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FINAL REPORT

CONTRACT NAS9-10960
APPENDIX H
12 November 1971



Space Division
North American Rockwell

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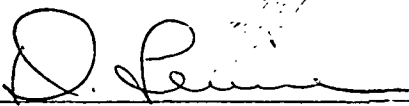
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12 Nov. 1971

SPACE SHUTTLE

ALTERNATE AVIONICS SYSTEM STUDY
AND PHASE B EXTENSION

FINAL REPORT

Approved by: _____


D. S. Levine, Director
Electrical & Electronic Systems
Research, Engineering, & Test

CONTRACT NAS9-10960

EXHIBIT H



Space Division
North American Rockwell



FOREWARD

This report presents the results of avionics system studies conducted under the Phase B Extension Study, Contract NAS9-10960, and the special studies conducted per Appendix H to that Contract, dated September 23, 1971, titled "Statement of Work, Space Shuttle Program Alternate Avionics System Study." A comprehensive summary is provided of all the configurations defined since the completion of the basic Phase B contract and a complete description of the new Avionics baseline configuration. This report supplements report SD 71-342, "Phase B Extension Final Technical Report" which describes the results of the Phase B Extension contract at the program and vehicle level.

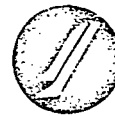


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1.0 SUMMARY

1.1 INTRODUCTION

The Avionics team formed originally for the Phase B study contract continued its efforts in performance of the Phase B-Extension period. This team includes North American Rockwell, General Dynamics/Convair Division, International Business Machines, Honeywell, Inc., and American Airlines. The primary challenge during this period was to identify means of drastically reducing the cost of the vehicle Avionics without incurring major off-setting cost on the ground. This Summary section outlines the candidate approaches considered, and the rationale leading to a new baseline configuration. Details of the subsystem trades and configuration changes are given in subsequent sections of this report.

The Avionics baseline that had evolved at the end of Phase B was an integrated system characterized by a centralized digital computer complex that performed the data control and management functions for all shuttle systems (except the main engines). The computer complex was linked to the other subsystems by a digital data bus, which provided the channels for both measurement and control signals. Adequate memory capacity was obtained by use of mass memories. Displays were multipurpose and featured variable formats. This approach provided significant advantages in flexibility, low weight (especially significant for the large orbiter of Phase B), and minimum number of electrical connections.

During the latter part of the Phase B contract, the Customer indicated that the estimated total costs for Avionics as well as the peak annual funding rate required appeared higher than the program would be able to support. Additionally, a cost uncertainty was recognized in the centralized system management approach primarily due to the amount of new development required. Consequently, Phase B' effort was directed toward defining an alternate Avionics approach that would reduce both the cost estimates and the development risk. A set of alternate configuration candidates was defined, and from them a new baseline has been selected. In the selection process, it was necessary to carefully assess the impact of the candidate configurations on other subsystems such as electrical power generation, environmental conditioning, ground support, and the crew workload in order to avoid the possibility of off-setting costs in other subsystems and operations. These factors are summarized in Figure 1.1-1.

PHASE B BASELINE

- CENTRALIZED, DIGITAL DATA BUS SYSTEM

FOR:

- FLEXIBILITY
- LOW WEIGHT
- MINIMUM ELECTRICAL CONNECTIONS

PROBLEM

- PROGRAM CAN'T AFFORD

CHALLENGE

- DEFINE NEW & DIFFERENT APPROACH TO DRASTICALLY REDUCE COST
- DETERMINE IMPACT ON OTHER SUBSYSTEMS

Figure 1.1-1. Introduction

1.2 OBJECTIVES

The objectives that were followed in defining alternate concepts to the Phase B baseline are shown in Table 1.2-1.

Table 1.2-1. Objectives of Alternate Configurations

- MINIMUM DEVELOPMENT
 - PHASED DEVELOPMENT
 - OFF-THE-SHELF COMPONENTS & SYSTEMS - NO MODS
 - SIMPLE CONVENTIONAL MINIMUM RISK CONCEPTS
- REDUCE MANAGEMENT/ INTEGRATION COMPLEXITY
 - SUBSYSTEM ACCOUNTABILITY
- DETERMINE REDUNDANCY BY INDIVIDUAL ANALYSES
 - CRITICALITY, EXPERIENCE, COST
 - MANUAL BACKUP FOR FLIGHT CONTROL FUNCTIONS
- UTILIZATION OF CREW
 - REDUCE AUTOMATED FUNCTIONS & ASSOCIATED SOFTWARE
 - SIMPLIFY REDUNDANCY MANAGEMENT - SINGLE STRING
- AVOID CONCURRENCY OF DEVELOPMENT & PRODUCTION
 - EMPLOY ALL LEVELS OF TEST FOR CERTIFICATION



In identifying the components of total Avionics cost, it is clear that development costs are by far the predominant factor. Also, as a function of time, the Phase B baseline Avionics costs peaked early; thus, contributing to a high early funding rate. Minimizing total development costs and spreading them as a function of time were therefore primary objectives. It was found practical to implement a phased approach wherein only the aircraft oriented Avionics need be developed and installed for horizontal flight tests. Development and test cost associated with vertical flight test and operations can be deferred. A fundamental means of reducing development cost is to employ available equipment and to make no modifications unless absolutely necessary. It has been shown in this study that a large amount of off-the-shelf equipment is available and can be used as-is. One of the factors that must be satisfied to accomplish this is to provide the environment required by the available equipment. It appears practical to do this.

