal between-flight servicing. Present space shuttle studies indicate that the SSV will either consist of:

- Low cross range system - Delta booster and straight winged orbiter, or
- High cross range system - Twin tank booster and delta wing orbiter

Figure 1-7 shows a typical low cross range launch arrangement. Figure 1-8 shows a system's arrangement for the delta booster configuration. Figure 1-9 shows a typical twin tank booster design. DMS requirements will vary with the configuration selected. Major changes will occur in flight control and propulsion programs with relatively minor changes in other programs such as communications and recording.

1.2.2 MISSION PROFILE

A mission profile is shown in Figure 1-10 and the top level events of a logistics mission sequence are outlined in Figure 1-11. Both the booster and the orbiter are manned and capable of landing at a standard airport after completion of their mission. The primary phases of the booster flight mission are:

- <u>Launch</u> - The booster and orbiter are launched together after an autonomous checkout using minimal ground checkout equipment.
- <u>Boost</u> - The booster hydrogen/LOX engines will provide thrust to lift both booster and orbiter above the atomsphere with sufficient terminal velocity (approximately 9000 fps) for the orbiter to achieve an earth orbit using its own engines and allowing for sufficient orbiter fuel reserves for orbital maneuvering and de-orbiting.
- <u>Staging</u> - Upon achieving sufficient altitude and velocity, the booster engines will cut off, and the booster and orbiter will separate.
- <u>Coast</u> - The booster will then do an unpowered coast maintaining correct attitude for reentry until it reenters the atmosphere.

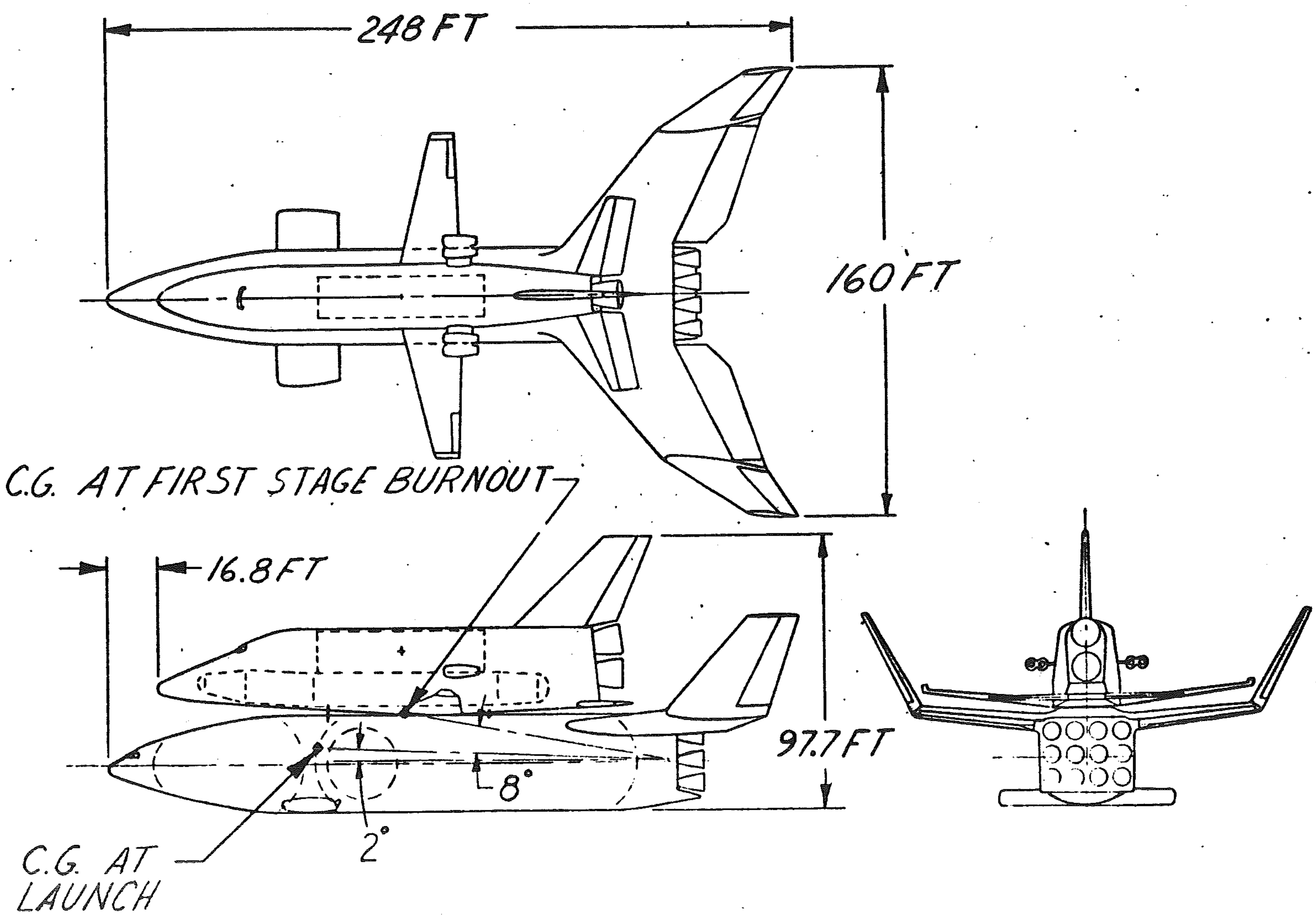


Figure 1-7
Typical Low Cross-Range Launch Arrangement

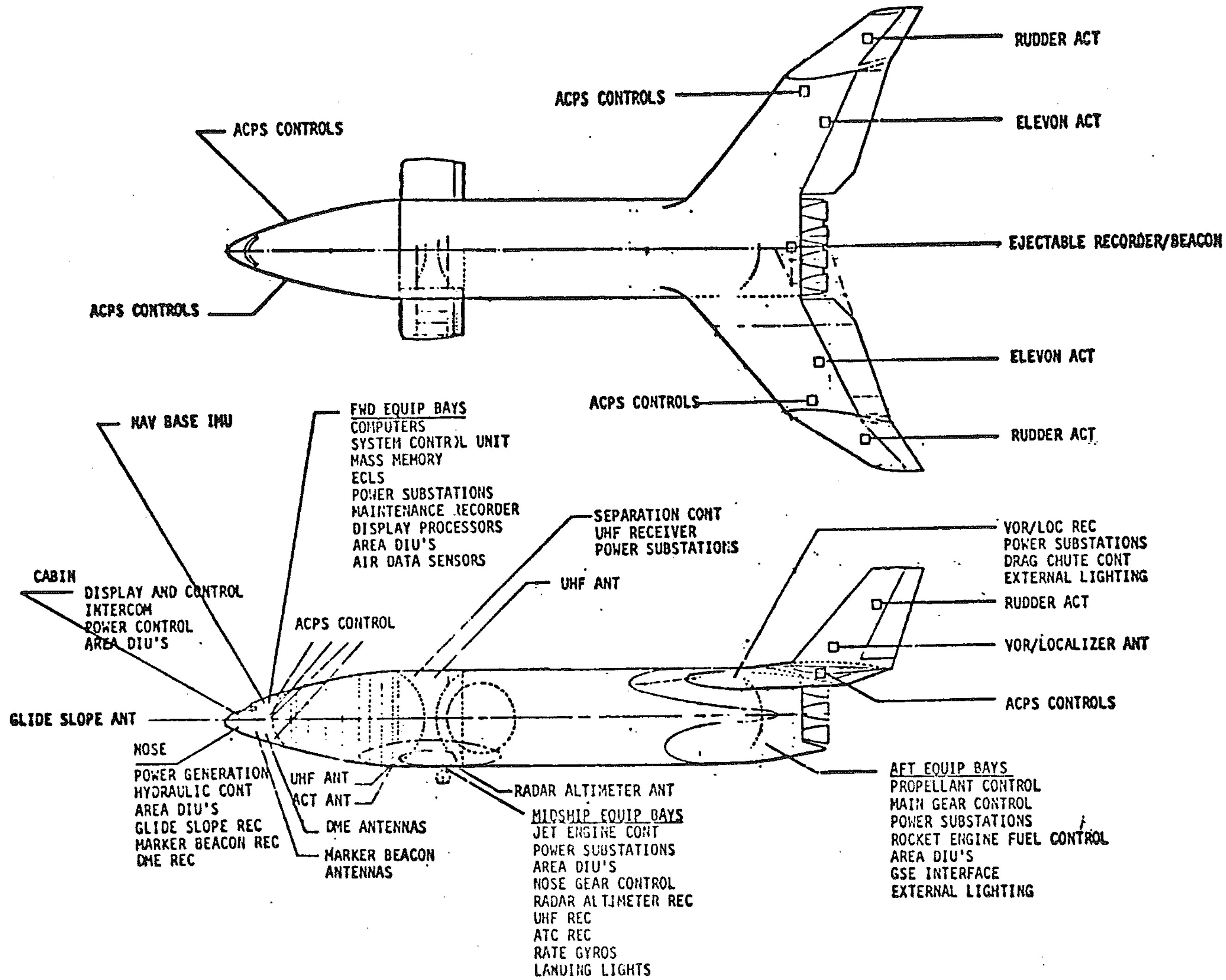


Figure 1-8
Booster

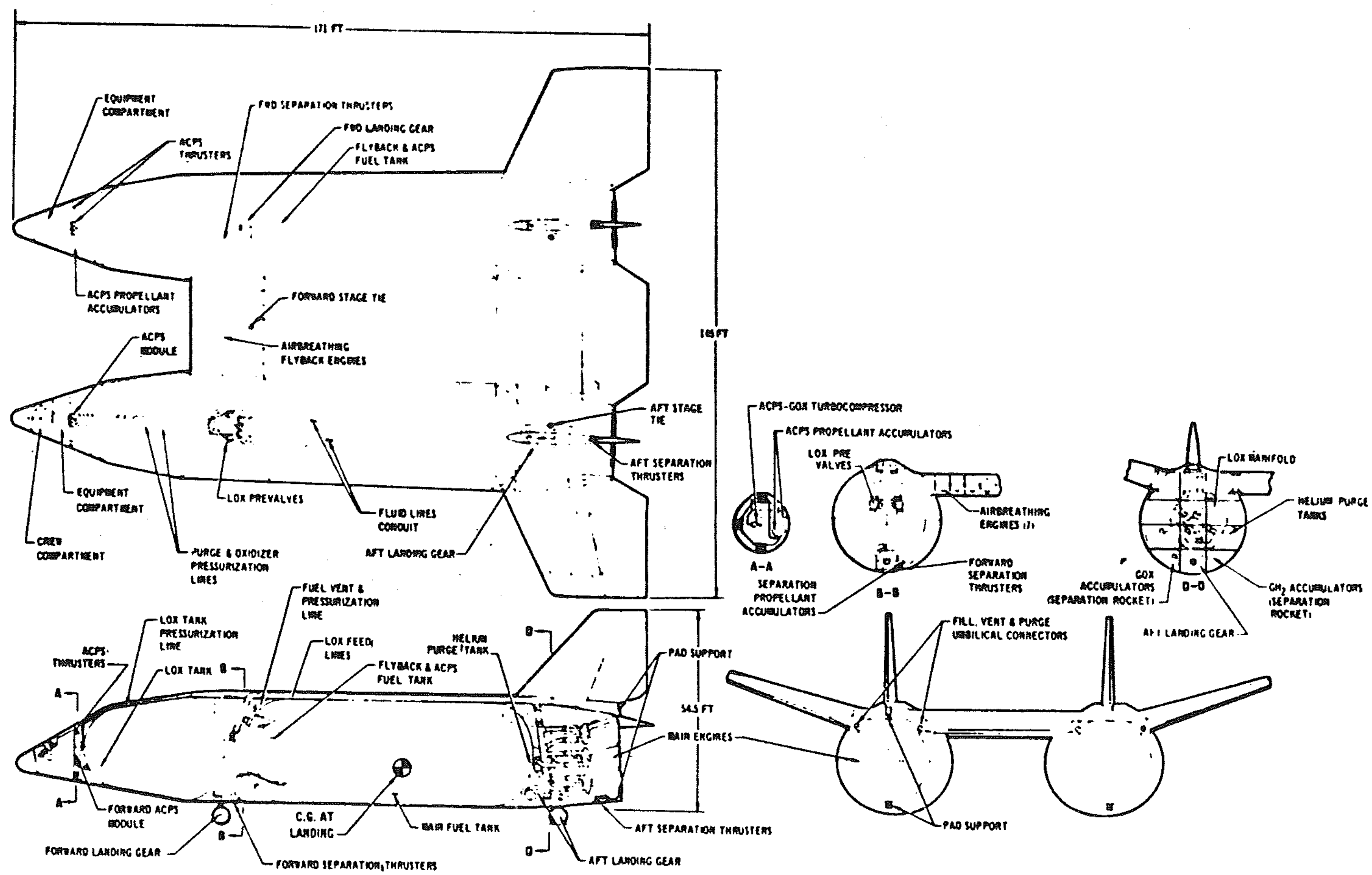


Figure 1-9
Twin Body Booster Arrangement

MISSION EVENT	EVENT INITIATION TIME	EVENT DURATION
	PRELAUNCH OPERATIONS	
Transport Booster to Pad	T_o - 24 hrs	1 hr
Erect Booster on Pad	T_o - 23 hrs	5 hrs
(Transport Orbiter to Pad)	T_o - 18 hrs	1 hr
(Erect Orbiter on Pad)	To - 17 hrs	4 hrs
Mate Orbiter to Booster	T_o - 13 hrs	3 hrs
Connect Holddowns	T_o - 10.5 hrs	30 mins
Connect Cyogenic Service Lines	T_o - 10 hrs	2 hrs
Perform Tank Leakage Test	T_o - 8 hrs	2 hrs
Power Up for Range Check and Navigational Input	T_o - 6 hrs	2 hrs
Propulsion System Operational Checkout	T_o - 6 hrs	2 hrs
Final Preparation and Inspection	T_o - 4 hrs	1 hr
Crew Ingress	T_o - 3.5 hrs	20 mins
Final Systems Checkout and Guidance Update	T_o - 3 hrs	1 hr
On Pad Abort (Engines Off)	To - 3 hrs	
Systems Monitoring	T_o - 2 hrs	1 hr 50 mins
Cryogenic Servicing	T_o - 2 hrs	1 hr 50 mins
(Passenger Ingress)	T_O - 40 mins	25 mins
GSE Removal	T_o - 15 mins	5 mins
Final Countdown	T_o - 10 mins	10 mins
	ASCENT PHASE	
Ignite Engines	T_o - 3 sec	-
Holddown Space Vehicles	T_o - 3 sec	3 sec
Lift Off (Release Holddowns)	T_o	-
Pre Separation Abort	T_o	206 sec
Translate Engine Nozzles	T_o + 54 sec	-
Max -q	T_o + 75 sec	-
Begin Constant 3_g Acceleration	T_o + 167 sec	39 sec
Shutdown Booster Engines	T_o + 200 sec	-
Separate Orbiter and Booster	$T_s = T_o + 206$ sec	-
	DESCENT AND LANDING PHASE	
Begin Booster Descent Phase	T_s	-
Reentry Abort I	T_s	100 sec
Begin Reentry Maneuver	T_s + 10 sec	90 sec
Begin Aerodynamic Braking	T_s + 100 sec	-
Reentry Abort II	T_s + 100 sec	5746 sec
Begin Unpowered Return	T_L - 5860 sec ($T_L = T_s$ + 2 hrs)	5860 sec
Decelerate to Approach Speed	T_L - 354 sec	-
Approach Abort	T_L - 354 sec	354 sec
Start Cruise Engines	T_L - 336 sec	336 sec
Holding	T_L - 336 sec	120 sec
Level Off	T_1 - 162 sec	-
Extend Landing Gear	T_L - 162 sec	-
Intercept Glide Slope	T_L - 144 sec	-
Landing (Normal)	T_L - T_s + 2 hrs	-
Landing (After Hold)	T_L - 120 sec	-
Taxi	T_L +	10 min

Figure 1-10 Sequence of Events

MISSION EVENT	EVENT INITIATION TIME	EVENT DURATION
	MAINTENANCE PHASE	
Crew Egress	T_L + 10 min	20 min
Install Safety Devices	T_L + 30 min	12 min
Make Visual Inspection	T_L + 42 min	15 min
Perform Cabin Switch Check	T_L + 48 min	15 min
Cool and Decontaminate Booster	T_L + 63 min	3 hrs
Move Booster to Post Flight Maintenance Area.	T_L + 4 hrs	1 hr
Position Emergency Equipment	T_L + 5 hrs	15 min
Provide Access to Required Areas	T_L + 5 hrs	30 mins
Deservice ACS	T_L + 7 hrs	7 hrs
Deservice Propulsion System	T_L + 14 hrs	12.5 hrs
Deservice APU System	T_L + 14 hrs	7 hrs
Deservice Fuel Cells	T_L + 14 hrs	7 hrs
Move Booster to Preflight Maintenance Area	T_L + 26.5 hrs	1 hr
(Go to Ferry Phase if Required)		
Install Booster on Handling/Transporting Vehicle	T_L + 27.5 hrs	1 hr
Position Maintenance AGE	T_M=T_L + 28.5 hrs	30 mins
Begin Quality Assurance Inspection	T_M + 30 mins	119.5 hrs
Checkout Booster Subsystems	T_M + 30 mins	4.5 hrs
Provide Access to Required Areas	T_M + 5 hrs	8 hrs
Perform Scheduled Maintenance	T_M + 13 hrs	80 hrs
Perform Unscheduled Maintenance	T_M + 13 hrs	80 hrs
Close up Access Areas	T_M + 93 hrs	3 hrs
Perform Post Maintenance Procedures	T_M + 96 hrs	24 hrs
Load Booster on Erector/Transporter	T_M + 113 hrs	1.5 hrs
Load Crew Provisions on Booster	T_M + 114.5 hrs	30 mins
Load Jet Fuel	T_M + 115 hrs	2 hrs
	FERRY PHASE	
Load Jet Fuel	T_F - 3 hrs	2 hrs
Crew Ingress and Ferry Pre-flight Check	T_F - 1 hr	45 min
Engine Start and Taxi	T_F - 15 min	15 min
Takeoff	T_F	1 min
Takeoff Abort	T_F	1 min
Climbout	T_F + 1 min	9 min
Climbout Abort	T_F + 1 min	9 min
Cruise	T_F + 10 min	1 hr 50 mins
Cruise Abort	T_F + 10 min	1 hr 50 mins
Approach	T_{LF} = 354 sec	354 sec

Figure 1-10 Sequence of Events (Continued)

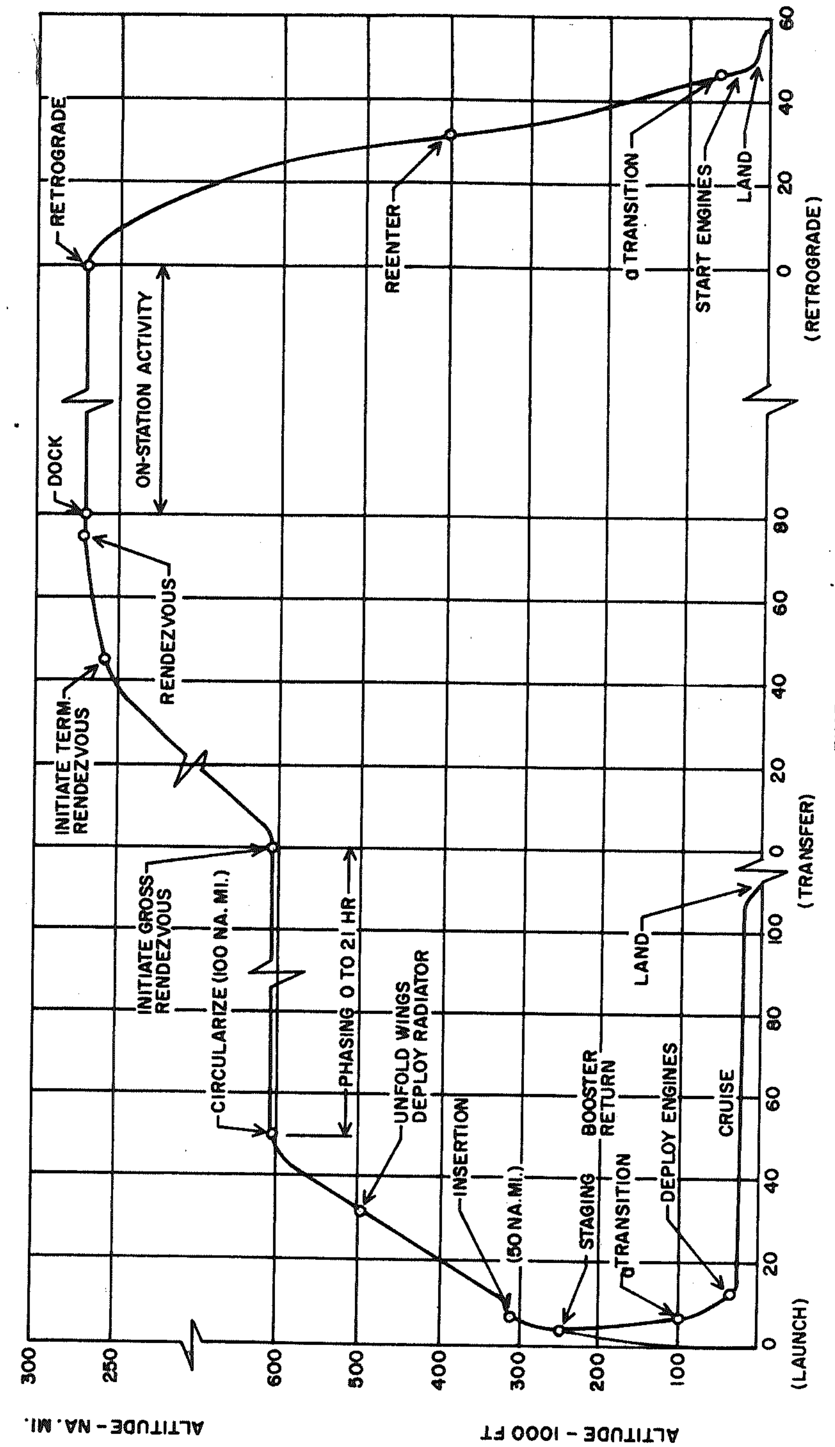


Figure 1-11 Baseline Logistics Mission Profile
Low X-Range System

- Reentry - The booster will maintain a large angle of attack (near 60°) as it is aerodynamically decelerated from hypersonic to subsonic velocity (approximately 300 fps at 40,000 feet).

- Transition to Aerodynamic Flight - At approximately 40,000 feet and 300 fps, the booster will make a transition to aerodynamic flight by deploying its airbreathing engines and assuming a nose down attitude until adequate velocity for level flight is reached. The turbojet engines will then be started.

- Cruise - After starting the turbojet engines and obtaining sufficient velocity to maintain equilibrium flight, the booster will cruise to a desired airport for landing. In cruise, the booster will be capable of following visual flight rules (VFR) or instrument flight rules (IFR).

- Approach and Landing - Cruise ends as the booster enters the airport's approach corridor; at which time, flight path and descent rate are controlled such that the booster achieves the proper velocity, heading, and altitude to initiate flare and final glide to touchdown. Go around and automatic landing capabilities are included.

- Ferry - After landing and refueling the booster will be capable of executing a standard airport take off and a climb to cruise altitude enabling it to return to an airport near the launch site.

The primary phases of the orbiter following separation from the booster are:

- Orbit Boost - The orbiter conducts a low energy, low temperature ascent into a parking orbit. The parking orbit is circularized and the orbiter then coasts until it is in phase with the space station.

- Orbit Maneuvers - The orbiter performs a gross rendezvous maneuver to move it to approximately 15 miles below and 75 miles behind the space station. A 200-foot-per-second impluse terminal maneuver brings the orbiter to 3 miles behind the station. A docking maneuver is performed to bring the orbiter and space station together. Support operations are conducted while the vehicles are mated.

- De-Orbit - After undocking at the optimum point, the orbiter performs a de-orbit maneuver in which its de-orbit motor is fired for a maximum time of ten seconds. The descent from orbit continues to about 400,000 feet where the entry maneuver is initiated. Mission phases from this point are similar to those described for the booster.

1.2.3 OPERATIONAL MODES

Operational modes cover a specified portion of the mission time line. Transition from one mode to the next is determined by some significant change in environment, system configuration or data management system activity. The booster mission consists of the following phases and operational modes:

- Prelaunch phase
    - Mechanical preparation and servicing mode
    - Initial testing and flight preparation
    - Countdown
    - Lift off ready.
- Ascent phase
    - Vertical boost
    - Trajectory establishment
    - Trajectory control
    - Separation initialization
    - Post separation
    - Coast
- Descent and Landing phase
    - Reentry maneuver
    - Flyback turn
    - Glide
    - Cruise
    - Initial approach
        - Holding pattern
    - Final approach
        - Go around
    - Landing rollout and taxi
- Post Flight Maintenance Phase
    - Post flight security
    - Booster system deservice
- Maintenance Cycle Phase
    - Pre maintenance preparation
    - Maintenance
    - Pre-pad servicing
- Ferry Phase
    - Ferry flight systems check
    - Taxi
    - Climbout
    - Cruise
    - Initial approach
        - Holding pattern

- Final approach
    - Go around
- Landing rollout and taxi
- Post Ferry flight inspection

Prelaunch Phase

The Mechanical Preparation and Servicing mode starts 24 hours prior to liftoff. This mode consists of booster transportation to and erection on the launch pad. The orbiter and the booster are mated, and holddowns and service lines are connected. Cryogenic, and other servicing, which could not be performed in the refurbishment and maintenance areas, are performed during this mode. No requirements for data management system usage is anticipated.

The Initial Testing and Launch Preparation mode performs those tests necessary for verifying the space vehicle integrity. Tank leakage tests, system calibrations, and propulsion system checkouts are conducted. Diagnostic tests are used to localize troubles if failures are detected by system verification tests.

During the Countdown mode the monitoring of all systems is continued. Methods are provided for the automatic return of the vehicle system to a safe configuration if the countdown mode is interrupted by a condition requiring a hold in the mission.

The Liftoff Ready mode occurs during the interval between engine ignition and release of the holddowns. Systems are monitored for any hazardous condition resulting from abnormal engine operation.

Ascent Phase

The Vertical Boost mode begins at liftoff. During this mode a programmed 20 second vertical rise and roll maneuver is executed. The roll program serves to rotate the launch configuration from the launch aligned azimuth to the desired flight azimuth.

The Trajectory Establishment mode supervises the vehicle flight during the subsonic, atmospheric portion of the ascent phase. Flight path, propulsion and aerodynamic data are displayed for crew evaluation.

The Trajectory Control mode continues the monitoring of flight dynamics into the hypersonic region. Flight path errors are minimized to provide an accurate orbital entry for the orbiter vehicle.

The Separation Initialization mode prepares vehicle systems for a separation of the booster and orbiter which will keep the orbiter on its proper flight path. Body rates and flight path rate are nulled. Booster engines are shut down and thrust decay is monitored to determine proper moment for separation.

The Post Separation mode performs and monitors the separation maneuver. Thrust engine performance and flight parameters are monitored to ensure that collision between booster and orbiter does not occur after separation.

The Coast mode concludes the ascent phase. The booster continues on a ballistic trajectory.

Descent and Landing Phase

The Reentry Maneuver mode controls the performance of a maneuver designed to initiate fly back of the booster to the launch site or other designated airport. Angle of attack and velocity are controlled to keep the booster within temperature and load factor constraints. When the proper altitude and velocity conditions are met, a pullout maneuver is controlled and monitored.

The Flyback Turn mode controls the turn which heads the booster towards the airport of intended landing. Velocity, altitude, bank angle, and angle of attack parameters are controlled and monitored to ensure safe transition to aerodynamic flight.

The Glide mode continues the transition to aerodynamic flight. Booster systems no longer needed are secured. Jet engines are deployed and prepared for use. Navigation and guidance systems provide flight path data for heading to facilities serving desired airport.

In the Cruise mode, the booster is under FAA control at an assigned altitude. Jet engines have been started and their performance monitored and controlled. Standard navigational aids for airways flight are used as the booster nears the airport facilities.

The Initial Approach begins at a specified navigational fix. A prescribed rate of descent is executed to an inner navigational fix where the landing gear is lowered, and level flight is maintained while final pre-landing checks are performed.

A Holding Pattern mode may be required if traffic or weather conditions make this necessary. Air traffic controllers will specify altitude, holding maneuver, and time of leaving pattern. Booster performance, fuel consumption, and navigational aids are monitored during this mode.

The Final Approach mode starts when the glide path to the airport runway is intercepted, and ends at touchdown. Proper speed and position on the glide path is maintained, and all systems continue to be monitored as the booster approaches the landing point.

The Go Around mode may occur at any time during the final approach, if required by the tower controllers because of traffic pattern conditions. The booster flies a go-around pattern and rejoins the traffic pattern. Rate-of-climb, altitude, air speed, and navigational aids provide data for the next landing attempt.

The Landing Rollout and Taxi mode monitors events occuring between touchdown and engine shutdown. Booster deceleration functions are supervised. Events following landing rollout will depend upon airport facilities and booster conditions. If booster fuel supply and airprot conditions permit, the booster may taxi under its own power to service area. Otherwise, provisions for towing the booster will be employed.

Post Flight Maintenance Phase

The Post Flight Security mode commences after the booster has been parked in the service area. The crew removes onboard checkout, flight recorder and flight log tapes. Maintenance personnel check switch positions, install safety devices, and make visual inspections of all outside surfaces. The booster is cooled before maintenance can continue.

The Booster Systems Deservice mode performs necessary tasks in deservicing tbe attitude control system, the main propulsion system, fuel cells and auxiliary power units. Proper procedures are carried out in storing or disposing of such material as helium, oxygen, and hydrogen.

Maintenance Cycle Phase

The Pre-maintenance Preparation mode performs the tasks of moving the booster from the service to the maintenance area, inspecting mechanical safety devices, and making all preparations necessary for maintenance operations.

The Maintenance mode consists of the detailed inspections and tests of the components of all booster systems. The flight discrepancies and component malfunctions detected by the onboard checkout system are reviewed. After equipment inspection, repair, or replacement, the onboard checkout system verifies systems performance and integration.

The Pre-pad Servicing mode is entered approximately 26 hours prior to launch. The booster is loaded on an erector/transporter vehicle and moved to the pad where the sequence of events of the prelaunch phase is repeated.

Ferry Phase

The Ferry Flight Systems Check mode is entered following the post flight maintenance phase if a ferry flight is required. The check emphasizes verification of jet engine, aerodynamic flight control, and airways navigation systems.

The Taxi mode includes events from leaving the service area to positioning on the takeoff runway.

The Climbout mode performs a flight procedure requrested by ATC clearance. Air data information is monitored and flight data situation is dipalayed. When the booster has reached its assigned altitude it enters the Cruise mode which is followed by Approach and Landing modes. These modes during the ferry flight are similar to those which occur following the booster launch.

The Post Ferry Flight Inspection mode follows landing of the booster at the airport servicing the launch pad. The booster is secured and inspected. It then starts the maintenance cycle phase in preparation for the next launch.

1.2.4 DMS OPERATIONAL MODE ACTIVITIES

This section presents an overview of the detailed programs described in Volume II. Activities occuring during prelaunch, boost, coast, reentry, cruise, landing, preferry, and takeoff operations are outlined. Flowcharts are used to indicate scheduling, assignment and decision functions under the control of the DMS software programs.

Figure 1-12 outlines the major executive tasks. The first event which occurs upon starting the DMS computers is the initialization of the executive routine. Initializing the executive routines includes descheduling all computer programs and scheduling the DMS checkout routine. After initialization the executive routine is entered. The executive program calls the only scheduled program, the DMS checkout program. Upon ascertaining DMS operability the checkout program schedules the keyboard entry program and deschedules the DMS checkout program. The program waits at this point for an entry from the pilot or operator indicating an operational, or ferry mission or a special request. In the special request branch a selective scheduling program is scheduled which allows the operator to schedule and deschedule any program in the DMS or to read and schedule any mass storage program. This allows the operator to selectively energize systems and checkout functions, to obtain post flight data reduction programs or to obtain diagnostic programs. Selecting the operational or ferry branches schedules the prelaunch or preferry programs which start the sequences for the respective missions.

Prelaunch Activities

Figure 1-13 is a flow diagram of the prelaunch activities. The prelaunch sequence is achieved by scheduling a master prelaunch checkout sequencing program. This determines the order in which subsystems are tested and the sequencing of all prelaunch operations. The booster system is progressively built up into an operational configuration. The test results are displayed as each system is tested. Test failures are catagorized into four groups:

1. Failure does not effect mission success or operational reliability. An example of this type failure would be a failure in the performance monitoring system.

2. Failure reduces system reliability but not operational capabilities. An example of this type failure would be the loss of one redundant electronic path.

3. Failure reduces system operation and reliability but does not make mission success impossible. An example of this type failure is the loss of a main engine or turbo jet engine. A successful mission is possible but at a much greater risk to the crew.

4. Primary system failure making it impossible to perform mission. An example of this type failure would be the failure of 4 strapdown accelerometers making it impossible to perform navigation.

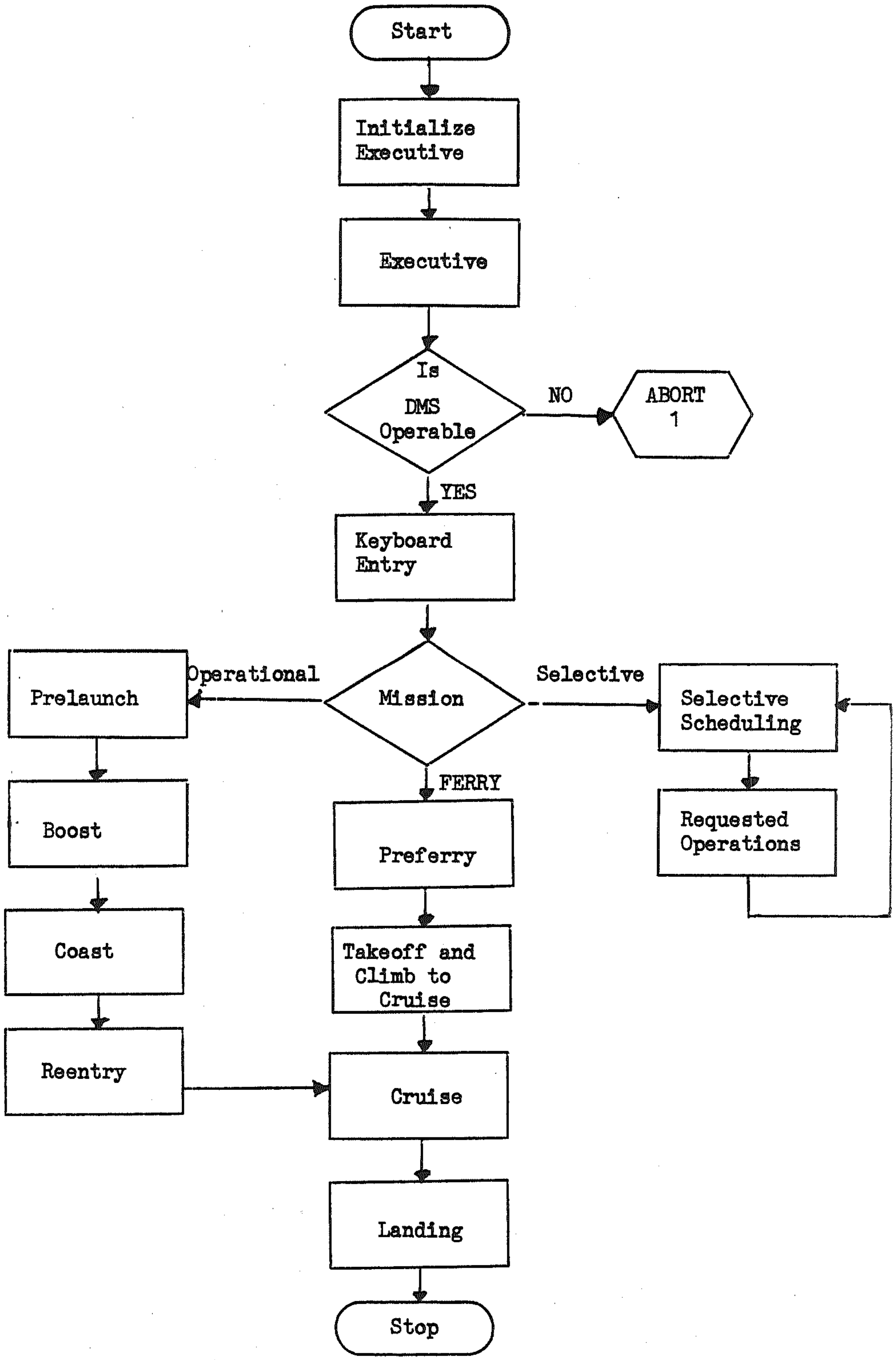


Figure 1-12 DMS Program

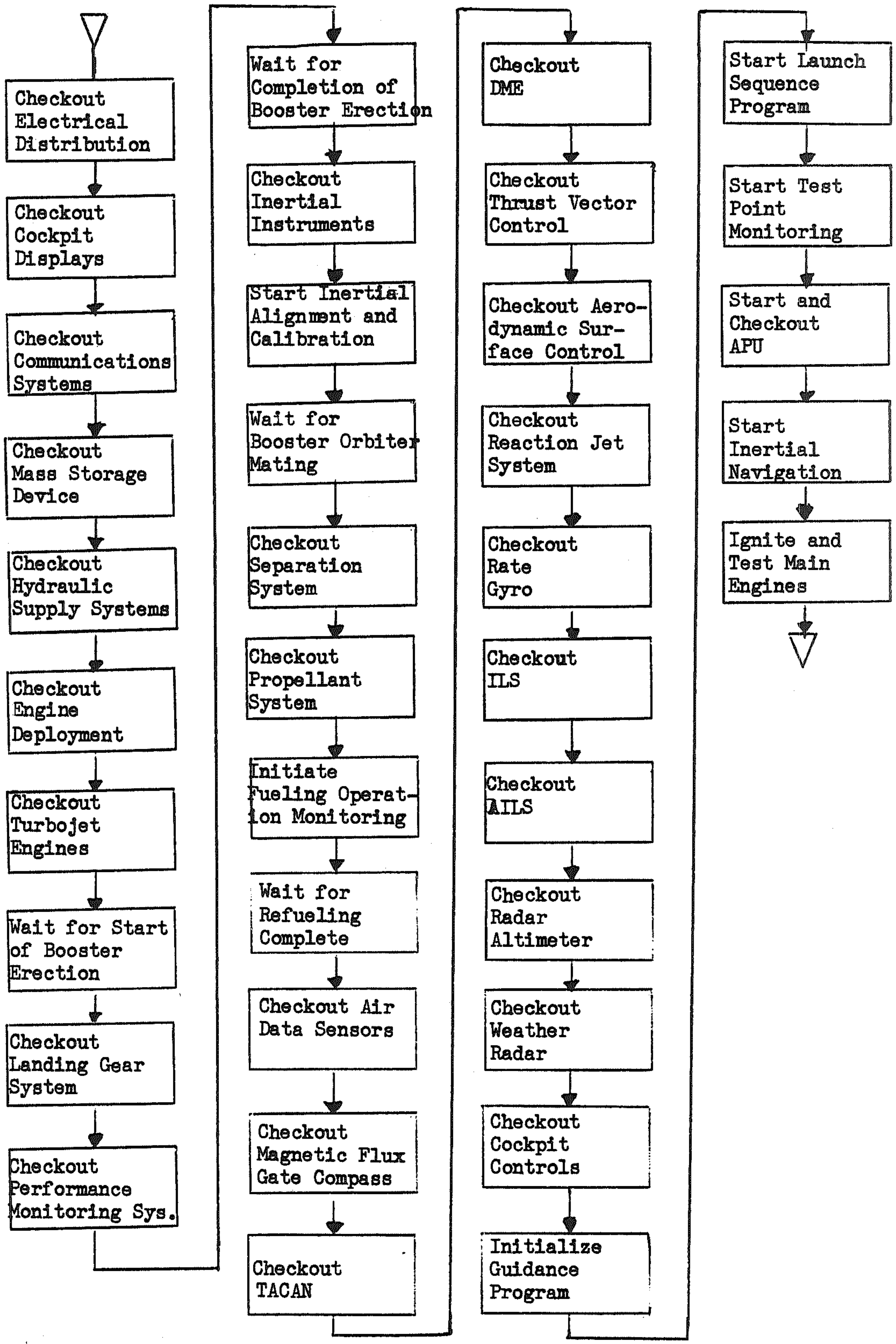


Figure 7-13 Prelaunch Program

It is assumed that the prelaunch function will be initiated before the booster is erected for launch. The first test performed is the checkout of the electrical distribution system in order to insure that power can be supplied to all subsystems. The cockpit displays are checked out next in order to insure that a means exists of displaying test results of the checkout of other systems. The communications system is tested next. This permits the booster crew to communicate with the ground crew so that ground and onboard operations can be coordinated. The mass storage device is then checked out to determine if diagnostic programs are available in the event of a subsystem failure. The hydraulic system is then activated and checked out in preparation for checkout of the turbojet engine deployment, landing gear system and control system actuator. The turbojet engine deployment mechanism is then tested and the engines started and tested while deployed. The engines are stowed after checkout. Prelaunch checkout is then halted until the booster erection process is started. As soon as the booster is lifted from the ground as part of the erection process the landing gear system is checked out. This test ends with the landing gear in a stowed condition. During the erection process the performance monitoring system is checked out. Performance monitoring is tested during booster erection because low amplitude vibration outputs from the vibration sensors can be expected during this activity. The prelaunch program is then halted until the erection process is completed. When the erection process is completed the inertial sensors are tested and the inertial alignment and calibration program started. Alignment and calibration accuracy depend on the time available. For this reason alignment and calibration are initiated as soon as possible after the booster is placed in the launch position. The prelaunch program is then halted until the orbiter has been mated to the booster. After mating the orbiter to the booster the propulsion system tanks, lines and valves are tested in preparation for fueling the booster. During the fueling operation the prelaunch program monitors the fueling operation. After fueling is completed the program checks the Air data sensors, the magnetic Flux Gate Compass, the TACAN, the DME, the thrust vector control system, the areodynamic surface control system, the reaction jet system, the rate gyros, the ILS, the AILS, the radar altimeter, the weather radar, and the cockpit controls. These checkout functions are shown sequentially in the flow diagram. Some of them, however, may be scheduled simultaneously if pilot load permits. The guidance program is initialized after the sensors and controls have been checked. This schedules the start launch sequence program which computes launch time. This program controls the start of all additional prelaunch programs as a function of the time left to launch time. Test points are monitored during this period to insure that all systems remain operational. At a fixed time before launch the auxiliary power units(APU) are started and checked out. Next the inertial navigation alignment program is descheduled and the inertial navigation program started. The main engines are then ignited and tested during the ignition sequence. The scheduling of main engine ingition ends the prelaunch activities and initiates boost activities.