The approaches defined in this report use conventional concepts, require minimum development and verification, and take full advantage of past experience on other programs. Single-point accountability for each Avionics subsystem is specified in order to simplify management and integration effort. For example, one source is accountable for GN&C hardware and software. This will simplify and shorten the decision-making processes, and simplify the interfaces.

The level of redundancy required for the alternate systems was determined by analyses of the various components and functional paths, considering criticality, experience, and cost factors, instead of attempting to apply a single criterion to the whole Avionics system. Consistent with current practice in aircraft, a manual backup for each flight control function is desirable and is provided.

A lower degree of automation of functions is seen in the new baseline. It was necessary to assess each candidate configuration carefully relative to crew workload and time criticality of functions in arriving at a practical but cost effective level of automation. It was found that software costs and some hardware costs (i. e., mass memories) could be reduced as a result. At the same time, the crew's redundancy management task was simplified by adopting a single-string approach in which no cross-strapping of redundant functional paths is normally done. Diagnostic programs for LRU fault isolation have been minimized. Some additional diagnostic effort on the ground is therefore required, as will be described later in this report.

1.3 ALTERNATE CONCEPTS

Table 1.3-1 summarizes the 11 alternate configurations that were considered as candidates in the Phase B¹ Avionics Study.



Table 1.3-1. Alternate Concept Descriptions

<u>SYSTEM</u>	<u>REMARKS</u>
1	øB BASELINE CONCEPT WITH HARDWIRED, DIGITAL GN&C BACKUP
1a	FEDERATED COMPUTER VERSION OF 1
2	2 DIGITAL GN&C STRINGS + ANALOG F/C BACKUP, CRT DISPLAYS, NO DATA MGMT COMPUTER
2a	3 DIGITAL GN&C STRINGS; DEDICATED DISPLAYS; NO DATA MANAGEMENT COMPUTER
3	2 DIGITAL GN&C STRINGS; ANALOG F/C BACKUP; DATA MANAGEMENT COMPUTER
3a	SIMILAR TO 3 BUT HAS 3 DIGITAL GN&C STRINGS
3b	3 DIGITAL G&N COMPUTERS; 3 ANALOG FLIGHT CONTROLS; DATA MANAGEMENT COMPUTER
3c	3 GN/DM COMPUTERS; 3 ANALOG FLIGHT CONTROLS
4	2 G&N COMPUTERS; 3 ANALOG FLIGHT CONTROLS; NO DATA MANAGEMENT COMPUTER
4a	3 GN&C COMPUTERS; DATA MANAGEMENT COMPUTER; ALPHA-NUMERIC DISPLAYS
5	3 ANALOG AERO STAB AUG SYSTEM; 3 SPACE GN&C STRINGS; ALPHA-NUMERIC DISPLAYS

The two group 1 systems both were similar to the Phase B baseline functionally and in the use of a data bus. However, they both had hardwired backup GN&C systems. The difference between the two is that one was centralized and the other was federated.

The group 2 systems had no data management computer - one had three digital GN&C strings; the other had two with an analog flight control backup.

The four group 3 systems each used a digital computer for data management, either dedicated or included in the GN&C computer. The main difference between the systems is the method of dividing the Guidance, Navigation, and Flight Control functions.

The group 4 systems were additional variations of all the above.

System 5 is the most similar to current aircraft and spacecraft approaches, and provides dedicated controls for the airplane and spacecraft modes.

These alternate concepts were jointly developed by all members of the NR team. The team also defined the criteria for comparing the eleven



candidates. Table 1.3-2 shows these criteria (characteristics) and a team evaluation of the alternate configurations using these criteria.

Table 1.3-2. Comparison of Alternate Configurations

CHARACTERISTIC	CONFIGURATION										
	1	1A	2	2A	3	3A	3B	3C	4	4A	5
ADAPTABLE TO PHASED DEVELOPMENT	NO	NO	NO	NO	NO	YES*	YES	YES	YES	YES*	YES
OFF-THE-SHELF EQUIPMENT - NO MODS	NO	NO	YES	YES	YES	YES	YES	YES	YES	YES	YES
REDUCED AUTOMATED FUNCTIONS	NO	NO	YES	YES	YES	YES	YES	YES	YES	YES	YES
MINIMUM RISK/SIMPLE/CONVENTIONAL	NO	NO	NO	NO	NO	NO	YES	YES	YES	NO	YES
MANUAL FLIGHT CONTROL BYPASS	NO	NO	YES	NO	NO	NO	YES	YES	YES	NO	YES
SIMPLIFIED REDUNDANCY MANAGEMENT	NO	NO	YES	YES	YES	YES	YES	YES	YES	YES	YES
SUBSYSTEM ACCOUNTABILITY	NO	YES	YES	YES	YES	YES	YES	NO	YES	YES	YES
PARALLEL DEVELOPMENT OF BACKUP	YES	YES	YES	NO	YES	NO	NO	NO	YES	NO	NO
DEDICATED AIRPLANE & SPACE-CRAFT CONTROLS	NO	NO	NO	NO	NO	NO	NO	NO	NO	NO	YES

* THROUGH SOFTWARE

SELECTED CONFIGURATION IS 5

With one exception, namely "Parallel Development of Backup," the desired characteristic is "Yes"; in the exception, a "No" is desired. Some explanation of the characteristics follows.

"Adaptable to Phased Development" means that a phased Kit approach to development can be used. For horizontal flight test only, the basic airplane Avionics is used; for vertical flight tests, the orbital spacecraft capability is added in a Kit; another Kit provides the operational configuration.

"Off-the-Shelf Equipment - No Mods" - Self-explanatory; satisfied by all candidate systems except the two employing data busses.

"Reduced Automated Functions" refers to subsystem management and fault isolation, not to GN&C control modes.

"Minimum Risk/Simple/Conventional" is self-explanatory.

"Manual Flight Control Bypass" means the manual flight control path (which each system has) bypasses the digital computer. Based on current aircraft configurations, this is a desired characteristic.



"Simplified Redundancy Management" refers to the single-string approach used in all systems except the group 1 data bus systems.

"Subsystem Accountability" is preserved by providing dedicated GN&C computer, not a centralized computer performing GN&C and other computations.

"Parallel Development of a Backup" means that providing the backup requires additional development over and above the primary mode development. It is undesirable because it increases the peak annual funding rate as well as the total cost.

"Dedicated Airplane and Spacecraft Controls" does not mean separate stations for crew members, but independent paths for the two modes of control.

It can be seen that the team consensus was that only System 5 had all the defined desirable characteristics (although System 3b was close). On this basis, the configuration selected for the new baseline is System 5.

1.4 DESCRIPTION OF THE NEW AVIONICS BASELINE

Figure 1.4-1 shows the Avionics configuration for horizontal flight test - the basic aircraft avionics. The diagram is drawn for the orbiter; the booster configuration will be shown later in a separate section.

The basic aircraft avionics contains a manual flight control system with stability augmentation. There are no autopilot functions, such as attitude hold, airspeed hold, etc. There is a 3-axis Stability Augmentation System (SAS) - analog - with 3 strings. Rate gyro, acceleration, and air data are provided to the SAS. Manual inputs from center sticks and pedals located at both the pilot and copilot seats provide parallel control of the aer-surfaces by a direct mechanical link to actuators and electrical signals to the SAS. Normally, both the direct mechanical links and the SAS are in operation.

The Communications and Tracking equipment in the basic aircraft avionics includes two redundant VHF AM voice links, two Audio Centers, two ATC transponders, and a single TACAN. An S-band transponder is used in this phase to provide downlink telemetry for Development Flight Instrumentation which is also recorded onboard. The Flight Log Recorder is essentially a crash recorder.

The instruments at the pilot and copilot stations include a vertical directional gyro package (VDGP) which is used to drive two axes of the electro-mechanical Attitude Director Indicator (ADI) and the Horizontal