Boost Activities

Figure 1-14 is a flow diagram of the boost program. Boost starts with the issuance of the main engine ignition signals. Some programs, originally scheduled during prelaunch, continue

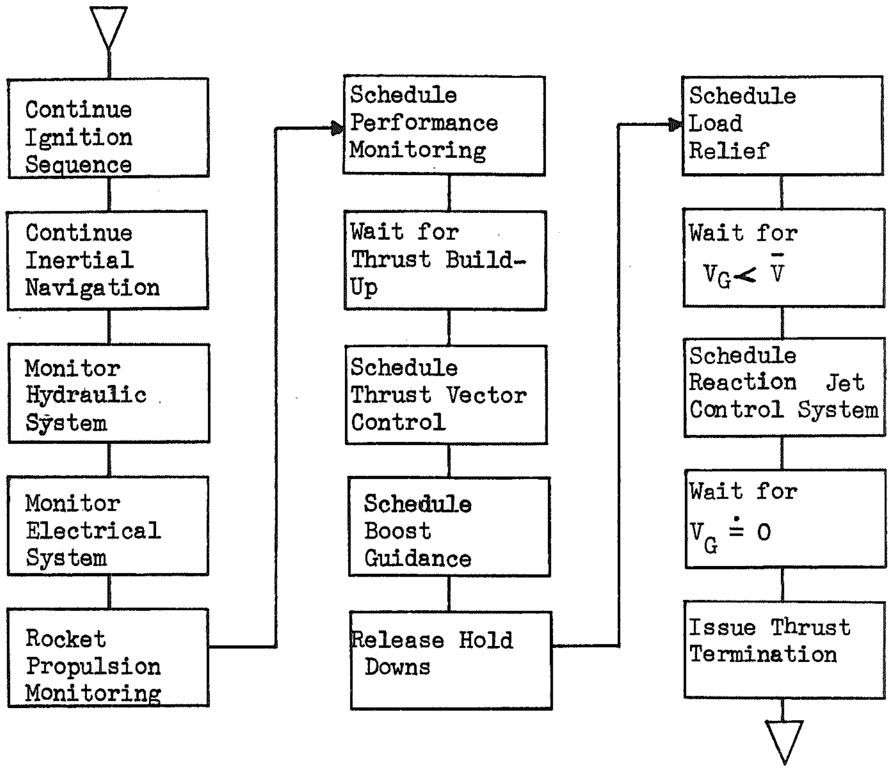


Figure 1-14 Boost Program

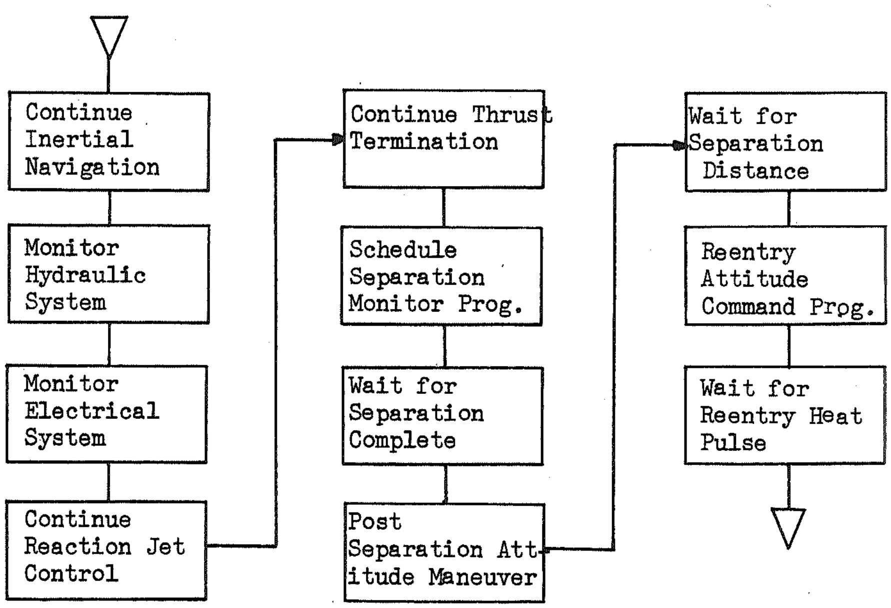


Figure 1-15 Coast Program

to function during boost. These programs are the ignition sequence program, Inertial Navigation, Hydraulic System Monitoring, Rocket Engine Propulsion System Monitoring, and Electrical System Monitoring. The Performance Monitoring program is initiated with the main engine ignition command. The boost guidance and control programs are scheduled when proper thrust build up is achieved. Boost control is achieved through thrust vectoring of the main engines. When the thrust reaches a predetermined magnitude with proper engine operation the release hold down command is issued. The load relief program is then scheduled. The load relief program controls the issuance of load relief commands only in the event of loading on the vehicle. The boost guidance program provides a computation of velocity to be gained. Toward the end of the powered boost flight when velocity to be gained has been reduced to a small value the reaction jet attitude control system is energized. A thrust termination signal is issued when the velocity to be gained is nearly zero. This ends the boost flight phase and starts the coast phase,

Coast Activities

Figure 1-15 is a flow diagram of the coast phase DMS programs. Programs which are scheduled prior to the coast phase but continue to function during coast are Inertial Navigation, Hydraulic System Monitoring, Electrical System Monitoring, Reaction Jet Control and Thrust Termination Sequencing. The orbiter will have control of the issuance of the separation commands. The booster monitors the separation operation. When separation is complete the booster will command an attitude control maneuver designed to minimize orbiter plume impingment. After achieving sufficient separation distance so that plume impingment is no longer possible, the Reentry Attitude Command Program is scheduled which computes the proper vehicle attitude for reentry. The boost phase is over when the temperature sensors on the booster skin register the occurence of the reentry heat pulse.

Reentry Activities

Figure 1-16 is a flow diagram of the reentry phase DMS programs. Programs which are scheduled prior to reentry but remain scheduled during reentry are Inertial Navigation, Hydraulic System Monitoring, Electrical System Monitoring and Reaction Jet Control. Immediately upon entry to the reentry flight phase Route Point Steering, Aerodynamic Control, Aerodynamic/Reaction Jet Mix and Reentry Guidance are scheduled. The high vehicle skin temperatures encountered during reentry prevent the transmission and reception of radio signals. The end of the high heat rate during initial reentry is awaited at which time the Radar Altimeter, Air Data Inputs, Magnetic Flux Gate Compass, TACAN, DMS, Navigation Update, and Weather Radar Programs are scheduled. At the same time the Reaction Jet Attitude Control Program is descheduled. The DMS then waits for an altitude/velocity criteria to be met which determines the point at which to deploy the cruise engines. The cruise engines are then deployed and started and climb to cruise altitude vertical guidance scheduled.

Cruise Activities

Figure 1-17 is a flow diagram of the cruise program. Programs which have been scheduled prior to cruise and remain scheduled during cruise are Inertial Navigation, Hydraulic System Monitoring, Electrical System Monitoring, Route Point Steering, Aerodynamic Control, Radar Altimeter, Air Data Inputs, Magnetic Flux Gate Compass, TACAN, DMS, Navigation Update, Weather Radar, and Cruise Engine Montoring. As soon as the cruise engines are deployed and started the landing gear up warning program is scheduled. This program sounds an alarm to the pilot if the vehicle landing configuration (flaps down or engines idle) is commanded with the landing gear up. A holding pattern program is scheduled and descheduled upon pilot demand. The DMS maintains this configuration until the scheduled or pilot designated landing airport is within a predetermined range. The landing mission phase is entered at this time.

Landing Activities

Figure 1-18 is a flow diagram of the landing program. Those programs scheduled prior to the landing phase which remain scheduled are Inertial Navigation, Hydraulic System Monitoring, Electrical System Monitoring, Route Point Steering, Aerodynamic Control, Radar Altimeter, Air Data Inputs, Magnetic Flux Gate Compass, TACAN, DME, Weather Radar, Cruise Engine Monitoring, and Landing Gear Up Warning Programs. Upon entering the landing program,data within the DMS is tested to determine if the selected landing airport has an operating AILS system. Either AILS or ILS guidance and navigation update are scheduled dependent upon the test results. The program controlling descent for landing is also scheduled. The DMS continuously tests range to the airport and when a preset range is reached the route point steering program is descheduled and landingpattern steering scheduled. A second range value is awaited at which time the lower landing gear program, nose wheel steering and main gear steering are scheduled. After touchdown and a shut down signal from the pilot all systems are shut down and the DMS returns to the selective scheduling mode.

Preferry Activities

Preferry is a selective subset of the prelaunch activities. Figure 1-19 is a flow diagram of the preferry program. The electrical distribution system, cockpit displays, and communications system are first checked out using umbilical ground power. This insures that all equipments required for the ferry mission can be supplied without electrical failures, the results of all checkout activities can be displayed to the pilot, and the pilot has communication with all ground facilities. The APU is then started and checked out and the mass storage device is tested. This allows the pilot to read and use any diagnostic programs in the event ofa failure indication. The hydraulic system is then turned on and tested to allow the use and checkout of those systems requiring hydraulic power. The inertial instruments are checked out and the alignment and calibration program started. The Air Data Sensors, Magnetic Flux Gate Compass, TACAN, DME, Aerodymamic Controls, Rate Gyro, ILS, AILS, radar altimeter, Weather Radar, and Cockpit Controls are then checked out. The Nose Wheel Steering, and Raise Landing Gear Programs are then scheduled. The Main Gear Steering program

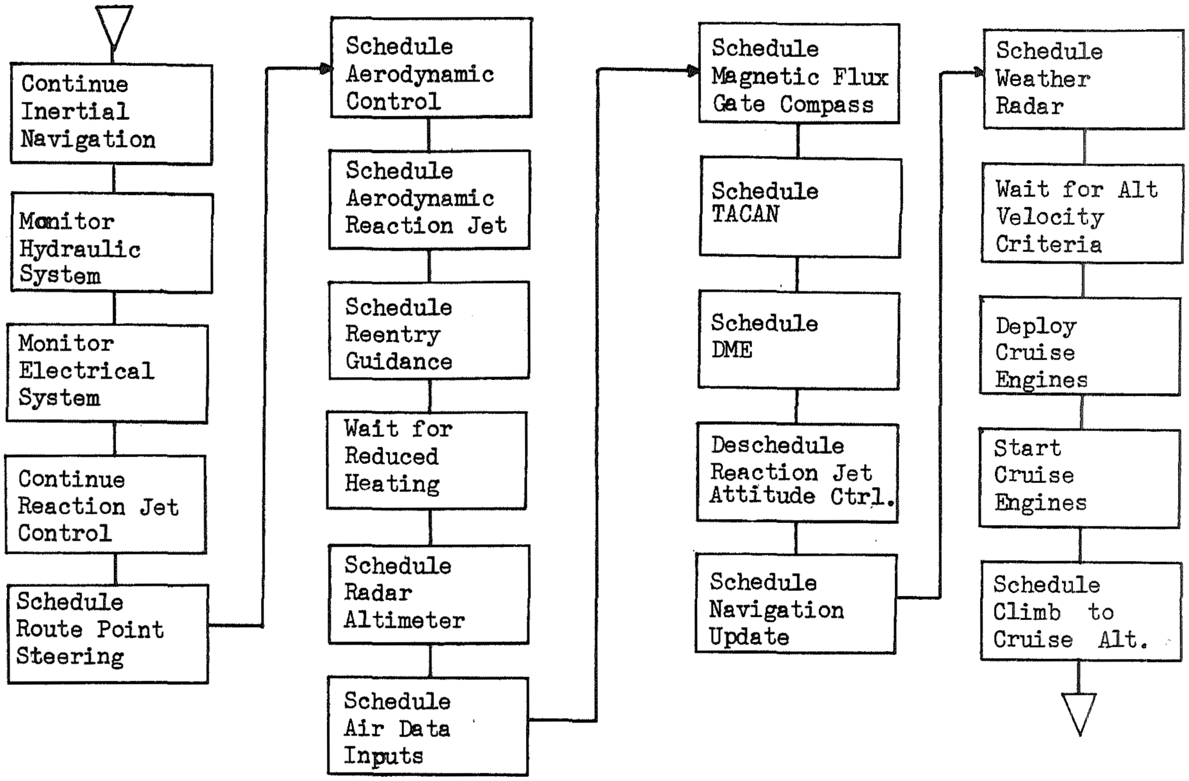


Figure 1-16 Reentry Program

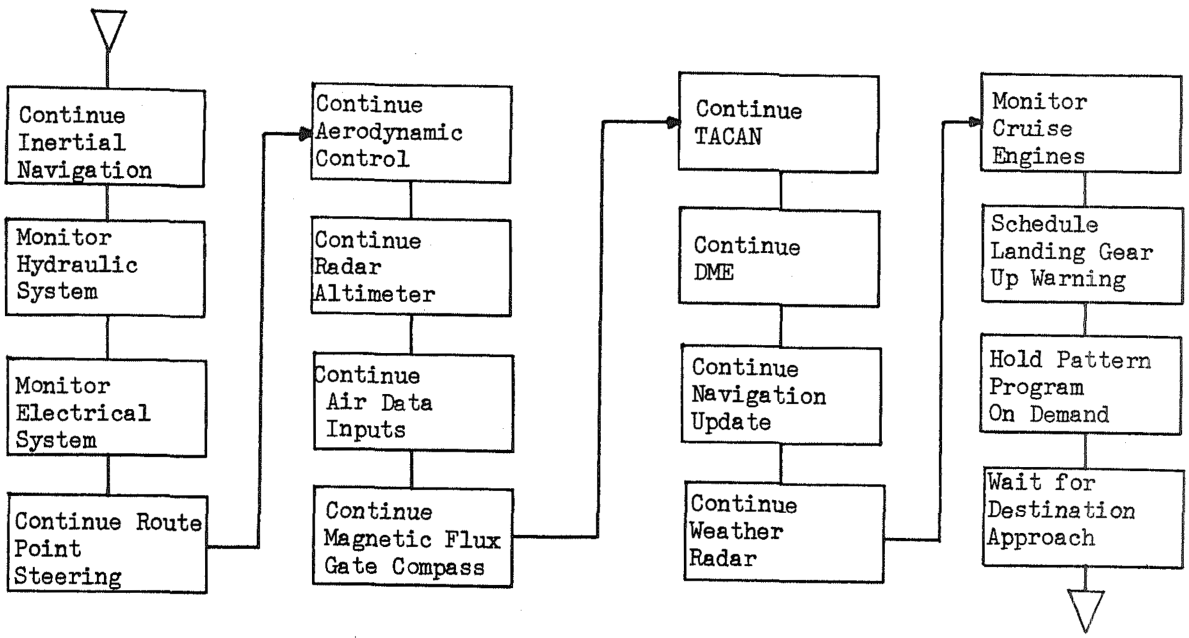


Figure 1-17 Cruise Program

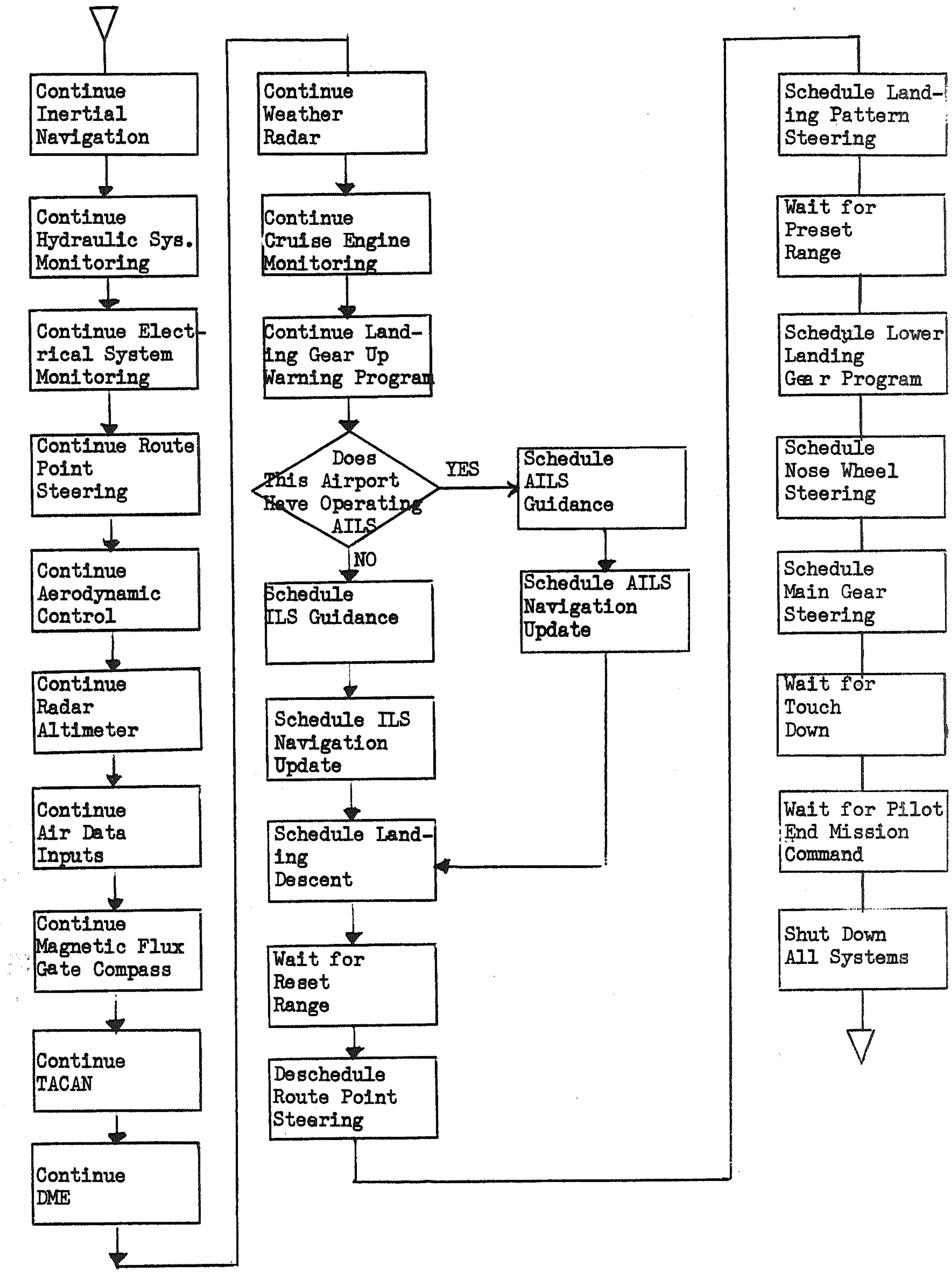


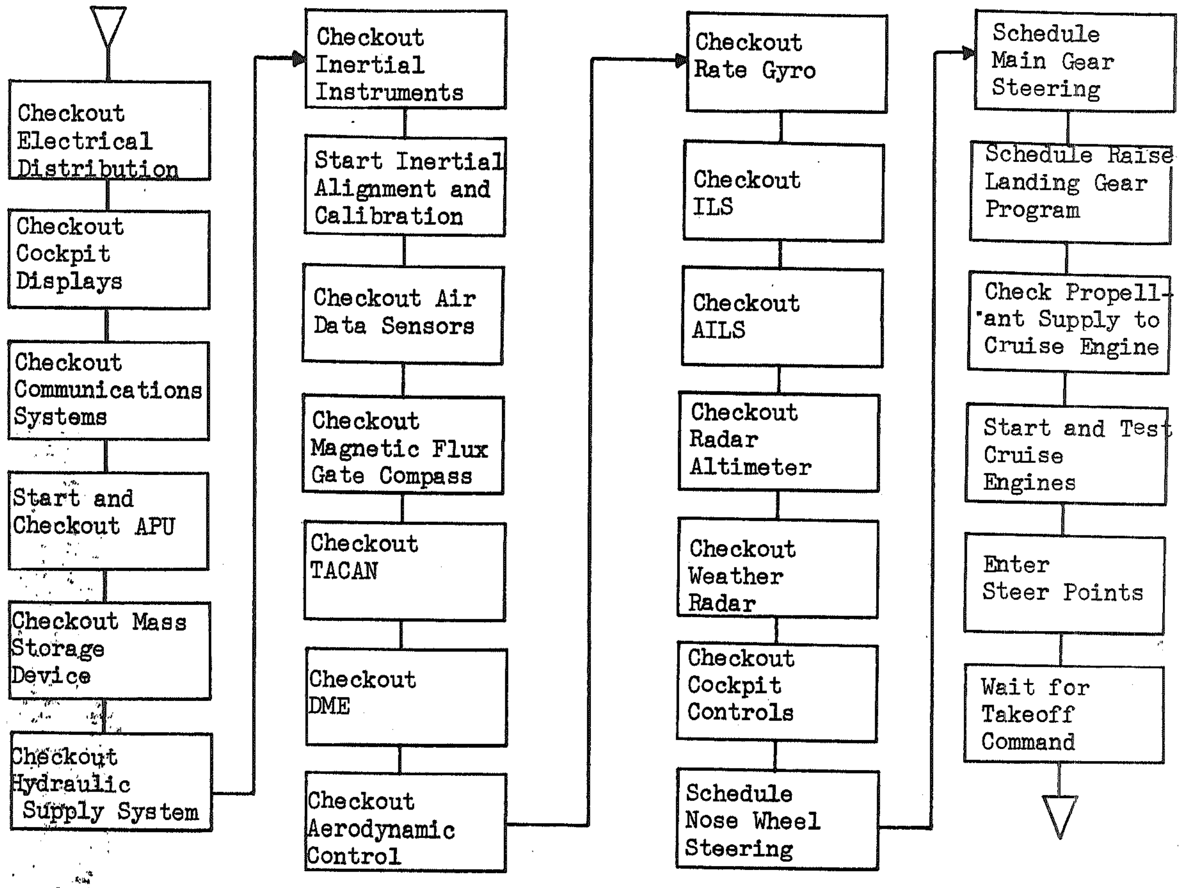
Figure 1-18 Landing Program

has a zero command during the takeoff maneuver. The Raise Landing Gear program is used to control the brakes prior to takeoff. The propellant supply to the cruise engines is then tested and the cruise engines started and tested. The pilot stores the ferry mission steering points in the DMS computer. The vehicle, assuming all tests were completed successfully, is now ready for takeoff. The pilot at this point can taxi the booster. When the pilot is at the end of the runway and prepared to takeoff he issues a command to the DMS which signals the end of preferry and starts the takeoff and climb to cruise activities.

Takeoff and Climb to Cruise Activities

Figure 1-20 shows the Takeoff and Climb to Cruise program. Programs scheduled during preferry which are continued during takeoff and climb to cruise are Main Gear Steering, Nose Wheel Steering, Raise Landing Gear, Monitor Cruise Engines, Monitor Hydraulic Supply and Monitor Electrical Supply. The inertial navigation alignment and calibration program is descheduled and the Inertial Navigation program scheduled. The required cruise sensor and control programs (Aerodynamic Control, Radar Altimeter, Aid Data Inputs, Magnetic Flux Gate Compass, TACAN, DMR, and Weather Radar) are scheduled. The DMS then waits for takeoff to occur at which time it schedules the navigation update program and the climb to cruise guidance.

Figure 1-19 Preferry Program



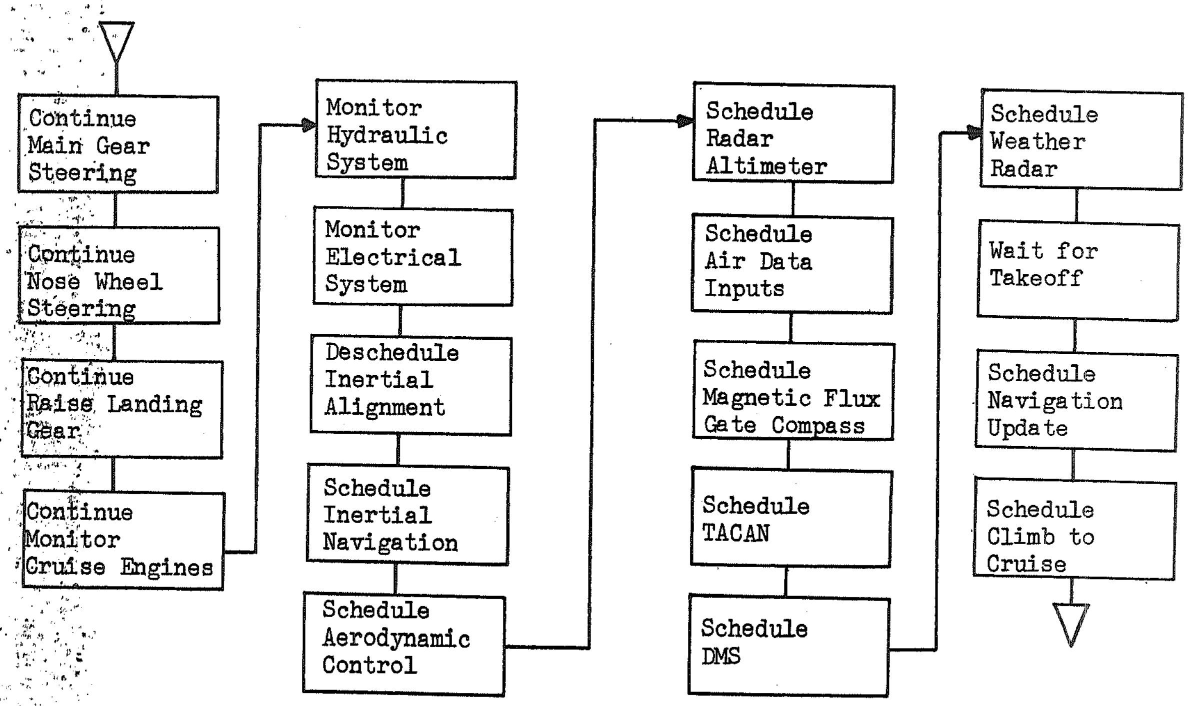


Figure 1-20 Takeoff and Climb to Cruise Program

1.2.5 SYSTEM CHARACTERISTICS AND STUDY ASSUMPTIONS

The following list of characteristics developed by NASA for the space shuttle program are basic assumptions for this study:

- The calendar year 1972 will be used as the materials technology base.
- Initial operational capability (IOC) is the second half of 1977.
- Flexibility will be maintained to incorporate technology advancement and alternate missions.
- Systems will be designed for a mimimum of maintenance with ease of removal and replacement; maximum use of aircraft design practice will be used.
- In systems where redundancy is needed, the space shuttle systems will be developed to provide redundant full mission capability and avoid minimum requirement, minimum performance backup system concepts.
- Multiple redundancy system techniques that minimize or eliminate system transients caused by system component failures will be adopted.
- All subsystems will be designed to fail operational after the failure of the most critical component and to fail safe for crew survival after the second failure. Electronic systems will be designed to fail operational after failure of the two most critical components and to fail safe for crew survival after the third failure.
- The vehicle will be designed for maximum onboard control, using onboard and ground capabilities as appropriate to maximize operational flexibility and minimize ground mission operations consistent with low cost.
- Survivability against hazards from radiation as specified in Joint DOD/NASA Survivability Characteristics document (S) dated June 1969, will be provided for.
- The space shuttle will have minimal assembly and checkout requirements at the launch pad.
- For the design reference mission, the space shuttle will be capable of launch from a standby status within two hours and nominally would be launched at the next acceptable in-plane opportunity. The vehicle should be capable of staying in a launch status until the second in-plane launch opportunity. The system must be capable of accommodating the time between insertion and rendezvous for a worst-case phasing situation.
- Systems sensitivity to weather conditions during assembly, checkout, and launch will be minimized.
- Total space shuttle turnaround time from landing to launch readiness should be less than two weeks. The removal and replacement time will be minimized with on-board checkout and module accessibility.
- All electronic displays and controls should be used, wherever practicable, to replace toggle switches and electromechanical gauges and motors.
- Service lines at the launch pad should be minimal, preferably only for the main propulsion system propellants.
- Maximum use of existing standards for the selection, design, packaging, and integration of hardware should be employed, consistent with program operational requirements.

2.0 DMS CONFIGURATION AND SIZING ANALYSIS

To determine the requirements of individual computers in a DMS multi-computer configuration, the computational speed, memory size and input/output data flow requirements must be determined for each computer in the configuration. Each computer in a multi-computer configuration will perform assigned tasks. This study ascertains individual computer requirements by breaking subsystem tasks into program module requirements, assigning program modules to individual computers and then, through the use of a computer program, determines the individual computer requirements as a function of the mission time line.

2.1 COMPUTATIONAL MODULE REQUIREMENTS ANALYSIS

2.1.1 COMPUTATIONAL REQUIREMENTS

The computational requirements descriptions in Section 5 define program modules (see Figure 2-1) and the computational tasks required of these modules. Computational tasks are given in the form of equations and flow diagrams. From these descriptions detailed quantitative estimates of module requirements are developed. Figure 2-2 shows the requirements for each program. The parameters used in estimating program module requirements are:

a. Total Instructions - This is an estimate of the total number of instructions in the program module. The memory storage requirements for the program module is this number plus the amount of required data storage.

b. Normal Path Instructions - In general, a single pass through a program will not execute every instruction in the program. Also some instructions if included within a loop will be executed several times for each pass through the program. Each program will include a path which under the condition of nominal booster operation and no equipment failures will be executed on each pass through the program. This path is defined as the normal path through the program. The number of slow and fast instructions which must be executed in this path are given. Slow instructions are multiplies, divides and any extremely long shift operations. All remaining instructions are fast instructions. Instructions required for standard function generation such as trigonometric functions, exponentials, logarithms, and square roots are not included in the estimate. These functions can be generated in the computer either by software subroutines or special hardware logic. This possible difference in the mechanization of function generation could have a major impact upon the computer speed requirements and thus the number of functions which must be generated in the program path are delineated separately as described below.

c. Longest Path Instructions - Most programs have a longest execution path which differs from the normal execution path. The longest path through the program is generally executed only in the event of some system failure or off-nominal flight condition. For worst case estimation purposes, the number of fast and slow instructions required in executing this longest path are estimated in the same manner as the normal path instructions were estimated.

ACSP	Aerodynamic Control Surface Program	MPSF	Main Propulsion System (4/sec)
ADCP	Air Data Computer Program	MPSS	Main Propulsion System (64/sec)
AILC	AILS Ground Checkout Program	MPST	Main Propulsion System (2/sec)
AILM	AILS Monitoring Program	NWSP	Nose Wheel Steering Program
AILN	AILS Navigation Program	PMCP	Performance Monitroing and Checkout
BCNT	Beacon Tester	PMFP	Performance Monitoring Flight Prog.
BCNX	Beacon Executive	PRMS	Propulsion Management System
BFCM	Boost Flight Control Monitoring Prog.	RAFL	Radar Altimeter Flight Program
BFCS	Boost Flight Control Gain & Filter Sched.	RALC	Radar Altimeter Ground Checkout
BGFP	Boost Guidance Flight Program	RECT	Recorder Tester
BIGP	Boost Iterative Guidance Program	REGP	Reentry Guidance Program
BOLG	Boost Open Loop Guidance Program	RJCP	Reaction Jet Control Program
CECM	Cruise Engine Control & Montoring Prog.	RJGG	Reaction Jet Gas Generator Control
CEQT	Communication Equipment Tester	RLGP	Raise Landing Gear Program
CFCM	Cruise Flight Control Monitoring Prog,	SDSU	Strapdown Start Up Program
CFGG	Pilot Command Filtering & Gain Genr. Prog.	SD01	Strapdown Instrument Temp Control
CRGP	Cruise Guidance Program	SD02	Navigation Parameters and Updating
COGP	Coast Guidance Program	SD16	Trend Analysis Model Generation
DCEX	Displays and Control Executive	SD32	Strapdown Attitude Reference
DLCI	Down Link Communications Initializer	SD64	Strapdown Instrument Integration
DITR	Display Tester	SSBA	Separation System Boost Abort
DMCT	Data Management Computer Tester	SSBM	Separation System Boost Monitoring
DMCX	Data Management Computer Executive	SSPC	Separation System Prelaunch Checkout
EDMC	Electrical Distribution Monitoring & Cont.	SSTA	Separation System Thrust Termination
EGCP	Electrical Generator Control Program	TCMO	Thrust Chamber Monitoring Program
EGSP	Electrical Generator Start Program	TCMP	Turbine Control and Monitoring
FCGC	Flight Control Ground Checkout	TCNC	TACAN Checkout Program
ILSC	ILS Checkout	TCNM	TACAN Monitoring and Temp. Control
ILSN	ILS Navigation	TCNN	TACAN Navigation
LGFC	Landing Gear Full Checkout	TGSP	Turbine Generator Shutdown
LGPC	Landing Gear Partial Checkout	TMCC	Telemetry Calibration
LGWP	Landing Gear Warning Program	TTEX	Tape Transport Executive
LLGP	Lower Landing Gear Program	TTFC	Trim, Flap and Throttle Control
MBFC	Main Boost Flight Control Program	UPEX	Uplink Executive
MCGC	Magnetic Flux Gate Compass Ground Checkout	VCET	Voice Communications Tester
MCHP	Magnetic Flux Gate Compass Heading Prog.	VCEX	Voice Comminications Executive
MCTM	Magnetic Flux Gate Compass Temp.Cont&Mont.	WRAC	Weather Radar Ground Checkout
MFIM	Miscellaneous Instrument Mont. Program	WRAF	Weather Radar Flight Program
MGSP	Main Gear Steering Program		
MPSE	Main Propulsion System (8/sec)		

Figure 2-1 Program Modules and Acronyms

	ACSP	ADCP	AILC	AILM	AILN	BCNT	BCNX	BFCM	BFCS	BGFP	BIGP	BOLG	CECM
TOTAL INSTRUCT.	209	235	145	36	353	635	155	304	48	148	707	63	208
NORMAL PATH $\pm$&HK	142	143	49	49	106	480	50	327	138	11	678	98	1217
NORMAL PATH $\bar{X}$ & ÷	34	54	1	0	19	4	0	2	16	0	210	4	288
LONG PATH $\pm$ &HK	142	243	49	49	106	680	95	327	138	21	678	114	1217
LONG PATH $\bar{X}$ & ÷	34	90	1	0	19	12	5	2	16	0	210	4	288
SIN & COS	0	0	0	0	6-6	0	0	0	0	0	12-12	0	0
ARCSIN & ARCCOS	0	0	0	0	0	0	0	0	0	0	0	0	0
ARCTAN	0	0	0	0	0	0	0	0	0	0	4-4	0	0
EXPONENTIAL	0	2-8	0	0	0	0	0	0	0	0	2-2	0	0
LOG	0	3-8	0	0	0	0	0	0	0	0	2-2	0	0
SQUARE ROOT	0	4-4	0	0	0	0	0	0	0	0	4-4	0	0
MATRIX MULTIPLY	0	0	0	0	0	0	0	0	0	0	0	0	0
CONSTANT (24-32BIT)	0	0	0	0	7	49	458	0	0	6	0	20	0
CONSTANT (12-16BIT)	31	80	45	21	59	16	16	68	141	23	21	31	216
VARIABLE (24-32BIT)	0	0	0	0	5	12	5	0	0	15	8	0	0
VARIABLE (12-16BIT)	33	32	13	9	25	23	13	2	17	18	60	9	48
ITERATIONS/SEC	32	8	4	1	8	1	1	1	4	2	2	2	4
QUAD REDUNDANCY	YES	YES	NO	YES	YES	NO	YES	YES	YES	YES	YES	YES	YES
SCHEDULING EVENT	19	30	0	50	50	0	10	9	9	9	18	11	40
DESCHEDULING EVENT	60	60	9	60	60	9	50	20	20	20	20	18	61

	CEQT	CFCM	CFGG	COGP	CRGP	DCEX	LLCI	DITR	DMCT	DMCX	EDMC	EGCP	EGSP
TOTAL INSTRUCT.	830	258	254	178	419	2962	225	5100	3200	620	200	56	185
NORMAL PATH $\pm$&HK	770	129	147	85	98	1195	83	3950	3000	150	172	81	155
NORMAL PATH $\bar{X}$&÷	8	4	58	31	12	15	2	25	10	0	0	7	42
LONG PATH $\pm$&HK	1090	129	175	93	116	2470	132	5300	3650	855	292	81	155
LONG PATH $\bar{X}$&÷	32	4	63	31	20	75	3	120	25	5	0	7	42
SIN & COS	0	0	0	0	1-4	0	0	0	0	0	0	0	0
ARCSIN & ARCCOS	0	0	0	0	2-6	0	0	0	0	0	0	0	0
ARCTAN	0	0	0	1-1	0	0	0	0	0	0	0	0	0
EXPONENTIAL	0	0	0	0	0	0	0	0	0	0	0	0	0
LOG	32-32	0	0	0	0	0	0	0	0	0	0	0	0
SQUARE ROOT	0	0	0	1-1	1-1	0	0	0	0	0	0	0	0
MATRIX MULTIPLY	0	0	0	0	0	0	0	0	0	0	0	0	0
CONSTANT (24-32BIT)	75	0	0	2	0	16830	208	4650	200	695	0	0	0
CONSTANT (12-16BIT)	15	61	424	20	34	210	12	60	15	30	96	19	74
VARIABLE (24-32BIT)	45	0	0	5	22	1890	1218	510	55	35	0	0	0
VARIABLE (12-16BIT)	15	4	21	17	8	2105	12	590	35	45	71	24	35
ITERATIONS/SEC	8	1	16	2	2	32	64	32	1	1	4	4	4
QUAD REDUNDANCY	NO	YES	YES	YES	YES	YES	YES	NO	NO	YES	YES	YES	YES
SCHEDULING EVENT	0	19	19	20	40	0	10	0	0	10	2	2	1
DESCHEDULING EVENT	9	60	60	30	50	60	50	9	9	60	61	61	2

Figure 2-2 COMPUTER PROGRAM REQUIREMENTS

	FCGC	ILSC	ILSN	LGFC	LGPC	LGWP	LLGP	MBFC	MCGC	MCHP	MCTM	MFIM	MGSP
TOTAL INSTRUCT.	1934	91	116	450	160	40	170	119	167	163	86	22	70
NORMAL PATH ± &HK	9	26	23	15	14	6	16	496	43	52	52	93	10
NORMAL PATH $\bar{X}$ & ÷	0	0	4	0	0	0	4	106	0	14	0	0	0
LONG PATH ± &HK	216	26	73	110	100	20	80	496	63	59	57	157	20
LONG PATH $\bar{X}$&÷	0	0	4	0	0	0	4	106	0	14	0	0	0
SIN & COS	0	0	0	0	0	0	0	0	0	0	0	0	0
ARCSIN & ARCCOS	0	0	0	0	0	0	0	0	0	0	0	0	0
ARCTAN	0	0	0	0	0	0	0	0	0-1	1-1	0	0	0
EXPONENTIAL	0	0	0	0	0	0	0	0	0	0	0	0	0
LOG	0	0	0	0	0	0	0	0	0	0	0	0	0
SQUARE ROOT	0	0	0	0	0	0	0	0	0	0	0	0	0
MATRIX MULTIPLY	0	0	0	0	0	0	0	0	0	0	0	0	0
CONSTANT (24-32BIT)	0	0	0	0	0	0	0	0	0	0	0	0	0
CONSTANT (12-16BIT)	330	29	43	60	15	5	40	344	35	390	20	65	12
VARIABLE (24-32BIT)	0	0	0	0	0	0	0	0	0	0	0	0	0
VARIABLE (12-16BIT)	14	6	14	15	0	0	30	89	16	18	9	0	8
ITERATIONS/SEC	32	1	1	4	4	1	4	32	1	4	1	1	4
QUAD REDUNDANCY	NO	NO	YES	NO	YES	YES	YES	YES	NO	YES	YES	YES	YES
SCHEDULING EVENT	0	0	50	0	50	40	51	9	0	30	0&30	10	60
DESCHEDULING EVENT	9	9	60	9	51	51	52	20	9	60	9&60	60	61

	MPSE	MPSF	MPSS	MPST	NWSP	PMCP	PMFP	PRMS	RAFL	RALC	RECT	REGP	RJCP
TOTAL INSTRUCT.	120	35	333	698	80	200	82	1161	144	110	1070	433	74
NORMAL PATH ± &HK	98	27	256	353	10	8	52	962	92	45	1625	197	50
NORMAL PATH $\bar{X}$&÷	20	8	65	69	0	0	0	0	0	0	0	81	6
LONG PATH ± &HK	98	27	256	353	25	480	52	1082	96	45	2105	1607	55
LONG PATH $\bar{X}$& ÷	20	8	65	69	0	0	0	0	0	0	0	565	6
SIN & COS	0	0	0	0	0	0	0	0	0	0	0	3-0	0
ARCSIN & ARCOS	0	0	0	0	0	0	0	0	0	0	0	3-0	0
ARCTAN	0	0	0	0	0	0	0	0	0	0	0	2-1	0
EXPONENTIAL	0	0	0	0	0	0	0	0	0	0	0	1-21	0
LOG	0	0	0	0	0	0	0	0	0	0	0	2-0	0
SQUARE ROOT	0	0	0	1-1	0	0	0	0	0	0	0	4-22	0
MATRIX MULTIPLY	0	0	0	0	0	0	0	0	0	0	0	0	0
CONSTANT (24-32BIT)	0	0	0	0	0	0	0	0	0	0	400	0	0
CONSTANT (12-16BIT)	31	20	178	333	10	71	19	1210	45	32	70	40	33
VARIABLE (24-32BIT)	0	0	0	0	0	0	0	0	0	0	320	0	0
VARIABLE (12-16Bit)	13	6	42	51	4	72	12	185	13	11	180	50	9
ITERATIONS/SEC	8	4	64	2	8	8	8	2	1	1	8	2	32
QUAD REDUNDANCY	YES	YES	YES	YES	YES	NO	YES	YES	YES	NO	NO	YES	YES
SCHEDULING EVENT	9	9	9	9	60	0	10	0	40	0	0	30	19
DESCHEDULING EVENT	19	19	19	19	61	9	60	60	60	9	9	40	32

Figure 2-2 COMPUTER PROGRAM REQUIREMENTS (Continued)

	RJGG	RLGP	SDSU	SD01	SD02	SD16	SD32	SD64	SSBA	SSBM	SSPC	SSTA	TCMO
TOTAL INSTRUCT.	167	170	372	30	1095	422	213	1780	85	65	150	60	21
NORMAL PATH ± & HK	212	16	435	105	535	510	73	517	16	24	24	14	164
NORMAL PATH X&÷	36	4	6	0	182	148	18	70	0	0	0	0	0
LONG PATH ± &HK	212	80	585	123	856	510	149	1450	60	75	80	65	164
LONG PATH X&÷	36	4	32	0	324	148	45	94	0	0	0	0	0
SIN & COS	0	0	0-12	0	5-5	2-2	4-4	0	0	0	0	0	0
ARCSIN & ARCCOS	0	0	0	0	0	0	0	0	0	0	0	0	0
ARCTAN	0	0	0	0	4-4	2-2	5-5	0	0	0	0	0	0
EXPONENTIAL	0	0	0	0	0	1-1	0	0	0	0	0	0	0
LOG	0	0	0	0	0	0	0	0	0	0	0	0	0
SQUARE ROOT	0	0	0	0	1-1	4-4	0	0	0	0	0	0	0
MATRIX MULTIPLY	0	0	0-4	0	0	0	0	3-3	0	0	0	0	0
CONSTANT (24-32BIT)	0	0	10	0	4	0	1	0	0	0	0	0	0
CONSTANT (12-16BIT)	225	24	61	28	17	123	0	92	18	10	30	20	19
VARIABLE (24-32BIT)	0	0	94	0	99	3	23	37	0	0	0	0	0
VARIABLE (12-16BIT)	122	10	47	6	7	85	0	63	0	0	45	0	17
ITERATIONS/SEC	2	4	16	1	2	16	32	64	16	16	8	4	8
QUAD REDUNDANCY	YES	YES	NO	YES	YES	YES	YES	YES	YES	YES	NO	YES	YES
SCHEDULING EVENT	19	70	0	0	10	20	10	9	80	10	0	19	19
DESCHEDULING EVENT	22	71	9	60	60	60	60	60	81	19	9	20	32

	TCMP	TCNC	TCNM	TCNN	TGSP	TMCC	TTEX	TTFC	UPEX	VCET	VCEX	WRAC	WRAF
TOTAL INSTRUCT.	117	152	46	247	50	30	460	114	340	195	40	105	89
NORMAL PATH ± &HK	140	13	52	78	41	140	90	61	345	420	75	13	84
NORMAL PATH X&÷	11	0	0	18	0	0	0	8	0	0	0	0	0
LONG PATH ± &HK	148	29	57	261	41	152	280	61	825	510	195	87	88
LONG PATH X&÷	13	0	0	47	0	0	0	8	8	0	0	0	0
SIN & COS	0	0	0	3-3	0	0	0	0	0	0	0	0	0
ARCSIN & ARCCOS	0	0	0	2-1	0	0	0	0	0	0	0	0	0
ARCTAN	0	0	0	2-0	0	0	0	0	0	0	0	0	0
EXPONENTIAL	0	0	0	0	0	0	0	0	0	0	0	0	0
LOG	0	0	0	0	0	0	0	0	8-8	0	0	0	0
SQUARE ROOT	0	0	0	3-2	0	0	0	0	0	0	0	0	0
MATRIX MULTIPLY	0	0	0	0	0	0	0	0	0	0	0	0	0
CONSTANT (24-32BIT)	0	0	0	0	0	8	75	0	297	75	935	0	0
CONSTANT (12-16BIT)	46	54	24	173	9	2	100	24	13	35	7	42	42
VARIABLE (24-32BIT)	0	0	0	0	0	6	4075	0	413	45	20	0	0
VARIABLE (12-16BIT)	38	8	11	37	11	4	150	10	18	15	8	12	11
ITERATIONS/SEC	4	1	1	4	4	64	8	8	8	1	1	1	1
QUAD REDUNDANCY	YES	NO	YES	YES	YES	NO	YES	YES	YES	NO	YES	NO	YES
SCHEDULING EVENT	2	0	0&40	30	61	0	10	19	10	0	10	0	40
DESCHEDULING EVENT	61	9	9&60	60	83	9	40	60	30	50	60	9	60

Figure 2-2 COMPUTER PROGRAM REQUIREMENTS (Continued)

d. Function Generation - The number of standard functions which must be generated in both the normal and longest execution path are estimated per function type; e.g., under the heading sin and cos the entry 4 - 6 means 4 sine or cosine functions must be generated in the normal path and 6 in the longest path.

e. Data Requirements - Data estimates are made separately for constant and variable data. Constant data are those data items which do not change during flight while variable data can change. Some computer designs are mechanized with two types of memory. A read only memory is used for program and constant data storage and read/write memory for variable storage. By separating data into two types, separate estimates of the size of each type of memory needed can be made. In addition, the constant and variable data is further separated into length categories; long or short. In making this categorization a computer having a word length of approximately 16 bits was assumed. Short word length data has accuracy requirements of 16 bits or less while long word length data requires an accuracy of greater than 16 bits. This is primarily navigation data which will require a minimum of 24 bits for accuracy.

f. Iteration Rate - Each program module is programmed so that each execution pass through the program updates the task to be performed. An executive program causes each program to be executed at a fixed iteration rate. The iteration rate is determined by the data rate requirements of the equipment controlled by the program module or by accuracy requirements in those programs performing integrations. The iteration rate for each program is listed.

g. Redundancy - Those program modules which perform critical tasks during flight must be mechanized quadruply redundant in four different computers. Noncritical programs such as ground checkout programs, do not require a redundant mechanization.

h. Scheduling - Most programs operate only over a portion of the total mission profile. A numeric code is listed to designate the mission time when each program is scheduled and descheduled.

0	Beginning of Prelaunch Checkout
1	Start up of the Vehicle Electrical Generators
2	Internal Electrical Power Achieved
9	Near End of Prelaunch Checkout (Go-Inertial)
10	Beginning of Launch
11	Initiation of Pitchover
18	End of Boost Significant Atmosphere
19	Beginning of Thrust Termination
20	Beginning of Coast
30	Beginning of Reentry
32	Entry into Significant Atmosphere
34	End of Survival Phase
40	Beginning of Cruise
50	Beginning of Landing

51 Initiate Lower Landing Gear
52 Landing Gear Down
60 Touchdown
61 End of Taxi
70 Ferry Take Off
71 Landing Gear Up
80 Beginning of Boost Abort
81 End of Boost Abort
82 Pilot Initiation
83 Program Completion

Several of the programs in Figure 2-1 must be given special considerations which are not indicated in the tabularized data. These considerations are:

ILSN - This program is scheduled only if the landing airport does not have AILS facilities.

MPSE, MPSF, MPSS, and MPST - These programs must be duplicated for each main propulsion system rocket engine.

REGP - The longest program path is used until the end of the survival phase (time point 34). The normal path is then used until cruise begins.

SD02 - During boost and coast the normal path contains 16 fast instructions, 5 slow instructions and 1 square root.

SD32 - The long path is normal during reentry, cruise and landing.

SD64 - From time 9 through 10 the program normal path contains 110 fast and 30 slow instructions.

The computer must contain an executive program and mathematical program subroutines in addition to the programs listed in Figure 2-1. The executive program is described in Section 5.7.3 and the subroutine package in Section 5.7.4.

A summary of the computer requirements imposed by the executive are:

A. Number of short instructions executed per second

$$6586 + 1024\,N_{64} + 576N_{32} + 320N_{16} + 176N_8 + 96N_4 + 52N_2 + 29N_1 + 192T_{64} + 96T_{32} + 48T_{16} + 24T_8 + 12T_4 + 6T_2 + 3T_1$$

where T_1, T_2,..., T_{64} are the total number of 1/sec, 2/sec, etc., iteration rate programs stored in the computer and N_1, N_2, ...,N_{64} are the total number of programs in each rate group scheduled at a particular time.

B. Total Number of Instructions

550

C. Total Number of Short Constants

25 + T

Where T is the total number of programs in the computer

D. Total Number of Short Variables

108

The subroutine functions of sine, cosine, arctangent, arcsine, exponential, logarithm and matrix multiply are described in Section 5.74. Total subroutine software requirements could be reduced if the booster computer uses hardware to mechanize some of these functions. The navigation and guidance programs require 32 bit precision while the other functions require only 16 bit precision. A summary of the subroutine package computer requirements is given in Figure 2-3 Matrix Multiply is used only by the navigation programs.

	16 BIT PRECISION							32 BIT PRECISION							
	SIN	TAN	ARC SIN	ARC TAN	EXP	LOG	SQR RT	SIN	TAN	ARC SIN	ARC TAN	EXP	LOG	SQR RT	MTX PLY
TOTAL INSTRUCT.	28	51	11	46	18	37	29	32	59	11	52	24	41	33	102
SHORT CONSTANTS	11	8	1	7	5	7	4	2	0	0	0	0	1	4	0
LONG CONSTANTS	0	0	0	0	0	0	0	13	12	1	10	8	8	0	0
SHORT VARIABLES	3	3	1	6	2	3	3	0	0	0	1	1	1	1	0
LONG VARIABLES	0	0	0	0	0	0	0	3	3	1	5	1	2	2	0
FAST INSTRUCT.	26	37	63	31	14	28	22	28	41	69	34	17	30	25	75
SLOW INSTRUCT.	4	8	13	8	4	6	4	6	12	17	11	7	8	5	27

Figure 2-3 Subroutine Requirements Summary

2.1.2 DATA FLOW REQUIREMENTS

The input/output requirements of the DMS computers are dictated by the data required and generated by each program module. The data flow to and from each program module is either between program modules or between external equipment and the program module. Figure 2-4 is a list of the data required by the program modules. In the first column is an acronym for the data element. In the second column is the iteration rate at which the data must be transferred. In the third column is the record length of the data in binary bits. In the fourth column is the program module or external source of the data. In some cases different program modules will act as the data source during different portions of the mission. Under this case additional data sources are listed on successive lines in the figure. The remaining columns list the destination of each data element. The data element acronyms listed in columns one are identified in Figure 2-5.. The data elements FEAEPS, FEAFPS, FEASPS, FEATPS, MPSETS, MPSFTS, [illegible]E, MPSTTE, MPSTTS, TEAEPS, TEASPS, and TEATPS represent the data flow for a single rocket engine. Each of these items must be repeated 12 times for the accumulated data flow for all 12 rocket engines.

DATA	RT	BIT	SRCE	DESTINATION			
AAGCE	16	192	SD16	SDSU	SD64		
ABEFU	16	576	NAV	SDSU			
ACFPIG	16	96	CFGG	ACSP			
AFMDIS	1	384	SDSU	NAV			
AILLLH	8	80	AILN	SD02			
AILPRC	8	32	AILN	TTFC	DAC		
AILSDO	1	4	AILC	NAV			
			AILM				
AILSOF	2	1	AILN	CRGP			
AILSSC	8	12	AILC	NAV			
			AILN				
AILSSI	8	46	NAV	AILC	AILN	TTFC	
AILSVT	1	34	NAV	AILC	AILM		
AIRDAT	8	76	FCS	ADCP			
AIRDOT	8	80	ADCP	SD02	ACSP	CFGG	TTFC
AIRDTH	1	13	FCS	ADCP			
ALCANG	1	10	VSS	NWSP	MGSP		
ALRMFL	32	16	DAC	DCEX			
ANKEYF	1	224	DAC	DCEX	TTEX	DMCX	
ADODIS	1	48	SDSU	NAV			
APSVOL	4	336	NAV	SDSU	SD64		
ATCIDC	1	50	COM	DCEX	BCNX		
ATCVCS	1	66	COM	DCEX	DCLI	TTEX	BCNX
ATTREF	32	48	SD32	MCHP	TCNN	AILN	BGFP
				COGP	REGP	CRGP	MBFC
				RJCP	ACSP	CFGG	
AXCELI	64	30	NAV	SD64			
BCNDSY	1	62	BCNX	DCEX	DAC		
BELECM	1	32	MBFC	BFCM			
BFPAGN	4	224	BFCS	MBFC			
BOSTIM	2	16	BGFP	BOLG			
BPARAC	2	32	BGFP	MBFC			
BPAYAC	2	32	BIGP	FCS			
CHAVEL	2	32	DAC	BGFP			
CMDMSG	1	47	UPEX	DCEX	DAC		
COMREC	1	16	UPEX	TTEX	DAC		
COMMOD	1	12	DCEX	DAC	UPEX	DCLI	VCEX
				BCNX			
CPRYAC	2	48	COGP	RJCP	ACSP		
CSSTRQ	1	162	COM	UPEX	DCEX		
CTETUP	1	30	DAC	UPEX	DCEX		
DACSWS	1	328	DAC	DCEX	DCLI	TTEX	
DBCDRC	1	3	PMCP	VSS			
			PMFP				
DBMSPM	1	113	DAC	UPEX	DCEX	DCLI	VCEX
DEKEYF	1	253	DAC	DCEX	DMCX		
DELGEN	4	16	TCMP	EGCP			
DIRCOS	32	288	SD64	SD32	SD16	SD02	COGP
				REGP	CRGP		
DMSDSY	1	64	DMCX	DCEX	DCLI	TTEX	DAC
DMSLRU	1	64	DMCX	DCEX	DAC		
ELARCC	1	48	ACSP	CFCM			
ELARDC	32	30	FCGC	FCS			
			BFCM				
			MBFC				
			ACSP				

DATA	RT	BIT	SRCE	DESTINATION			
ELPWRS	1	24	DAC	DCEX	DCLI	TTEX	
EMCLWD	32	32	DMCX	TTEX	DCEX	DAC	
ENGOPD	2	12	FCS	BGFP	BOLG		
ENGPAR	16	48	PRMS	SD16			
ENGTHC	8	66	FCGC	FCS			
			TTFC				
ERFPAV	1	60	FCS	FCGC	BFCM	CFCM	
ERSNAC	32	64	SD32	SD02			
ETPRIP	8	138	FCS	FCGC	TTFC		
FCACIN	32	24	FCS	FCGC	MBFC	CFGG	
FCPOS1	2	42	DAC	DCEX	DCLI	TTEX	
FCPOS2	4	53	DAC	DCEX	DCLI	TTEX	
FCSTMP	2	68	PRMS	PPS			
FDMSTE	4	128	CECM	PPS			
FEAEPS	8	48	PPS	MPSE			
FEAFPS	4	16	PPS	MPSF			
FEASPS	64	60	PPS	MPSS			
FEATPS	2	80	PPS	MPST			
FETDMS	4	336	PPS	CECM			
FGGTRJ	2	66	RJGG	PPS			
FLAPCC	1	16	TTFC	CFCM			
FLAPCM	8	8	FCGC	FCS			
			BFCM				
			TTFC				
FMPTCS	2	462	PPS	PRMS			
FMPTFC	8	16	TCMO	RJCP			
FRJTGG	2	312	PPS	RJGG			
FTCTMP	8	256	PPS	TCMO			
GATEMP	4	84	NAV	SDSU			
GNDCMD	8	22	COM	UPEX	DCEX		
GNVOLT	4	42	EGS	EGSP	TCMP	EGCP	
GRAVIT	2	32	SD16	SD02			
GROTSP	1	54	NAV	SDSU	SD64		
GYROOT	64	90	NAV	SD64			
ILSPRC	1	64	ILSN	NAV			
ILSPTD	1	4	ILSC	NAV			
			ILSN				
ILSVTS	1	51	NAV	ILSC	ILSN		
INEALT	2	16	SD02	BGFP	REGP	MPST	
INEVAP	32	192	SD64	SD32	SD16	SD02	BGFP
				BIGP	COGP	MPST	
JETCOM	32	16	FCGC	FCS			
			RJCP				
JETSTA	32	16	FCS	FCGC	RJCP		
LATLON	2	64	SD02	SD32	MCHP	TCNN	AILN
				REGP			
LGBCOM	4	84	LGFC	VSS			
			LLGP				
			RLGP				
LGBHTS	4	475	VSS	LGFC	LGPC	LLGP	RLGP
LGCOMM	1	39	LGFC	VSS			
			LLGP				
			RLGP				
LGHESW	4	168	VSS	LGFC	LGPC	LLGP	LGWP
				RLGP			

Figure 2-4 DATA FLOW SUMMARY

DATA	RT	BIT	SRCE	DESTINATION				
LGSCOM	8	30	LGFC	VSS				
			NWSP					
			MGSP					
LGSFEE	8	30	VSS	LGFC	LLGP	RLGP	NWSP	
				MGSP				
MAGADG	4	16	MCTM	SD02				
MEMWRD	1	32	DMCX	TTEX	COM			
MFCFDI	4	26	NAV	MCGC	MCHP			
MFCHCM	1	1	MCTM	NAV				
MFCPOD	1	2	MCGC	NAV				
			MCTM					
MFCTDO	4	16	MCGC	NAV				
			MCHP					
MFGCDI	1	49	NAV	MCGC	MCTM			
MFIDIP	1	105	DAC	MFIM				
MGTPFW	8	42	COM	DCEX	DCLI	TTEX		
MGTPSW	1	15	COM	DCEX	TTEX			
MRSETS	8	48	MPSE	MPSS				
MPSFTS	4	32	MPSF	MPSS				
MPSSTE	8	16	MPSS	MPSE				
MPSTTE	2	64	MPST	MPSE				
MPSTTS	2	80	MPST	MPSS				
MSGVFI	8	74	UPEX	DAC	DCLI			
MTPWRS	1	27	COM	DCEX	DCLI	TTEX		
O2H2PP	4	14	EGS	EGSP	TCMP	TGSP		
O2H2SO	1	8	TCMP	EGS				
			TGSP					
			EGSP					
O2SRVC	4	9	EGSP	EGS				
			TCMP					
			TGSP					
OAGQUA	4	33	EGS	EGSP	EGCP			
OCTRVL	4	7	EGSP	EGS	EGCP			
			TCMP					
			TGSP					
OPATIN	4	45	NAV	SDSU				
PBINPT	4	20	VSS	LLGP	RLGP			
PBSDIS	1	48	EDMC	EGS				
PDSINP	4	103	EGS	EDMC				
PFLAG	1	32	SDSU	NAV				
PHSCNG	8	32	DCEX	DAC				
PMVSTS	8	64	VSS	PMCP	PMFP			
PSHAAC	2	24	DAC	CRGP	ACSP			
PSINPT	4	10	VSS	NWSP	MGSP			
PTACPT	4	49	EGS	EGSP	TCMP			
PWSTRG	1	99	COM	DCEX	DCLI	DMCX	TTEX	
RADALT	1	20	RALF	SD02				
RADDIS	1	4	RALC	NAV				
			RALF					
RADOUT	1	82	NAV	RALC	RALF			
RATAPD	1	47	FCGC	FCS				
			BFCM					
RATGYI	32	48	FCS	FCGC	MBFC	RJCP	ACSP	
READRQ	1	16	DCEX	TTEX	COM			

DATA	RT	BIT	SRCE	DESTINATION				
REATTC	2	32	REGP	RJCP	ACSP			
RECBST	1	124	COM	DCEX	DCLI	TTEX	BCNX	
RECDSY	1	27	COM	DCEX	TTEX			
REENTM	2	17	NAV	COGP	REGP			
REGBVC	4	14	EGSP	EGS				
			TCMP					
RJFPIG	16	96	CFGG	RJCP				
RTGYTI	1	88	FCS	FCGC	BFCM	CFCM		
SBDISC	8	17	COM	UPEX	DCLI			
SCLATL	2	128	SD02	SD32	TCNN	AILN	REGP	
				CRGP				
SDITHC	1	96	NAV	SD01				
SEPDIS	2	1	VSS	COGP				
SHAADS	2	13	DAC	CRGP				
SPCMD1	1	60	DAC	UPEX	DCLI	DMCX		
SPCMD2	1	30	DAC	UPEX				
SSBCOM	1	8	SSPC	VSS				
			SSBA					
			SSTA					
SSBSTS	16	42	VSS	SSPC	SSBA	SSBM		
SSCINP	16	93	FCS	FCGC	CFGG			
SSSTAT	1	36	VSS	SSPC	SSTA	SSBM		
SYMCRT	32	64	TTEX	DCEX	DAC			
TCNBCS	4	51	NAV	TCNC	TCNN			
TCNCSC	4	13	TCNN	NAV				
TCNLLH	4	64	TCNN	SD02				
TCNPOR	1	15	TCNM	NAV				
TCNTAV	1	42	NAV	TCNC	TCNM			
TEAEPS	8	20	MPSE	PPS				
TEASPS	64	20	MPSS	PPS				
TEATPS	2	15	MPST	PPS				
THROTL	4	96	TTFC	CECM				
TMSYNC	64	74	COM	DCLI	TTEX			
TMYDSY	1	27	DCLI	UPEX	COM			
TPTNCT	2	288	BGFP	BIGP				
TRBEXT	4	7	EGS	TCMP				
TRBSPD	4	7	EGS	EGSP	TCMP	EGCP	TGSP	
TURBVB	4	6	EGS	TCMP				
TVCCOM	32	40	FCGC	FCS				
			MBFC					
TVCFDB	32	40	FCS	FCGC	MBFC			
TVCSTV	1	126	FCS	FCGC	BFCM			
UPDDSY	4	32	UPEX	DAC	DCEX			
VCFSEL	1	133	VCEX	DCEX	COM			
VCOMMS	1	45	COM	DCEX	VCEX	TTEX		
VFSDRQ	1	89	UPEX	VCEX	COM			
VRECSW	1	11	DAC	UPEX	VCEX	TTEX		
VTPCTL	8	50	DCEX	DAC	DCEX	DCLI	TTEX	
WHORNS	8	7	DAC	DCEX	DCLI	TTEX		
WHAPHT	1	4	WRAC	NAV				
			WRAF					
WRATOT	1	72	NAV	WRAC	WRAF			
WRITRQ	1	16	UPEX	DMCX	TTEX	COM		

Figure 2-4 DATA FLOW SUMMARY (Continued)

AAGCE	Accelerometer and Gyro Acceleration Errors	ENGTHC	Engine Thrust and Trim Motor Commands
ABEFU	Accelerometer Bias Error Functions	ERFPAV	Elevon, Rudder and Flap Position and Voltage
ACFPIG	Aerodynamic Controls Filtered Pilot Inputs and Gains	ERSNAC	Sin and Cos of Earth Rate
AFMDIS	Accelerometer and Gyro Scale Factor and Failure Discretes	ETPRIP	Engine Throttle and Trim System Inputs
AILLLH	AILS Latitude, Longitude, and Altitude	FCACIN	Flight Control Accelerometer Inputs
AILPRC	AILS Pitch and Roll Command	FCPOS1	Flight Controller Position #1
AILSDO	AILS Power On & Test Config. Discretes	FCPOS2	Flight Controller Position #2
AILSOF	AILS Operating Flag	FCSTMP	Propulsion Mgmt System Controls
AILSSC	AILS Slot Center Command	FDMSTE	Cruise Engine Control Inputs
AILSSI	AILS Signal Outputs	FEAEPS	Rocket Engine LOX and Preburner LH2 Valve Position, etc.
AILSVT	AILS Voltages and Temperature	FEAFPS	Rocket Engine Main Pump Inlet Temp.
AIRDAT	Air Data Instrumentation Outputs	FEASPS	Rocket Engine LOX Valves and Turbine Speed
AIRDOT	Air Data Computation Results	FEATPS	Rocket Engine Temperatures
AIRDTH	Temperature and Voltage from Air Data Sensors	FETDMS	Cruise Engine Outputs
ALCANG	Aircraft Landing Crabb Angle	FGGTRJ	Reaction Jet Gas Generator Controls
ALRMFL	Alarm Flag	FLAPCC	Flap Computer Commands
ANKEYF	Alphanumeric Key Board Function Codes	FLAPCM	Flap Command
APODIS	Accelerometer and Gyro Power On Discretes	FMPTCS	Propulsion Management System Monitoring Outputs
APSVOL	Accelerometer and Gyro Power Supply Voltages	FMPTFC	Reaction Jet Failure Indicators
ATCIDC	ATC Identifier Code, Noise Ranging, Radar, and Recovery Beacon Switches	FRJTGG	Reaction Jet Gas Generator Outputs
ATCVCS	ATC Transponder Voltage-Current Selector Status	FTCTMP	Reaction Jet Thrust Chamber Monitoring Outputs
ATTREF	Attitude Reference	GATEMP	Gyro and Accelerometer Temperature
AXCELI	Accelerometer Output	GNDCMD	Guidance and Navigation Data Command
BCNDSY	Beacon Subsystem Status Display	GNVOLT	Generator Voltages and Currents
BELECM	Boost Elevon Commands	GRAVIT	Local Acceleration Due to Gravity
BFPAGN	Boost Filer Pointer and Gains	GROTSP	Gyro Rotor Speed
BOSTIM	Boost Time	GYROOT	Gyro Output
BPARAC	Boost Pitch and Roll Attitude Commands	ILSPRC	ILS Pitch and Roll Command and Steering Errors
BPAYAC	Boost Pitch and Yaw Attitude Commands	ILSPTD	ILS Power On and Test Configuration Discretes
CHAVEL	Characteristic Velocity and Vehicle Mass Estimate	ILSVTS	ILS Voltage Temperature and Signal Outputs
CMDMSG	Command Message for Displays	INEALT	Inertial Altitude
COMREC	Communication Status Recording	INEVAP	Inertial Velocity and Position
COMMOD	Communication Mode Settings	JETCOM	Reaction Jet Commands
CPRYAC	Coast Pitch, Roll, and Yaw Attitude Commands	JETSTA	Reaction Jet Status
CSSTRQ	Communication Switches, Status and Test Request	LATLON	Latitude and Longitude (Geocentric)
CTETUD	Central Timing Equipment Time Up Date	LGVCOM	Landing Gear Brake Commands
DACSWS	Displays and Controls Switches	LGBHTS	Landing Gear Brake Hydraulic Temp.
DBCDRC	Data Bus Controller and Data Recorder Commands	LGCOMM	Landing Gear Commands
DBMSPM	Margin Summary Parameters	LGHESW	Landing Gear Hydraulic Actuator and Status Words
KEKEYF	Data Entry Key Board and External Function Codes	LGSCOM	Landing Gear Steering Commands
DELGEN	Change in Generator Electrical Load	LGSFEE	Landing Gear Steering Feedback
DIRCOS	Direction Cosine Matrix	MAGHDG	Magnetic Heading Reference
DMSDSY	Computer Subsystem Status Display	MEMWRD	Computer Word for TM Memory Dump
DMSLRU	Card Replacement Message Information	MFCFDI	Compass Word for TM Memory Dump
ELARCC	Elevon and Rudder Computed Commands	MFCHCM	Compass Heater Command
ELARDC	Elevon and Rudder Commands	MFCPOD	Compass Power ON Discretes
ELPWRS	Electroluminescent Instrument Power Status	MFCTDO	Compass Torquer Output
EMCLWD	Emergency Check List Word	MFGCDI	Voltage, Temperature and Frequency from Compass
ENGOPD	Main Engine Operating Discretes	MFIDIP	Miscellaneous Instrument Data Input
ENGPAR	Engine Parameters, Thrust & Fuel Expend.	MGTPFW	Magnetic Tape Function and Status Wd.
		MGTPSW	Magnetic Tape Switches and Temp.
		MPSETS	Data from MPSE to MPST
		MPSFTS	Data from MPSF to MPSS

Figure 2-5 Data Elements and Acronyms

MPSSTE	Data from MPSS to MPSE
MPSTTE	Data from MPST to MPSF
MPSTTS	Data from MPST to MPSS
MSGVFI	Mode and Data Status Verification and Error Message
MTPWRS	Magnetic Tape Power Status
O2H2PP	Oxygen and Hydrogen Preheater Pressure
O2H2sO	Oxygen and Hydrogen Shut-Off Valve
O2SRVC	Oxygen Servo and Heater Valve Commands
OAGQUA	Oil Pump Quantities and Generator Speed
OCTRLV	Oil Control Valve and Generator Field Current Control
OPATIN	Optical Attitude Inputs
PBINPT	Pilot Brake Inputs
PBSDIS	Power Bus System Discretes
PDSINP	Power Distribution System Inputs
PFLAG	P Matrix Selection Flag
PHSCNG	Phase Change Word
PMVSTS	Performance Monitoring Vibration and Temperature Sensors
PSHAAC	Pilot Selected Heading and Altitude Command
PSINPT	Pilot Wheel Steering Inputs
PTACPT	Preheater Temperature and Combuster Temperature and Pressure
PWSTRG	Power Status and Functional Registers for Computer Units
RADALT	Radar Altitude and Signal Present
RADDIS	Radar Power On, Heater, and Test Discretes
RADOUT	Radar Outputs
RATAPD	Rate Gyro, Accelerometer, TVC and Aero Power On Discretes
RATGYI	Roto Gyro Inputs
READRQ	Request for Magnetic Tape Read Operation
REATTC	Reentry Attitude Commands
RECBST	Recovery Beacon, C-Band Radar and PRN Ranging, Monitor
RECDSY	Recorder Status Display
REENTM	Reentry Temperature Sensor Output
REGBVC	Regenerator Bypass and Hydrogen Servo Valve Command
RJFPIG	Reaction Jet Filtered Pilot Input and Gains
RTGYTI	Rate Gyro and Accelerometer Test Status
SBDISC	S-Band Functional Discretes
SCLATL	Sin and Cos of Geocentric Latitude and Longitude
SDITHC	Strapdown Instrument Temperature and Heater Commands
SEPDIS	Separation Discretes
SHAADS	Selected Heading, Altitude and Cruise Steering Discretes
SPCMD1	Stored Program Command for Memory Dump
SPCMD2	Stored Program Command for Event Changes
SSBCOM	Separation System Bolt Command
SSBSTS	Separation System Bolt Status
SSCINP	Side Stick and Rudder Pedal Inputs
SSSTAT	Separation System Status
SYMCRT	Symbol Characters for CRT and Checklist
TCNBCS	Tacan Bearing, Channel, Range, etc.
TCNCSC	Tacan Center of Slot Command
TCNLLH	TACAN Navigation Output
TCNPOR	Tacan Power On and Test Command Discretes
TCNTAV	Tacan Temperature and Voltage Inputs
TEAEPS	Main Chamber LOX and Preburner LH_2 Valve Commands
TEASPS	Preburner and Low Speed Inducer LOX Valve Commands
TEATPS	Rocket Engine Solenoid Valve Commands
THROTL	Cruise Engine Throttle Commands
TMSYNC	Telemetry Synchronization, Form and Data Identification
TMYDSY	Telemetry Status Display
TPTNCT	Target Plane to Navigation Coordinate Transformation
TRBEXT	Turbine Exhaust Temperature
TRBSPD	Turbine Speed
TURBVB	Turbine RMS Vibration
TYCCOM	Thrust Vector Control Commands
TYCFDB	Thrust Vector Control Feedback
TVCSTV	Thrust Vector Control Status and Voltage
UPDDSY	Uplink Data Display Information
VCFSEL	Voice Communications Frequency Selector and Volume Control
VCOMMS	Voice Communication Switches
VFSDRQ	Frequency Table Look Up Request and Voice Status Display
VRECSW	Voice Recorder Switches
VTPCTL	Vertical Tape and Film Slide Control
WHORNS	Warning Horn Switches
WRAPHT	Weather Radar Power On, Heater, and Test Discretes
WRATOT	Weather Radar Monitoring Signal Outputs
WRITRQ	Request for Magnetic Tape Write Operation

Figure 2-5 Data Elements and Acronyms (Continued)

2.2 COMPUTER TASK ASSIGNMENT BY SUBSYSTEM FUNCTION

This section summarizes computational requirements, execution times and memory storage estimates for the program modules assigned to the booster subsystems. The impact of this data on a centralized multiprocessor, distributed linked unit, and a centralized linked unit system are investigated in section 2.3.

2.2.1 SUBSYSTEM COMPUTATIONAL REQUIREMENTS

The computational and data flow requirements summaries in Section 1.2 establish the DMS requirements for the space shuttle booster. The DMS will be composed of several computers, each performing a segment of the tasks listed in the summary. The apportionment of tasks to individual computers can be made in many different ways. For the purposes of this study, computer assignment will generally be based upon subsystem function. Dividing the tasks in this manner should minimize intercomputer data flow and best match present space shuttle plans to mechanize some of the subsystem functions (such as propulsion control and/or strapdown navigation) external to the DMS. The program modules are grouped into the following subsystems with assigned acronyms:

- Communications Systems (COM)
- Displays and Controls Systems (DAC)
- Electrical Generation System (EGS)
- Flight Control System (FCS)
- Navigation and Guidance System (NAV)
- Propulsion System (PPS)
- Vehicle Structures Systems (VSS)

The program modules assigned to these systems are listed in Figure 2-6

Figure 2-7 is a summary of the computational requirements for each subsystem. There are two tables given for each subsystem function. One table shows the accumulated requirements for the <u>normal</u> computational path and the other the <u>longest</u> computational path. The <u>longest</u> path data represents a worst case condition executed with multiple subsystem failures. The first column in each table designates an initiating event for the requirements defined by each row. The second column lists the number of add, subtract and housekeeping (i.e., <u>short</u> execution time) instructions which must be executed every second to perform the subsystem function during the indicated mission time period. The count given in this column does not include the generation of transcendental, square root, and matrix multiply functions using software subroutines. This generation must be added to these values. The count does include a portion of the executive function, however. The portion of the executive function included is the portion that is dependent upon the number of programs being executed and mechanized in a single processor. The third column indicates the number of multiply and divide instructions (i.e., <u>long</u> execution time instructions) which must be executed every second to perform the subsystem functions during the indicated mission time period. Again, the estimates do not include the software generation of the previously mentioned math functions. The requirements for the math functions (using software subroutines) is given in successive columns whenever a requirement exists. All instruction counts are given in units of thousands of instructions per second (KIPS).

Communication System

BCNT	Beacon Tester
BCNX	Beacon Executive
CEQT	Communication Equipment Tester
DCLI	Down Link Communications Initializer
RECT	Recorder Tester
TMCC	Telemetry Calibration
TTEX	Tape Transport Executive
UPEX	Uplink Executive
VCET	Voice Communication Tester
VCEX	Voice Communication Executive

Displays and Control System

DCEX	Display and Controls Executive
DITR	Display Tester
DMCT	Data Management Computer Tester
DMCX	Data Management Computer Executive
MFIM	Miscellaneous Instrument Monitoring

Electrical Generation System

EDMC	Electrical Distrib. Mont. & Control
EGCP	Electrical Generator Control Program
EGSP	Electrical Generator Start Program
TCMP	Turbine Control and Monitoring
TGSP	Turbine Generators Shutdown Program

Flight Control System

ACSP	Aerodynamic Control Surface Program
ADCP	Air Data Computer Program
BFCM	Boost Flight Control Monitoring
BFCS	Boost Flight Cont. Gain & Filter Sched.
CFCM	Cruise Flight Control Monitoring
CFGG	Pilot Command Filtering & Gain Gen.
FCGC	Flight Control Ground Checkout
MBFC	Main Boost Flight Control
RJCP	Reaction Jet Control
TTFC	Trim, Flap and Throttle Control

Propulsion System

CECM	Cruise Engine Control and Monitoring
MPSE	Main Propulsion System (8/sec)
MPSF	Main Propulsion System (4/sec)
MPSS	Main Propulsion System (64/sec)
MPST	Main Propulsion System (2/sec)
PRMS	Propulsion Management System
RJGG	Reaction Jet Gas Generator
TCMO	Thrust Chamber Monitoring

Navigation and Guidance System

AILC	AILS Ground Checkout
AILM	AILS Monitoring
AILN	AILS Navigation
BGFP	Boost Guidance Flight Program
BIGP	Boost Iterative Guidance
BOLG	Boost Open Loop Guidance
COGP	Coast Guidance
CRGP	Cruise Guidance
ILSC	ILS Checkout
ILSN	ILS Navigation
MCGC	Mag. Flux Gate Compass Ground Checkout
MCHP	Mag. Flux Gate Compass Heading Prog.
MCTM	Mag. Flux Gate Compass Temp Cont.&Mon.
RAFL	Radar Altimeter Flight Program
RALC	Radar Altimeter Ground Checkout
REGP	Reentry Guidance
SDSU	Strapdown Start Up Program
SD01	Strapdown Instrument Temp.Control
SD02	Navigation Parameters and Updating
SD16	Trend Analysis Model Generation
SD32	Strapdown Attitude Reference
SD64	Strapdown Instrument Integration
TCNC	TACAN Checkout
TCNM	TACAN Monitoring
TCNN	TACAN Navigation
WRAC	Weather Radar Ground Checkout
WRAF	Weather Radar Flight Program

Vehicle Structures System

LGFC	Landing Gear Full Checkout
LGPC	Landing Gear Partial Checkout
LGWP	Landing Gear Warning Program
LLGP	Lower Landing Gear
MGSP	Main Gear Steering
NWSP	Nose Wheel Steering
PMCP	Performance Monitoring and Checkout
PMFP	Performance Monitoring Flight Program
RLGP	Raise Landing Gear
SSBA	Separation System Boost Abort
SSBM	Separation System Boost Monitoring
SSPC	Separation System Thrust Prelaunch Ck.
SSTA	Separation System Thrust Termination

Figure 2-6 Program Module Assignments to Booster Subsystems

The memory storage requirements for the subsystem are also listed. Two storage requirements are listed for each subsystem. The first number represents fixed storage words including a single word for each short (16 bit or less) variable and two words for each long (17-32 bits) variable. The fixed storage requirements can be mechanized using read only memory whereas the variable storage requirements need read-write memory.

Propulsion system sizing estimates assume that all 12 main rocket engine control systems will be mechanized in the same computer. Thus the basic engine control programs need to be stored only once rather than once per engine. The storage requirements include only a single copy of the main engine control programs; executive requirements include execution of the control programs twelve times.

The data flow requirements for each subsystem are tabulated in Figure 2-8 The first two columns indicate the source and destination of the data flow path. An E or I have been added to the three letter subsystem acronyms to designate between external subsystem equipment and internal subsystem software. All data flow paths have a single source but may have one or two destinations. Each data flow path has data transfer rate requirements which vary as a function of mission time. The third and fourth columns list the required data transfer rate in bits per second and the mission time line period during which the requirement exists.

2.3 CONFIGURATION SIZING ANALYSIS

The computational and data flow requirements summarized in Figures 2-7 and 2-8 are based upon raw sizing estimates of the program and interface descriptions given in volume II. There are many reasons for increasing the computer estimates of memory size and computational speed above the raw sizing estimates. Previous avionics system experience shows that the typical 15 to 20% increase above raw sizing estimates is grossly inadequate. Technical management is well aware of the lead time required for the generation of production hardware. Initial management pressures on subsystem groups is to commit hardware designs to production. This results in giving software requirements definition secondary priority. When time allows a closer scrutiny of software requirements, original sizing estimates are in general inadequate.

Many instances occur where production hardware does not perform according to original design specifications. Inadequate hardware designs can often be corrected by "software fixes"; a much less costly procedure than the reworking of completed production units if sufficient computing capability is available.

Software development is often unrecognized as a long lead time item. This results in a tight software development schedule which in combination with a minimal capacity computer system leads to excessive software costs. Minimal capacity demands the use of tricky programming shortcuts which makes program generation modification and checkout a costly chore. On the other hand, when computer capacity is not a constraint upon software development, cost and time saving aids such as higher order languages may be employed. Rather than a 15 to 20% excess capacity over raw sizing estimates a 50 to 100% excess is recommended. Specifying a computer which initially appears to have $1\frac{1}{2}$ to 2 times the required computing capacity would seem to be an

Communications System - Memory 9436 Fixed Plus 12756 Variable Words

EVENT	+&HK	X&+	LOG	
			+&HK	X&+
Prelaunch	30.7	.1	7.2	1.5
Go Inertial	.9	.0	.0	.0
Launch	11.3	.1	1.8	.4
Reentry	8.4	.1	.0	.0
Cruise	7.5	.1	.0	.0
Landing	.6	.0	.0	.0
Touchdown	.0	.0	.0	.0

Normal Path

EVENT	+&HK	X&+	LOG	
			+&HK	X&+
Prelaunch	38.4	.3	7.2	1.5
Go Inertial	1.0	.0	.0	.0
Launch	20.0	.3	1.8	.4
Reentry	13.3	.2	.0	.0
Cruise	10.8	.2	.0	.0
Landing	.7	.0	.0	.0
Touchdown	.0	.0	.0	.0

Long Path

Displays and Controls System - Memory 57039 Fixed Plus 7755 Variable Words

EVENT	+&HK	X&+
Prelaunch	169.0	1.3
Go Inertial	39.0	.5
Launch	39.0	.5
Touchdown	.0	.0

Normal Path

EVENT	+&HK	X&+
Prelaunch	253.7	6.3
Go inertial	79.8	2.4
Launch	80.9	2.4
Touchdown	.0	

Long Path

Electrical Generation System - Memory 857 Fixed Plus 179 Variable Words

EVENT	+&HK	X&+
Start Generators	.8	.2
Internal Power	1.9	.1
End Taxi	.3	.0
Program	.0	.0

Normal Path

EVENT	+&HK	X&+
Start Generators	.8	.2
Internal Power	2.4	.1
End Taxi	.3	.0
Program Complete	.0	.0

Long Path

Flight Control System - Memory 5095 Fixed Plus 231 Variable Words

EVENT	+&HK	X&+	LOG		EXP		SQ RT	
			+&HK	X&+	+&HK	X&+	+&HK	X&+
Prelaunch	1.4	.0	.0	.0	.0	.0	.0	.0
Go Inertial	18.0	3.5	.0	.0	.0	.0	.0	.0
Thrust Terminate	28.7	5.7	.0	.0	.0	.0	.0	.0
Coast	11.3	2.3	.0	.0	.0	.0	.0	.0
Reentry	12.6	2.7	.7	.1	.2	.1	.7	.1
Enter Atmosphere	10.4	2.5	.7	.1	.2	.1	.7	.1
Touchdown	.0	.0	.0	.0	.0	.0	.0	.0

Normal Path

EVENT	+&HK	X&+	LOG		EXP		SQ RT	
			+&HK	X&+	+&HK	X&+	+&HK	X&+
Prelaunch	1.4	.0	.0	.0	.0	.0	.0	.0
Go Inertial	18.0	3.5	.0	.0	.0	.0	.0	.0
Thrust Terminate	29.4	5.8	.0	.0	.0	.0	.0	.0
Coast	11.3	2.4	.0	.0	.0	.0	.0	.0
Reentry	12.6	3.1	1.8	.4	.9	.3	.7	.1
Enter Atmosphere	10.4	2.9	1.8	.4	.9	.3	.7	.1
Touchdown	.0	.0	.0	.0	.0	.0	.0	.0

Long Path

Figure 2-7 Computer Requirements by Subystem

Navigation and Guidance System - Memory 9562 Fixed Plus 1205 Variable Words

EVENT			SIN		ARC SIN		ARCTAN		EXP		LOG		SQ RT		MTRX MPY	
	±&HK	X&÷	±&HK	X&÷	±&HK	X&÷	±&HK	X&÷	±&HK	X&÷	±&HK	X&÷	±&HK	X&÷	±&HK	X&÷
Prelaunch	8.7	.1	.0	.0	.0	.0	.0	.0	.0	.0	.0	.0	.0	.0	.0	.0
Go Inertial	8.8	1.9	.0	.0	.0	.0	.0	.0	.0	.0	.0	.0	.0	.0	.0	.0
Launch	37.8	5.1	3.6	.8	.0	.0	5.4	1.8	.0	.0	.0	.0	.0	.0	14.4	5.2
Pitchover	38.0	5.1	3.6	.8	.0	.0	5.4	1.8	.0	.0	.0	.0	.0	.0	14.4	5.2
End Atmosphere	39.2	5.5	4.3	.9	.0	.0	5.7	1.8	.1	.0	.1	.0	.2	.0	14.4	5.2
Coast	46.5	7.5	4.5	1.0	.0	.0	6.6	2.1	.3	.1	.0	.0	1.6	.3	14.4	5.2
Reentry	53.8	9.9	5.1	1.1	.8	.1	7.3	2.4	1.0	.4	.0	.0	3.0	.6	14.4	5.2
End Survival	51.0	8.9	5.3	1.1	1.0	.2	7.3	2.4	.3	.1	.1	.0	2.2	.4	14.4	5.2
Cruise	51.1	8.8	5.2	1.1	.8	.2	7.2	2.3	.3	.1	.0	.0	2.0	.4	14.4	5.2
Landing	51.9	8.9	6.4	1.4	.6	.1	7.2	2.3	.3	.1	.0	.0	2.0	.4	14.4	5.2
Touchdown	.0	.0	.0	.0	.0	.0	.0	.0	.0	.0	.0	.0	.0	.0	.0	.0

Normal Path

EVENT			SIN		ARC SIN		ARCTAN		EXP		LOG		SQ RT		MTRX MPY	
	±&HK	X&÷	±&HK	X&÷	±&HK	X&÷	±&HK	X&÷	±&HK	X&÷	±&HK	X&÷	±&HK	X&÷	±&HK	X&÷
Prelaunch	11.2	.5	5.4	1.2	.0	.0	.0	.0	.0	.0	.0	.0	.0	.0	4.8	1.7
Go Inertial	8.8	1.9	.0	.0	.0	.0	.0	.0	.0	.0	.0	.0	.0	.0	.0	.0
Launch	97.6	6.6	3.6	.8	.0	.0	5.4	1.8	.0	.0	.0	.0	.0	.0	14.4	5.2
Pitchover	97.9	6.6	3.6	.8	.0	.0	5.4	1.8	.0	.0	.0	.0	.0	.0	14.4	5.2
End Atmosphere	99.0	7.0	4.3	.9	.0	.0	5.7	1.8	.1	.0	.1	.0	.2	.0	14.4	5.2
Coast	106.2	9.0	4.5	.9	.0	.0	6.6	2.1	.3	.1	.0	.0	1.6	.3	14.4	5.2
Reentry	114.9	11.8	5.1	1.0	.3	.1	7.0	2.3	1.0	.4	.0	.0	3.0	.6	14.4	5.2
End Survival	112.1	10.9	5.3	1.1	.7	.2	7.1	2.3	.3	.1	.1	.0	2.2	.4	14.4	5.2
Cruise	112.2	10.8	5.3	1.1	1.1	.3	6.9	2.2	.3	.1	.0	.0	1.8	.4	14.4	5.2
Landing	113.1	10.9	6.4	1.3	.3	.1	6.9	2.2	.3	.1	.0	.0	1.8	.4	14.4	5.2
Touchdown	.0	.00	.0	.0	.0	.0	.0	.0	.0	.0	.0	.0	.0	.0	.0	.0

Long Path

Propulsion System - Memory 4983 Fixed Plus 1704 Variable Words

EVENT	±&HK	X&÷	SQ RT ±&HK	SQ RT X&÷
Prelaunch	2.3	.0	.0	.0
Go Inertial	219.4	53.9	.5	.1
Thrust Terminate	4.3	.1	.0	.0
Enter Atmosphere	2.3	.0	.0	.0
Cruise	7.2	1.2	.0	.0
End Taxi	.0	.0	.0	.0

Normal Path

EVENT	±&HK	X&÷	SQ RT ±&HK	SQ RT X&÷
Prelaunch	2.5	.0	.0	.0
Go Inertial	219.6	53.9	.5	.1
Thrust Terminate	4.5	.1	.0	.0
Enter Atmosphere	2.5	.0	.0	.0
Cruise	7.5	1.2	.0	.0
End Taxi	.0	.0	.0	.0

Long Path

Vehicle Structures System - Memory 2129 Fixed Plus 212 Variable Words

EVENT	±&HK	X&÷
Prelaunch	1.0	.0
Launch	1.6	.0
Thrust Terminate	1.0	.0
Coast	.9	.0
Cruise	.9	.0
Landing	1.0	.0
Lower Gear	1.0	.0
Gear Down	.9	.0
Touchdown	.7	.0
End Taxi	.0	.0

Normal Path

EVENT	±&HK	X&÷
Prelaunch	5.6	.0
Launch	2.4	.0
Thrust Terminate	1.2	.0
Coast	.9	.0
Cruise	.9	.0
Landing	1.4	.0
Lower Gear	1.3	.0
Gear Down	.9	.0
Touchdown	.8	.0
End Taxi	.0	.0

Long Path

Figure 2-7 Computer Requirements by Subsystem (Continued)

SRCE	DEST	BPS	TIME
COME	COMI	4883	10-50
		11	50-60
COME	COMI	1127	10-30
	DACI	789	30-40
		747	40-50
		45	50-60
COME	DACI	1127	0-10
		338	30-40
		380	40-50
		1082	50-60
COMT	COME	116	10-30
		27	30-50
COMI	COME	149	10-30
	DACI	133	30-60
COMI	DACE	608	10-30
COMI	DACE	2285	10-30
	DACI	2110	30-40
		62	40-50
DACE	COMI	30	10-30
DACE	COMI	1131	10-30
	DACI	1101	30-40
		877	40-50
		113	50-60
DACE	DACI	1836	0-9
		1900	9-10
		934	10-20
		870	20-30
		900	30-40
		1124	40-50
		1888	50-60
DACE	FCSI	48	19-40
		48	50-60
DACE	FCSI	48	40-50
	NAVI		
DACE	NAVI	26	40-50
DACI	COME	16	0-10
		48	40-60
DACI	COME	48	10-40
	COMI		
DACI	COMI	1500	10-40
	DACE	475	40-50
		12	50-60

SRCE	DEST	BPS	TIME
DACI	DACE	668	0-10
		320	10-40
		1344	40-50
		1808	50-60
EGSE	EGSI	580	1-2
		1044	2-61
		84	61-83
EGSI	EGSE	128	1-2
		176	2-61
		72	61-83
FCSE	FSCI	6962	0-9
		3858	9-19
		6962	19-20
		5556	20-30
		6177	30-32
		5667	32-60
FCSE	NAVI	24	9-20
FCSI	FCSE	3391	0-9
		2278	9-19
		3391	19-20
		2064	20-32
		1552	32-60
FCSI	NAVI	640	30-60
FCSI	PPSI	384	40-60
NAVE	FCSI	368	19-60
NAVE	NAVI	12232	0-9
		9174	9-20
		9208	20-30
		9565	30-40
		9727	40-50
		10180	50-60
NAVI	DACE	256	50-60
	FCSI		
NAVI	FCSE	64	18-20
NAVI	FCSI	64	9-10
		1600	10-20
		1632	20-30
		1600	30-40
		1536	40-60
NAVI	NAVE	657	0-9
		118	30-40
		141	40-50
		309	50-60

SRCE	DEST	BPS	TIME
NAVI	PPSI	6144	9-10
		6176	10-19
PPSE	PPSI	924	0-9
		54300	9-19
		3596	19-32
		924	32-40
		2268	40-60
		1344	60-61
PPSI	FCSI	128	19-32
PPSI	NAVI	768	20-60
PPSI	PPSE	136	0-9
		15776	9-19
		268	19-40
		648	40-60
		512	60-61
VSSE	VSSI	4032	0-9
		1220	10-19
		548	19-20
		514	20-30
		512	30-40
		1184	40-50
		3324	51-52
		3164	51-52
		512	52-60
		290	60-61
		2892	70-71
		672	80-81
VSSI	VSSE	626	0-9
		3	10-19
		11	19-20
		3	20-51
		378	51-52
		3	52-60
		240	60-61
		375	70-71
		8	80-81

Figure 2-8 Major Subsystem Data Flow Summary

unwarranted luxury on a program which is striving to reduce orbital payload costs. Closer analysis shows that the recommended capacity will minimize the combined costs of hardware plus software development.