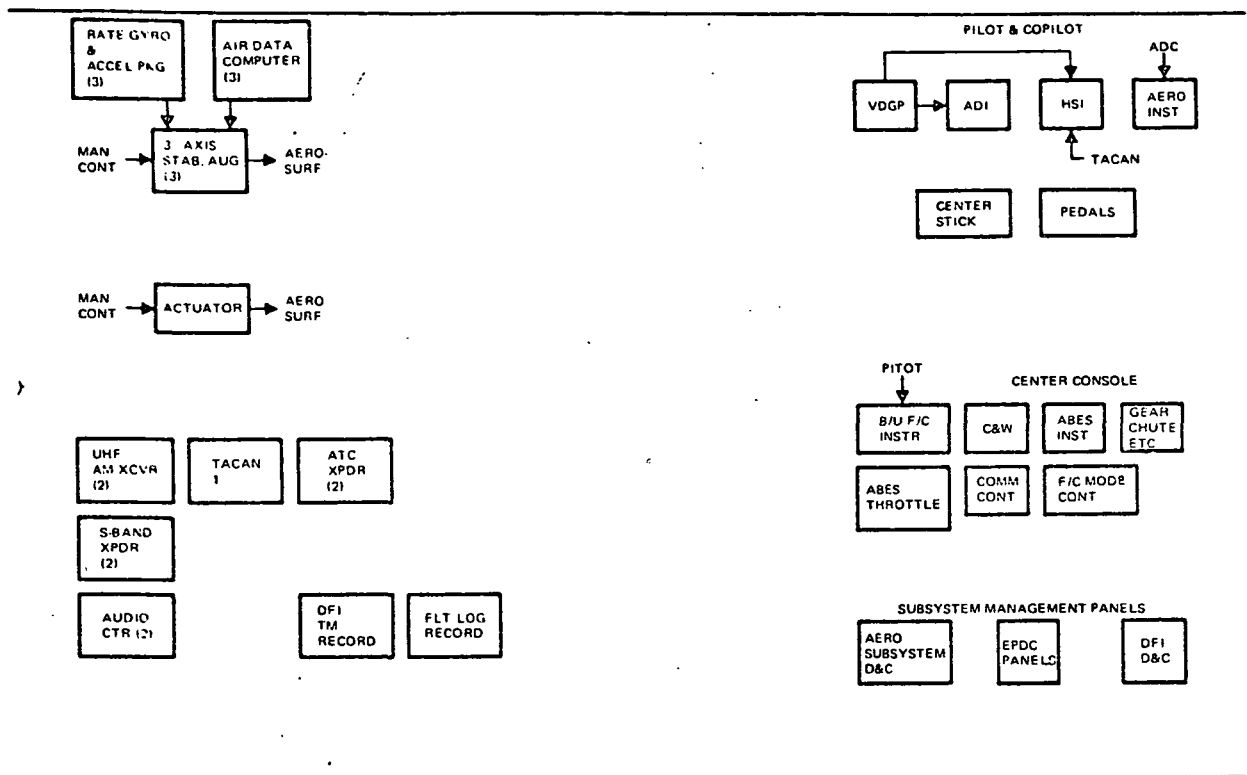


Figure 1.4-1. Interim Configuration A/C Avionics - Orbiter

Situation Indicator (HSI). The ASI also receives TACAN data. Both stations provide the usual aero instruments, driven by the Air Data Computer.

The center console, between the pilot and copilot and accessible to both, contains backup flight control instruments that are self-contained; e. g., aero instruments that operate directly off probes.

The subsystem management panels contain other secondary subsystem Displays and Controls, the DFI displays and controls, and the Electrical Power Distribution and Control switches, circuit breakers, and instruments.

Figure 1.4-2 shows the equipment added in Kit 1 to provide the capability for orbital flight.

In this Kit, three GN&C strings are provided. Each string consists of a dedicated digital computer, a 4-gimbal Inertial Measurement Unit, and a fixed base Star Tracker. The Star Tracker's primary function is to determine IMU platform orientation. The inertial system is updated during orbital operation via an updata link, using a state-vector determined on the ground from MSFN tracking data. The MSFN tracking uses the same S-band transponders which in the basic system were provided for DFI telemetry. During

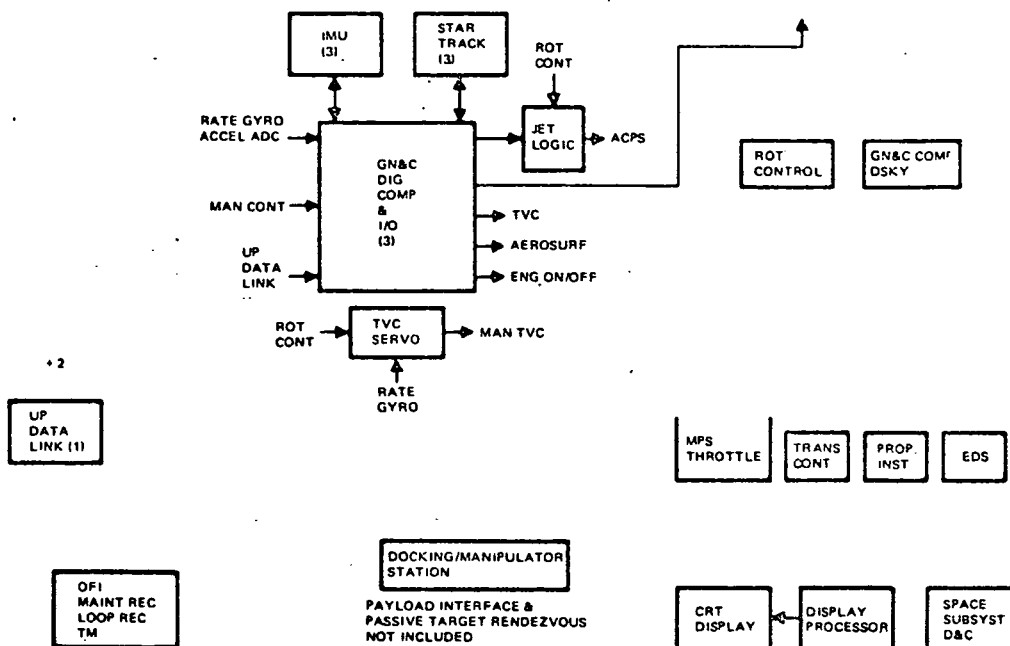
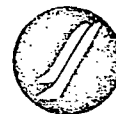


Figure 1.4-2. Kit 1

approach, the inertial system is updated using TACAN data, and in Kit 1 two additional TACAN units are added for redundancy.