Raw estimation values will be used in the following development of the size requirements of individual computers which form a part of various DMS configurations. The addition of excess capacity estimates will be made when relating sizing requirements to hardware implications.

2.3.1 CENTRALIZED MULTIPROCESSOR SYSTEM

A centralized multiprocessor system mechanizes all computational requirements in a single computer having multiprocessor capability. Figure 2-9 is the accumulated computational requirements of all subsystem functions. In analyzing this data, effort will be concentrated on the mission profile from just prior to launch to touchdown. Computational and data flow requirements occurring prior to launch can be time distributed so that they do not influence maximum computational speed requirements. Assuming the mathematical functions are mechanized using hardware algorithms, the maximum computational speed requirements occur from the end of significant atmosphere during boost until boost thrust termination.

The computer input/output requirements are given in Figure 2-10 The maximum bit rate input/output requirements occur during the same time period as maximum computational speed requirements with a total data transfer rate of 101,057 bits per second. This is a raw estimate of the data flow only and does not include addressing bits which would have to accompany data on a multiplexed data bus line. Numerous schemes are available to move data stored in memory to the multiplexed data bus line or from the data bus to memory. All methods being considered for the space shuttle application access memory in a cycle steal mode in order to minimize any reduction in processing capacity caused by input/output operations. The input/output transfer efficiency must be known in order to assess processing degradation due to input/output operations. Transfer efficiency is the ratio between the number of data bits transferred to the total number of memory bits accessed. A data bus system requires that each data field be accompanied by a unique coding bit field to distinguish it from all other data fields. The coding bit fields must be stored in a memory. If the memory used is the DMS memory then each data transfer requires access to both the data bits and coding bits.

The maximum total data rate requirements according to Figure 2-10 is 101,057 bits per second. If this number is doubled for contingencies and it is assumed that each data field is 16 bits long then there are 12,632 unique data fields which must be transferred. This requires a coding bit field of 14 bits. The transfer efficiency equals 16/(14+16) or 53%. To transfer 101,057 data bits per second requires 11842 memory cycle steals if 16 bits are obtained with each cycle steal. A cycle steal will interrupt the computer processor only if the request for a cycle steal occurs simultaneously with a processor read or write request on the same memory module. The interference between processor and input/output memory requests can be reduced to an insignificant value by carefully assigning program and data to memory modules. Even if, in the above case of 11842 cycle steals, each input/output cycle

Total Storage Locations Required are 89855 Fixed and 24958 Variable Words

EVENT	+&HK	X&+	SIN +&HK	SIN X&+	ARCSIN +&HK	ARCSIN X&+	ARCTAN +&HK	ARCTAN X&+	EXP +&HK	EXP X&+	LOG +&HK	LOG X&+	SQRT +&HK	SQRT X&+	MTRX MPY +&HK	MTRX MPY X&+
Prelaunch	219.6	1.4	.0	.0	.0	.0	.0	.0	.0	.0	7.2	1.5	.0	.0	.0	.0
Start Generators	220.4	1.6	.0	.0	.0	.0	.0	.0	.0	.0	7.2	1.5	.0	.0	.0	.0
Internal Power	221.5	1.5	.0	.0	.0	.0	.0	.0	.0	.0	7.2	1.5	.0	.0	.0	.0
Go Inertial	294.6	59.8	.0	.0	.0	.0	.0	.0	.0	.0	.0	.0	.5	.1	.0	.0
Launch	335.9	63.1	3.6	.8	.0	.0	5.5	1.8	.0	.0	1.8	.4	.5	.1	14.4	5.2
Pitch Over	336.1	63.1	3.6	.8	.0	.0	5.5	1.8	.0	.0	1.8	.4	.5	.1	14.4	5.2
End Atmosphere	337.3	63.5	4.3	.9	.0	.0	5.7	1.8	.1	.0	1.9	.4	.7	.1	14.4	5.2
Thrust Terminate	132.4	12.0	4.3	.9	.0	.0	5.7	1.8	.1	.0	1.9	.4	.2	.0	14.4	5.2
Coast	122.0	10.5	4.5	1.0	.0	.0	6.6	2.1	.3	.1	1.8	.4	1.7	.3	14.4	5.2
Reentry	127.2	13.3	5.1	1.1	.5	.1	7.3	2.4	1.2	.5	.7	.1	3.8	.7	14.4	5.2
Enter Atmosphere	123.5	13.1	5.1	1.1	.5	.1	7.3	2.4	1.2	.5	.7	.1	3.8	.7	14.4	5.2
End Survival	120.7	12.1	5.3	1.1	1.0	.2	7.3	2.4	.6	.2	.8	.2	2.9	.6	14.4	5.2
Cruise	124.9	13.2	5.2	1.1	.8	.2	7.2	2.3	.5	.2	.7	.1	2.7	.5	14.4	5.2
Landing	119.0	13.2	6.4	1.4	.6	.1	7.2	2.3	.5	.2	.7	.1	2.7	.5	14.4	5.2
Lower Gear	119.0	13.2	6.4	1.4	.6	.1	7.2	2.3	.5	.2	.7	.1	2.7	.5	14.4	5.2
Gear Down	118.9	13.2	6.4	1.4	.6	.1	7.2	2.3	.5	.2	.7	.1	2.7	.5	14.4	5.2
Touchdown	9.2	.1	.0	.0	.0	.1	.0	.0	.0	.0	.0	.0	.0	.0	.0	.0
End Taxi	.0	.0	.0	.0	.0	.0	.0	.0	.0	.0	.0	.0	.0	.0	.0	.0

Figure 2-9 Total Computational Requirements (KIPS)

EVENT	INPUT	OUTPUT	TOTAL
Prelaunch	27.1	5.5	32.6
Start Generator	27.7	5.6	33.3
Internal Power	28.2	5.7	33.8
Go Inertial	71.4	18.9	90.4
Launch	77.7	23.3	101.0
End Atmosphere	77.7	23.3	101.0
Thrust Terminate	29.9	8.9	38.8
Coast	28.4	7.5	36.0
Reentry	28.7	6.6	35.3
Enter Atmosphere	26.1	6.1	32.3
Cruise	28.3	4.6	33.0
Landing	26.0	4.9	31.0
Lower Gear	25.9	5.3	31.2
Gear Down	23.2	4.9	28.2
Touchdown	2.9	.7	3.6
	.0	.0	.0

Figure 2-10 Total Input/Output Requirements (Thousand Bits Per Second)

steal interrupts the computer for one add time it would represent less than a 2% load upon the computer. Figure 2-11 shows the increase in computational requirements as the functions are removed. The first two columns show the total computational requirements when the matrix multiply function must be performed using a software subroutine. The next two columns show the total computational requirements when the logarithmic and exponential functions plus the matrix multiply function must be generated with software subroutines. In the next two columns the trigonometric function generation ia added to the software requirements. The last two columns represent the computational requirements with no special hardware function generation. The order in moving the function generation from hardware to software mechanization was chosen according to the complexity of mechanizing the function using hardware. The matrix multiply function would require the greatest amount of hardware to mechanize and the square root the least.

The percentage increase in computational capacity required during boost due to software function generation is:

	±&HK	X&÷
Matrix Multiply	4.3%	8.2%
Log - Exponential	0.6%	0.7%
Trigonometric	3.0%	4.3%
Square Root	0.1%	0.2%
All Functions	7.9%	13.4%

Thus software function generation requires 8 to 13% of the computational capacity of the centralized multiprocessor system. Most of the function generation capacity is used in the generation of matrix multiplies.

Figure 2-12 shows a graphical representation of the maximum computational requirements of the centralized multiprocessor system and several candidate computer configurations. The graph plots ordinate values of the number of add, subtract and housekeeping operations per second against absicssa values of multiply and divide operations per second. On the graph are four points differentiated by the symbols +, X ⊙, ⊡ . These four points represent the maximum computational requirements during boost. The + point represents the raw data computational estimates assuming hardware function generation. The X point represents the raw computational estimates assumirg software function generation. The ⊙ and ⊡ points have been generated by adding a 75% excess capacity to the raw computational estimates. The four straight lines, A, B, C, and D, represent the maximum computational capacity of a particular computer system. Consider a computer which will add in 1 microsecond and multiply in 10 microseconds. Such a computer would be fully occupied in performing 1 million adds per second or 400,000 adds per second and 60,000 multiplies per second, etc. Line A of Figure 2-12 represents the fully occupied computer with 1 microsecond add/10 microsecond multiply rates. The computer could also perform any computational task represented by a point to the left or below line A without using its full computational capacity. Any task above or to the right of the line A exceeds the computer's capacity.

EVENT	MATRIX MPY		LOG & EXP		TRIG		SQ RT	
	+&HK	X&÷	+&HK	X&÷	+&HK	X&÷	+&HK	X&÷
Prelaunch	219.6	1.4	226.8	3.0	226.8	3.0	226.8	3.0
Start Generators	220.4	1.6	227.6	3.2	227.6	3.2	227.6	3.2
Internal Power	221.5	1.5	228.7	3.1	228.7	3.1	228.7	3.1
Go Inertial	294.6	59.8	294.6	59.8	294.6	59.8	295.1	59.8
Launch	350.3	68.3	352.0	68.7	361.2	71.2	361.7	71.3
Pitch Over	350.5	68.3	352.3	68.7	361.4	71.2	361.9	71.3
End Atmosphere	351.7	68.7	353.6	69.1	363.6	71.9	364.3	72.0
Thrust Terminate	146.8	17.2	148.8	17.6	158.7	20.4	158.9	20.4
Coast	136.4	15.7	138.5	16.2	149.5	19.3	151.2	19.6
Reentry	141.6	18.5	143.5	19.1	156.4	22.7	160.1	23.4
Enter Atmosphere	137.9	18.3	139.8	18.9	152.7	22.5	156.4	23.2
End Survival	135.1	17.3	136.4	17.7	150.0	21.4	152.8	22.0
Cruise	139.3	18.3	140.5	18.7	153.7	22.3	156.3	22.8
Landing	133.4	18.3	134.6	18.7	148.8	22.5	151.5	23.0
Lower Gear	133.4	18.4	134.6	18.7	148.8	22.5	151.5	23.0
Gear Down	133.3	18.3	134.4	18.7	148.8	22.5	151.3	23.0
Touchdown	9.2	.1	9.2	.1	9.2	.1	9.2	.1
End Taxi	.0	.0	.0	.0	.0	.0	.0	.0

Figure 2-11 Computational Requirements With Software Subroutines

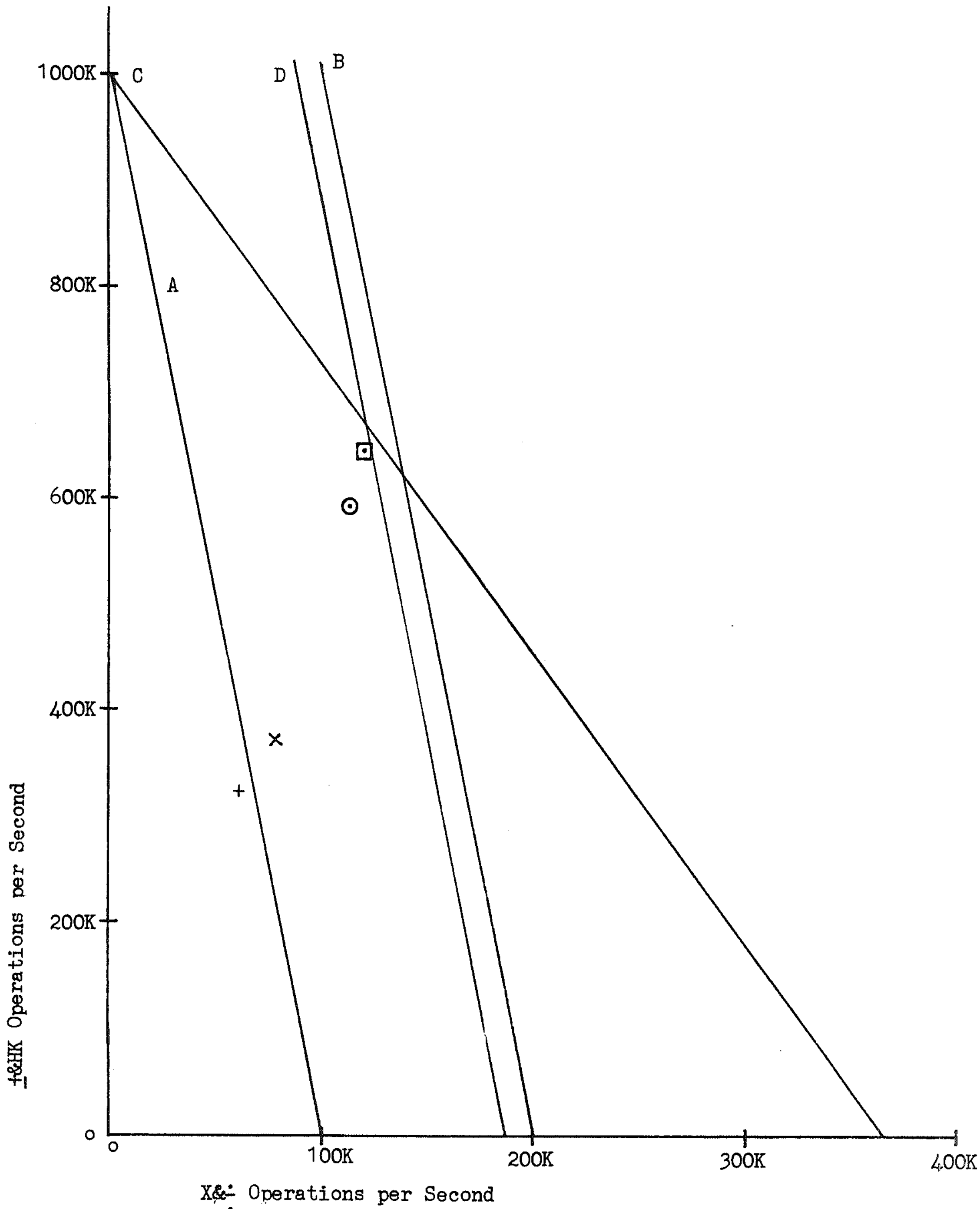


Figure 2-12 Centralized Multiprocessor System Computational Requirements

Line A is fairly representative of state of the art of high speed single processor computers. There are a few single processor computers manufactured which exceed the computational capacity of line A. However, these are exceptions which use special high speed scratch pad memories and special high speed logic design. The high reliability demanded by the space shuttle mission can more easily and economically be met by using conservative state of the art hardware designs. The computer represented by line A barely has the computational capacity to perform the raw data estimates with hardware function generation. Lines C and D represent two single processor computer capacities which will meet the raw data plus 75% excess capacity requirements. Line C represents a computer having a 1 microsecond add and a 2.8 microsecond multiply. Multiply logic which would operate at this speed would be very complex. A computer of this computational capacity could be considered if reliable, high speed LSI multiplication logic could be economically developed. Line D represents a computer having a .5 microsecond add and a 5.4 microsecond multiply. This computer requires a less complex multiply mechanization. A very fast memory is required to achieve a .5 microsecond add. The execution of an add instruction requires reading an instruction and data from memory. If the data and instructions are stored in separate memory modules, and overlap capabilities exists, the .5 microsecond add instruction is beyond the present speed capabilities of core memories. Plated wire memories have been operated at speeds faster than .5 microsecond but not in avionics applications. One of the primary considerations in achieving highly reliable hardware is to operate the hardware well below its maximum performance limits.

Line B represents a multiprocessor computer having two processors. Each processor has the capacity of the single processor computer represented by line A. A dual processor normally does not have twice the capacity of a single processor due to executive overhead. In an avionics application such as the space shuttle booster this is not the case. The programs executed by the computer in a normal application are a variable which continuously changes. In an avionics application the programs are well defined and fixed for each flight. The executive overhead can be greatly reduced when the programs to be executed are not a variable. A dual processor computer will closely approach twice the capacity of a single processor computer in an avionics application. From Figure 2-12 it can be seen that the dual processor will perform the desired excess capacity requirements.

The raw data estimates show a total memory requirement of 90,000 fixed 16 bit words which could be stored in read only memory and 25,000 variable 16 bit words which must be stored in read/write memory. Applying the 1.75 excess capacity figure yields a requirement of approximately 157,000 fixed and 44,000 variable storage words. This is a total memory requirement of 201,000 words of memory. This does not include any special memory for storing data field codes for the multiplex data bus system.

2.3.2 DISTRIBUTED LINKED UNIT SYSTEM

There are many ways of distributing the various program modules between a number of small unit computers. The configurations considered in the study have been obtained by separating candidate subsystems from the centralized system.

Referring to Figure 2-9 it is seen that the major computational load occurs during boost. Figure 2-7 shows that the subsystem which produces the major contribution to the boost computational load is the propulsion system. Placing the propulsion system into a separate computer yields computational and memory requirements for both the central and propulsion computer as shown in Figure 2-13. The figures for computational speed have been rounded and expressed in thousands of operations per second. An executive program is included in both computers. Boost has the heaviest loading on the propulsion computer while reentry into the atmosphere produces the heaviest loading on the central computer (excluding ground checkout).

EVENT	PROPULSION		CENTRAL	
	±&HK	X&÷	±&HK	X&÷
Prelaunch	9	0	225	3
Internal Power	9	0	226	3
Go Inertial	226	54	75	6
Launch	226	54	142	17
End Atmosphere	226	54	144	18
Thrust Terminate	11	0	155	20
Coast	11	0	147	20
Reentry	10	0	156	23
Enter Atmosphere	9	0	154	23
End Survival	9	0	151	22
Cruise	14	1	149	22
Landing	14	1	144	22
Touchdown	11	1	9	0
End Taxi	0	0	0	0

COMPUTER SPEED REQUIREMENTS

	PROPULSION	CENTRAL
FIXED	5586	84872
VARIABLE	2527	22542
TOTAL	8113	107414

MEMORY REQUIREMENTS

Figure 2-13 Computational and Memory Requirements with Separate Propulsion Computer

Figure 2-14 shows a graph of the maximum requirements of the propulsion and central computer. Both the raw data estimates and the 1.75 excess capacity estimates are plotted for both computers. The plotted points are marked C and P to distinguish between the central and propulsion computer. The points for each computer which are closest to the original are the raw data estimates. Line A represents a 1 microsecond add/10 microsecond multiply computer. A computer of this capacity is capable of performing the central computer requirements. However, it will not perform the excess capacity propulsion requirements. Two computers of this capacity will perform the excess capacity propulsion requirements.

The numerical excess capacity estimates are 395,000 adds per second and 95,000 multiplies per second for the propulsion computer and 273,000 adds per second and 40,000 multiplies per second for the central computer.

To meet these requirements, the propulsion system computer must satisfy:

$$.395\ A_p + .095\ M_p \leq 1 \qquad (1)$$

and the central computer must satisfy:

$$.273\ A_c + .040\ M_c \leq 1 \qquad (2)$$

where A_p, A_c, M_p, and M_c are the add and multiply times in microseconds for the propulsion system and central computers. Figure 2-15 shows a plot of

$$.395\ A_p + .095\ M_p = 1 \qquad (3)$$

$$.273\ A_c + .040\ M_c = 1 \qquad (4)$$

On this graph a computer is represented by a single point. On the graph the point representing a 1 microsecond add/10 microsecond multiply computer is marked with an X. An application requirement is represented by a straight line. A computer capable of performing an application requirement will lie to the left or below the application line. As stated before the 1 microsecond add/10 microsecond multiply computer can meet the central computer requirements but not the propulsion system computer requirements.

The large computational load in the propulsion computer is primarily caused by the control of the 12 main rocket engines. There are four program modules associated with the main rocket engines; MPSE, MPSF, MPSS and MPST. Each of these programs must be executed once for each of the 12 main rocket engines during boost. Configuring the DMS so that two computers are used to control the main rocket engines and one computer for all other functions will balance the computational speed requirements between the three computers. Figure 2-16 shows the computational requirements of the central computer and two identical main rocket engine computers.

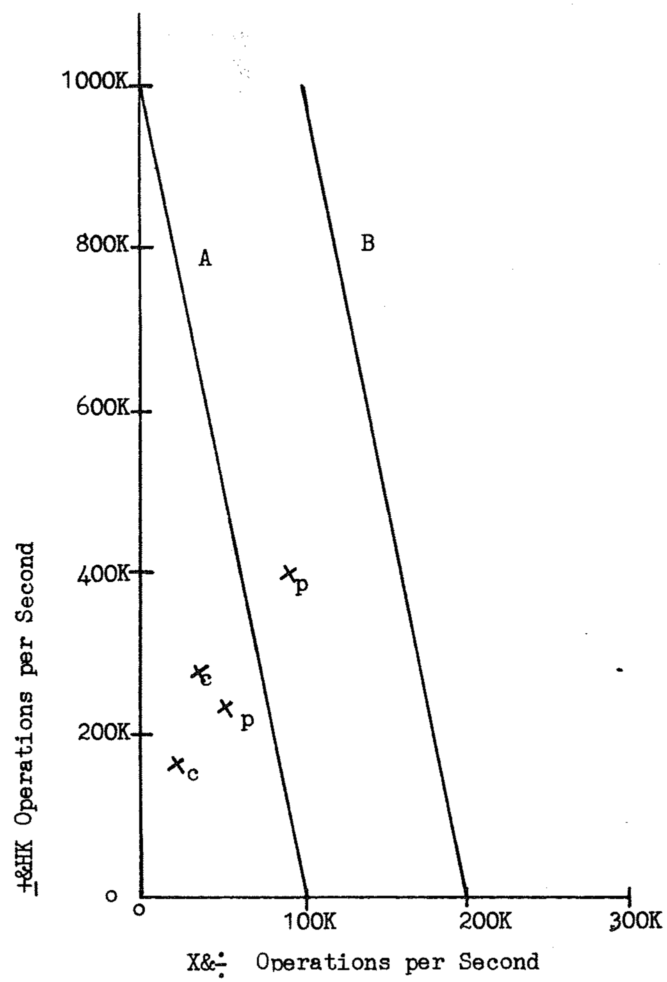


Figure 2-14 Propulsion System Distributed

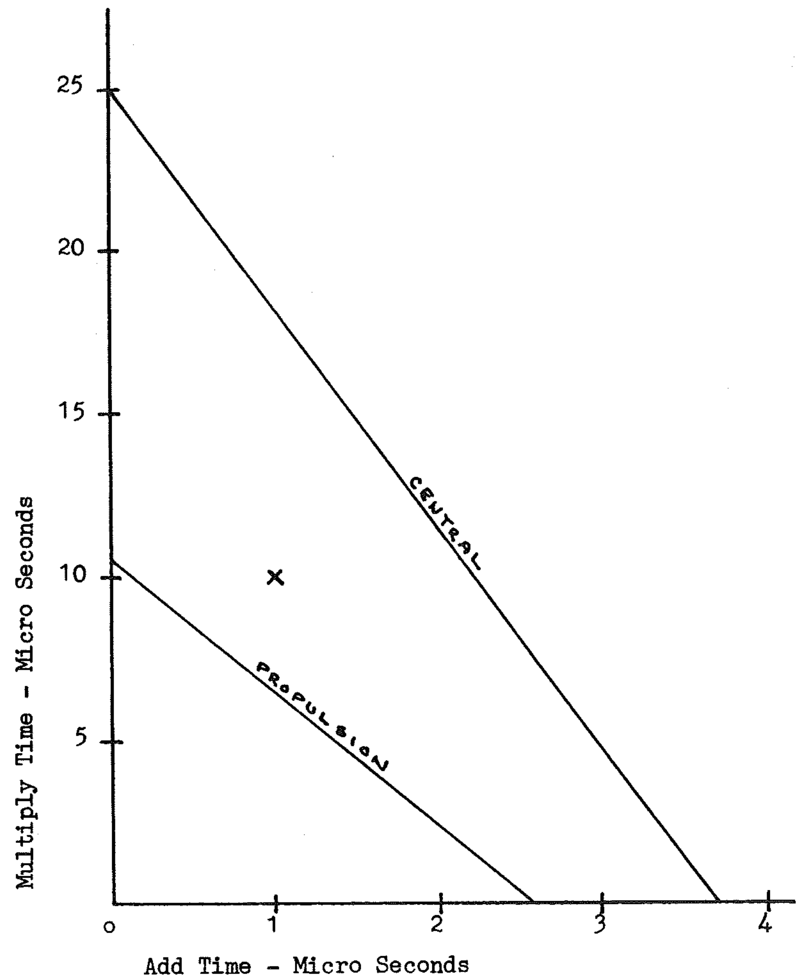


Figure 2-15 Computer Limits for Separate Propulsion Mechanization

EVENT	CENTRAL		PROPULSION	
	±&HK	X&÷	±&HK	X&÷
Prelaunch	227	3	0	0
Internal Power	228	3	0	0
Go Inertial	77	6	115	27
Launch	144	17	115	27
End Atmosphere	146	18	115	27
Thrust Terminate	159	20	0	0
Coast	151	20	0	0
Reentry	159	23	0	0
Enter Atmosphere	156	23	0	0
End Survival	153	22	0	0
Cruise	156	23	0	0
Landing	151	23	0	0
Touchdown	14	1	0	0
End Taxi	0	0	0	0

COMPUTER SPEED REQUIREMENTS

	PROPULSION	CENTRAL
FIXED	2360	88103
VARIABLE	777	22914
TOTAL	3737	111017

MEMORY REQUIREMENTS

Figure 2-16 Computational and Memory Requirements
With Two Main Engine Control Computers

Each of the propulsion computers controls six of the main rocket engines. All of the other propulsion system functions (those functions required of the cruise engine, reaction jets and fuel system) are performed by the central computer. Excluding preflight checkout functions, the maximum central computer load occurs in the initial reentry phase. Using the .75 excess capacity figure the central computer must be capable of performing 278 thousand adds per second plus 40 thousand multiplies per second. Each of the propulsion computers must be capable of performing 201 thousand adds per second plus 47 thousand multiplies per second. Figure 2-17 is a graphical representation of the requirements of the central and dual propulsion computers. The requirements for each computer lie well below line A which represents a 1 microsecond add/10 microsecond multiply computer.

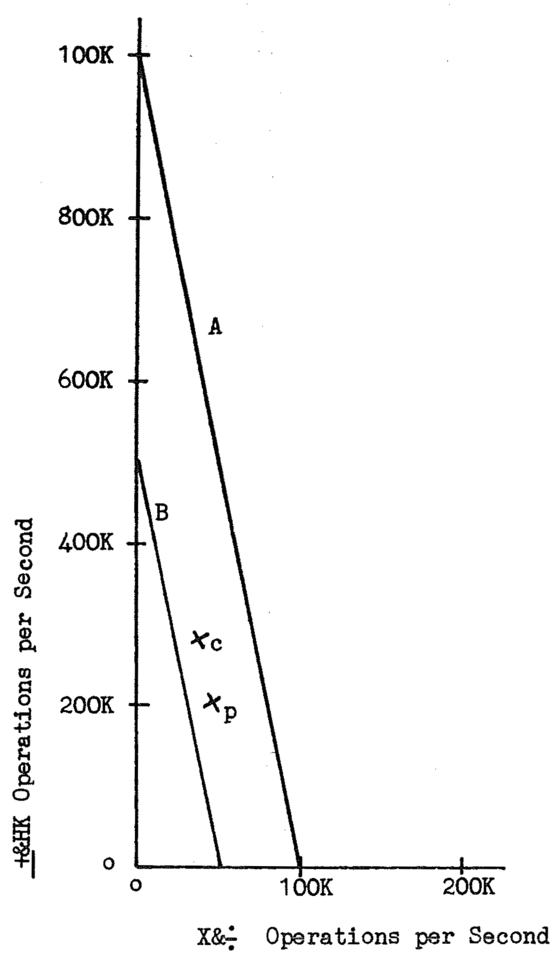


Figure 2-17 Computational Requirements of Central Plus Dual Propulsion Computers

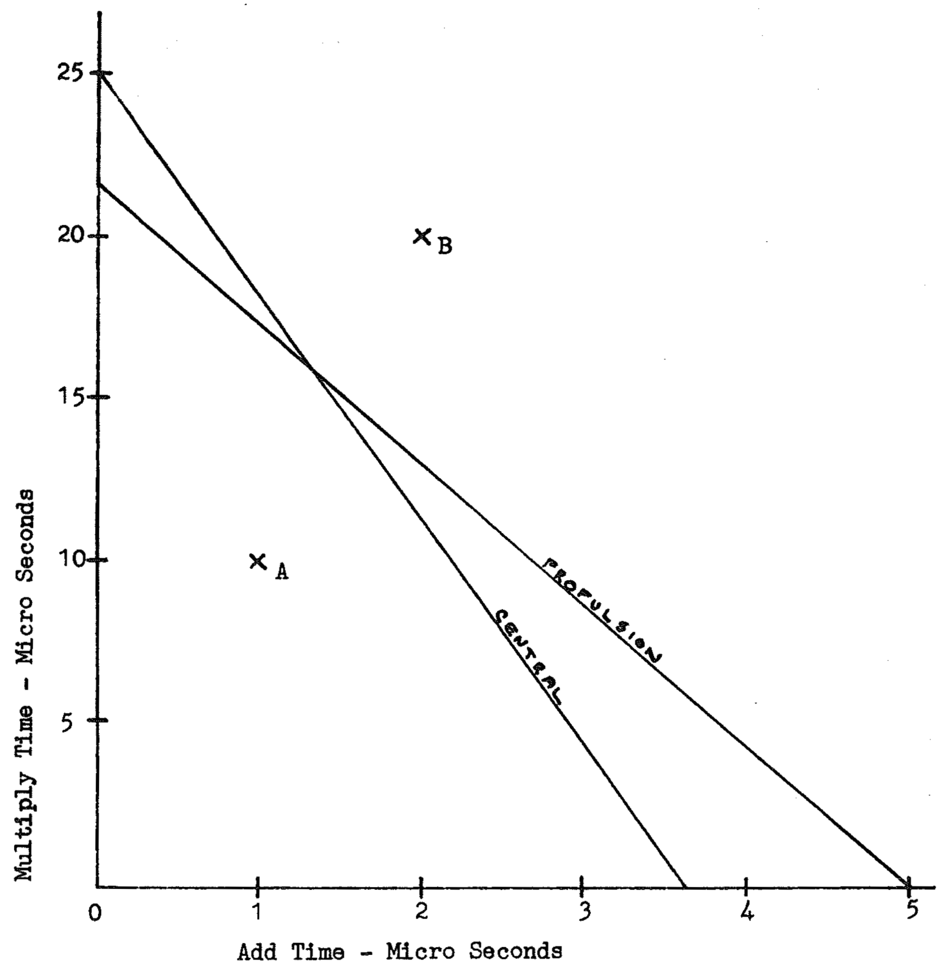


Figure 2-18 Computer Limits for Central Plus Dual Propulsion Computer System

Line B represents a computer having half the computational capability of the computer of line A. (Computer B is a 2 microsecond add/20 microsecond multiply computer). The computer represented by line B will not meet the requirements of the central or propulsion computers. Reduction of computer speed capacity by $\frac{1}{2}$ generally results in a significant reduction in hardware sophistication. The 2 microsecond add/20 microsecond multiply computer represents a good tradeoff point for the next lower level of computer hardware sophistication. Figure 2-18 shows the add/multiply time limits for the computer required to perform the central and propulsion functions. Point A represents the 1 microsecond add/10 microsecond multiply computer and is within both limits. Point B, representing the 2 microsecond add/20 microsecond multiply computer, is not within the requirements limits.

Because the 2 microsecond add/20 microsecond multiply computer represents a reduction in hardware sophistication per computer, the question arises as to the number of this computer type required in mechanizing the DMS and the functions to be performed by each individual computer in the mechanization. The next logical step in the reduction of the "per computer" requirements of the propulsion computer is to use three identical propulsion computers with each computer controlling four of the main rocket engines. The functions being performed by the central computer must be divided into two computers. The navigation and guidance subsystem is the largest single central computer function and is thus a prime candidate for mechanization in a separate computer. Figure 2-19 gives the raw data estimates for each computer in a DMS partitioned as indicated above. The maximum computational requirements of the central computer occurs at the very end of boost.

EVENT	CENTRAL		NAVIGATION		PROPULSION	
	±&HK	X&÷	±&HK	X&÷	±&HK	X&÷
Prelaunch	218	3	15	0	0	0
Start Generator	219	3	15	0	0	0
Internal Power	220	3	15	0	0	0
Go Inertial	68	4	15	2	79	18
Launch	82	5	68	13	79	18
End Atmosphere	82	5	71	14	79	18
Thrust Terminate	95	7	71	14	0	0
Coast	77	3	80	16	0	0
Reentry	75	4	92	20	0	0
Enter Atmosphere	71	4	92	20	0	0
End Survival	71	4	88	18	0	0
Cruise	75	5	87	18	0	0
Landing	69	5	89	18	0	0
Lower Gear	68	5	89	18	0	0

COMPUTATIONAL SPEED REQUIREMENTS

	CENTRAL	NAVIGATION	PROPULSION
FIXED	78554	10595	2360
VARIABLE	21628	1394	666
TOTAL	100182	11989	3026

COMPUTER MEMORY REQUIREMENTS

Figure 2-19 Computer Requirements for Navigation and Propulsion System Distributed

This is primarily due to an overlap in the boost and coast flight control computations. The maximum computational requirements of the navigation computer occur during the initial reentry phase when survival energy management computations are being performed. The propulsion computers are again only used during boost when the main propulsion engines must be controlled.

Figure 2-20 is a graphical representation of the excess capacity computational requirements of the central, navigation and one of the three identical propulsion computers. A line representing the maximum capacity of a 2 microsecond add/20 microsecond multiply computer is also plotted. This computer just meets the navigation requirements and exceeds the propulsion and central computer requirements. This point is reiterated by Figure 2-21 which shows the computational limits for the three computer types with the navigation requirements intersecting the 2 micro-

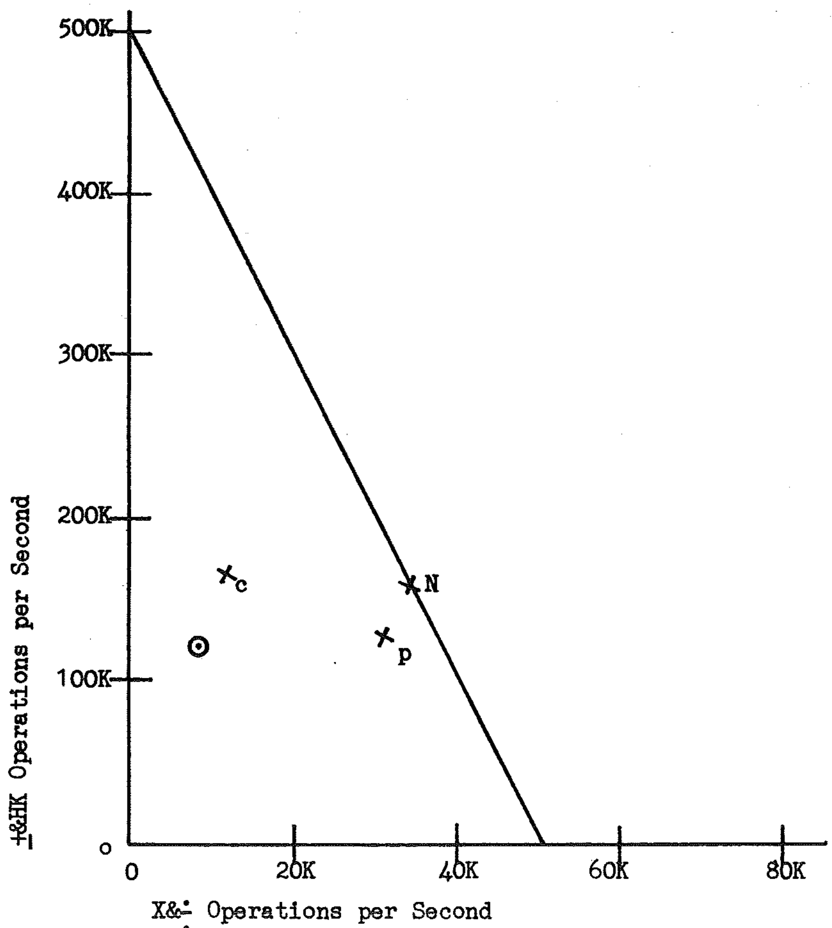


Figure 2-20 Computational Requirements of Central Plus Navigation Plus Triple Propulsion Computers

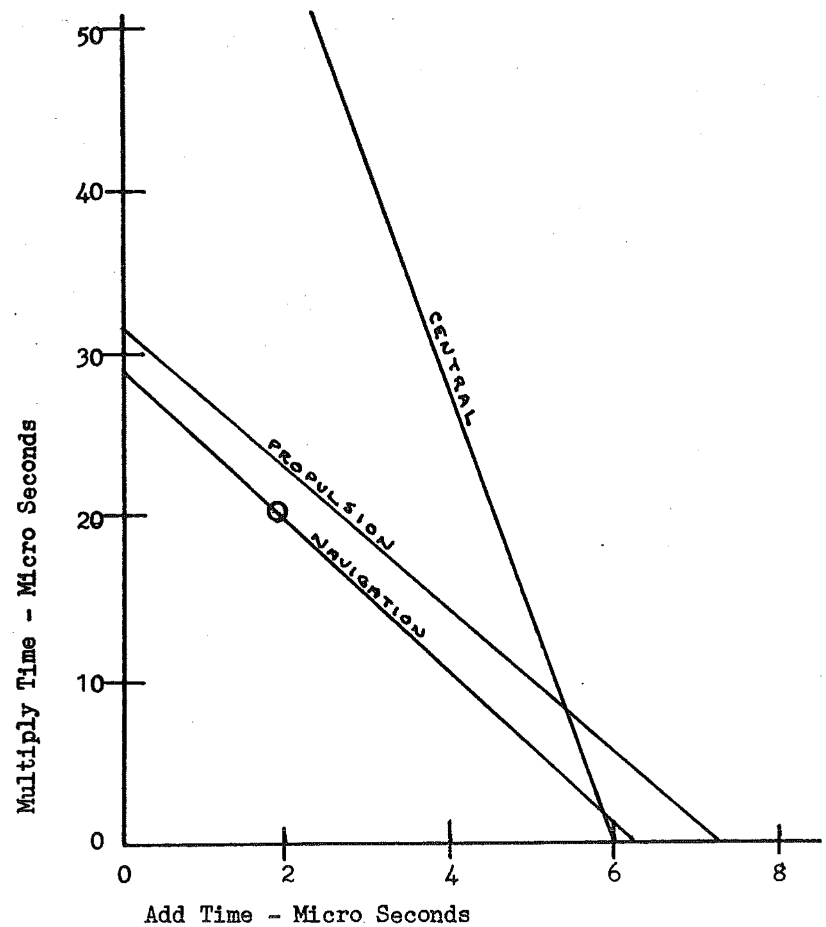


Figure 2-21 Computer Limits for Central Plus Navigation Plus Navigation Plus Triple Propulsion Computer System

second add/20 microsecond multiply point. The central computer speed requirements are significantly less than the navigation computer tasks. During reentry, when the navigation computer is operating at its maximum loading, the load on the central computer has dropped to the point marked with ⊙ on Figure 2-20 A more balanced loading could be achieved between the computers in the system by moving some of the navigation tasks to the central computer.

2.3.3 CENTRALIZED LINKED UNIT SYSTEM

The Centralized Linked Unit System is a set of individual computers each performing a portion of the total computational task. In the centralized system there is no reason to devote single computers to subsystem functions. This allows for an even distribution of computational loads among the computers. Dividing computational tasks according to subsystem function minimizes the data flow between computers. The data flow requirements for the space shuttle booster DMS have a maximum data rate between the computer and external subsystem equipment of 101,000 bits per second. If this number is doubled to allow for the addition of address coding bits and parity and then doubled again for contingencies, the resultant 404,000 bits per second is still well within the state of the art of multiplexer system capabilities. The low data rate requirements will allow intercomputer data transfers to be mechanized using the data bus system. Analysis of the interprogram data flow requirements given in Figure 2-8 produces a maximum data flow estimate of 228,000 bits per second. Using the same multiplication factor of 4 to account for addressing, parity and contingencies produces a maximum upper limit on data flow rate of 912,000 bits per second. This includes both interprogram and computer-external equipment data flow estimates.

Figure 2-11 indicates that the maximum computational speed requirements occurs just prior to thrust termination with 364,338 adds per second and 72,028 multiplies per second required using raw data estimates. By subtracting the executive requirements from the above total and distributing the program load among N computers with each computer having its own executive yields individual computer requirements of

$$A_s = 1.75 \times \left(\frac{357,752}{N} + 6586 \right)$$

$$M_s = 1.75 \times 72,028/N$$

where A_s is the number of required adds per second per computer and M_s the required multiplies per second per computer. The 1.75 multiplier in the formulas is the desired excess capacity ratio. The table below gives the values of A_s and M_s in units of thousands of operations per second for values of N from 1 through 10.

N	A_s	M_s
1	638	126
2	325	63
3	220	42
4	168	32
5	137	25
6	116	21
7	101	18
8	90	16
9	81	14
10	74	13

Figure 2-22 graphically illustrates the number of computers required in a Centralized Linked unit system when given the individual computer add and multiply times. A computer is represented by a single point on the graph. For example a computer having an add time of 2 microseconds and a multiply time of 20 microseconds is indicated in the figure by the point marked with the symbol +. The point lies above the line marked 3 and below the line marked 4 indicating that 4 computers of this type are required to perform all of the space shuttle booster tasks.

The memory requirements for each of the linked unit computers is found from

$$W_F = 1.75 \times \left(\frac{89280}{N} + 575\right)$$

$$W_V = 1.75 \times \left(\frac{24850}{N} + 108\right)$$

$$W_T = 1.75 \times \left(\frac{114130}{N} + 683\right)$$

where W_F is the number of 16 bit fixed storage words, W_V the number of 16 bit variable storage words, and W_T the total number of 16 bit storage words required. These values have been adjusted by the 1.75 desired excess capacity factor. The table below gives the memory requirements in thousands of words for individual computers for values of N between 1 and 10.

N	Fixed	Variable	Total
1	157.2	43.7	199.7
2	79.1	21.9	99.9
3	53.1	14.7	66.6
4	40.1	11.1	49.9
5	32.3	8.9	39.9
6	27.0	7.4	33.3
7	23.3	6.4	28.5
8	20.5	5.6	25.0
9	18.4	5.0	22.2
10	16.6	4.5	20.0

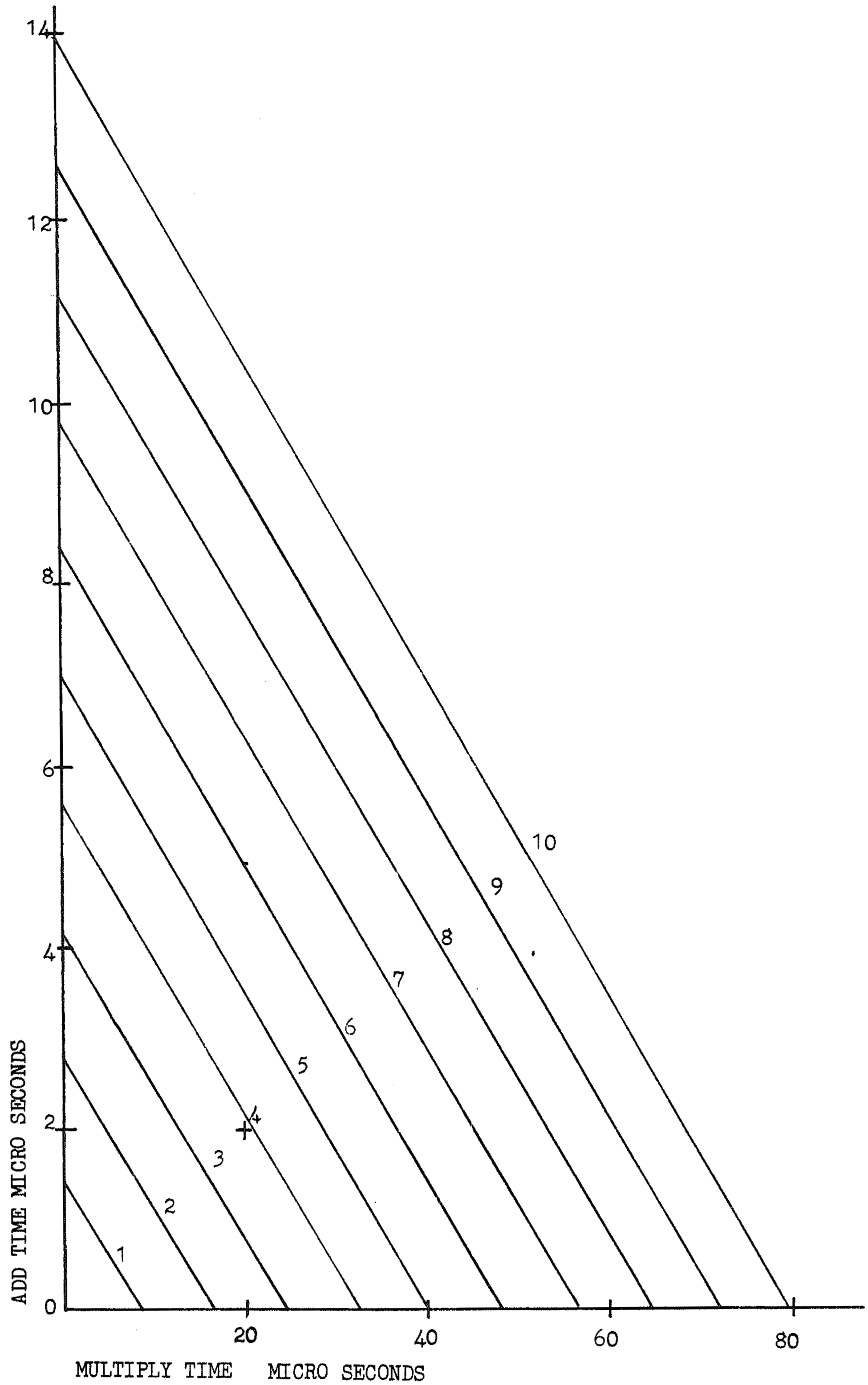


Figure 2-22 Centralized Linked Unit System Number of Computer Requirements

3. FUNCTIONAL REQUIREMENTS

The tasks performed by the DMS are divided into the following subsystem functions:

- Structures
- Propulsion
- Electrical Power Generation and Distribution
- Navigation and Guidance
- Flight Control
- Communications
- Operations Management

This section discusses the functional requirements of each of these subsystems selecting specific mechanizations to perform the required functions.

3.1 STRUCTURES

There are three primary functions performed by the space shuttle booster structures:

1. To mechanically couple all of the booster subsystems with structural integrity sufficient to maintain this coupling while experiencing the various forces and stresses experienced during the mission. The structure must also provide a coupling to the space shuttle orbiter during the periods of prelaunch, launch and boost. The landing gear must be lowered and locked just prior to touchdown. The booster configuration may possibly use wings which are stowed during boost and deployed just prior to reentry. It is assumed for this study that the booster will have fixed wings.

2. To provide an aerodynamic shape to the configuration which minimizes drag on the vehicle while providing sufficient stability during boost, maintain controllability, and provide the required lift and stability during atmospheric cruise flight.

3. To provide a protective covering for all systems and in particular to provide protection against the large heat pulse encountered during reentry.

All three primary functions of the booster structures interface either directly or indirectly through other systems with the data management system.

<u>Mechanical Structural Performance</u>

Direct coupling of the mechanical performance of the structures with the DMS will be strictly a monitoring function. During early flights this monitoring will be used to assess the structural performance of the vehicle providing engineering data required for vehicle modifications and redesigns. During later operational flights this data will determine any mechanical overstresses and provide a criteria for determining the useful life of the vehicle. The equipment required specifically for this task are vibration and stress sensors plus multiplex and encoding electronics to place the sensor outputs on the data bus.

During the prelaunch and boost flight phase the DMS will be informed of the status of the mechanical coupling between the booster and orbiter and during separation the DMS will be informed of the success of the mechanical disconnect system. The DMS will continuously monitor the landing gear system to determine if the gear is up and locked, lowering, or down and locked during each applicable mission phase. During flight, stresses and loads on the vehicle must be kept below design limits. There are several systems which function during different portions of the mission which are specifically included to keep loads below the design limits. In each case, these systems couple to the DMS through the flight control system. Specifically these systems are the load relief system used during boost, the energy management system used during reentry, and the automatic landing system used during landing. The functional operation of these systems are described in the flight control system sections.

Aerodynamic Performance

The aerodynamic performance of the structures is highly coupled and integrated with booster flight control and navigation and guidance systems. The aerodynamic performance is not directly monitored or controlled by the DMS but only indirectly through the other booster systems. This indirect coupling will be discussed under the other booster systems.

Protective Covering Performance

The most critical protection provided by the structure is against aerodynamic heating during reentry. This heating will be monitored on the structural skin by the DMS to provide engineering data required for vehicle modification or redesign as well as for determining useful vehicle life. The temperature within the vehicle at the location of critical temperature sensitive subsystems will also be monitored by the DMS.

3.2 PROPULSION SYSTEM

The booster contains three propulsion systems, main rocket engines, the attitude control propulsion system (ACPS), and cruise air breathing engines. The major characteristics of these systems are shown in Figure 3-1.

Main Propulsion System Description

The type of main propulsion systems under consideration for the space shuttle vehicles involves closed loop propellant utilization techniques in place of the open loop systems used in previous programs. Throttleable engines that permit proportional throttling as opposed to selective engine shutdown to obtain the desired acceleration program are also under consideration. Figure 3-2 shows a nominal LOX and LH_2 pressurization system for the main engine; Figure 3-3 is a schematic representation of a possible variable thrust control loop.

The engine described here employs the staged combustion cycle which uses the preburner to generate the gases needed to drive the propellant turbopumps. Four rather than two turbopumps are used. Basic operation is described in the paragraphs that follow. LH_2 from the fuel tank enters the engine through the low pressure turbopump. This pump draws fuel from the tank without cavitation and feeds it into the main high speed (high pressure) field turbopump. LOX from the oxidizer tank is similarly brought through a low pressure pump into the high speed oxidizer turbopump and then routed into the main oxidizer and preburner valves. With the opening of the main fuel valves, most of the hydrogen is diverted through the regenerative, cooling passages surrounding the forward half of the fixed primary nozzle and then into the preburner. The remainder of the fuel flows through a separate cooling system surrounding the aft half of the primary nozzle. This portion of the fuel then goes through a small turbine that drives the speed fuel inducer. It is then routed through the wall of the combustion chamber and nozzle throat which it cools by transpiration.

For the LOX, when the main and preburner oxidizer valves are opened, most of the flow is diverted to the small turbine driving the low speed oxidizer inducer. From this turbine, the main oxidizer flow enters the main combustion chamber through the main injector. The remainder of the oxidizer flows directly into the preburner where it mixes with the main portion of the fuel flow and is ignited by a hydrogen-oxygen torch lit by a spark plug. The fuel rich combustion gases exhaust through the two turbines that drive the main fuel and oxidizer turbopumps. Most of the exhaust flows through the larger opening of the main fuel turbopump drive and a smaller part exhausts through the turbine that drives the main oxidizer pump. The two fuel rich exhaust streams recombine in a plenum chamber in the main case and then flow into the main combustion chamber through the main injector. Here they combine and burn with the main oxidizer flow.

Attitude Control Propulsion System (ACPS) Description

The booster's auxiliary propulsion requirements for reentry attitude control and for separation involve thrust magnitudes that are most efficiently obtained with H_2/O_2 engines. The hypergolic bipropellant systems used for attitude control in previous spacecraft could more readily achieve the short firing pulses associated with attitude stabilization but are

MAIN PROPULSION

Engine	
Thrust	400K (SL)
Number	12
Expansion Ratio	65:1
Mixture Ratio	6.0
Specific Impulse	449.3 (nominal vac)
TVC Concept	Gimbal - Fail Null Actuators
TVC Angles	5° Pitch, 4° Yaw
Pressurization	
Pressurant	GO_2/Gh_2
Pressure	29 psia O_2, 28 psia H_2
Ideal Velocity	16,000 FPS

ACPS AND ORBIT MANEUVER

Engine	
Number	16
Thrust	1600 lb
Chamber Pressure	500 psi
Expansion Ratio	40:1
Specific Impulse	385 (nominal)
Mixture Ratio	5.0
Pressurization	Turbo pump
Propellant Conditioning	Heat Exchangers GO_2/GH_2 Gas Gen
Feed and Distribution	Regulated supply from high pressure accumulators
Maneuver Rates	
Roll	1.1 deg/sec^2
Pitch	0.5 deg/sec^2
Yaw	0.5 deg/sec^2
Translation	
Total Impulse	942,000 lb-sec

CRUISE/GO-AROUND

Engine	
Type	Low Bypass Turbofan
Number	8
Thrust (SLS)	18,000 lb
Thrust-to-Weight	0.27g
Range	485 nm plus Go-Around

Figure 3-1 Propulsion System Characteristics

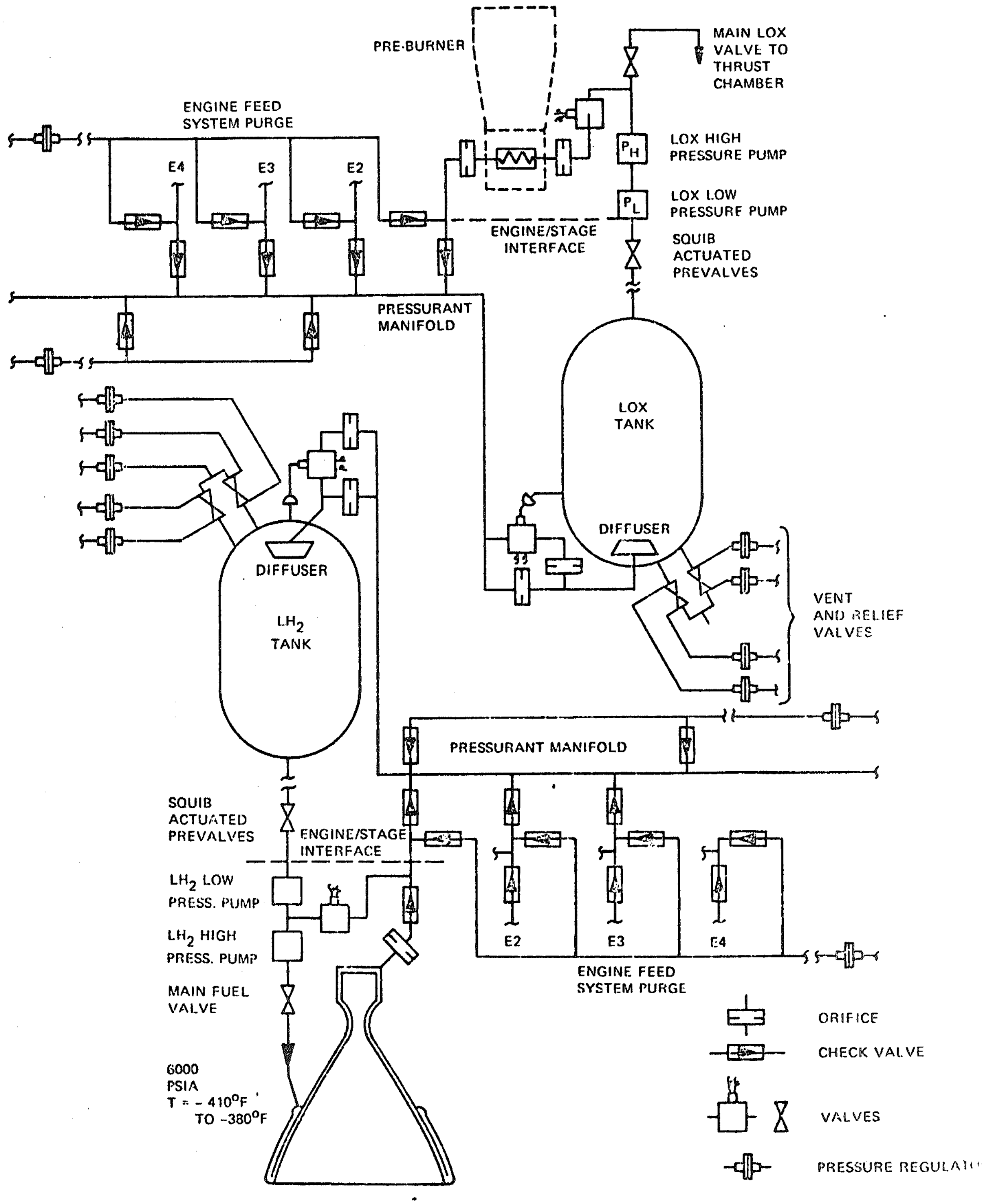


Figure 3-2
Main Propulsion Pressurization System

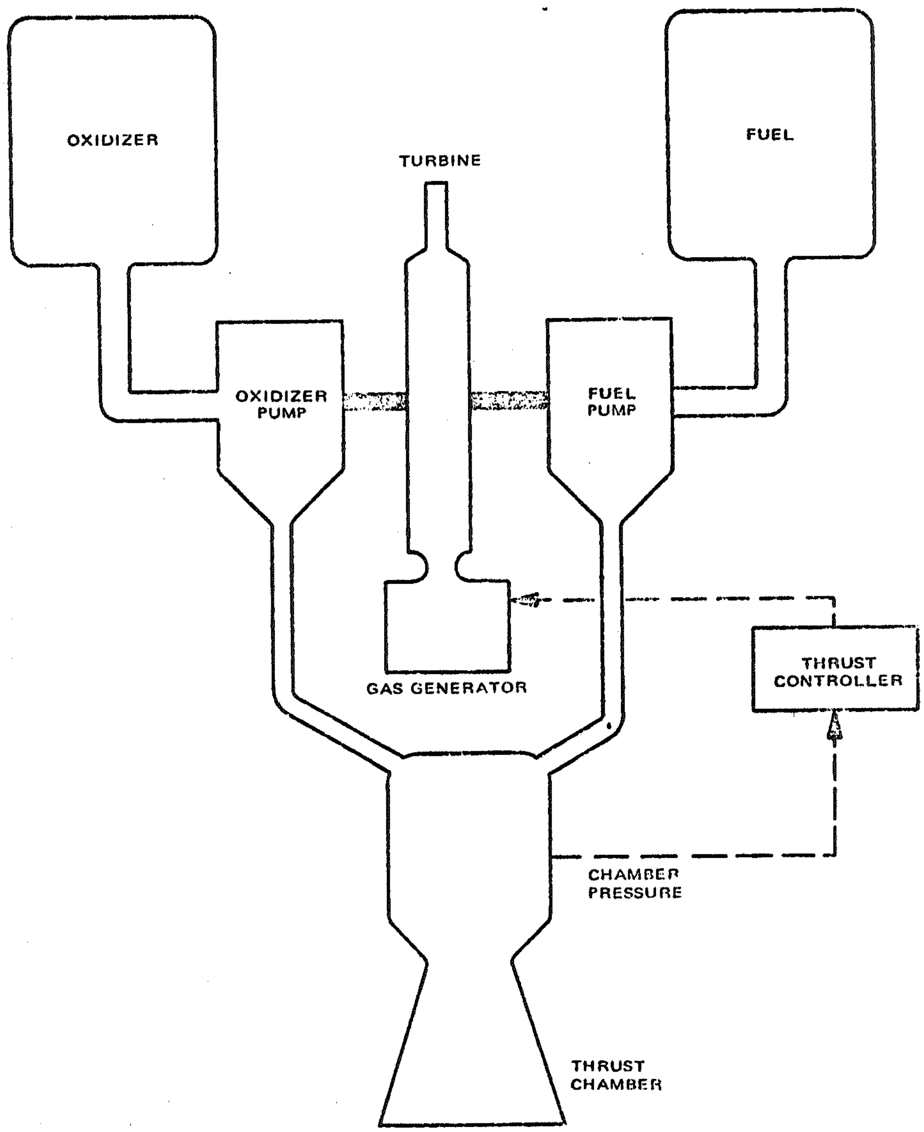


Figure 3-3
Closed Loop Thrust Control Schematic

apparently being rejected for shuttle booster applications because of size and weight penalties. A system based on LH_2/LO_2 that also uses a turbopump rather than a pressure feed system is shown in Figure 3-4. This system is attractive if the airbreathing turbofan engines can be made for LH_2 fuel and if the ACPS turbopump can match the flow and pressure characteristics of the airbreathing engines. However, the rapid and short pulse firing of these engines in the attitude control modes of reentry may pose some new problems.

Air Breathing Engine

A hydrogen fueled, low bypass ratio turbofan is the leading contender for this function. Its control system would therefore be closely related to the propulsion pressurization and feed systems associated with the ACPS and main engine systems. Thrust control could be achieved in an open loop sense as in conventional jet engines. Commands from the throttle transducer (manual) or from the guidance and control computers (automatic) would adjust fuel/air mixtures to vary engine pressure ratio and RPM.

Each of these propulsion systems must be controlled by the DMS or a special control computer. The functional control requirements for each system are:

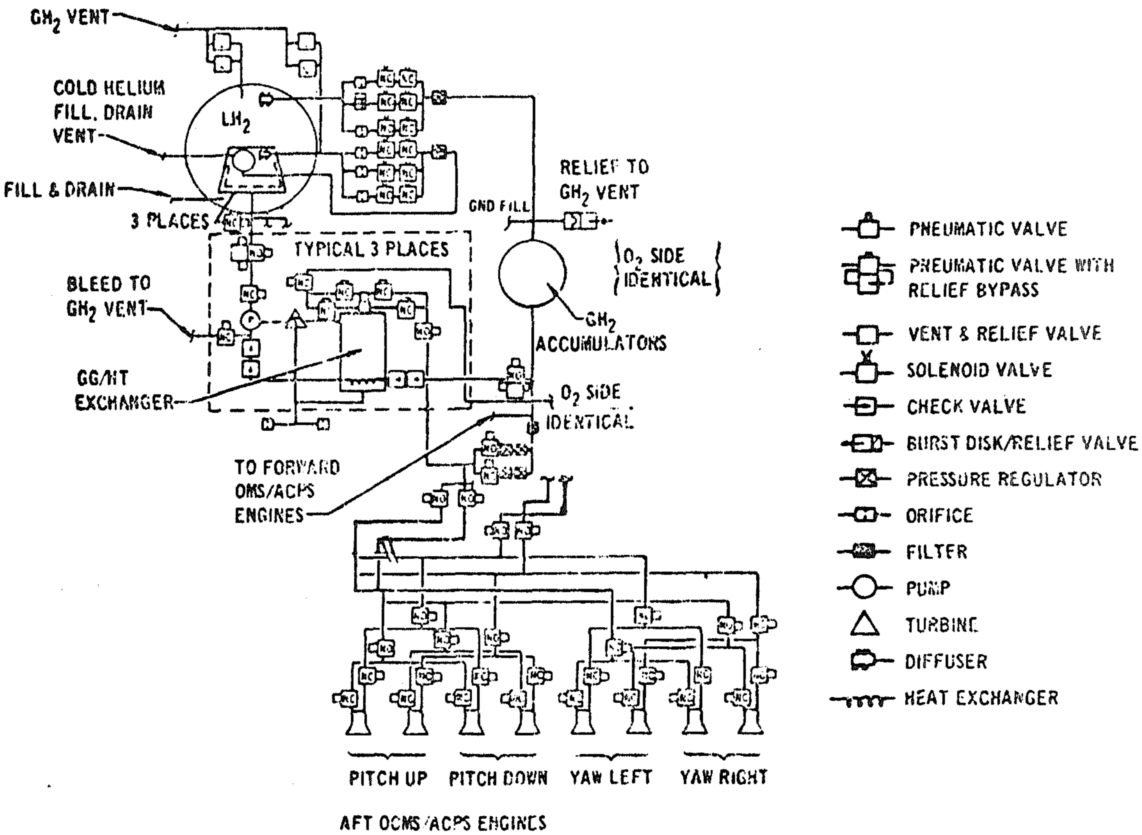
Main Engine Controls - The functions involved in the control of the booster's main engines are the following:

- Thrust Level
- Propellant Utilization
- Start/Shutdown
- Operational Readiness Checks
- Malfunction Detection and Diagnostics

Although the design of the required staged combustion cycle, high pressure, liquid propellant engine is not sufficiently well advanced to define the details of its control system, most of the requirements can be identified in general. The basic main engine control system block diagram is shown in Figure 3-5. Note that the thrust vector controls are assumed to be part of the flight control system. The control devices are the various fuel and oxidizer valves and the igniters. These valves include the ones that control oxidizer/fuel ratios and thrust as well as the many valves associated with propellant management (bypass and vent valves and so on). The engine parameter sensors are those associated with the closed loop thrust controls and those associated with performance monitoring and operational readiness testing.

The propulsion control computations include the stored calibration information needed to relate the main chamber oxidizer valve position with thrust level for various conditions of mixture ratio (turbopump rpm), total oxidizer flow, oxidizer valve area, and so on. A typical control loop for driving the main oxidizer control valve actuator in order to achieve the desired thrust (as measured by chamber pressures P_c) is shown in Figure 3-6. The required valve position is a nonlinear function of thrust error and flow error. The calibrations are achieved within the computer using table look-up techniques for the required functions. Associated with this thrust control loop is a simultaneous control of turbopump (preburner) rpm and LH_2 flow (consequently affecting mixture ratio).

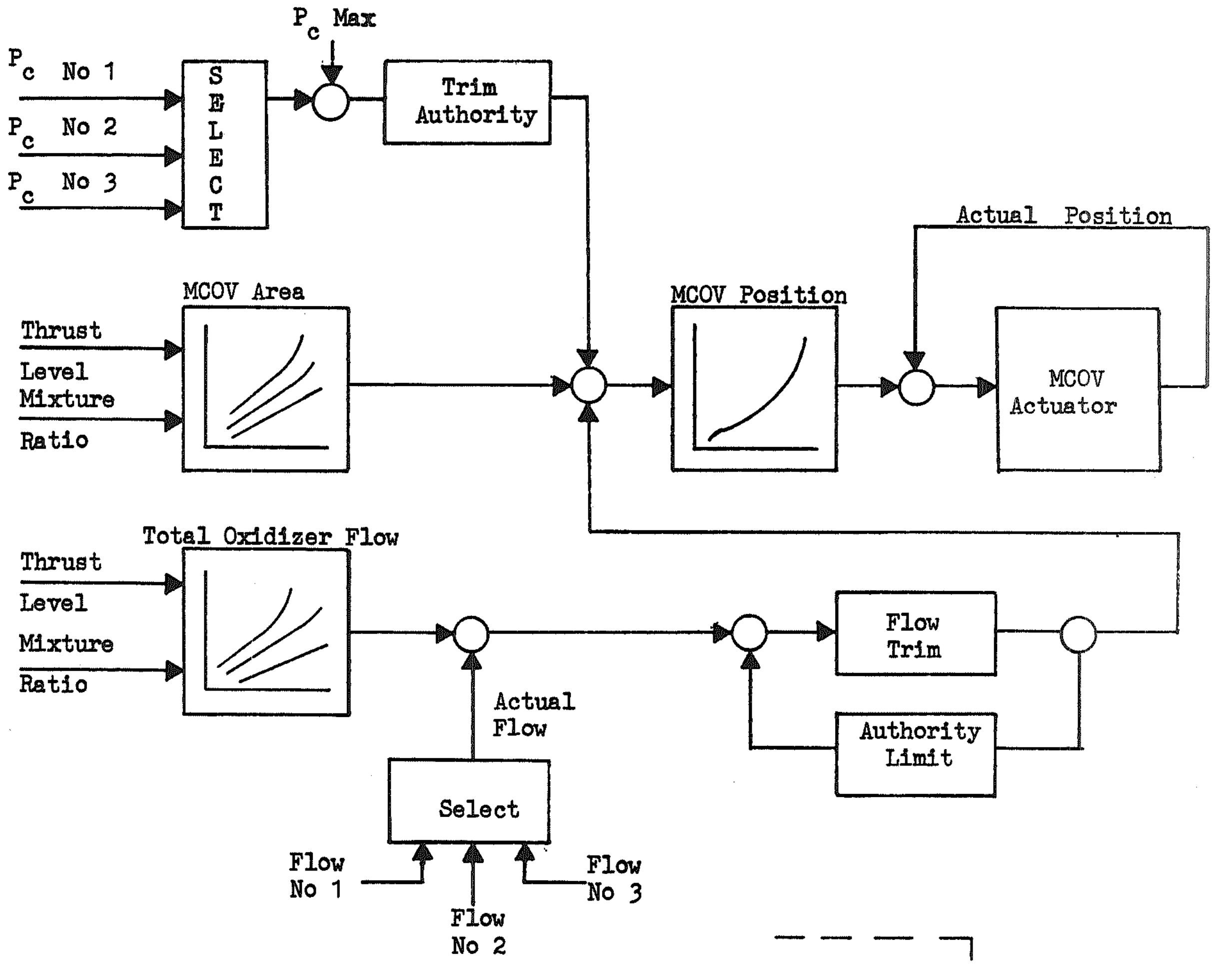
VEHICLE	CRITICAL REQUIREMENT	ACCELERATION NOMINAL REQUIRED	 ACTUAL (COMMON ENGINE)	 DOUBLE FAILURE
BOOSTER	MANEUVER–NONE ATTITUDE CONTROL–REENTRY (R) –REENTRY (P&Y)	 1.0 °/SEC² 0.5 °/SEC²	 1.1 °/SEC² .5 °/SEC²	 56 °/SEC² .26 °/SEC²

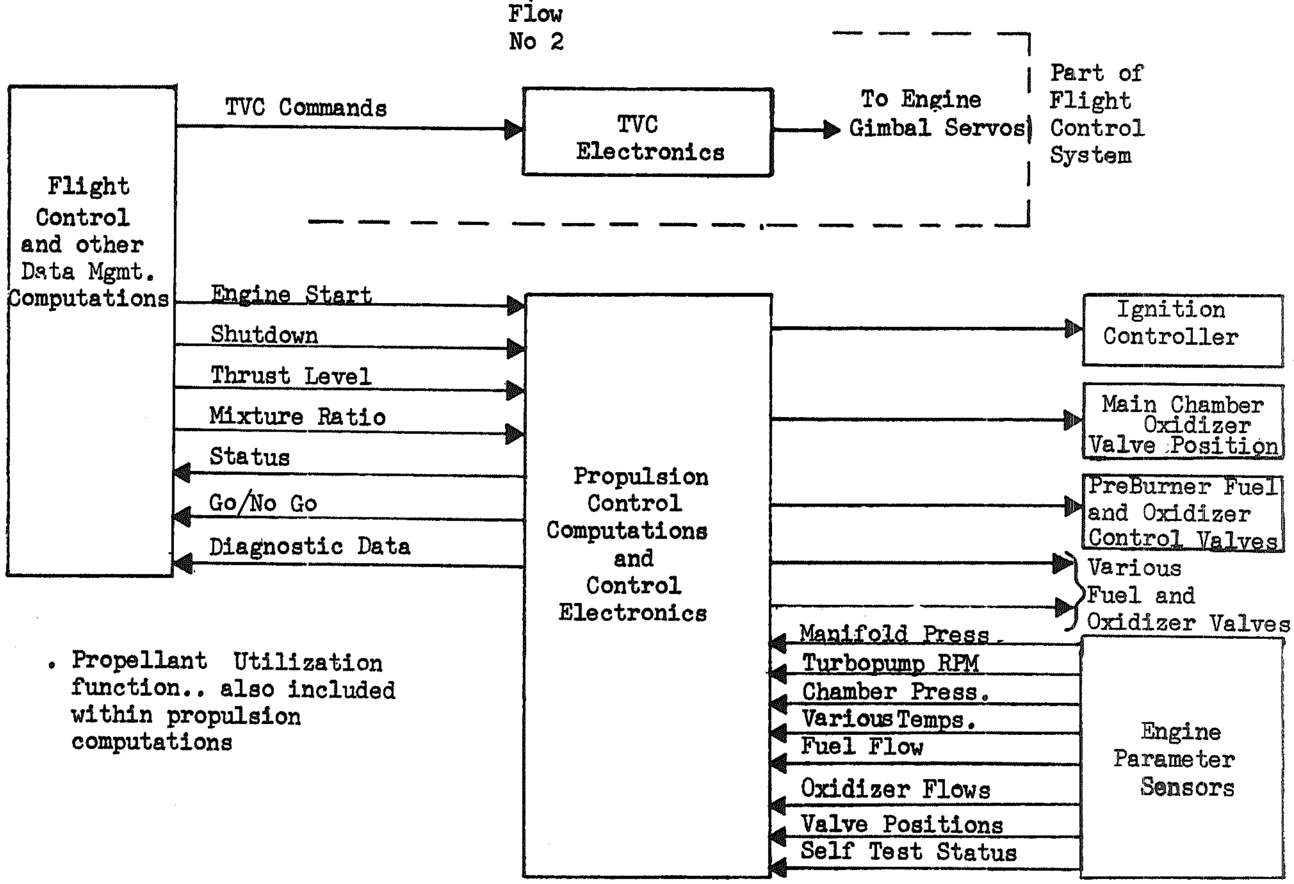


- PURE THRUST COUPLES ABOUT PRIMARY AXES: MINIMAL CROSS-COUPLING.
- START TANK PROPELLANT QUALITY UNEFFECTED BY MAIN TANK PRESSURANT COLLAPSE OR FEED LINE HEAT LEAKS.
- TURBOPUMP FEED SYSTEM: POTENTIAL MATCHING OF LH_2 PUMP FLOW RATE –ΔP CHARACTERISTICS WITH AIRBREATHING ENGINE REQUIREMENTS.

Figure 3-4
Delta Booster Auxiliary Propulsion

Figure 3-6 Typical Control Loop Main Chamber Oxidizer Valve Control





• Propellant Utilization function.. also included within propulsion computations

Figure 3-5 Basic Block Diagram Main Engine Controller

Engine and propellant system parameters that are used by the propulsion control system computers include:

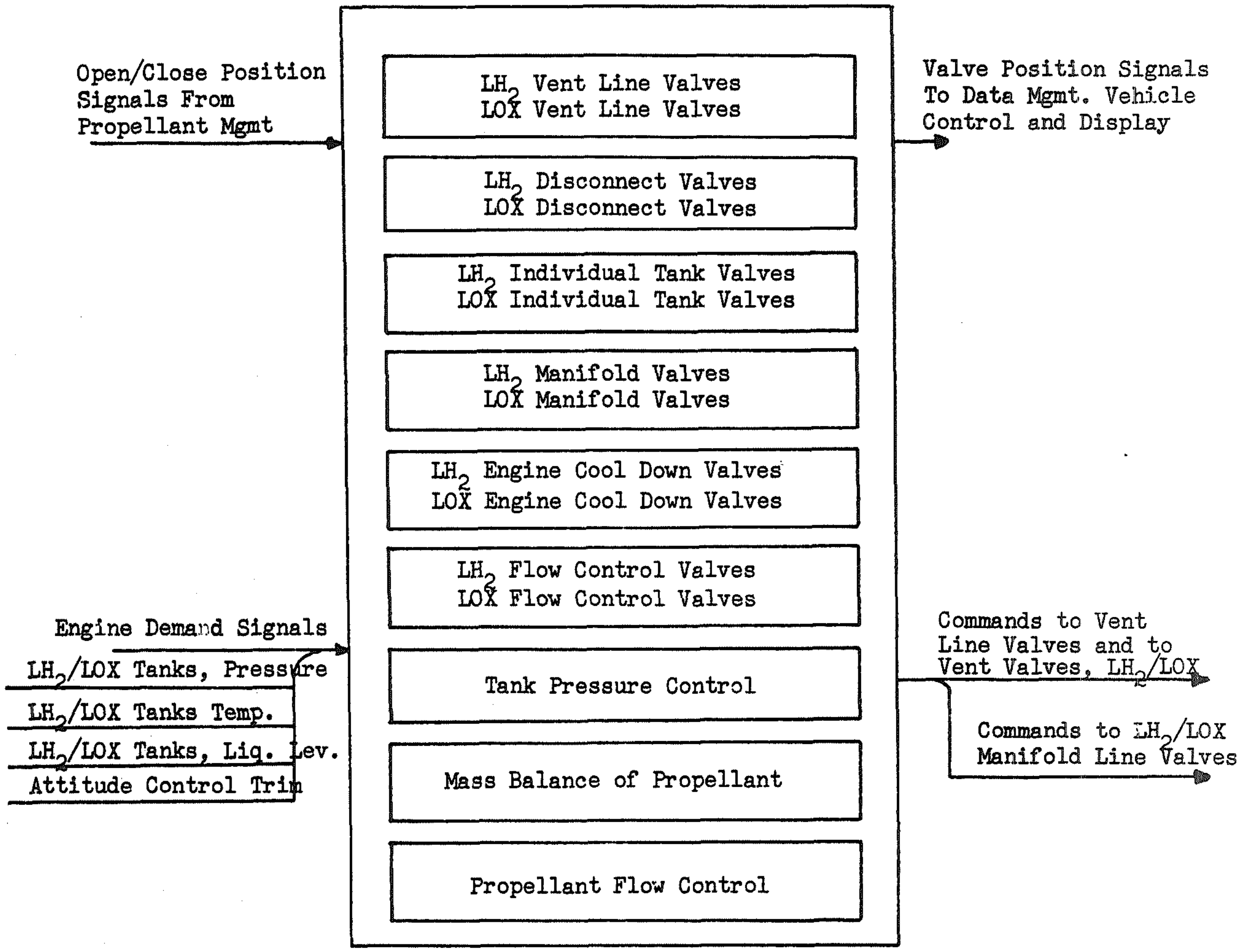
Nozzle coolant temperature	LH_2 level (main tank)
Main burner temperature	LOX level (main tank)
Preburner chamber temperature	LH_2 temperature
Heat exchanger temperature	LOX temperature
Fuel chamber temperature	LH_2 pressure
Oxidizer chamber temperature	LOX pressure
Fuel pump pressure (high)	Oxidizer flow
Fuel pump pressure (low)	Fuel pump rpm
Oxidizer pump pressure (high)	Oxidizer pump rpm
Oxidizer pump pressure (low)	Ignition voltage
Fuel low	Ignition current

The use of some of this data for the propellant control and manangement is shown in Figure 3-7.

ACPS Control System - The firing of the individual attitude control thrusters is a function of the flight control system. That is, the interface with individual fuel and oxidizer control valves is contained within the flight control electronics. The propellant management and control functions are part of the Propulsion Control System. These functions are similar to those shown in Figure 3-7 for the main engine controls.

Air Breathing Engine Control System - The air breathing or cruise engines are high bypass ratio turbofans. Hydrogen derived from residual fuel in the main engine and ACPS is a candidate fuel but the propellant management problem is rather complicated. The scope of the cruise engine control problem, however, can be estimated from other vehicle requirements. Because of the complication involved in engine deployment, the engine controls will be fly-by-wire with no mechanical linkages from the throttle lever to the engines. A summary schematic of signal inputs to the engine controller and the various functions required for thrust control and performance monitoring is shown in Figure 3-8.

Figure 3-7 Propellant Control and Management Block Diagram



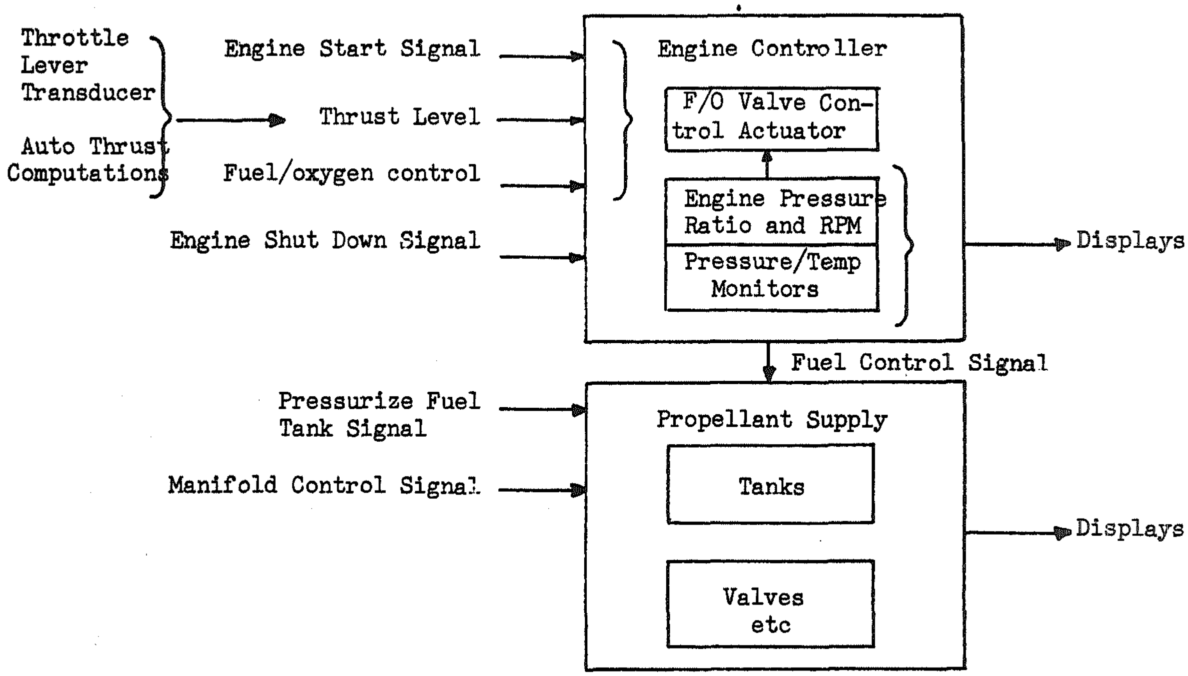


Figure 3-8 Air Breathing Engine Controller

3.3 ELECTRICAL POWER GENERATION AND DISTRIBUTION

The electrical power for the space shuttle booster could be supplied from different sources during different mission phases. These possible electrical sources are:

1. umbilical ground power
2. rechargeable batteries
3. APU driven by the rocket engine hot gas generator/turbine
4. APU driven by the turbojets
5. APU driven by a turbine and hot gas generator supplied specifically for electrical power.