The digital computer also receives inputs from the sidestick Rotation Controller, added in Kit 1, and the rate gyro, accelerometer, and air data computer packages. The outputs are commands to all three control effectors: engine gimbals, aerosurfaces, and ACPS jets. The TVC outputs go to the gimbal actuators. ACPS outputs, which are pitch yaw, and roll commands, go to analog Jet Logic boxes which generate individual ACPS commands. The aerosurface outputs go to the basic aircraft Stability Augmentation System. Note that the computer normally provides all control functions automatically, with two exceptions: the Phase B' baseline does not provide automatic landing or docking. In addition to the auto modes, the Rotation Control input to the computer provides a manual, rate damped ACPS mode for attitude control and for a manual TVC mode. Finally, the chart shows a direct manual ACPS control (acceleration command) and a rate-damped MTVC mode, both of which bypass the digital computer.

At the pilot and copilot seats, the ADI's are now driven by computer outputs derived from platform gimbal angles. Also, a DSKY is provided for computer control and display of computer data.



At the center console, provision is made for Main Propulsion System throttling. A Translation Control is added. Propulsion Instruments and the Emergency Detection System displays are located here.

On the subsystem panels, Kit 1 adds an alphanumeric CRT display driven by a dedicated digital Display Processor to assist the crew during orbital operations by simple subsystem performance monitoring.

Kit 1 also adds the Docking/Manipulator Station, located behind the pilot. It will be manned by one of the crew while the other remains at his flight station.

For Operational Flight Instrumentation (OFI), a loop recorder and a maintenance recorder are added. In case of an anomaly in flight, the loop recorder contents would be stored in the Maintenance Recorder to provide data to assist in ground resolution of the anomaly after the flight. The OFI telemetry is used only during ascent.

For the rendezvous operation, the orbiter TACAN is used in conjunction with a TACAN transponder located on a cooperative target for providing range data. Bearing data is obtained from the star tracker by tracking a light source located on the cooperative target. (For passive targets, the rendezvous aids would be part of the payload.)

Figure 1.4-3 shows Kit 2 which is added to give the operational configuration. Most important is the addition of three horizon scanners - one for each GN&C string - used operationally to provide state vector updates and thereby provide an autonomous on-orbit navigation capability (although the ground tracking capability is maintained as a backup). Note that the DFI and the VDGP's are deleted when the vehicle becomes operational.

Figure 1.4-4 is a somewhat more detailed diagram of the new baseline depicting signal flows.

The preliminary Panel arrangement for the orbiter is shown in Figure 1.4-5.

The panel is shown in the operational configuration. The pilot and copilot seats and the center console show the functional concepts described. Note the "Basic T" of flight instruments, DSKY location, center stick, rotation, and translation hand controls. The Docking/Manipulator station is also shown. It contains two TV displays, one viewing the target and the other the manipulators. Two Manipulator Controls are provided. In addition, for maneuvering the orbiter from this station, a third Rotation Controller and a second Translation Controller are located here.

HORIZON
SCANNER
(3)