One of the major parameters effecting the choice of electrical power generation is the method used to supply hydraulic power to gimbal the rocket engines and to move the aerodynamic control surfaces. The electrical power required by all of the electrical systems on the booster including the avionics system, controls and displays, lighting, etc., is approximately 5KW; the hydraulic pumps (if driven electrically) will require 50KW or more. Factors used in making electrical source tradeoffs in the Phase A reports vary greatly; as an example Martin Marietta uses a factor of 7 watt hours per pound in estimating the weight of AgO-Zn batteries required while McDonnell Douglas uses a factor of 55 to 60 watt hours per pound for the same battery.

The electrical power source assumed in this study is a hot gas generator fueled from the primary H_2-O_2 fuel source used to power a hot gas turbine which will drive a 115/208 volt, 400 hertz, alternating current generator and a hydraulic pump. For redundancy there will be two identical systems each feeding an essential AC bus. The essential AC buses feed non-essential AC buses and power converters to generate 28 volts DC which is fed to two essential DC buses and in turn fed to two non-essential DC buses. During pre-launch and ground check-out mission phases the electrical system will be supplied from ground power through an umbilical connection. Synchronization of all AC sources (i.e., umbilical power and redundant AC generators) will be maintained in order to eliminate excessive transients when switching from ground to vehicle power or from one to the other flight supply systems in event of a failure.

The DMS will derive its power from this supply source as well as continously monitor all functions of generation and distribution of electrical power. The DMS will initiate and monitor the supply of H_2 and O_2 to the gas generator, monitor the gas generator, turbine, AC generator, essential and non essential DC buses, and the supplied power at critical distribution points.

3.4 NAVIGATION AND GUIDANCE

The navigation, guidance and flight control systems combined function is to keep the vehicle upon the desired mission flight path. To accomplish this task the navigation system measures the vehicle's instantaneous position and velocity, the guidance system generates the desired position and velocity; and the flight control system controls the various forces on the vehicle in a manner to reduce errors between the actual and desired vehicle position and velocity.

3.4.1 NAVIGATION

The navigation system produces a major influence upon the size and speed of the DMS computer. The type of navigation solution required of the DMS is highly dependent upon the mission phase.

Strapdown Inertial Navigation - During boost the navigation sensor will be a strapdown inertial measuring unit (IMU). The IMU consists of redundant sensors capable of measuring the vehicle's instantaneous angular rate and translation acceleration in a body fixed coordinate system. For redundancy there will be six of each type of sensor (i.e., six single degree of freedom gyros and six linear accelerometers). Each instrument set will be mounted in a hexad configuration, i.e., the six instruments of each type will be mounted so that their sensitive axes are normal to the faces of a regular dodecahedron.

During flight, the outputs of the gyros and accelerometers must be processed such that the instantaneous vector value of the vehicle's velocity and position in inertial space is determined. This computational process is shown in block diagram form in Figure 3-9.

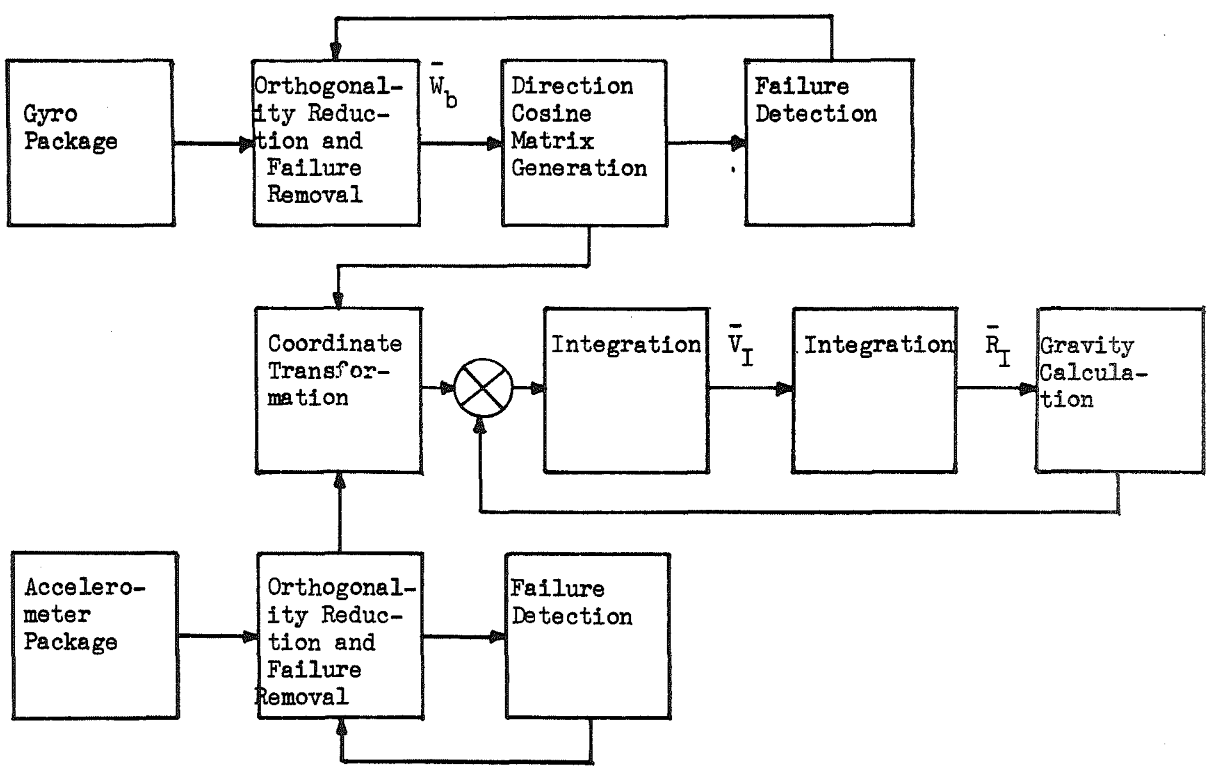


Figure 3-9 Strapdown Inertial Navigation Functional Flow

The outputs of the six single degree of freedom gyros are transformed to three orthogonal body axis rates ($\bar{W}_b$) after eliminating outputs of any failed gyro. The orthogonal body axis rates are processed to form a direction cosine matrix of the transformation between the body and inertial coordinate systems. Orthogonality relationships of the direction cosine matrix are investigated to determine which gyros, if any, have failed. The outputs of the accelerometers which pass the tests are transformed to form a body axis orthogonal set. Subsets of the body axis orthogonal set are investigated to determine any failed accelerometers. The orthogonal body axis accelerations are transformed to inertial axis by multiplication by the direction cosine matrix developed from the gyro outputs. The acceleration due to gravitational forces is subtracted from the inertial acceleration vector and the resultant difference integrated to obtain inertial vehicle velocity. The inertial vehicle velocity is integrated to determine the inertial vehicle position. The inertial position vector is used to determining the gravitational accelerations.

The strapdown inertial navigation system is used throughout the flight mission. In addition to providing inputs for the direction cosine matrix generation the gyro system is used by the control system to obtain aircraft attitude. During boost the strapdown inertial navigation system provides vehicle three dimensional position and velocity for control of the desired guidance trajectory and determination of the engine cutoff point. During coast it provides attitude information from the gyros. The point of reentry into the earth's atmosphere can be approximated from the navigation outputs at the time of engine cutoff (the accelerometer outputs during coast will be zero except for instrument drift if pure torque couples are assumed for the attitude control system). During reentry the strapdown inertial navigation system will again determine the vehicle's three dimensional velocity and position vector. After the initial reentry phase has passed the navigation system will receive supplemental inputs from a radar altimeter. During the final reentry phase and cruise the strapdown inertial navigation system will be used to provide two dimensional position data (latitude and longitude) with the third dimension (altitude) being supplied by a radar or barometric altimeter. Radio navigation aids will be used to update the inertial navigation position data. During the landing phase the inertial navigation system will provide only short term position and velocity data being continously updated by landing navigation aids.

During mission ground phases the DMS must perform checkout, calibration, and alignment of the strapdown system. Checkout will involve measurements of the operation of each instrument and its torquing loop. Accelerometer or gyro normal failure modes will produce either a hardover or a zero output. Abnormal failure modes will appear as excessive drift rates or large scale factor changes. Calibration consists of measuring instrument drift rates and scale factors which must be accounted for during flight. Alignment is a significant problem in a strapdown system. The basic alignment problem in a strapdown system is to determine the initial transformation matrix between the vehicle body axis and inertial space. With the vehicle stationary in a launch position the strapdown gyros should only be measuring earth rate and the accelerometers only gravitational acceleration. Vehicle latitude, longitude, altitude, and attitude will be entered from a keyboard as determined from survey data. The transformation matrix will be adjusted until the instrument outputs coincide with the keyboard entries. This procedure is complicated by the vehicle not remaining stationary upon the launch pad and thus the instruments are measuring forces and motions in addition to earth rate and gravitation. These extraneous movements are caused by wind forces and the loading of fuel and passengers into the vehicle. These movements could be measured by accurate ground based optical equipment and compensated for in the DMS computer. This would require a data link with the ground either through the umbilical or an RF link. These are high frequency movements and could be removed from the sensor outputs with filtering in the DMS computer at the expense of increased alignment time requirements. For the purposes of this study it will be assumed that the ground based optical equipment will be used. This equipment will be available only at launch sites and not at local airports for use in alignment prior to ferry missions.

A strapdown inertial navigation system has several error sources. The primary sources of error are: alignment errors, gyro and accelerometer drift, gyro and accelerometer scale factor errors, communication errors, integration truncation errors, and computer roundoff error. Because of these errors the difference between the vehicle's true and measured position and velocity will grow with mission time. The mission boost phase is of short enough duration (approximately 3 minutes) that this error propagation will be acceptable to mission success. After the time for coast,reentry, and transition to cruise has elapsed the error propagation will be significant. These errors will have to be corrected, especially any significant error in altitude. This error correction will be accomplished by using other navigation instruments for a navigation update of the inertial system. One method of accomplishing this update would be through the use of a Kalman filter, however, it is assumed that such a complicated updating mechanization will not be required on the space shuttle. Kalman filtering is in general reserved for applications where pin point accuracy is required and ground based radio navigation aids are unavailable. In cruise the space shuttle navigation task is to determine the vehicle position with enough accuracy to allow the pilot to guide the vehicle into the desired airport approach pattern. To accomplish this the pilot will have available all military and civilian radio navigation aids and ground control facilities. Updating will be accomplished by predetermining which navigation input will be most accurate during each mission phase and through preselected weighting factors adjusting the inertial navigation direction cosine matrix and position and velocity vector integration results.

Air Data Instruments - The vehicle will contain sensors which measure parameters of the surrounding air mass. These sensors outputs are processed and used by various vehicle subsystems including the navigation system. The parameters determined from air data instruments for use by the navigation system are true air speed and barometric altitude. Barometric altitude is determined primarily from static pressure. Static pressure is measured as the absolute pressure of the air ported into a chamber where the port is on a wall of the chamber parallel to the vehicle velocity, i.e., the absolute pressure of the surrounding undisturbed air mass. The raw sensor data must be corrected for several error sources when computing altitude. These are installation and instrument anomalies which produce a nonlinear relationship between raw sensor data and true static pressures. These anomalies are removed by curve fitting empirical data developed from flight tests. Secondly, air flow patterns around the vehicle, and thus the sensed static pressure, vary with vehicle Mach number and angle of attack. These relationships are also measured empirically from flight test data. True static pressure is then obtained from a polynomial curve fit function of raw static pressure, sensor input, Mach number, and angle of attack. Altitude is then computed as a logarithmic function of true static pressure. Mach number and angle of attack are computed from other air data instruments and true static pressure. Mach number is a complex function of the ratio between true dynamic pressure and true static pressure. Raw data used in determining dynamic pressure is derived from the pressure in a chamber supplied by an open tube (pitot tube) pointing directly out of the nose of the vehicle. This raw data is used in a polynomial curve fit function to derive true dynamic pressure. Angle of attack is sensed by a cylindrical probe in the airstream having two slots spaced 90° apart and oriented such that equal pressures occur behind the two slots when the vehicle is at zero angle of attack. Pressures are unequal when the angle of attack is not zero. Angle of attack is then computed as a function of these two pressures. True air speed is a function of Mach number and indicated temperature. Indicated temperature is the temperature of the air in the chamber supplied by the pitot tube True air speed is then computed as a simple function of indicated temperature and Mach number. If true air speed is to be used to update the inertial navigation system an estimate of wind velocity must be obtained. This estimate will have to be supplied by the pilot through a keyboard entry. The pilot will have to receive wind velocity data through ground communication from a weather station.

In the case of a total failure of the inertial navigation sensor the air data inputs will be used to provide dead reckoning navigation data. The DMS will supply the reduction of all raw sensor data to useful air data parameters; will provide the inertial navigation updates; will compute dead reckoning navigation; and will provide inflight and ground checkout of the air data sensors.

Magnetic Flux Gate Compass - During the cruise mode, magnetic heading will be obtained from a flux gate compass. A flux gate compass operates by using a vertical seeking gyro to stabilize a soft iron core in a horizontal plane. The earth's magnetic field within the soft iron core is modulated by cyclically driving the core in and out of saturation. Secondary windings on the soft iron core are oriented such that they couple only with the modulated earth magnetic field and provide a typical synchro output. The DMS computer will interface with the synchro output and through an arctangent calculation obtain magnetic heading. Magnetic heading can be used to update the inertial navigation system if the local magnetic variation is known. There are two ways in which magnetic variation can be provided to the DMS computer. A table of magnetic variation as a function of vehicle location could be stored in the computer. This could require a significant amount of computer memory if magnetic variation was to be stored for the entire earth with good accuracy. A disadvantage of this method is that magnetic variation is continously changing and thus must be periodically updated within the computer. The other method available for supplying the DMS with magnetic variation data is to allow the pilot to periodically insert the data during flight.

It is assumed that for the space shuttle application a combination of the available methods will be employed. An accurate magnetic variation table will be stored in the computer covering that portion of the earth in which the expected mission will occur. This table will have to be changed periodically to adjust for changes in magnetic variation and will have to be changed if the mission launch or landing sites change. An additional table will be stored containing coarse magnetic variation data for the entire earth. This table will have to be changed less often due to changes in magnetic variation. The DMS will automatically choose the data from the more accurate table unless the vehicle is outside the normal mission area. In addition to the two tables, the pilot will be given override capability, allowing for the insertion of magnetic variation values through the keyboard.

Other interfaces between the DMS and the magnetic flux gate compass include those required for checkout and correction of the vertical seeking gyro.

TACAN Receiver - The Tactical Air Navigation System (TACAN) gives continuous range and bearing from the vehicle to a TACAN ground station. The TACAN ground station transmits a reference signal and variable signal. These two signals are controlled such that the phase of the variable signal at the time of transmission of the reference signal pulse is proportional to the bearing from the receiver to the transmitter. The TACAN receiver automatically converts the received signal to bearing which will be read by the DMS computer. By knowing magnetic variation and the location of the transmitting station this data can be used for inertial navigation updating or dead reckoning navigation.

There are 126 frequency channels between 1025 and 1150 MHz assigned for use by TACAN stations. The useful range of a TACAN transmitter is 195 nautical miles. Channels are assigned to TACAN stations such that they will not interfere with one another. By knowing the channel selected and approximate vehicle position the transmitter station location could be found from a look up table stored in the DMS computer. It is conceivable in a fully automated system that the DMS could automatically select the TACAN channel. Because the TACAN receiver will be used primarily as a backup instrument providing navigation data to the pilot in the event of inertial navigation or DMS failure, it is assumed that automatic channel selection by the DMS does not exist for the shuttle booster. The DMS will be provided with the channel selected by the pilot and automatically determine the coordinates of the transmitter from a look-up table computing bearing to the station and to any selected route point.

Another function of the TACAN system is to determine range to the transmitting station. To accomplish this, the vehicle transmits an interrogation pulse which is transponded by the ground station. The time between the transmission of the interrogation pulse and its return to the vehicle is proportional to the slant range between the vehicle and the ground station. In order to make TACAN range determination simultaneously available to more than one vehicle, the transmitted interrogation pulse repetition rate is randomly varied. The vehicle's TACAN range determination equipment operates in two modes: Search and Track. In the search mode, initially a time delay interval is set. After transmission of the interrogation pulse and waiting the set interval, the return signal is strobed and a determination made as to the presence of the transponded pulse during the strobe. This process is repeated a fixed number of times after which the ratio of returns received to transmissions sent is formed. If this ratio is low the time delay interval is lengthened and the process repeated until an acceptable ratio is obtained. After receiving an acceptable ratio the track mode is entered. During track, the time delay interval is continuously adjusted to maintain the return pulse in the center of the strobe, thus tracking the return pulse. The tracking velocity is maintained even through short periods of return signal loss. During track, slant range is computed. In a typical TACAN receiver the search and track functions are mechanized along with a slant range and status display capability. For the space shuttle application, the DMS will have control over setting the time delay interval, computing the ratio of returns to transmissions, controlling the selection of track and search modes, and setting of the tracking velocity. The DMS also performs checkout and monitoring of the TACAN system.

Landing Systems - The minimum visibility and ceiling criteria for landing produces a large influence upon landing system equipment requirements and DMS computational requirements. The primary space shuttle mission is to provide logistic support for a manned orbiting space station. In an emergency situation, an immediate launch, regardless of weather conditions at possible landing sites, may be required. This assumption implies that the booster and orbiter have the capability of all weather landing, i.e., zero-zero visibility and ceiling. Adding the requirements that all major commerical and military air fields should be available for use by the space shuttle vehicles restricts the ground equipment available for all weather landing to that used or planned for use in the near future at these air fields.

Automatic landing systems require navigation accuracies of a few feet at touchdown. Present inertial navigation systems are incapable of these accuracies and thus must have numerous high accuracy updates during the final approach and landing phases. The ground systems considered for providing automatic landing capability are

- ILS (Instrument Landing System)
- AILS (Advanced Instrument Landing System)
- GVC (Ground Vector Control Systems)

The ILS operates by broadcasting an RF beam emanating from the end of the runway directed down the extended runway center line and vertically at an angle of 2.5 to 3 degrees from the horizontal. The transmitters used in forming the beam are modulated in a manner which allows the aircraft to determine if it is above, below, right, or left of the beam. Two marker beacons are used with ILS to inform the aircraft when it reaches 4 to 7 miles of the runway end and when it reaches approximately 3500 feet of the runway end. Inaccuracies in the beam position are generated from topographical effects and other aircraft in the vicinity. Beam distortion from these effects can be large. This distortion becomes particularly bad as the runway edge is approached due to parabolic distortion of the glide slope beam. It is at this point, right before touchdown that accuracy is most important. The present ILS system does not meet FAA standards for zero-zero landings and it is not anticipated that it will meet these standards within the space shuttle time frame.

GVC requires a sophisticated ground based radar and data link system for its operations. The ground based radar accurately measures the vehicle's position and velocity with respect to the runway position. This data can then be either transmitted directly to the booster for use as navigation updates and thus allow the DMS to compute accurate landing navigation and guidance, or the data can be processed in a ground base computer generating commands for use by the booster's control system and these commands then transmitted to the booster. This system has been successfully flight tested and in a modified form is operational on Navy carriers. Its availability at commercial air fields during the space shuttle time frame is not anticipated, however. For this reason and because the AILS system will meet the requirements of zero-zero landing, the GVC system is rejected as a possible candidate for the space shuttle automatic landing system.

The AILS is similar to the ILS system in that a beam emanating from the end of the runway is used. The beam accuracy is much greater than that used with ILS. Transmitted with the beam is data which can be used to accurately determine where within the beam the vehicle is located. The beam is generated from two ground based antenna scanning arrays operating at frequencies from 15.4 to 15.7 GHz. Elevation and azimuth scans are not simultaneously transmitted. Each are transmitted for 1/30th of a second at a 5 per second repetition rate. During the 1/30th of a second transmission period for azimuth and elevation the transmitted beam sweeps a 10 degree arc. The sweep is from 10 degrees to 0 degrees for elevation and 5 degrees left to 5 degrees right of the runway centerline for azimuth. The booster will receive the transmission only as the beam sweeps by it. The transmissions are coded to enable the receiver to distinguish between azimuth and elevation scans. The transmission consists of a series of pulse pairs. The time between the two pulses forming the pulse pair is a constant value of 12 microseconds for the elevation scan and 14 microseconds for the azimuth scan when the aircraft is left of the runway center line and 10 microseconds when right of the center line. In elevation the time between each pulse pair group is 40 microseconds when the elevation angle is 0 degrees and increases by 8 microseconds for each azimuth degree right or left of the center line up to a maximum of 5 degrees in either direction. AILS also includes Distance Measuring Equipment (DME) operating at the same frequency. The DME function can be interrogated during the 1/30th of a second following the azimuth scan. DME returns are coded pulse pairs with the separation code equal to 8 microseconds.

Because of its expected availability and accuracy it is assumed that the AILS will be the system used for automatic and semi-automatic pilot assist landings on the space shuttle booster. In the normal automatic mode of operation TACAN assisted inertial navigation will guide the vehicle into the AILS acquisition window approximately 4 to 8 nautical miles from the runway. Upon acquisition the AILS system will provide positional and velocity data to the avionics system. The inertial navigation system will be updated by the AILS derived data. The inertial navigation system will function as a smoothing filter and a backup sensor in case of AILS loss of signal or failure in this mode.

Because a critical portion of this function is ground equipment, the Fail Operational/Fail Operational/Fail Safe requirements for this mission phase will have to be deleted. All other navigation equipment on the vehicle will be used to continuously monitor the AILS system duringthis critical mode with automatic abort procedures immediately initiated upon any indication of AILS failure.

The DMS shall provide the computations of coordinate transformations, position and velocity determination, AILS acquisition, final approach control, flare control, decrab-control, abort procedures, in-flight monitoring and checkout.

Though it is anticipated that AILS will be installed in major airports in time for use by the space shuttle, it can be anticipated that there will be many airfields capable of handling a space shuttle booster landing which will not be equipped with an AILS but will have an ILS. In order to assist the booster in landing at these airports under poor weather conditions, a normal ILS will be included on the space shuttle. Functionally the DMS-ILS interface will be the same as the DMS-AILS interface during the initial approach phase. When making an ILS approach, if the pilot does not take over the landing control before category II minimums (1200 feet visibility and 100 feet altitude) are reached, the DMS will initiate an automatic go around.

Radar Sensors - There are three general categories of radar sensors which can be considered for use on the space shuttle booster; altimeter, doppler radar, and weather avoidance radar. Without a radar altimeter the only sources of altitude on the space shuttle booster are the inertial navigation system, the pressure altitude from the air data system, and the AILS during the final landing phase. After reentry the inertial navigation system has operated for a sufficient period to have accumulated drift errors large enough to make the inertial navigation altitude undependable. Pressure altitude will be of sufficient accuracy to perform the reentry and reentry-cruise transition flight phases. Air Traffic Control (ATC) procedures demand that a constant pressure altitude be maintained during cruise in order to insure vertical separation from other aircraft. During the approach phase the pressure altimeter can be set to local barometric pressure which will give good correlation between absolute and pressure altitude if the atmospheric conditions match the standard atmosphere mode. It is possible under certain atmospheric conditions for pressure altitude errors to be large enough to cause insufficient ground clearance during approach. Because the space shuttle must have all weather landing capabilities, it is assumed for this study that a radar altimeter will be included in the avionics system. A radar altimeter consists of a radar transmitter and receiver mounted in the belly of the aircraft. The transmitter radiates a cone shaped beam with a 20° half cone angle. Distance is computed from the time between the transmitted pulse and the first received return. This provides a vertical altitude measurement over smooth terrain for vehicle attitude variations up to ± 20° pitch or roll. The pilot will be given the provision of inserting local terrain elevation so that the radar altimeter output can be used for inertial navigation updates. The DMS interfaces with the radar altimeter for the purpose of inertial navigation update, approach trajectory control, checkout, and operation monitoring.

Doppler radar systems perform the task of dead-reckoning navigation by measuring the average velocity of the aircraft with respect to the terrain. Velocity is determined by measuring the doppler shift on several radar beams directed at different angles on the terrain. Aircraft heading must be included with the doppler velocities in order to determine position and velocity in an earth fixed coordinate system. The doppler system could only function at those flight phases starting with the end of reentry through landing. The accuracy of the doppler system is less than the inertial navigation system though using both systems together would result in better accuracy than either could give alone. The major advantage gained from the inclusion of a doppler radar navigation system would be the addition of functional redundancy to the self contained navigation system during cruise. This is not believed to be a great enough advantage to require the inclusion of the doppler radar system. The most critical period for self contained navigation is during boost. The doppler system is not capable of providing adequate navigation data during boost, thus the inertial navigation system will be mechanized to meet the fail operational/fail operational/fail safe redundancy criteria. Adequate redundancy and reliability are thus built into the inertial navigation systems such that it can be assured that operation will be maintained during cruise. During cruise other navigation aids such as TACAN, the Air Data Computer, magnetic flux gate compass and ATC communication are available, making any self contained navigation system, either inertial or doppler radar, a non-critical system for mission success. For these reasons it is assumed that the space shuttle booster will not contain a doppler radar navigation system.

There are several mission criteria of the space shuttle booster which dictate the inclusion of a weather radar. A weather radar will allow the booster during the cruise mode to fly between thunderstorm cells thus increasing the number of available landing fields during bad weather conditions. The launch window is necessarily constrained by the space station orbital parameters and launch site weather conditions. To, at least partially, remove booster reentry area weather condition constraints from the launch window could mean the difference between launching and not launching an emergency recovery or servicing mission. It is, therefore, assumed that the booster will contain a weather radar system. A weather radar includes an X or C band transmitter and receiver with a parabolic gimbaled dish antenna, capable of forming a pencil beam. The radar returns are processed through two threshold circuits with the outputs of both thresholds displayed on a typical PPI display. The resultant displays shows storms as bright areas on the screen surrounding dark holes representing areas of severe storm activity. The DMS interface with the weather radar will be to initiate and assist checkout, to accumulate checkout results, to monitor the signal processing electronics during operation and to provide control commands to the antenna gimbaling system.

Miscellaneous Flight Instruments - In addition to the navigation instruments already described, there are other flight instruments included which act as backup instruments in the event of a major system failure such as aircraft electrical power on the DMS. These include such common flight instruments as the turn and bank indicator and the artifical horizon. The DMS will interface with these instruments only for the purpose of checkout and operation monitoring.

3.4.2 GUIDANCE

The space shuttle booster will use four different guidance programs during different mission phases. These are boost, reentry, cruise, and landing guidance.

Boost Guidance - There are numerous constraints upon the selection of the guidance equations for the boost phase. Some of the major considerations influencing the boost trajectory are:

1. Meet mission objectives - The primary mission objective assumed for this study is the rendezvous of the space shuttle orbiter with the orbiting space station. With this objective in view the boost trajectory must have terminal altitude and velocity conditions which will allow the orbiter to assume the space station's orbit at the correct time for rendezvous.

2. Range safety objectives - If a catastrophic failure occurs during boost resulting in a crash of the booster and/or orbiter, the trajectory must be such that the crash site will not occur in any populated area.

3. Vehicle loading requirements - During the atmospheric portion of boost the trajectory selected must keep normal aerodynamic forces on the vehicle below a minimum value.

4. Return capabilities - After separation the booster coasts (i.e., is non-thrusting) until it reaches the atmosphere where it uses aerodynamic forces for braking and path control. After reaching the appropriate altitude and velocity the booster deploys turbo jet engines and cruises to a landing site. The total possible return area can be determined by starting at the point of entry into the atmosphere and developing all possible reentry trajectories and adding maximum cruise range to the reentry trajectory end points. This total possible return area is designated as the return footprint. The boost trajectory must provide the booster with a reentry point that will result in a return footprint containing at least one acceptable landing site. An additional constraint that could be added to the requirements is that an acceptable landing site exist in the much narrower return footprint resulting from the failure of the cruise turbojet engines.

Guidance equations bounded by the first three of the above constaints have been developed and flight tested on Apollo and Gemini missions. The major reason for the inclusion of turbojet engines and thus the cruise mode is to eliminate any major influence of the fourth constraint upon the boost trajectory. It is believed that the guidance equations for the boost trajectory will assume an evolutionary process as the space shuttle develops. Initially the guidance equations will be hand selected and finely tuned for each mission. The first missions will be designed for test purposes with the boost trajectory chosen so that a straight in reentry path will lead to a landing site optimally located for a booster landing under the condition of the turbojet engines failing. These trajectories and guidance equations will be developed on ground based computers and verified with thorough simulations. At the end of the evolutionary process it can be visualized that the pilot or flight crew will insert the orbital parameters of the space station plus the weight, fuel and any other special characteristics of the orbiter into the DMS computer. The DMS then selects the desired trajectory based upon some optimization scheme and generates a complete mission plan and profile including booster fuel requirements, launch time, and selected landing site and alternates. The DMS then enters the determined flight plan into the flight program. It is beyond the scope of this study to attempt to predict this evolutionary development to any degree of accuracy. It is necessary in this study to determine fairly accurate estimates of memory and speed allocations within the DMS for the solution of the guidance equations. As stated above guidance equations bounded by the first three constraints have been developed and used and that the fourth constraint has been effectively removed by inclusion of cruise engines. Section 5.4 gives the boost guidance equations which are the same set of equations used for the Saturn V Apollo mission guidance. This set of equations was selected because the Saturn V due to its size and construction has severe aerodynamic loading constraints. It can be anticipated that the space shuttle booster will also have severe aerodynamic loading constraints due to the aerodynamic surfaces necessary for the reentry and cruise mission phases and due to the large off-axis drift of the combined vehicle center of gravity caused by piggy-back attachment of the orbiter. Normal preflight initialization equations and time to launch equations are included;however, those preflight mission planning features visualized for inclusion at the end of the evolutionary guidance system development have not been included. This program could require up to 32000 words of memory (a typical size for similar functions now used on ground based systems) but do not require a real time solution and are not contemplated to require excess computer time for their solution. It has already been assumed that a mass memory storage device will be available for programs of this nature and thus this particular program in itself will not influence the DMS size or speed.

If boost is defined as that period during which the booster rocket engines are firing then the guidance function for boost is divisible into the following operational modes:

1. Prelaunch - During prelaunch the guidance equations are initialized and a continous countdown to engine start performed. When the countdown is zeroed the engines are ignited if the checkout function shows all systems are operational.

2. Ignition - Ignition is defined as that phase between the start of the ignition sequence until lift off. During this time the engines are started and engine thrust allowed to buildup before hold downs on the vehicle are released. The guidance system measures this time so that accurate measurements of thrust termination time can be transferred to the orbiter at separation time.

3. Liftoff - After the hold down mechanism is released and for the first few seconds of flight the guidance system commands a vertical flight and no roll. This is done in order to inhibit any maneuvering transients of the vehicle until all launch site obstructions such as the launch tower are cleared.

4. Roll Maneuver - As soon as the launch site obstructions are cleared the guidance system issues a roll command to align the vehicle pitch plane with the initial trajectory reference plane. The trajectory reference plane is initially determined from the space station orbital plane, the launch site location, launch time, and range safety constraints. As time progresses during flight this plane will rotate until it becomes coincident with the space station orbital plane. Rotation of the trajectory reference plane is caused by yaw steering after pitch over.

5. Pitch Over Maneuver - As soon as the roll maneuver is complete the guidance system commands a change in pitch attitude. The pitch trajectory during boost uses gravity forces to turn the vehicle velocity vector from the initial vertical direction toward the horizontal. This type of trajectory is called a zero-g trajectory with simplified vector equations of motion of:

$$\dot{\bar{V}} = (T/M - g \cos\gamma)\ 1_v + g \sin\gamma\ 1_n$$

$$\dot{\gamma} = g \sin\gamma \ / \left| \bar{V} \right|$$

1_v is a unit vector parallel to the vehicle instantaneous velocity, and 1_n normal to this velocity. It is assumed that the engine thrust (T) is always directed along the instantaneous velocity vector. From the equations it can be seen that without the pitch over manuver, $\dot{\gamma}$ would remain zero since $\sin \gamma$ is zero for vertical flight. The magnitude of the pitch over command is chosen such that the final values of V and γ are those desired for the orbiter to achieve rendezvous with the space station. The pitch over maneuver is completed as readily as possible. During the pitch over maneuver the vehicle develops an angle of attack producing normal aerodynamic forces on the vehicle. These aerodynamic forces must be kept small in order to avoid over-stressing the vehicle structure. The aerodynamic forces are proportional to the square of the vehicle velocity making it desirable to complete the pitch over maneuver before the vehicle velocity becomes large.

6. Atmospheric Boost - The above equations describing a zero-g trajectory are developed with two major assumptions; 1) there are no aerodynamic forces and 2) the thrust vector always points along the velocity vector. Aerodynamic forces are normally resolved into two vector components which are drag and lift. Drag is defined parallel to and in the opposite direction to the velocity vector. Lift is defined as being normal to the vehicle X body axis. It is the lift force which produces loads upon the vehicle which must be kept below an established value in order to keep the vehicle structure from becoming overstressed. The equation for the lift force is given by:

$$L = 1/2 \; pSC_{l\alpha} \; V^2 \alpha$$

where p is the atmospheric density, S the vehicle reference area (a constant), $C_{l\alpha}$ an aerodynamic lift coefficient, V total vehicle velocity with respect to the air mass, and α the angle of attack. The angle of attack is the only variable available for controlling lift once the trajectory has been established. Using the zero-g trajectory defined above would result in α remaining zero except for a residual angle of attack produced by the pitchover maneuver not occuring at zero velocity, any center of gravity offset, a thrust misalignment, or any atmospheric winds. There are two basic methods used to reduce loads. Normally the largest single contributor to loading is winds. By knowing the winds to be experienced by the vehicle at all altitudes, it is possible to modify the zero-g trajectory equations so that the angle of attack remains small. It is impossible to have exact knowledge of the wind velocity at altitude prior to launch. Loading can be significantly reduced however, by using measurements made shortly before launch. The second method used to achieve load relief is to measure either the lift force with a normal accelerometer or measure the angle of attack with the air data sensors and apply an attitude control command such that the sensed values are reduced. Angle of attack or normal acceleration can be reduced only at the expense of generating deviations from the planned zero-g trajectory. These errors in the trajectory can be removed after the booster is above the atmosphere where normal lift forces are no longer produced. It is assumed that the second method will be used for the space shuttle booster because wind measurements prior to launch require significant ground support capabilities and because thrust misalignments and center of gravity offsets will probably be significant. In a typical booster it is impossible to reduce the loads produced by a thrust misalignment or center of gravity offset because an angle of attack must be maintained to offset the torques on the vehicle produced by the misalignment or offset. With the aerodynamic surfaces available on the space shuttle booster the angle of attack can be reduced. In order to reduce short duration angle of attack buildups caused by wind gusts the load relief system must have a wide bandwidth. This wide bandwidth causes significant coupling between the load relief system and the attitude control system necessitating a change in loop gains and filter coefficients in the attitude control system when load relief is active.

7. Exo-atmospheric boost - When dynamic pressure has dropped to the point where loads are no longer significant the load relief system will be disabled. This point will be determined by measuring either velocity or altitude. At this time it is possible to tighten up the guidance trajectory control loop removing those errors generated by the load relief system during atmospheric boost. Engine cutoff will be based upon velocity measurements.

The boost guidance system must be formulated so that the terminal boost conditions can be achieved with large deviations from nominal flight conditions. For example, the booster will have clustered rocket engines. If one of the engines fails during flight there will be a discrete reduction in the vehicle thrust, and this can occur at any arbitrary point in time. The decrease in the thrust does not alter the total available energy available since the fuel for the failed engine will be available for use by the other engines. Because the rate at which the energy is available has been reduced, the trajectory shape must change. The guidance system must be capable of accommodating this change in thrust magnitude and automatically generate commands to produce the trajectory. Since the space shuttle mission is logistical support of a space station, payload weights may vary greatly from mission to mission; for example from a maximum design weight supply mission to an empty payload for an emergency mission to remove space station personnel. The guidance system should have sufficient flexibility to handle missions with extreme payload variations. Obviously, an explicit guidance scheme is required.

Reentry Guidance

Immediately after thrust termination the orbiter will separate from the booster. The booster attitude is then oriented so that the effects of impingment upon the booster of the orbiter's engine plumes is minimized. The guidance system is responsible for generating and maintaining this attitude command for sufficient duration to insure that the orbiter-booster separation is great enough for no further plume impingment. If the booster-orbiter RF data link is still available verification of sufficient separation can be obtained from received orbiter navigation output values. Upon completion of the separation sequence the booster guidance will command an attitude for reentry.

In order to visualize the reentry guidance function, motion in the trajectory plane and normal to the trajectory plane is considered separately. The trajectory plane is defined as the plane containing the center of the earth and the instantaneous vehicle total velocity vector with respect to inertial space. This plane will rotate with respect to inertial space if the booster turns during reentry. The only forces used to control the vehicle trajectory during reentry are aerodynamic forces on the vehicle. If roll and yaw remain zero (the vehicle pitch axis is maintained normal to the trajectory plane) then there will be no aerodynamic forces generated normal to the trajectory plane. The primary task of the reentry guidance system is flight path control generation in such a manner that the vehicle does not experience excessive heating rates or deceleration forces. The space shuttle booster will have non-ablative high temperature protection on its underside and leading surface edges only. During periods of high heating rates the vehicle's attitude must be controlled such that heating of the upper vehicle surface is minimized.

During the initial reentry phase heating rates build up much more rapidly than deceleration forces. Therefore temperature sensors are more sensitive in measuring the initial atmospheric conditions of reentry than accelerometers. The heating rate during reentry can be approximated by

$$\dot{Q} = K_1 \sqrt{p} \; V^3 \quad \frac{\text{BTU}}{(\text{ft}^2)(\text{sec})}$$

where $\dot{Q}$ is the heating rate per unit area, K_1 is a dimensionality constant, p is the free stream atmospheric density, and V the vehicle velocity with respect to the air mass. The aerodynamic forces on the vehicle are normally separated into the components of lift and drag which are approximated by the formulas

$$L = K_2 pV^2 \sin\alpha \cos\alpha$$

$$D = pV^2(K_3 + K_4 \sin^2\alpha)$$

where α is the vehicle angle of attack and K_2, K_3, and K_4 are constants. The constants K_2 and K_4 will be approximately equal while K_3 is much less than K_2 or K_4. The total deceleration force can then be approximated by

$$F \approx K_2 pV^2 \sin\alpha \approx K_4 pV^2 \sin\alpha$$

when α is large. Present plans call for the space shuttle booster to enter with large angles of attack of 45 to 60 degrees. Loading constraints dictate that F must remain below some maximum value which is determined either by vehicle structural strength or by human limitations. The heating constraint takes on two forms, first the total heat absorbed must be kept below a maximum value thus

$$\int_{t_o}^{t} (\dot{Q} - \dot{Q}_r)\, dt < Q\text{max}$$

where $\dot{Q}_r$ is the radiated heat rate from the vehicle which will normally be very small compared to $\dot{Q}$ and Qmax is the maximum allowable total heat absorbed. Second the heat rate Q must remain below a maximum value. If the heat rate is too high then various "hot spots" on the vehicle such as wing tips, nose, etc., will absorb heat faster than it can be conducted away to the cooler aircraft structure.

The flight path in the trajectory plane is controlled through the angle of attack. In a return from a ballistic trajectory, as in the case with the space shuttle booster, the entry angle (the angle between local level and the velocity vector at reentry) will be fairly large. Low angles of attack will cause rapid penetration of the atmosphere with low deceleration forces resulting in high heating rates. Large angles of attack, near 45°, will generate maximum lift forces causing the velocity vector to be turned toward the level. This reduces the atmosphere penetration, generates significant deceleration forces and reduces the velocity magnitude. This reduces the heating rate but extends the total heat exposure time and the vehicle range. Larger angles of attack, i.e., above 45 degrees, reduce the lift force causing again more rapid atmosphere penetration but at the same time increasing deceleration forces thus more rapidly reducing total velocity magnitude. These larger angles of attack will bring the deceleration forces close to or beyond the acceleration limit and will reduce the range.

Rotation of the trajectory plane is accomplished by rolling the vehicle. This causes the lift force to rotate out of the trajectory plane producing a force component normal to the plane. Vehicle yaw is commanded at the same time in order to maintain the velocity vector in the X-Z body axis plane and to coordinate the turn.

Reentry guidance systems are normally mechanized as a model system where a nominal trajectory path is propagated through a simulation program of the vehicle and atmosphere. This simulation occurs at speeds much more rapid than real-time. Each propagation through the model starts with initial conditions equal to the instantaneous position, velocity, and attitude of the vehicle. The model simulation then generates the end conditions of reentry and compares these with the desired end position, velocity and heading. Differences between the desired and simulated end conditions are used by simple guidance laws to generate changes in the vehicle attitude commands.

Reentry guidance is terminated when the vehicle reaches cruise altitude and cruise velocity at which time the turbojet engines are deployed and started.

Cruise Guidance

Cruise is characterized by flight legs of constant altitude, heading and velocity starting at a point where reentry ends and ending with the beginning of approach descent to the landing field. The route flown during cruise will be determined before launch. The points along the route, where heading or altitude change, will be entered into the DMS computer prior to launch. These points are defined as steer points. At the beginning of cruise the DMS will compute the constant heading route between the desired reentry end point and the first steering point. It will then issue attitude commands to fly the vehicle to the computed route. It will compute constant turning rate curves so that a smooth transition to the desired route path is achieved. As it nears each steer point it will compute a smooth constant rate turn to the constant heading route to the next steer point until the point for approach descent to the airfield is reached. As the vehicle progresses along the route the computer will continuously determine the estimated time to the next steer point and the estimated time of arrival at the airfield and issue these values for display.

Also in the table will be points where the vehicle is most likely to be requested to enter a holding pattern. The pilot may select one of these points and the holding pattern function via keyboard entry. The guidance system will automatically cause the vehicle to enter the holding pattern and remain there until instructed to leave by pilot instruction through the keyboard. The DMS will also compute range to the airfield and using values of vehicle velocity, wind velocity, fuel consumption rates and predicted rates during other route legs and landing determine an estimate of fuel reserves at landing. Landing fuel reserve estimates will be displayed and a warning issued if they drop below safety minimums.

The pilot will be able to modify the route by inserting, deleting or changing any steer point in the table. The table will also contain routes to alternate fields which the pilot can select at any point. In addition the pilot will be able to change the computation from constant heading paths (these are normally specified by air traffic control) to great circle routes in order to choose a shorter range path. The pilot will also be able to select one of several velocity control functions. He will be able to select maximum cruise velocity, maximum range velocity or short periods of increased or reduced velocity so that he may phase his vehicle into the route with proper spacing from other vehicles flying the same route. Stored with each route leg is an altitude which will be maintained by the guidance system. When altitude changes are commanded either automatically at steer points or by the pilot the guidance system will command a smooth transition to a constant ascent or descent rate.