Figure 1.4-3. Kit 2

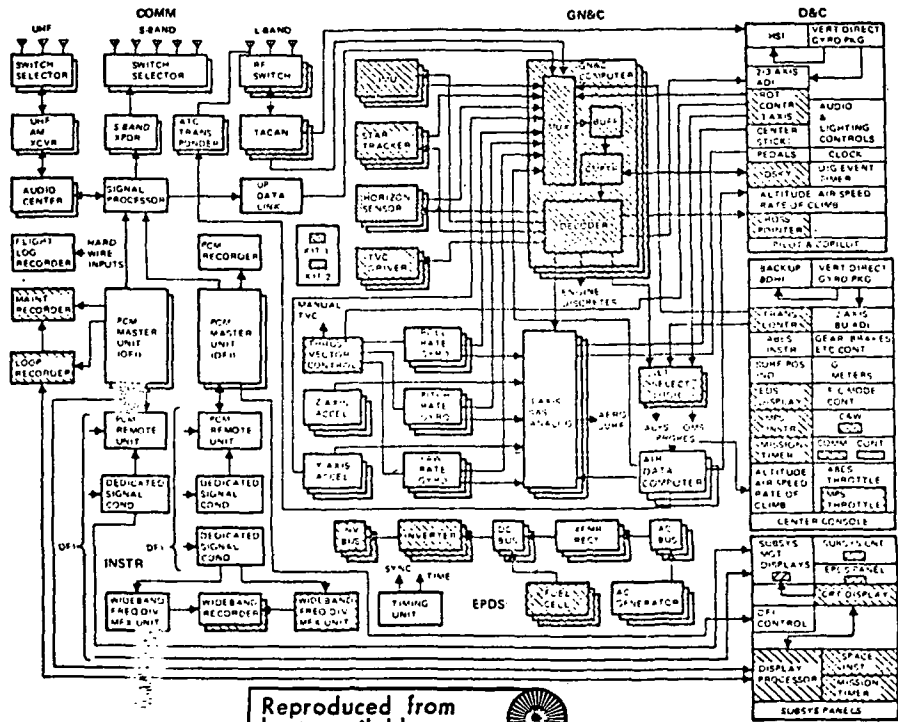


Figure 1.4-4. Space Shuttle Avionics Interim Configuration - Orbiter

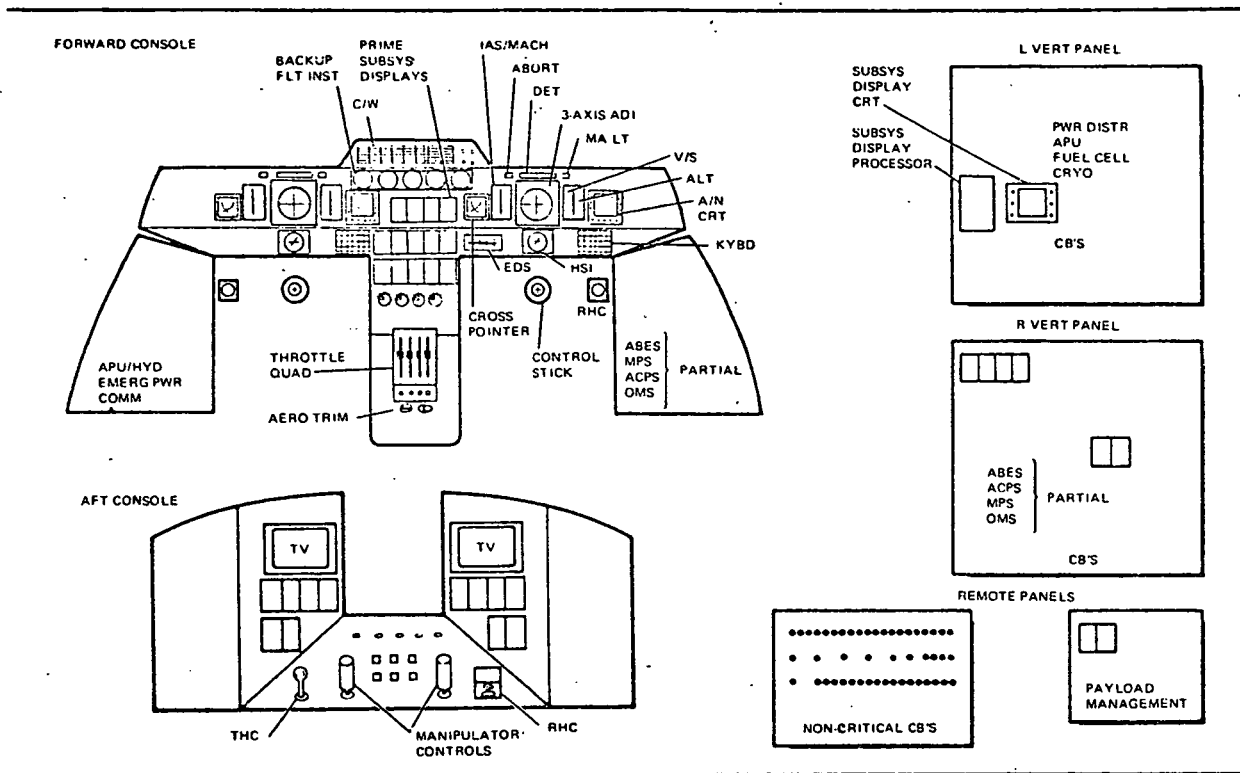


Figure 1.4-5. Preliminary Panel Arrangement - Orbiter

The functional capabilities of the new baseline are summarized in tabular form in Tables 1.4-1 and 1.4-2.

Table 1.4-1. Functional Capabilities

	A/A	AA + KIT 1	AA + KIT 1 + KIT 2
GN&C			
ATMOS FLT			
MANUAL, ANALOG SAS	X	X	X
AUTOMATIC, DIGITAL OUTERLOOP		X	X
ANALOG INNERLOOP			
ACPS			
AUTOMATIC DIGITAL		X	X
MANUAL, RATE COMMAND, DIGITAL		X	X
MANUAL, ACCELERATION COMMAND		X	X
TVC			
AUTOMATIC, DIGITAL		X	X
MANUAL, ANALOG EQUALIZATION		X	X
NAV			
TACAN	X	X	X
INERTIAL		X	X
GROUND TRACKING/UPDATES		X	X
ONBOARD UPDATES			X
COMM & TRACKING			
UHF VOICE	X	X	X
S-BAND TM	X	X	X
S-BAND TRACKING		X	X
UP DATA LINK		X	X
ATC XPDR	X	X	X



Table 1.4-2. Functional Capabilities (Cont)