Landing Guidance

The landing phase can be divided into approach, final approach, flare, decrab, and touchdown. The approach mode consists of a constant descent rate from cruise altitude to the final approach window at 4 to 8 nautical miles from the runway end. During this time the vehicle turns onto the horizontal flight path defined by the extended runway centerline, and reduces its air speed to approximately 1.3 times stall speed. Entry into the final approach window is defined as the point where reliable AILS signals are received. If AILS signals are not received by the time other navigation equipment indicates a minimum altitude or distance from the runway end, the pilot is given a warning. If the warning is not responded to by the pilot taking over control of the vehicle within a prespecified time the guidance system will initiate an automatic fly around maneuver. Upon reception of reliable AILS signals the guidance system will follow a zero azimuth and a 2 to 3 degree glide path as determined from the AILS system. At 500 feet ground range and 50 feet altitude the flare phase will be initiated slowing the descent rate to 2 to 3 feet per second. If a cross wind exists the vehicle will be flying at a crabbed attitude, i.e., the angle between the vehicle velocity vector and x body axis will not be zero in yaw. At 20 feet altitude, approximately 3 seconds from touchdown the guidance system will issue a yaw command to zero the crab angle. At touchdown, as determined by sensors on the landing gear, the guidance system will be disabled and control given to the pilot. It will be assumed that automatic taxi will not be available on the space shuttle booster.

3.5 FLIGHT CONTROL

The flight control system receives attitude and/or attitude rate commands from the guidance system or from pilot controls and converts these to commands to the rocket engine vectoring actuators, reaction jet solenoid valves and the aerodynamic surface actuators. The flight control systemcan be functionally separated by its performance during boost, coast, reentry and cruise. This primarily separates the means available for generating attitude control torques, i.e., during boost torques are obtained by thrust vectoring and possibly aerodynamic surfaces, during coast by reaction jets, during reentry by reaction jet/aerodynamic surface mix, and during cruise by aerodynamic surfaces.

Boost Flight Control

There are several factors affecting the boost flight control system which cannot be determined until the booster/orbiter configuration have been established and thoroughly analyzed. Figure 3-10 shows various pitch plane coordinate systems, vectors and angles required to describe the boost flight control function. The coordinate system X-Z is an inertial set with the Z axis locally vertical at the launch site. The coordinate system X_b - Z_b is fixed to the booster body centered at the combined booster/orbiter center of gravity. The vector V is the vehicle's velocity with respect to the inertial coordinate frame; for the purposes of flight control system analysis it is generally assumed that the earth is spherical and non-rotating. The angle of attack, α, is the angle between the vehicles X_b body axis and the velocity vector. The force F_n is an accumulation of all aerodynamic forces on the vehicle that are normal to the X_b axis. This force is proportional to the angle of attack, α , and goes to zero when α is zero. The accumulated aerodynamic force, F_n, does not necessarily act through the center of gravity, but through a point on the X_b axis defined as the center of pressure (cp). The center of pressure on a booster having no wings and small fins or skirts is forward of the center of gravity. This has been the case with most large boost vehicles such as the Saturn V, Atlas, and Minuteman. The force F_n produces a torque on the vehicle which is destabilizing if the center of pressure is forward of the center of gravity. Destabilization produces a torque on the vehicle in a direction to cause the angle of attack to increase. The booster is designed for stable flight during cruise meaning that the center of pressure will be behind the center of gravity during cruise. The center of gravity during boost cannot be predicted until the boost configuration is known. The orbiter when attached to the booster will change the combined vehicle cg location. The orbiter will shade some of the booster aerodynamic surfaces and add aerodynamic surfaces of its own, which will affect the center of pressure location. Also during boost the fuel and oxidizer in the booster tanks will effect the cg location.

The vector T is the force vector from the thrusting rocket engines and is at an angle of β from the X_b axis. The angle Θ is the angle between the X_b axis and the X inertial axis and is the angle commanded by the guidance system during boost. In order to size the impact of the boost attitude control system on the DMS an assumption will have to be made as to whether the booster's and/or orbiter's aerodynamic control surfaces will be used. There are numerous advantages and disadvantages to using the aerodynamic control surfaces during boost. The linearized moment equation resulting from only those forces shown in Figure 3-10 is

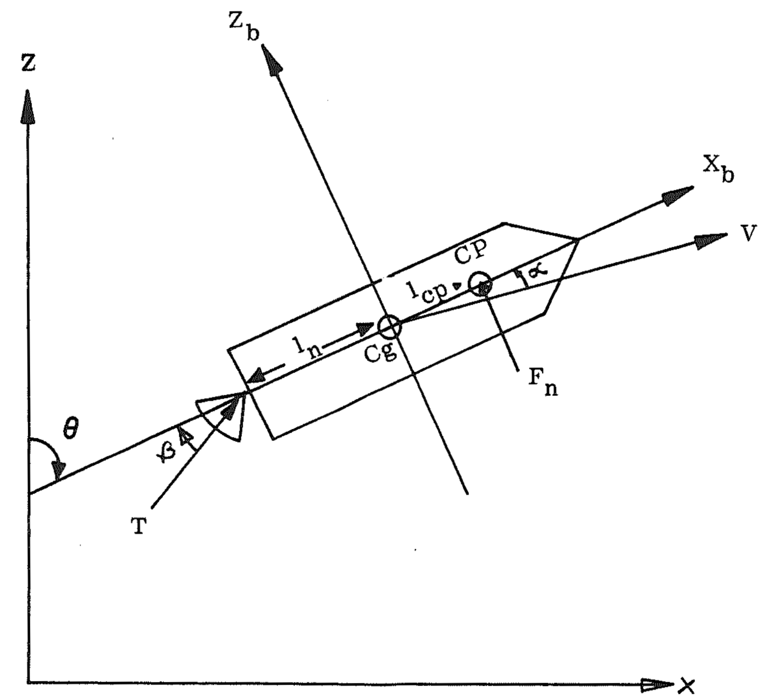


Figure 3-10 Pitch Plane Flight Control Variables

$$I\ddot{\theta} = -qS1_{cp}\, C_{m_{\alpha}}\, \alpha + T1_n\beta$$

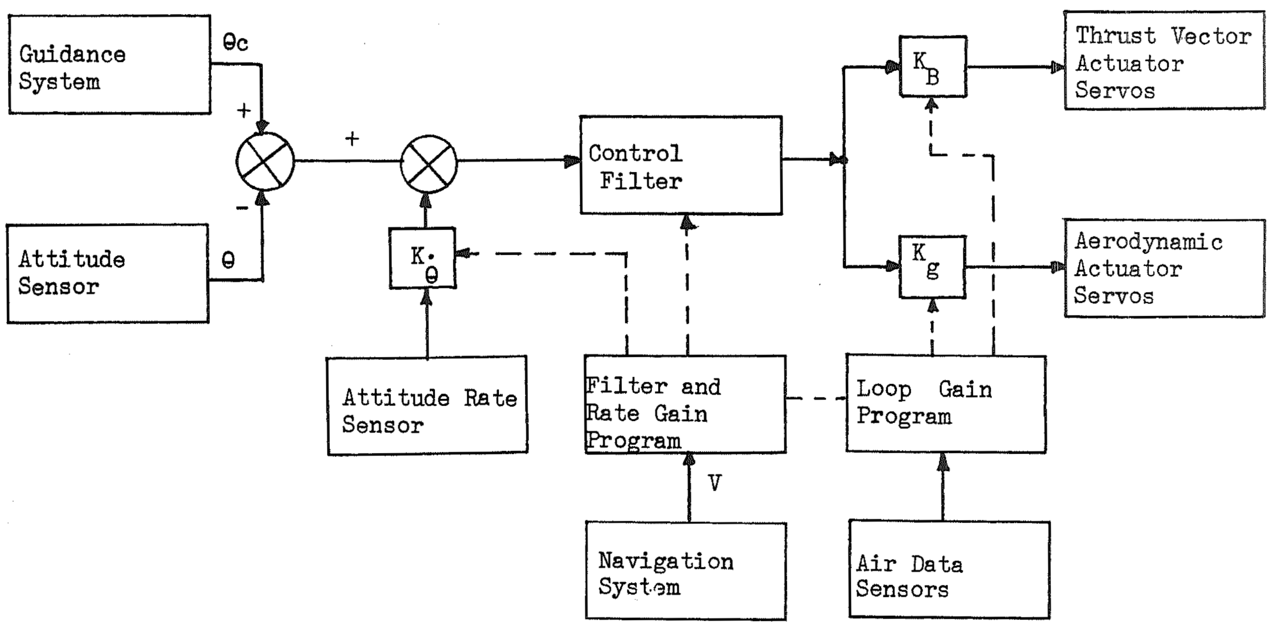
where $qSC_{m_{\alpha}}\alpha$ is F_n and q is dynamic pressure, S reference area and $C_{m_{\alpha}}$ an aerodynamic coefficient. The dynamic pressure, q, goes through a large variation during the boost flight being zero at lift off, reaching a maximum some time after the velocity reaches Mach 1 and decaying back to zero as the booster leaves the atmosphere. This means that the control system loop gain must also vary in order to produce torques to counteract the aerodynamic torque. The control effectiveness of aerodynamic surfaces will also be proportional to dynamic pressure providing an inherent gain variation in the control loop. If only thrust vector control is used this gain variation will have to be programmed within the control loop. The gain variation programming could be an open loop function of velocity or air data sensor measurements or could be an adaptive system where the gain variation is based upon the immediate dynamic response of the vehicle. Thrust vector control is required when dynamic pressure is zero or near zero. If aerodynamic control is used, the ratio of thrust vector and aerodynamic control will have to be programmed during boost in order to maintain proper stability margins. The linearity of the control obtained from aerodynamic control surfaces may depend upon the aerodynamic shading of the control surfaces by the attached orbiter vehicle.

There are two unique features of the space shuttle booster that have never previously been present on any large boost vehicle. These are the aerodynamic control surfaces and a complete air data sensor system. It would be unrealistic to assume that the final control system design would not take advantage of these unique features. A block diagram of the boost pitch axis control system is shown in Figure 3-11 . The figure shows the control loop gains K_B and K_g being changed through a loop gain program driven as a function of air data sensor outputs.

To this point the discussions of the boost control system have assumed that the boost vehicle is a rigid body. The boost vehicle will not be a rigid body but will bend in response to aerodynamic and thrust forces. Figure 3-12 shows typical vehicle normalized bending curves for the first three bending modes and a typical bending time response due to a step force input. The curves on the left, A, C, and E, represent typical deflections of the vehicle from the rigid body centerline (represented by the X axis) along the length of the vehicle. The total instantaneous deflection of the vehicle at a body station is obtained by adding together the deflections from each curve multiplied by their appropriate scale factor. The response of each bending mode to a step force input, e.g., a step thrust deflection, is shown in curves B, D, and F, as a poorly damped sinusoidal response to a level. The second bending mode frequency will be approximately twice the first bending mode frequency and the third bending mode frequency approximately three times the first.

In order to stabilize the attitude control loop andobtain the desired short period response, attitude rate feedback is required. This rate feedback could be obtained by differentiating the attitude gyro outputs. Because of the physical size of the attitude gyros (there will be six single degree of freedom attitude gyros for redundancy purposes) physical constraints regulate where the gyros can be mounted. The pitch gyro output will measure the attitude

Figure 3-11 Boost Pitch Attitude Control System



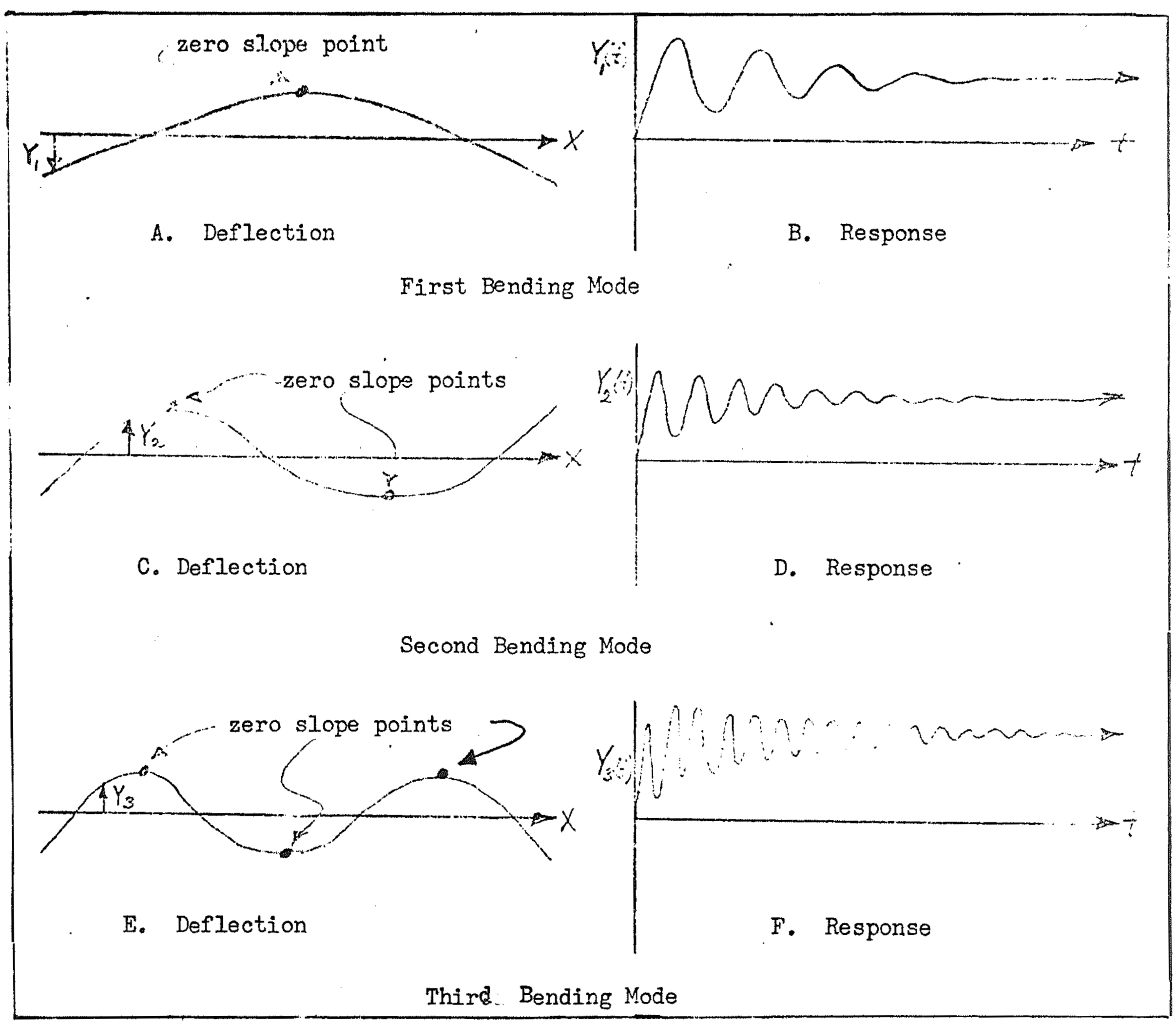


Figure 3-12 Typical vehicle bending mode deflections and response

of the rigid body X axis with respect to inertial space and also will measure the slope of the bending deflection at the body station where the gyro is mounted. If the sinusoidal portion of the bending response in the gyro output is

$$A \sin w t$$

then in the differential gyro output this signal will change to

$$A w \cos w t.$$

Bending frequencies are generally larger than one radian/second, thus more bending is fed into the control loop through the differentiated gyro output than through the direct gyro output. The control system filter and actuator servos will generally cause significant lag phase shift at the bending frequencies. The combination of lag phase shift and gain through the rate loop on bending signals is generally sufficient to drive the bending modes unstable.

For a large booster it is difficult to correct this stability problem by either correcting the filtering phase shift or by adding a filter to band reject bending frequencies. The first bending mode frequency is generally 2.5 to 5 times the desired short period frequency at boost. This frequency increases as the boost flight phase progresses due to the total vehicle mass decreasing as fuel and oxidizer is expended. Any filter with a wide enough phase or gain characteristic to properly control the vehicle bending over its entire range of variation will adversely interfere with the short period response. Narrow notch filters which track the bending mode frequencies are not feasible if tracking is controlled as a function of time or vehicle velocity due to preflight uncertainties in the bending mode frequencies. Methods to automatically track the bending mode frequencies are crude, complicated and not flight proven.

The standard method of obtaining an attitude rate signal which possesses little bending signal is to mount a rate gyro on the vehicle at a point where the slope of the bending deflection curve is at or near zero for those bending modes producing stability difficulties. It is impossible to find a point on the vehicle where all the bending deflection slopes are zero. Because rate gyros are physically much smaller than position gyros the restrictions as to their mounting position on the vehicle are much less severe.

As fuel and oxidizer are expended the vehicle total mass and inertia become less. This causes an increase in the attitude torque available from a unit thrust vector deflection, a change in the bending frequencies, and a change in the normalized bending deflection curves. As these changes occur it is generally desirable to change the rate gain, control filter and loop gains through the thrust vector and aerodynamic loops. In a nominal vehicle the mass flow rate from the fuel and oxidizer changes could be programmed as a function of time. The space shuttle booster must be capable of surviving the failure of a boost rocket engine in which case the mass flow rate from the rocket engines will decrease. The thrust variations of the vehicle are proportional to the vehicle velocity. Vehicle total velocity which is measured by the navigation system provides an easily available parameter to use to correct control system variations caused by vehicle mass changes.

The sloshing of fuel and oxidizer in their tanks results in another dynamic mode which may affect the flight control system. It is assumed that tank baffling will be used to control this slosh and thus no specific flight control functions are required for fuel and oxidizer slosh.

In Figure 3-11 the effects of the load relief system upon the boost control system have been ignored. Incorporation of the load relief system will cause discrete changes in the control system gains and filter time constants as the load relief system is switched in and out.

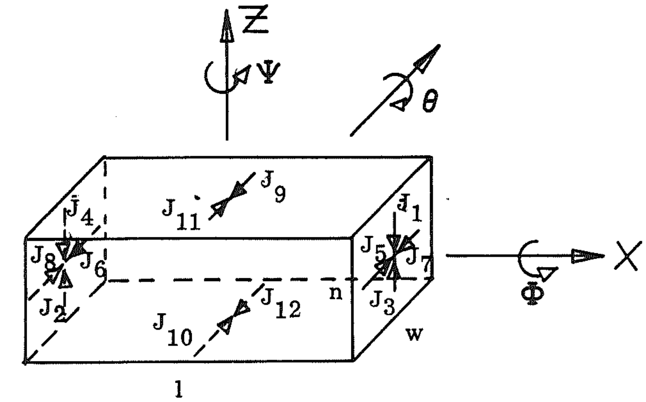
In the past boost vehicle control systems have been mechanized for each mission as mission payload weights changed. With the space shuttle booster this procedure will have to be eliminated in the interest of reducing ground support activities for each flight. The orbiter weight which comprises the booster payload can vary from empty weight to maximum loaded weight. It is probable that these weight variations will require control system changes. These changes will be stored in the DMS computer. Prior to launch the pilot inserts payload weight by keyboard entry and the proper control system constants for the flight are automatically selected.

The above functional description of the boost flight control system has been restricted to the pitch system only. In the past, boost vehicles have been symmetrical in pitch and yaw and thus identical control systems have been used for pitch and yaw. Since the space shuttle booster is designed also for cruise flight it will not be symmetrical in pitch and yaw. The Phase A shuttle designs indicate that the yaw axis control system restrictions will be less severe than those for the pitch axis. As the yaw axis will possess higher structural rigidity than the pitch axis, the maximum aerodynamic load values increase, bending magnitudes are reduced, and bending frequencies increase. This should eliminate requirements for a yaw load relief system and a yaw rate gyro. Attitude rate feedback will be accomplished by differentiating the yaw attitude gyro output. The Phase A configurations indicate that the boost rocket engines will be mounted in a horizontal double row on the aft end of the booster. A failure in one of the outboard rocket engines will result in a large yaw torque which will have to be rapidly corrected by the other engines. This will require a large control authority ability in yaw. Yaw normal aerodynamic forces will be less than those of pitch; the wings will contribute large aerodynamic forces in pitch. The yaw system will probably not employ the yaw aerodynamic surfaces for control during boost. In order to rapidly respond to an engine failure the yaw short period response will be much higher than the pitch short period response.

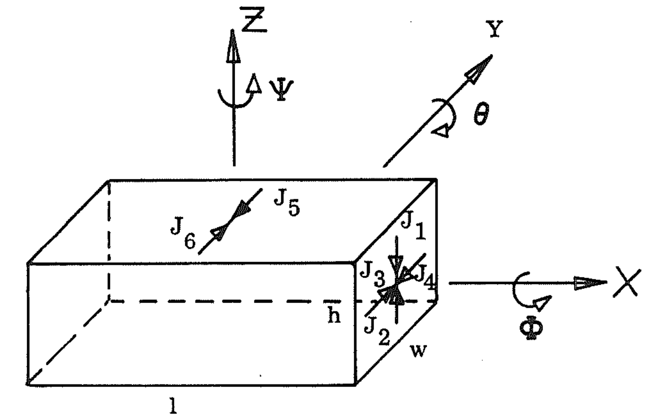
The roll attitude control system has even less restrictive requirements than yaw. Torsional aerodynamic loading on the vehicle can be neglected along with torsional bending. Previous booster systems have had simple attitude plus derived attitude rate systems for roll control. The space shuttle booster differs from other boost vehicles in that significant yaw-roll aerodynamic coupling will exist because of the wings and rudder. Large yaw turns are not contemplated during boost and thus coordinated turn roll-yaw coupling within the control system will not be required.

Coast Flight Control

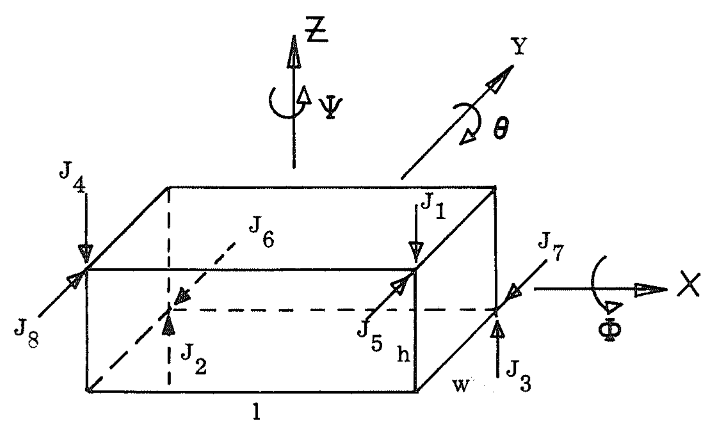
The boost vehicle will be above any effective atmosphere when rocket engine cutoff occurs. At this point the only system available for attitude control will be a reaction jet system. Figure 3-13 shows some of the possible arrangements of reaction jets on a vehicle for a



A. Torque Couples, Axis-Uncoupled



B. No Torque Couples, Axis-Uncoupled



C. Torque Couples, Axis-Coupled

Figure 3-13 Reaction Jet Attitude Control Configurations single non-redundant attitude control system. The vehicle is represented as a rectangular prism and the reaction jets by their thrust vectors. Figure 3-13A shows a system in which pure uncoupled torques can be obtained and there is no coupling between each axis. The thrust magnitude out of each jet is represented by the sumbols J_1, J_2,.....J_{12}. The mounting dimensions represented by the rectangular prism are l, h, and w. The mass and inertias are represented by M, I_{xx}, I_{yy}, and I_{zz}. The vehicle's center of gravity is at the center of the rectangular prism. Then the linear and angular accelerations obtained from the system can be represented by the matrix equation shown in Figure 3-14A . With this system each attitude control acceleration command will activate two jets. The jets activated and the resultant accelerations generated for each desired control torque are shown in Figure 3-15 under System A. It is assumed that the thrust produced by each jet is identical and equal to J. It can be seen for this system that only the desired accelerations are produced.

Figure 3-13B shows the reaction jet arrangement for a system in which pure torque couples are not produced. Figure 3-14B shows the control matrix relating jet thrust magnitude to linear and angular accelerations.

System B of Figure 3-15 indicates the control logic required for this configuration. With this system each axis is controlled independently. When using pitch acceleration torques neither roll nor yaw angular accelerations are produced. Each torque producing the desired angular accelerations does, however, also produce a linear acceleration. These linear accelerations will have a negligible effect upon the mission. If there are no external torques on the vehicle such as torques produced by radiation, separation, or rarefied atmosphere, then the time integral of all linear acceleration caused by the reaction jet system will be zero. If there are external torques on the vehicle, the sources of these torques will most probably also produce linear accelerations. Under this condition the velocity change produced by the reaction jet system in countering these external torques can be expected to be the same order of magnitude or smaller than the velocity created by the external forces. The velocities created by reaction jet forces can thus be neglected.

System C of Figure 3-13 produces torque couples and has coupling between the pitch-roll and yaw-roll axis. Figure 3-14C shows the control matrix relating the thrust magnitude to linear and angular accelerations for this configuration. System C of Figure 3-15 indicates the control logic required for this configuration and the linear and angular accelerations resulting from each desired attitude control torque. Figure 3-15 shows that there are no linear accelerations created with this system. Whenever a pitch or yaw angular acceleration is requested a roll angular acceleration is also generated. Figure 3-16 shows the sequence required to obtain a change in pitch attitude. The jets J_1 and J_2 are turned on to produce a pitch angular acceleration. Jets J_1 and J_2 are left on until a desired pitch rate is achieved. When the desired pitch angle is achieved, jets J_3 and J_4 are turned on until the pitch rate is zeroed. When jets J_1 and J_2 were turned on a roll angular acceleration was produced in addition to the desired pitch angular acceleration. This produces a roll rate and roll position error. Jets J_5, J_6, J_7, and J_8 were turned on immediately after jets J_1 and J_2 were turned off in order to first cancel the roll rate and then to drive the roll rate negative in order to reduce the roll position error. At the proper instant when the roll rate and roll position were of proper magnitude, jets J_5, J_6, J_7 and J_8 were turned off and jets J_1, J_2, J_3

$$
\begin{bmatrix} \ddot{\theta} \\ \ddot{\phi} \\ \ddot{\psi} \\ \ddot{X} \\ \ddot{Y} \\ \ddot{Z} \end{bmatrix}
=
\begin{bmatrix}
\ell/2I_{yy} & \ell/2I_{yy} & -\ell/2I_{yy} & -\ell/2I_{yy} & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 \\
0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & h/2I_{xx} & h/2I_{xx} & -h/2I_{xx} & -h/2I_{xx} \\
0 & 0 & 0 & 0 & \ell/2I_{zz} & \ell/2I_{zz} & -\ell/2I_{zz} & -\ell/2I_{zz} & 0 & 0 & 0 & 0 \\
0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 \\
0 & 0 & 0 & 0 & 1/m & -1/m & -1/m & 1/m & -1/m & 1/m & 1/m & -1/m \\
-1/m & 1/m & 1/m & -1/m & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0
\end{bmatrix}
\times
\begin{bmatrix} J_1 \\ J_2 \\ J_3 \\ J_4 \\ J_5 \\ J_6 \\ J_7 \\ J_8 \\ J_9 \\ J_{10} \\ J_{11} \\ J_{12} \end{bmatrix}
$$

A. Torque Couples, Axis- Uncoupled

$$
\begin{bmatrix} \ddot{\theta} \\ \ddot{\phi} \\ \ddot{\psi} \\ \ddot{X} \\ \ddot{Y} \\ \ddot{Z} \end{bmatrix}
=
\begin{bmatrix}
\ell/2I_{yy} & -\ell/2I_{yy} & 0 & 0 & 0 & 0 \\
0 & 0 & 0 & 0 & h/2I_{xx} & -h/2I_{xx} \\
0 & 0 & \ell/2I_{zz} & -\ell/2I_{zz} & 0 & 0 \\
0 & 0 & 0 & 0 & 0 & 0 \\
0 & 0 & 1/m & -1/m & -1/m & 1/m \\
-1/m & 1/m & 0 & 0 & 0 & 0
\end{bmatrix}
\times
\begin{bmatrix} J_1 \\ J_2 \\ J_3 \\ J_4 \\ J_5 \\ J_6 \end{bmatrix}
$$

B. No Torque Couples, Axis-Uncoupled

$$
\begin{bmatrix} \ddot{\theta} \\ \ddot{\phi} \\ \ddot{\psi} \\ \ddot{X} \\ \ddot{Y} \\ \ddot{Z} \end{bmatrix}
=
\begin{bmatrix}
\ell/2I_{yy} & \ell/2I_{yy} & -\ell/2I_{yy} & -\ell/2I_{yy} & 0 & 0 & 0 & 0 \\
w/2I_{xx} & w/2I_{xx} & w/2I_{xx} & w/2I_{xx} & -h/2I_{xx} & -h/2I_{xx} & -h/2I_{xx} & -h/2I_{xx} \\
0 & 0 & 0 & 0 & \ell/2I_{zz} & \ell/2I_{zz} & -\ell/2I_{zz} & -\ell/2I_{zz} \\
0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 \\
0 & 0 & 0 & 0 & 1/m & -1/m & -1/m & 1/m \\
-1/m & 1/m & 1/m & -1/m & 0 & 0 & 0 & 0
\end{bmatrix}
\times
\begin{bmatrix} J_1 \\ J_2 \\ J_3 \\ J_4 \\ J_5 \\ J_6 \\ J_7 \\ J_8 \end{bmatrix}
$$

C. Torque Couples, Axis-Coupled

Figure 3-14 Reaction jet attitude control matricies for three Configurations

SYSTEM	DESIRED CONTROL	JETS ACTIVATED	RESULTANT ACCELERATIONS $\ddot{\theta}$	$\ddot{\phi}$	$\ddot{\psi}$	$\ddot{X}$	$\ddot{Y}$	$\ddot{Z}$
A	$+\ddot{\theta}$	$J_1\ J_2$	$\ell J/I_{yy}$	0	0	0	0	0
	$-\ddot{\theta}$	$J_3\ J_4$	$-\ell J/I_{yy}$	0	0	0	0	0
	$+\ddot{\phi}$	$J_9\ J_{10}$	0	hJ/I_{xx}	0	0	0	0
	$-\ddot{\phi}$	$J_{11}\ J_{12}$	0	$-hJ/I_{xx}$	0	0	0	0
	$+\ddot{\psi}$	$J_5\ J_6$	0	0	$\ell J/I_{zz}$	0	0	0
	$-\ddot{\psi}$	$J_7\ J_8$	0	0	$-\ell J/I_{zz}$	0	0	0
B	$+\ddot{\theta}$	J_1	$\ell J/2I_{yy}$	0	0	0	0	$-J/m$
	$-\ddot{\theta}$	J_2	$-\ell J/2I_{yy}$	0	0	0	0	J/m
	$+\ddot{\phi}$	J_5	0	$hJ/2I_{xx}$	0	0	$-J/m$	0
	$-\ddot{\phi}$	J_6	0	$-hJ/2I_{xx}$	0	0	J/m	0
	$+\ddot{\psi}$	J_3	0	0	$\ell J/2I_{zz}$	0	J/m	0
	$-\ddot{\psi}$	J_4	0	0	$-\ell J/2I_{zz}$	0	$-J/m$	0
C	$+\ddot{\theta}$	$J_1\ J_2$	$\ell J/I_{yy}$	wJ/I_{xx}	0	0	0	0
	$-\ddot{\theta}$	$J_3\ J_4$	$-\ell J/I_{yy}$	wJ/I_{xx}	0	0	0	0
	$+\ddot{\phi}$	$J_1\ J_2\ J_3\ J_4$	0	$2wJ/I_{xx}$	0	0	0	
	$-\ddot{\phi}$	$J_5\ J_6\ J_7\ J_8$	0	$-2hJ/I_{xx}$	0	0	0	
	$+\ddot{\psi}$	$J_5\ J_6$	0	$-hJ/I_{xx}$	$\ell J/I_{zz}$	0	0	
	$-\ddot{\psi}$	$J_7\ J_8$	0	$-hJ/I_{xx}$	$-\ell J/I_{zz}$	0	0	
D	$+\ddot{\theta}$	J_1	$\ell J/2I_{yy}$	$wJ/2I_{xx}$	0	0	0	$-J/m$
	$-\ddot{\theta}$	J_3	$-\ell J/2I_{yy}$	$wJ/2I_{xx}$	0	0	0	J/m
	$+\ddot{\phi}$	$J_1\ J_3$	0	wJ/I_{xx}	0	0	0	
	$-\ddot{\phi}$	$J_5\ J_7$	0	$-hJ/I_{xx}$	0	0	0	
	$+\ddot{\psi}$	J_5	0	$-hJ/2I_{xx}$	$\ell J/2I_{zz}$	0	J/m	
	$-\ddot{\psi}$	J_7	0	$-hJ/2I_{xx}$	$-\ell J/2I_{zz}$	0	$-J/m$	

Figure 3-15 Attitude Control Logic for Four Configurations

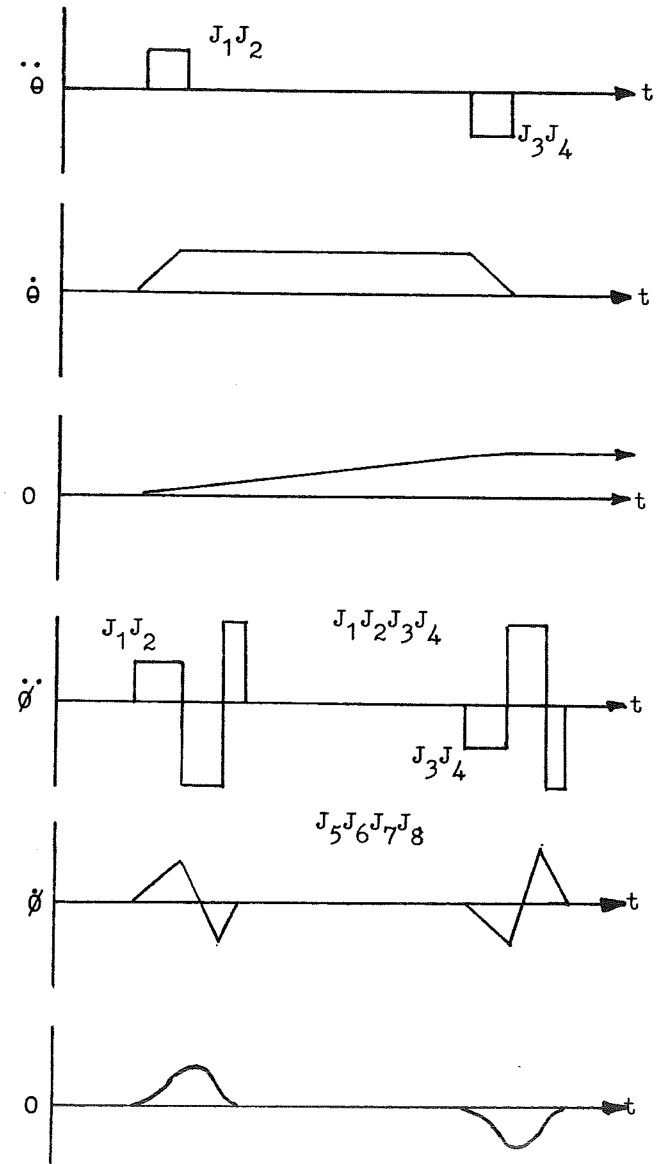


Figure 3-16 Typical Response to Step Pitch Command for Axis Coupled System

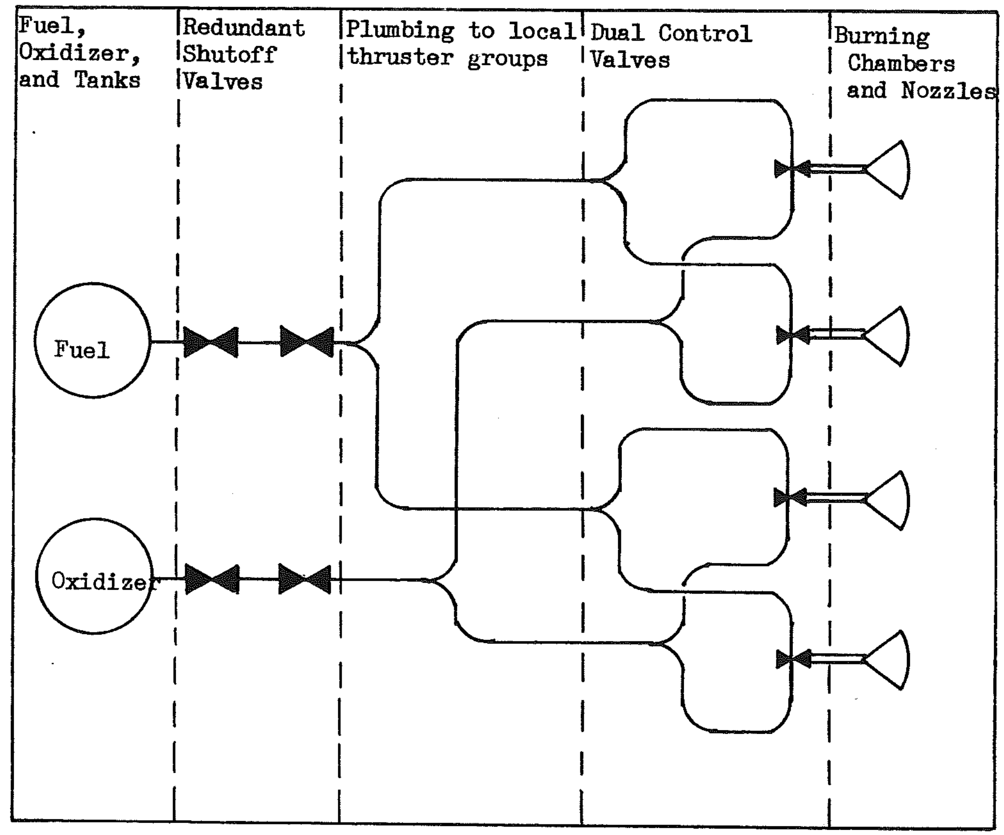


Figure 3-18 Typical Reaction Jet Control System Hardware

System	Jets	Thruster Groups
A	12	4
B	6	2
C	8	4
D	4	2

Figure 3-17 Jets and Thruster Groups

and J_4 turned on so that both roll rate and roll position would go to zero at the same time. When jets J_3 and J_4 are turned on the same type of coupling into the roll axis occurs. This causes same type of roll axis activity, though of opposite polarity, to cancel the roll rate and position errors.

System D of Figure 3-15 is the same as System C with jets J_2, J_4, J_6, and J_8 removed. For attitude control System D operates identically to System C, the only difference being that linear accelerations are produced when ever pitch or yaw accelerations occur.

There are numerous advantages and disadvantages to each system. The most obvious tradeoff parameters between the four systems is the number of jets required to mechanize the system. Figure 3-17 lists the number of jets for each system.

If system selection is based upon weight, reliability and redundancy, it would at least on the surface, appear that System D would be the selection. The reaction jet system includes fuel tanks, fuel, valves, and plumbing. Figure 3-18 shows a typical arrangement of various reaction jet attitude control system components. The plumbing required between the fuel tanks and shutoff valves to the local thruster groups can be a significant portion of the total system weight. As shown in Figure 3-18 this plumbing consists of a single fuel and a single oxidizer line to each local thruster group. Figure 3-17 lists thruster group numbers for each system.

Another parameter which figures heavily in the total system weight is the amount of fuel and oxidizer required by the mission. If the control logic used to generate Figure 3-16 were used for System C or D and control logic developed for Systems A and B to give similar pitch and yaw response, then System C and D would require 4.45 times as much fuel as System A or B. By tolerating roll and roll rate errors, this ratio can be reduced to a theoretical minimum of 2.

Redundancy is probably the greatest single factor affecting the final system selection. Since the reaction jet attitude control system is primarily not an electronic system its redundancy requirements, at least upon the fuel tanks, valves, plumbing and thrusters is fail operational-fail safe. The reaction jet attitude control system is critical to crew safety; if the booster is at the wrong attitude or tumbling at the time of reentry, it will be destroyed. Thus, for the reaction jet attitude control system, a fail safe criteria is in general, equivalent to a fail operational requirement. The redundancy requirement thus reduces to a requirement for at least three operational reaction jet attitude control systems. In order to ascertain the mechanization needed to fulfill the redundancy requirement, the types of failure modes within the system must be known. Figure 3-19 lists possible failure modes.

The final design of the reaction jet attitude control system will be based upon parameters which are beyond the scope of this study to develop. The final design will probably be very similar to three independent system D mechanizations unless it is determined that the additional fuel required from axis coupling is prohibitive. In this case, three independent System B mechanizations will be used.

Failure	Results
Fuel or Oxidizer Tank Rupture; Pressurization System Failure	If the fuel of oxidizer tank ruptures or the pressurization system used to force the fuel and oxidizer from the tanks fails the entire reaction jet system fed from the failed tanks will fail in a zero thrust mode. A reaction jet system may use hypergolic bipropellants thus the fuel and ozidizer tanks must be located on the vehicle such that if a rupture of both tanks occurs the two fluids will not have a chance of mixing.
Shutoff Valves Fail Open	The system will continue to operate normally, unless certain types of second failures occur.
Shutoff Valves Fail Closed	Shuts down the entire reaction jet system fed through the failed shutoff valve in a zero thrust mode.
Plumbing Rupture	Shuts down the entire reaction jet system in a zero thrust mode. Hypergolic bipropellant fuels and oxidizers are very corrosive. Fluid coming from a ruptured line must be stopped rapidly. This is the reason for dual redundant shutoff valves. If a line ruptures and at the same time, its shutoff valve fails open, a continuous flow of propellant will leak from the ruptured line until the entire tank supply is exhausted. The corrosiveness of the propellant leaking on other critical systems could cause their failure resulting in multiple sequential failures totally disabling the booster.
Plumbing Stoppage	Shuts down these thrusters being supplied by the stopped pipe in a zero thrust mode.
Dual Control Valve Fails Closed	The single thruster supplied by the failed valve will fail to operate in a zero thrust mode.
Dual Control Valve Fails Open	Associated thruster continues to thrust until the fuel supply is exhausted or until shutoff valves are closed on the fuel or oxidizer tanks.
Burning Chamber or Nozzle Failure	Degraded thrust from the failed nozzle.

Figure 3-19 Reaction Jet Attitude Control System Failure Modes

Reentry Attitude Control System

Reaction jet and aerodynamic control are both used during reentry. During the initial reentry period before dynamic pressure has reached a value sufficient for effective aerodynamic control, the reaction jet system will be used. After dynamic pressure reaches a sufficient value aerodynamic control will be used. The aerodynamic attitude control system during reentry consists of a stability augmentation system (SAS) with outer loop control provided through the reentry guidance system. Several configurations are possible for the reentry attitude control system. In order to determine which configuration should be used the following two questions must be answered:

1. How should the change from reaction jet control to aerodynamic control be accomplished?
2. Should the system be a fly-by-wire system?

There are two basic methods of changing from reaction jet to aerodynamic attitude control. There can be a discrete point at which the attitude control is switched from reaction jet to aerodynamic control or there can be a slow phasing from one type of control to the other. The point at which the changeover is accomplished by switching can be determined automatically by

- Measurements of dynamic pressure from the air data computer
- Measurements of total acceleration from the navigation system linear accelerometers

or can be performed manually by the pilot. Control slowly phased from reaction jets to aerodynamic surfaces can be easily accomplished by continuously using both systems throughout the reentry phase. If this is done as dynamic pressure increases the control from the aerodynamic surfaces will completely overshadow those torques generated by the reaction control system. To date the only vehicle that used both reaction jet and aerodynamic control and has flown through reentry type flight environments is the X-15 aircraft. The X-15 used discrete switching to transfer from reaction jet to aerodynamic control. During flight testing it was discovered that reaction jet control was preferred by the pilots. Reaction jet control was preferred because torques about each flight axis were produced while aerodynamic control produced inter-axis coupling. The system assumed for use on the space shuttle booster will use automatic discrete switching with the switching determined by total acceleration and with override capabilities by the pilot. This method is chosen because:

1. The DMS is not required to perform both reaction jet and aerodynamic control computations simultaneously.
2. The aerodynamic control can be disabled if the pilot prefers to use reaction jet control later than normally planned.

Total acceleration was chosen over air data sensor outputs to determine the automatic switching point because pitot-static sensor probes are inaccurate at high angles of attack. These probes may have to be disabled and covered during reentry for protection from reentry heating.

The attitude control system for each axis of an aerodynamically controlled vehicle can be separated functionally into a stability augmentation system (SAS) and into outer loop control functions. Figure 3-20 shows a basic SAS loop. The handling characteristics of an aircraft changes greatly with vehicle velocity. The constraints on vehicle design imposed by mission requirements are generally incompatible with desirable handling characteristics. The SAS is mechanized in order to artificially generate desirable handling characteristics. Many studies have been conducted to determine the optimum short period response of a vehicle. The SAS system endeavors to achieve optimum short period response throughout all flight regions. This is accomplished by closing a rate loop around the vehicle axis being augmented. Compensation is mechanized within the rate loop to achieve the desired short period response. On most vehicleseach aerodynamic surface has more than one hydraulic or electrical actuator controlling its position. One actuator is driven from the high bandwidth SAS and the other from a low bandwidth trim system. The value of trim required is determined from a low gain integrator receiving its input from the SAS error signal. There are two places in which the pilot commanded stick signal can be inserted into the system. It can be added to the rate sensor output to form a rate command to the system or it can be added directly into the loop at the actuator. If the pilot stick command is added to the rate sensor output the command will be filtered and shaped first in order to further enhance the aircraft handling qualities. The mechanization can be accomplished entirely with mechanical couplings if the pilot stick command is inserted directly at the actuator. The pilot will still have direct control of the vehicle aerodynamic surfaces and will be able to control the vehicle even if a failure in the SAS loop or in electrical power occurs. If the vehicle dynamics are greatly different than the desired dynamics implemented with the SAS system then the SAS loop gain will be large. If the SAS loop gain is large, inserting the pilot stick command at the control actuator will result in a very sluggish system. In general, inserting the pilot stick command as a rate command allows the SAS to handle a much greater variation in vehicle dynamics than if the pilot stick command is inserted at the control actuator. To insert the pilot stick command as a rate command requires that control of the vehicle have an electronic path such that failure of the electronic path will result in loss of control of the vehicle. It is assumed that the space shuttle booster SAS will have the pilot stick command inserted as a rate command by addition with the rate sensor output. The variation of the aerodynamic environment of the space shuttle booster during reentry will be large. It is probable that the vehicle in some of its operational aerodynamic regimes will be uncontrollable without an operating SAS. Thus the major advantage of a non fly-by-wire system is negated in the space shuttle application.

The variation in vehicle dynamics throughout reentry will be large enough to require changes in SAS gains as the mission progresses. The SAS gains can be changed either as a function of a sensed atmospheric variable or adaptively by sensing system response. Since the availability of air data sensors during reentry is debatable atmospheric parameters will have to be generated from other systems measurements. Vehicle velocity and altitude can be determined from inertial navigation outputs. Atmospheric density can be determined by using altitude in an atmosphere model. Atmospheric density and total vehicle velocity can be used to produce dynamic pressure. The SAS gains can then be changed as a polynominal function of dynamic pressure. This is the assumed method of SAS gain variation to be employed on the space shuttle booster. Adaptive systems to date have only been

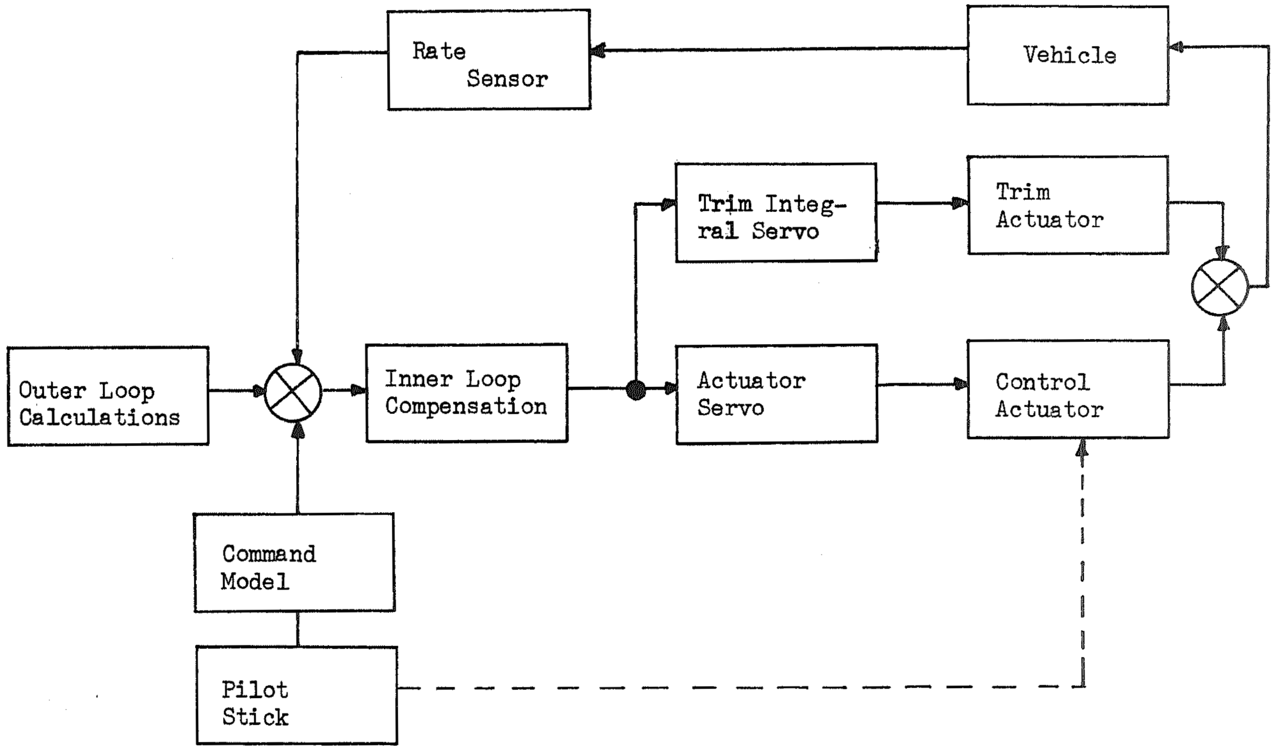


Figure 3-20 Stability Augmentation System (SAS) Block Diagram

	Hold Altitude (Init)	Hold Altitude (Set)	Hold Attitude	Hold Heading (Init)	Hold Heading (Set)	Hold Mach (Throttle)	Hold Mach (Altitude)	Hold Position
Hold Altitude (Init)	-			X	X	X		X
Hold Altitude (Set)		-	X	X	X	X		X
Hold Attitude		X	-	X	X	X		X
Hold Heading (Init)	X	X	X	-		X	X	
Hold Heading (Set)	X	X	X		-	X	X	
Hold Mach (Throttle)	X	X	X	X	X	-		X
Hold Mach (Altitude)				X	X		-	X
Hold Position	X	X	X			X	X	-

Figure 3-21 Cruise Outer Loop Control Compatibility

flown on experimental applications and are necessary only on vehicles with unpredictable dynamics. Even though the space shuttle booster dynamics will experience a wide range of variation during reentry, the dynamics for all flight conditions are sufficiently predictable to allow for a non-adaptive SAS design. Conventional non-adaptive SAS mechanizations employing programmed gain and compensation changes as a function of dynamic pressure have the advantage over adaptive mechanizations of being able to more closely control vehicle handling characteristics. In an adaptive system constraints are imposed upon the control loop compensation by the adaptive mechanization. In a non-adaptive system contraints are imposed upon the control loop compensation and gain by lack of knowledge of the vehicle dynamics. It is felt for the space shuttle booster the non-adaptive system will impose the least constraint upon the control system mechanization.

Cruise Attitude Control System

The cruise attitude control system will be an extension of the reentry aerodynamic attitude control system. The SAS for each axis will be fixed at a final configuration. The automatic outer loop mode will be switched from reentry guidance to cruise guidance. The pilot will be able to override the automatic outer loop function through his control stick. The pilot will also be able to override the normal cruise guidance outer loop function by selecting one or more other outer loop modes. These additionally available outer loop modes are:

Hold Altitude: There are two hold altitude submodes, in the first submode the booster will continue to fly at the pressure altitude present when the hold altitude was engaged. In the second submode an altitude value must be provided to the DMS. When this hold altitude mode is engaged the booster will smoothly ascend or descend to the stored altitude and then maintain that altitude. The ascent or descent rate will be a stored constant unless the hold attitude mode is simultaneously selected. In this case the descent or ascent rate determined by the hold attitude mode will be used.

Hold Attitude: The hold attitude mode will cause the booster to maintain the pitch attitude present when the hold attitude mode was engaged. This mode can be disengaged manually or will automatically disengage by engaging the hold present altitude mode or by arriving at the selected altitude with the hold stored attitude mode selected.

Hold Heading: There are two hold heading submodes, in the first submode the booster will continue to fly at the heading present when the hold heading mode is engaged. In the second submode a heading must be provided to the DMS. When this submode is engaged the booster will execute a smooth coordinated turn to the stored heading and then maintain that heading. The heading reference for the hold heading mode is selectable either from the attitude gyros, magnetic flux gate compass or TACAN receiver.

Hold Mach: There are two hold Mach submodes. In each case the automatic system will hold the Mach number present when the hold Mach mode is engaged. In one submode, constant Mach will be maintained by automatic throttle adjustments. In the other , by altitude adjustments.

Hold Position Mode: When this mode is engaged the booster will immediately execute a near constant turning rate right turn. The vehicle will continue to circle an earth fixed point.

Some of the above outer loop modes are compatible with one another while others are not. Figure 3-21 is a table showing compatibility modes. If incompatible outer loop modes are engaged the one engaged last will override the other.

3.6 COMMUNICATIONS

The communications system provides the equipment necessary for effective transmission and reception of information between the booster and orbiter, between the booster and ground facilities, and between the various booster systems. The communications system consists of the following subsystems:

- Voice Communications
- Command
- Telemetry
- Recording
- Beacon
- Data management systems link

3.6.1 VOICE COMMUNICATIONS

The function of the voice communication subsystem is to provide two way voice communication between the booster and orbiter crews, between the booster crew and ground support facilities or between the pilot and co-pilot of the booster vehicle. Equipment required to meet voice communications functions consists of:

- UHF transceiver
- VHF transceiver
- Intercom
- Navigation Aids

The UHF system uses a multichannel transceiver system and omnidirectional antennas. Any of 3500 channels in the UHF band between 225 and 400 MHZ can be selected. However, frequencies expected for use on the mission would be preset for ease of channel selection. Channel tuning is done electronically. RF power output of 20 to 100 watts is achieved by all solid state circuitry. The antenna system includes automatic antenna switches and flush mounted omnidirectional antennas. Antenna switching is required to select the antenna that maximizes the received signal. If required, two transceivers can be operated simultaneously at 2 different sets of operating frequencies. Antenna switches are then used to connect both transceivers to a common antenna or to connect the two transceivers to different antennas. That is, each transceiver is connected to an antenna that will provide an adequate receiver signal level.

The VHF system is used to communicate with the ground stations, ships and aircraft of the Manned Space Flight Network (MSFN). MSFN maintains two VHF voice networks. One, in the frequency range between 259.7 - 296.8 MHZ, is available during the period from liftoff through reentry. The second, in the VHF frequency range between 118 - 136 MHZ, will be monitored from the flyback turn through landing.

The intercom system is used for necessary voice communication between the pilot and co-pilot.

Navigation aids, in addition to performing their primary functions, provide voice channels for identification and necessary transmissions from ground facilities to the booster. The advanced instrument landing system (AILS) provides a UHF communication link for approach information. Enroute TACAN stations also transmit voice and identity tones on their assigned frequencies.

3.6.2 COMMAND

The function of the command subsystems is to provide a capability for ground control stations to send command, maintenance or information signals to the booster. The booster command subsystem is compatible with the Apollo unified S-band communication link, with current ground facilities available for data transmission. The command message typically includes words devoted to synchronization, sequencing, updating, antenna positioning, decoder address, and command instructions. Commands may be delayed (stored) until an accurate time reference indicates proper elapsed time, or they may be transmitted in real time for immediate execution. The DMS interprets the uplink data, initiates command execution, or stores information in proper storage locations.

Equipment required for command subsystem operation includes antennas, S-band receivers, demodulators, multiplexers, decoders and distributors with necessary connections to the system data bus. Command receivers are characterized by low sensitivity, high selectivity, and wide dynamic range to avoid saturation. Selectable fixed frequencies, each with narrow band width, permit operational flexibility and make available a choice of several command bands for alternate channel arrangements. The decoder checks incoming signals and commands with regard to satisfying encoding and decoding algorithms, procedural correctness, adequate signal strength, and compatibility with mode of operation. Timing and synchronization controls, and delay and filter circuits prevent acceptance of erroneous commands or the accidental activation or deenergization of booster systems.

3.6.3 TELEMETRY

The function of the telemetry subsystem is to provide an additional capability for the booster to supply status information to ground facilities. The telemetry data, from the booster sensors, is received by the nearest line-of-sight remote tracking site, and relayed via the NASA Communications Network (NASCOM), at Goddard Space Flight Center to Mission Control Center, Houston which monitors space shuttle missions. The telemetry subsystem can be considered as completing the command/control link in its transmission of verification of commands status of equipment, or actual diagnostic data. The DMS assists in telemetry subsystem functioning by formatting, conversion, error protection, and data compaction routines.

The equipment required for the booster telemetry PCM downlink consists of sensors, multiplexers, signal conditioners, amplifiers, oscillators, encoders, analog-to-digital converters, and the telemetry transmitter. Types of telemetry data consist of sampled 8 to 10 bit coded analog waveforms, bilevel "on" or "off" event information, computer memory words, or special requested data. Sampling rates may range from .5 to 200 samples per second. DMS data compression algorithms will provide a complete and accurate representation of all booster system outputs.

3.6.4 RECORDING

The function of the recording subsystem is to provide a permanent record of the booster mission for use in post flight analysis and maintenance cycle operations. The DMS interfaces with the recording subsystem in obtaining, processing, formatting and converting data for recording. Types of recordings used consist of:

- Telemetry data storage
- Flight recorder
- Mass memory (maintenance) recorder

Telemetry data storage techniques require extended life, minimum power consumption, minimum size and weight, high capacity, and long-term retentivity. As these characteristics are met by magnetic tape recorders, the booster telemetry recorder is magnetic tape. The common tape recording processes are; direct (analog), frequency modulation (FM), pulse duration modulation, and digital recording. The digital recording process is a binary coded PCM system employing PCM quantizing and encoding techniques. It is the most likely process for telemetry recording as it is directly compatible with off-line computer systems when playback is desired. All telemetry transmissions are copied onto the tape. This recording can be used following mission completion as a source for post-flight analysis and mission simulation.

The flight recorder is an airborne magnetic flight data recorder installed in accordance with Federal Aviation Agency regulations. A system consisting of a recorder unit, accessory unit, and sensor equipment record air speed, altitude, acceleration and heading as a function of time. All voice communications are also preserved. The magnetic flight data recorder is a primary source of information in event of a crash. It also provides valuable information for flight history, analysis and maintenance data.

The mass memory (maintenance) recorder provides a detailed record of the results of all onboard checkouts of booster systems. Magnetic tape control, writing, and formatting routines record time tagged check point information during all operational modes. This data is then used for quality assurance and scheduled maintenance tests during the maintenance cycle which recertifies the booster for flight operations.

3.6.5 BEACON

The function of the beacon subsystem is to provide positive identification and tracking of the booster to ground support facilities, or to provide positive navigation fixes to the booster. The equipment included in the transponder subsystems are:

- Radar beacon
- Tracking beacon
- Recovery beacon
- Marker beacon

Radar beacons provide air traffic controllers with a positive means of identifying traffic in the vicinity of airports, and also aid in reducing the amount of voice communication required. The ground based radar is furnished with an auxiliary pulse transmitter and rotating antenna, synchronized to the radar pulse-repetition rate and the radar antenna-rotation rate. The frequency of the transmitter is 1030 kHZ; its power output, about 1kW. In the booster, a pulse transponder replies at 1090 MHZ with about 500 watts peak power and with a pulse code having up to 4096 possible combinations. The ground equipment transmits an interrogation signal to the booster and, if the transponder in the booster is set to respond to the interrogation, the transmitter portion of the transponder sends a signal which is processed by the ground equipment and presented on a radar display. The interrogation pulse groups are in two types; mode 3/A and mode C. The selection of these modes is either controlled by the ground operator or automatically selected in a mixed sequence. Mode 3/A constitutes a request for identification information, and mode C asks for altitude data. Before the transmitter portion of the airborne transponder will reply to an interrogation, the proper pulse pairs must have been received and processed within the transponder. Two framing pulses spaced 20.3 u sec apart with 12 information pulses between them provide the basic reply code. Thus, the code system is capable of producing 4096 (2^{12}) different identification reply codes. The identification number is assigned by the ground controller and is used by the pilot to set the radar beacon control box. When mode C is received by the transponder, it automatically reports the altitude of the aircraft. Altitude is reported in increments of 100 feet up to a maximum of 126,700 feet.

Tracking beacons function during the boost and reentry phases to extend the tracking range and accuracy of ground based radars. As radar "skin" tracking is not suitable for space craft at orbital altitudes, an active on-board tracking aid is required. A receiver-transmitter combination (radar transponder) is generally used for this function. The transponder requires a ground radar transmitter for interrogation and a sensitive ground receiver to accept the reply signals. A locked phase-coherent transponder transmits a signal with an exact frequency and phase relationship to its received signal and thus permits accurate two-way Doppler measurements after lock is established. The input signal generated by the radar is either a single pulse or a pulse group coded by the time interval between pulses. The primary components of a typical microwave beacon transponder are a sensitive super heterodyne receiver, a high-power modulated transmitter to send a single reply pulse of selected pulse width, a pulse decoder, a demodulator to make audio command signals available to other equipment, preselector and duplexer. For a C-band transponder, operating in the 5.4 to 5.9 kHz ranges at least 40 Hz separation between the duplexer transmit and receive frequencies is desirable. With 100 Hz separation, the preselector alone can provide sufficient protection of the mixer crystal. The decoder, which may accept an interrogation of one or several pulse code groups at varying repetition rates, must compensate for variations in beacon reply pulse delay over a wide dynamic range of input signal levels. The command signal demodulator can be connected to a command circuit to permit remote control of on-board functions.

The recovery beacon is associated with the standard aircraft flight recorder described in the recording subsystem section. The beacon is activated, if a crash occurs, as an aid in recovery operations. It operates at a frequency of 243 MHZ. The antenna and transceiver may be activated by crash forces, hydrostatic pressure or under pilot control. The AILS presently under development and scheduled for installation at some airports during the next few years is the primary system for instrument landing, if required, following reentry. However compatibility with the present ILS facilities now available is required to give the capability to land at a larger number of airports. The current ILS provides voice and identification transmissions to aid the booster flight crew. The localizer of the instrument landing system provides an audible identification signal and voice channel from the control tower. There are twenty localizer channels at odd tenth MHZ from 108.1 to 111.9 MHZ with each channel being paired with a glide-slope channel.

Marker beacons provide fixes during instrument landing approaches. All current marker beacons operate at 75 MHZ. The fan-marker antenna pattern is generated by a ground array consisting of four horizontal half-wave radiators in line with the airway. They generate a narrow vertical beam which is reinforced by a wire-mesh counterpoise placed a few feet above the ground. The transmitter is crystal controlled, delivers up to 100 watts, is tone modulated, and identifies itself by gaps in its tone signal. The booster receiver is a crystal-controlled superheterodyne with its output supplying a tone to head phones or lighting a lamp. The airborne antenna comprises a quarter wave element recessed into the belly of the aircraft, parallel to the axis of flight, and covered by a dielectric sheet.

In the instrument landing system, the marker beacons provide distance spot checks along the glide path. The outer marker is identified by a tone modulated at 400 Hz, two dashes per second, and by a flashing purple lamp. The middle marker, placed at the Category I decision height, is modulated at 1300 HZ with one dash-dot each 2/3 second and has an amber lamp associated with it. The inner marker, placed at the Category II decision height, identifies itself with six dots per second modulated at 3000 HZ and a white lamp.

3.6.6 DATA MANAGEMENT SYSTEMS LINK

The function of the data management systems link is to provide an interface for data transfers between the booster and the orbiter. The two vehicles will share information necessary for mission termination decisions, optimizing mission guidance and control functions, and management of vehicle systems. The interface between the vehicles is a RF link as NASA specifications do not allow hard wire connections between the booster and orbiter.

3.7 OPERATIONS MANAGEMENT

The function of the operations management system is to acquire data and sensor information, provide distribution paths for the data, process the data and execute computational algorithms, analyze processing and computational results, test and monitor the status of booster systems, and present the information to the crew in a clear, accurate manner as an aid to decision procedures and mission completion. These functions are satisfied by the hardware and software associated with the

- Data Management Computer
- Data Bus
- On-Board Checkout
- Display and Control

subsystems and the associated interface units and implementation techniques.

3.7.1 DATA MANAGEMENT COMPUTER

The functions of the Data Management Computer Subsystem (DMS) are:

- Perform the computational requirements of the mission
- Provide for the exchange of data between systems
- Control system performance
- Monitor system performance
- Detect, analyze and isolate any malfunctions

To meet the requirements of a reusable vehicle, with redundant, self-check and autonomous operation capabilities, the DMS provides the central management and control procedures for the booster electronics, structural, and propulsion systems. The DMS functions are met by a combination of hardware and software. The hardware consists of:

- Digital Computers
  - Main memories
  - Processors
  - Input/Output Controllers
- Mass Storage Units

The software includes:

- Executive Control
- Computational Routines
- Preflight, inflight, and maintenance routines
- Data processing, formatting, and recording

There are many ways to configure the digital computers comprising the data management system. For the purposes of this study three configurations are selected:

- Centralized Multiprocessor
- Centralized Linked Unit Processor
- Decentralized Linked Unit Processor

A short review of some of the limitations of a single unit processor is helpful in understanding the reasons for the selected configurations. The single unit computer is composed of four basic sections:

- Memory
- Control
- Arithmetic
- Input/Output

In the simplest unit computer design every computer task, whether an internal computation or an input/output function, disrupts the computer from performing any other parallel function. This limits the total computational and input/output ability of the computer in a real time application. A measurement of this computational and input/output capability is termed the computer throughput. The throughput requirements of the data management system during periods of peak loading are beyond the state-of-the-art capabilities of a single unit computer design. Modifications to the simple single computer design can be made so that some tasks can be performed in parallel but standard designs of this type will still not meet the data management system throughput requirements. In a single unit computer a single component failure will generally disable the total computer. The space shuttle redundancy requirement by itself negates the consideration of a single unit computer.

The simplest conceptual solution to meeting both throughput and redundancy requirements is to mechanize a computer system composed of a number of single unit computers. Each computer would do a portion of the total task and the computers would be linked together through their input/output sections in order to supply other computers within the system with their partial problem solutions. For cost effectiveness each computer would be identical (except possibly in special input/output functions) and the total computational task would be portioned between the computers such that the throughput operation of each computer was nearly identical. This type of mechanization is termed a centralized linked unit processor.

Each space shuttle functional subsystem requires somewhat different computational capabilities. For example, the navigation and guidance function requires a high percentage of arithmetic computations while the displays and controls function requires mainly logic and data handling abilities. This leads one to speculate that by proportioning the tasks between the unit computers on a basis of subsystem functions and using different computer designs for the various functions, a total reduction in required computer hardware might be realized. This type of mechanization is termed a decentralized linked unit processor.

In the above discussion of centralized and decentralized linked unit processors, the redundancy requirement has not been heavily weighted. It is assumed that redundant mechanizations of each computer, along with majority voting, self test routines, and other failure detection mechanizations, would be sufficient to meet the redundancy requirements. A multiprocessor has certain advantages when redundancy is considered. A multiprocessor consists of several processors, memories, and input/output sections linked together. Each processor is capable of communicating with any memory and each input/output section has its own controller receiving instructions from one of the memories. The total operation of the processor is controlled through software by what is called an executive routine. The executive routine schedules software tasks and keeps track of the real time clock. Each processor as it completes a task returns to the executive for an assignment of the next task to be performed. In a multiprocessor a single component failure reduces the system by a single memory, processor, or input/output section, but does not eliminate a full single computer system. An additional advantage to the multiprocessor is that computational tasks are proportioned automatically by the executive and not rigidly by a preprogrammed functional distribution. The price paid for these advantages is additional executive overhead within the system.

Mass Storage Units are required for:

- Check lists for preflight, inflight and postflight securing tests
- Duplicates of operational programs
- Recording of inflight data for maintenance and postflight analysis.

These requirements can be satisfied by currently available devices such as:

- Computer memories
- Magnetic drums and disks
- Magnetic tapes

Parameters important to DMS program considerations are the capacity, access time, and information flow rate of these devices. A brief description of the characteristics of these mass storage units follows:

Computer Memories - May consist of magnetic core, wire, thin film, or LSI circuitry. Memory units of the DMS computers may be reserved to satisfy the mass storage requirements. For these devices,access time is defined as the delay between the time the control unit interprets the memory and data becomes available for use. In some memories such as magnetic core memories, an additional recovery time is required before a new request can be honored. The time between honoring requests is defined as a cycle time. The bandwidth of a memory unit represents the maximum rate at which the device can deliver or accept information. The bandwidth is defined as the number of data bits obtained with each read divided by the cycle time.

Magnetic Drums and Disks - Both consist of rotating recording surfaces running at uniform speeds. They are not started or stopped during access operations. Access time involves waiting for the desired position of the device to reach the read/write head, at which time the information can be routed to storage registers.

Magnetic Tape Units - Are sequential access devices. Data are recorded across the width of the tape in a fixed number of bits (a typical example is a byte of 9 bits with one bit being a check bit and the other 8 being information bits). Corresponding to each bit position is a read/write head used for recording or sensing information. The magnetic tape unit is started when access to its information is desired. The time it takes for the tape to accelerate to recording or reading speed is its access time. Information is then available at byte flow rates indicated by the unit's specifications. As the magnetic tape is sequential, preplanning of the positioning of data to be read during the booster mission will minimize the requirements for rewinding or search routines.

Executive Control - The executive program in an avionics computer must:

- Schedule all program modules per mission phase
- Insure all tasks are performed on time
- Respond to system interrupts

In addition, in a redundant mechanization the executive must reconfigure the system in the event of a system failure, and in a multiprocessor must assign loading to each processor. These assigned tasks must be performed efficiently so that a maximum of processor time is available to the operational programs.

Dependent upon the computer program organization the executive program may also be responsible for the routing of all input/output data from the DMS. With a data bus system and a centralized multiprocessor, it will be most efficient to have data transferred by independent data channels. These data channels will operate out of the main computer memory on a cycle steal basis. The program modules can pick up and deposit refreshed data directly into or from their operating memories.

3.7.2 DATA BUS

A large portion of the total avionics system weight can be the cabling and wiring used to electrically interconnect the various avionics subsystems. The cabling weight can be greatly reduced by using a time multiplexed data bus system. In a multiplex system all subsystems share a data bus for the transmission and reception of data. In a time multiplexed system each subsystem is given a short time interval in which to transmit its data. In a time multiplexed system no two subsystem equipments can use the data bus simultaneously. One of the major considerations in a time multiplex system is the method used to inform each subsystem when it should receive or transmit data. There are two basic methods by which each subsystem can be informed of its turn on the data bus.

One method is to use a single clock source and a clock bus. During each clocked word time each subsystem is programmed to be in one of three states dependent upon the count delivered by the clock. These three states are transmit, receive, and wait. Only one subsystem is assigned a particular clock count for transmitting. This system has the advantage of the data bus continuously handling only data and thus having the highest possible data rate capacity. In order to make the clock bus simple the clock signal is generally limited to a single pulse each word time plus a sync pulse arrangement at the beginning of each data frame.

Figure 3-22 shows the typical components of a clocked multiplex system. Each subsystem has a counter which counts clock pulses and is cleared whenever a sync pulse occurs. The counter in each subsystem should thus be in synchronism. Each subsystem has one or more registers containing a code to be compared against the clock. When agreement between the clock and clock code is achieved a gate opens and either receives data from the data bus or places data on the data bus. Transmission or reception periods are always made modules 2, 4, 8, 16, etc., of the basic clock period. One method of requesting a subsystem to stop or start transmission is to route data to the clock code registers from one or more of the data bus input comparator gates. Placing a code in the clock code register which is unobtainable by the clock counter will stop all transmissions or receptions controlled by the register.

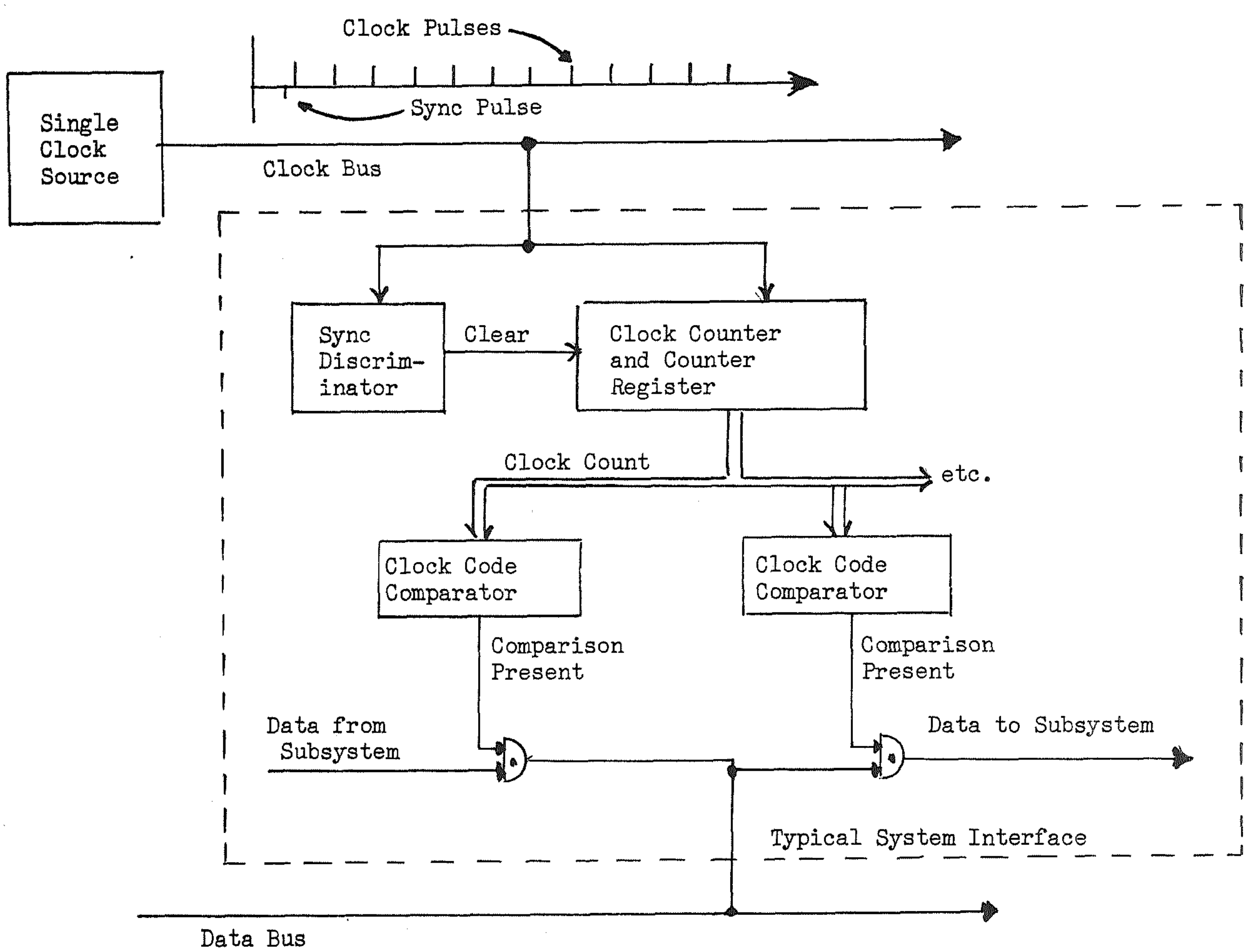


Figure 3-22 Clocked Data Bus System

The second multiplex method has the advantage of being completely under control of the central processor. In this method a coded command is issued by the central processor or a central controller. This command is received and interpreted by each subsystem. The interpretation given by each subsystem is to either transmit, receive or ignore the next data field (the data field may be one or more words). The command transmitted can contain

1. A source address
2. A destination address
3. A field length

The command can be transmitted over a separate bus similar to the clock bus or can be transmitted over the data bus. This system has the disadvantage of requiring the central processor or a central controller to provide the needed command requests.

There are numerous variations available in each of these basic methods. The method selected for use on the space shuttle booster is contingent upon required data rates from the individual subsystems and the degree of control which must be exercised by the central processor and is thus dependent upon the selected data management system configuration.

3.7.3 ON-BOARD CHECKOUT

The functions of the on-board checkout system (OCS) are to

- perform confidence tests which provide assurance that the booster is ready for launch and can perform its assigned mission

- conduct monitor and test operations during flight which will detect or predict system failures, and provide information to the flight crew which will assist in alternative or recovery procedures

- acquire in flight performance data for maintenance activities

- assist in the postflight securing and deactivating of booster systems

- assist in the maintenance cycle mode through use of data accumulated in flight to isolate failures within a subsystem, and to predict potential system failures.

The above functions are outlined in Figure 3-23

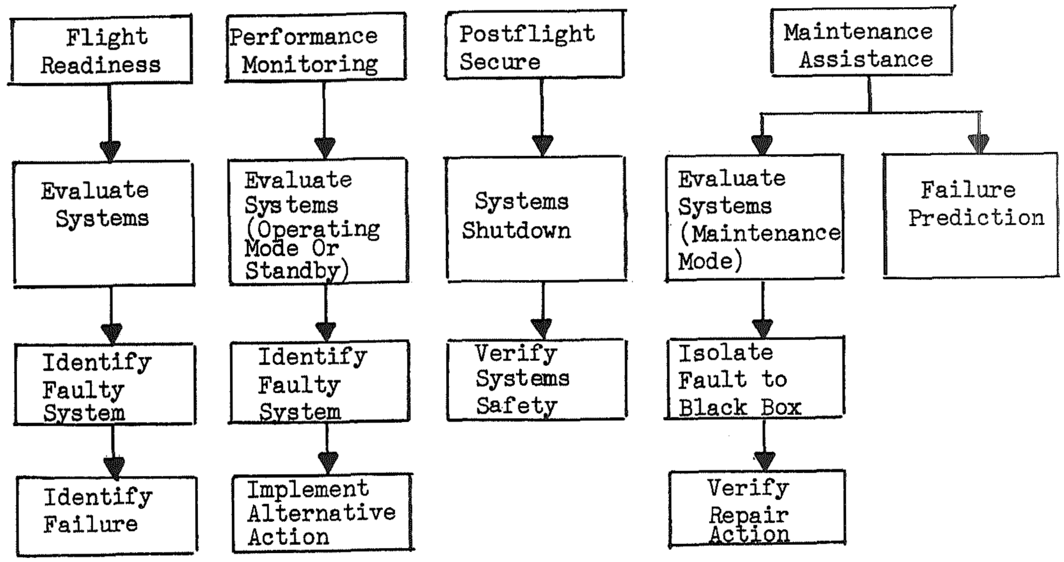


Figure 3-23 Functions of Onboard Checkout

The OCS cannot be identified as existing in any particular locality, equipment or processor. It, basically, consists of hardware associated with the other vehicle systems, and software under DMS control.

The OCS is designed to

- minimize crew participation in the checkout function so that checks are accomplished without interfering with normal mission procedures, or displaying to the flight crew a vast amount of data which has little significance

- perform every level of equipment performance evaluation from continuous on-line monitoring through fault isolation, fault re-calibration and post re-installation checkout.

- minimize equipment duplication, data format conversion, and processing procedures in conjunction with the DMS which controls checkout and maintenance processes.

The basic concept for self-check during flight is exception reporting. Limit checking of noncritical measurements will be performed on a schedule determined by characteristics of the measurements (such as sample rate, signal bandwidth, etc.). As long as measurements are within allowed tolerances, only an indication of this fact is recorded. If a quantity exceeds the pre-established limit, a time tag and the parameter value will be recorded on the maintenance history tape. The flight crew can request that certain data be recorded if concern over system performance exists or an anomalous condition is reported. Out-of-tolerance quantities will require an immediate display of the situation to the crew and entrance into a failure isolation routine. The objectives of the failure isolation routine will be to determine the failed line replacement unit (LRU), switch in a replacement, and restore or reconstruct data lost, if any, during recovery. An objective method for ensuring recovery is to conduct reasonableness tests in the DMS on input data and to verify reasonableness of outputs before data transfer. In the booster, it will be possible to verify unreasonable flight critical measurements through quantities available from redundant sources and to discard unreasonable quantities.

Alarms are presented to the crew on appropriate displays. Alarms may arise as a result of fault detection or as a result of a hazardous condition being detected by a sensor intended for that purpose. An alarm is a recognition by equipment of some unsafe condition requiring correction, and initiation of some process intended to alert the crew or bring about automatic compensation. Three categories of alarm are:

- Warning - hazardous conditions requiring immediate corrective action. Indication is effected by visually emphasized displays and audible alarm.
- Caution - potentially hazardous conditions requiring immediate attention, but not necessarily immediate action. Indication on the displays is by appropriate graphics and alphanumerics. Caution alarms may becoming warning alarms if action is not taken in sufficient time.
- Advisory - abnormal conditions in which reaction time may or may not be a constraint. The displays give appropriate emphasis depending on the particular parameter and situation.

Present estimates are that approximately 750 test and data points will be sampled and processed by the DMS. A subset of these points are regarded as critical to crew safety or mission success. Evaluation of these data sources is conducted throughout the mission, and the results of processing of these critical measurements are displayed. The current status of critical parameters such as temperatures, pressures, and consumables associated with the crew environment and power system will be displayed in a warning, caution, advisory, or normal operations mode.

Various techniques can be used to provide on-line performance evaluation. These include:

- Fault prediction - in which measured information is stored over a period of time for use in trend extrapolation and forecast of failure likelihood.

- Monitor - which involves equipment providing evaluative information about other equipment without interfering with the operation of that equipment.

- Superposition Testing - which introduces a small calibrated disturbance stimuli the effect of which can be observed in equipment operation or output, but which has negligible or compensatible effect on completion by the tested equipment of its intended function.

- Test Stimuli Testing - which verifies equipment by substitution of test inputs for normal inputs through automatic or manual switching.

In summarizing this functional description of the OCS, the general goals were found to be:

- Provide crew controlled preflight and flight capability
- Provide rapid booster turnaround capability
- Improve probability of mission success.

These goals can be met by an OCS with characteristics such as:

- Automatic continuous monitor
- Capability for crew initiation of supplemental tests
- Availability of all failure data for crew display
- Provisions for permanent record of malfunctions
- Capability for monitoring trend data.

3.7.4 DISPLAY AND CONTROL

The functions of the display subsystem are to

- Manage data for the flight crew, providing information for a particular mode of operation when it is required

- Provide uniformity of operation of the various modes by locating information in the same place and in the same way

- Combine independent data from various sensors, and, using the processing capabilities of the data management computer system (DMS), provide the crew with higher level information

- Allow the centralized display of various cues derived from independent sensors, but related to each other for flight operations in particular modes

- Permit the convenient display of symbolized alphanumerics

- Perform automatic tasks with the flight crew having manual override and supervisory capabilities.

Display and control requirements for the booster arise from its operational mission. Each phase and operational mode dictates the required functional demands such as flight control, communications, operations management, navigation and guidance, and propulsion. As the booster is both a space craft and an aircraft, with requirements for autonomous operation, on-board checkout, and redundant systems, a high degree of display automation is required to provide an acceptable crew work load and timeline. The required display information compression is provided by the use of CRT devices. These programmable devices allow the display of only that data pertinent to the current operational mode, with other data being monitored for status displays and analyzed for caution or warning classification. The design concept is based on current practice in high performance aircraft cockpits and includes additional displays and controls for boost, and reentry operations.

The CRT units may be designed to serve as special purpose devices such as in a Heads-Up Display (HUD), or may combine many system parameters to form a Multifunction Display (MFD).

The HUD is used during approach and landing. It is a projected, collimated light, see-through display with vertical situation displayed on the windshield in the pilot's line-of-sight. (An additional HUD is also supplied for the co-pilot's use or information). Special symbols denoting aircraft altitude, attitude, velocity, and command cues are projected on the windshields. The computer performs the required calculations and defines symbol locations. HUD thus provides essential orientation information when the crew is flying with eyes out of the cockpit.

MFD's are used to display system status, vehicle configuration, flight information, and navigation data. Controls provide the capability for selective display of computer or sensor data. MFD's can act as a back-up to HUD, and can present flight control information when the crew is flying in a head-down mode (eyes in the cockpit).

MFD's also provide critical cautionary, warning and advisory information. In addition to alphanumerics, the CRT will use graphics, color-coded bars, and other simple shapes so that the flight crews attention can be focused on the critical limits and tolerances.

Secondary control and display equipment is located in the overhead panel. It consists of power circuit breakers, switches, instruments and gauges which require infrequent monitoring or use during the mission.

Controls are those devices which provide for attitude and velocity control, central computer access, and subsystems selection or mode control. Cluttering of control devices is partially eliminated by mounting the jet aircraft engine throttles on the pedestal between the crewmen. The standard aircraft control yoke and rudder pedals is provided for aircraft flight control and a fly-by-wire hand controller is provided for spacecraft attitude control.

Access to the central processor is provided by way of a computer keyboard. This allows data insertion for mission parameter update, subsystem commands via computer control, or control of data recording and transmission.

Subsystem selection and mode control is provided primarily through several control panels containing switches, push buttons, thumb wheels, and twist knobs. Examples of selection and control are: CRT display mode control, checkout test override, communication channel selection, manual antenna slewing, landing gear, flap and trim tab actuation.

The capabilities of the DMS are used to implement human factor considerations for an optimum Display and Control Subsystem. The flight crew can perform their functions better if they can focus their attention on a small number of areas (one area being best). As display needs will vary with time and mission phase, the displays need not be static or dedicated to a specific task throughout the mission. In addition, the use of individual displays for each booster parameter would occupy a large area within the tight confines of the crew compartment. A programmable display configuration is used to overcome these difficulties and provide a legible, graphic, and easily understood portrayal of current information.

The need for particular parameters to be displayed is not continuous, and can be programmed to a large extent, well in advance of the mission. Displays can then be initiated by mission time-line or event occurrence. 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 DMS.

The programmable displays consist of several major devices which require use of DMS capabilities (19):

- CRT - requires refreshing and on-board checkout
- Keyboard - requires display formatting, decoding the requested function, and verification of keyboard input
- Display Parameter Selector - selects the appropriate parameters from its associated memory
- Display Control Processor - originates display requests based upon passage of time or occurrence of certain events, formats the material to be displayed, determines the screen location for every part of the display, performs the addition of title blocks and format lines, provides emphasis to portions of displays by brightness, color, or size changes, and generates the alphanumeric codes to supplement the selected parameter list
- Display Refresh Memory - accepts alphanumeric and other inputs from the processor and converts character, symbol and line codes to a form suitable for refresh memory, stores the complete display picture, and refreshes the display at the rate required for a flickerless picture on the CRT.

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Section II lists references which were surveyed during the literature search in preparation for this study. These references were also helpful in the preparation of the final report by providing general background information.

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