	A/A	AA + KIT 1	- AA + KIT 1 - KIT 2
INSTRUMENTATION			
DFI			
TM	X	X	X*
RECORD	X	X	X*
OFI			
FLIGHT LOG RECORDER	X	X	X
MAINTENANCE RECORDER		X	X
LOOP RECORDER		X	X
TM		X	X
D&C			
PILOT/COPILOT STATIONS	X	X	X
SUBSYSTEM MGMT PANELS	X	X	X
DOCKING/MANIPULATOR STATION		X	X
ELECTRO MECHANICAL ADI	X	X	X
VDGP- 2 AXIS	X	X	X
IMU - 3 AXIS		X	X
AERO INSTR (AIR DATA COMP)	X	X	X
BACKUP AERO INSTR	X	X	X
MPS, OMS, ACPS INST & CONTROL		X	X
ABES INST & CONTROLS	X	X	X
A/N DISPLAY FOR SUBSYSTEM MGMT		X	X

*VEHICLE 3 DEVELOPMENT FLIGHTS ONLY

1.5 NEW BASELINE EQUIPMENT

One of the objectives of the new baseline is maximum use of off-the-shelf components with no mods. Table 1.5-1 summarizes the status of this objective. It should be noted that no final hardware choices, or sources, have been established; the off-the-shelf items were identified to establish reasonableness and credibility, and to obtain realistic cost data. The definition used here for Minor Mod is a mod adding no more than 10% to the cost of the existing equipment.

Of most concern for costing are the new designs or major mod requirements. In the GN&C system, an unmodified central Processor Unit will be used, but the Input/Output unit will be customized. Our tentative decision is to use an unmodified MUX for converting analog to digital data, but to develop the buffer and decoder for the I/O.

The TVC drivers and the manual TVC are unique to this vehicle and will probably be new items. The same can be said for the analog Stability Augmentation System, though it is planned to attempt to build up the SAS from components now used in airplanes. In the Displays and Control System, only the required coupler boxes appear to require new designs. Of course,



Table 1.5-1. Avionics Equipment Types By Subsystem

SUBSYSTEM	NUMBER OF EQUIPMENT TYPES				OFF-SHELF	MINOR MOD	NEW DESIGN (OR MAJOR MOD)
	AA	KIT NO. 1	KIT NO. 2	TOTAL			
GN&C	7	9	1	17	4	8	5
D&C	15	11		26	15	9	2
COMM	10	4	3	17	8	7	2
INSTR	8	2		10	5	5	0
EPDC	10	4	1	15	5	4	6

- GN&C (5) BUFFER & DECODER, TVC DRIVERS, MANUAL TVC, & SAS
- D&C (2) DISPLAY/CONTROL COUPLER BOXES, -BUILD FROM AVAILABLE MODULES
(DOES NOT INCLUDE 19 PANELS)
- COMM (2) L-BAND & UHF ANTENNAS
- EDCP (6) RELAY BOXES & DISTRIBUTION BOXES

the L-band and UHF antennas must be built specifically for this vehicle. In the Electrical Power Distribution and Control, all the functional components are off-the-shelf, but the relay boxes and distribution boxes will have to be developed. The conclusion is that the majority of major equipment items require either no mods or minor mods - IMU, star tracker, horizon scanner, DSKY's, etc. The major mods or new designs are limited to vehicle unique items.

In order to implement the Avionics effort in Phase C and D, making maximum utilization of off-the-shelf hardware, much remains to be done as can be seen in Figure 1.5-1. At the end of Phase B Prime, we have completed the requirements of the "Design Selection" block. The next task is to identify the Phase C/D statement of work requirements for test analyses/certification and technology. An off-the-shelf hardware candidate will obviously be acceptable or rejected according to the NASA imposed requirements. The urgency for the definition of these requirements by the Customer is emphasized.

1.6 KEY SELECTIONS

A detailed description of all Avionics subsystems is found in subsequent sections of this report. Certain key points relative to the GN&C and

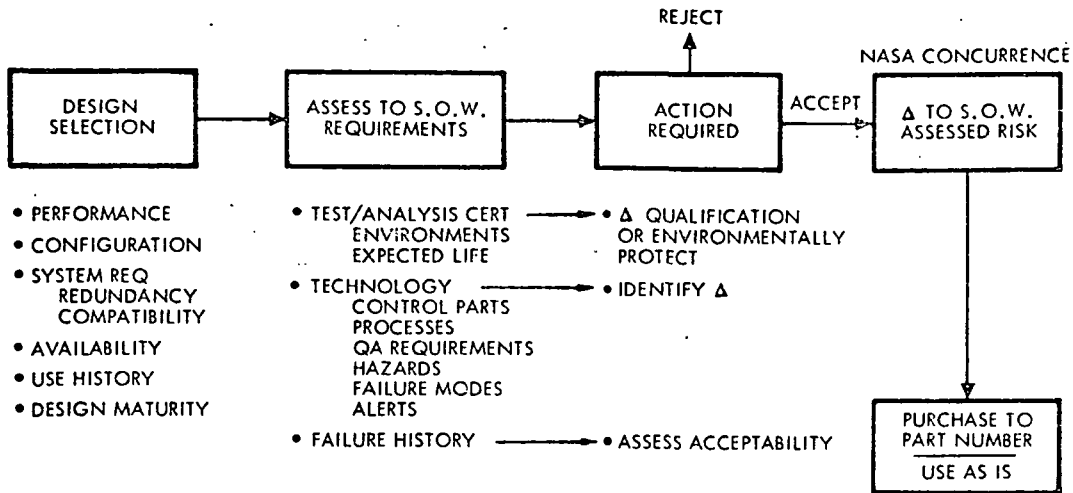


Figure 1.5-1. "Off-The-Shelf" Hardware Use Criteria (Purchase To P/N)

Communications and Tracking subsystems will be described here, along with some alternate approaches that deserve further study. Tables 1.6-1 and 1.6-2 summarize these points and are the basis for the discussion that follows.

Table 1.6-1. GN&C Selections

ISSUE	SELECTION	ALTERNATES
AIRPLANE AVIONICS	ANALOG SAS	DIGITAL
NO. OF SINGLE STRINGS	3	-
AUTOMATIC RECONFIGURATION	REQUIRED FOR 3 CRITICAL PHASES: VOTE FIRST FAILURE	SECOND FAILURE RECONFIGURATION METHODS
FUNCTIONAL BACKUP- DIFFERENT THAN PRIMARIES	MANUAL VEHICLE CONTROL	ALTERNATE WAYS OF ENERGY MGMT
PRIMARY TVC	DIGITAL	-
ATTITUDE REFERENCE SYSTEM	IMU	-
BETA CONTROL DURING ENTRY	ACPS	-
MANUAL CONTROL OF AEROSURFACES	SAS & MECHANICAL	FLY-BY-WIRE



Table 1.6-2. Communications & Tracking Selections

FUNCTION	SELECTION	ALTERNATES
SPACE-GROUND COMMUNICATION	USB TO MSFN UHF TO MSFN & ATC	RELAY SATELLITE
STATE VECTOR DERIVATION MK I MK II	MSFN-GND DERIVED HOR SENSOR - PRIMARY MSFN - BACKUP	RELAY SATELLITE PREC RANGING SYSTEM
STATE VECTOR DELIVERY	MSFN UDL PRIMARY MSFN VOICE BACKUP	
APPROACH & LANDING G&N UPDATE	TACAN	VOR/DME, PRS
LANDING AIDS	TACAN, VFR AT 2000 FT GLIDESLOPE LIGHTS	ILS, MLS, GCA, PRS
COOPERATIVE RENDEZVOUS	TACAN AIR/AIR RANGING STAR TRACKER ANGLES	VHF/AM, USB, PRS

Referring to Table 1.6-1, the basic airplane avionics employs an analog Stability Augmentation System for atmospheric flight. This is a requirement and is the result of both analyses and simulation programs related to the baseline vehicles. This is a consistent approach with that generally found in contemporary aircraft. However, a digital SAS is still considered a possibility and will be further studied.

There are three primary GN&C strings; in certain flight phases (ascent, entry, and landing) it has been established that automatic reconfiguration is required because the time factors involved in manual reconfiguration could be catastrophic. The new baseline uses voting for the first failure to detect a faulty string. Several methods for automatic reconfiguration on the second failure are still being evaluated, as will be discussed in the GN&C section. A manual backup flight control path is provided for aero surfaces, ACPS, and TVC, providing additional redundancy. However, use of these control techniques for a safe return requires development of alternate energy management techniques (independent of the computer).

Sideslip control during entry is achieved with the ACPS. In fact, the baseline uses ACPS control in entry and for 3-axes until the angle of attack is reduced, at which point all 3-axes are switched to aerosurface control. There is no blending of controls (although pitch trim is accomplished with



elevons while the vehicle is still under ACPS control). Table 1.6-1 also shows the selection of mechanical manual surface controls augmented by the electrical SAS. A full fly-by-wire manual control rather than mechanical control is still a possible alternate.

Referring to Table 1.6-2, the Communications and Tracking Subsystem includes an S-band data link and a UHF voice link to the MSFN ground stations. An ATC transponder is provided for traffic control during horizontal flight test and during approach and landing in orbiter flights. If a relay satellite becomes available, a voice and data link to the ground via the satellite would be desirable; however, such capability is not now shown or required. Similarly, a satellite system, if it is available, could be used as a nav aid; so could a long range Precision Ranging System, if it becomes available. However, the new baseline does not require either.

For delivery of a ground-derived state vector update to the vehicle, an updata link to the computer is provided rather than depending on voice communication and manual entry of the state vector data. This is based on the limited time over MSFN stations and the time required for manual updating (see Communications and Tracking Section).

Table 1.6-2 also shows that the new baseline uses TACAN for three purposes: for G&N updates during approach and landing; as a landing aid displayed on the HSI (supplemented by glideslope lights), and thirdly as a source of range data for rendezvous with a cooperative target. The listed alternates for these three functions remain potential candidates, when and if they become available.

1.7 REDUNDANCY MANAGEMENT

Redundancy Management is discussed in detail in both the GN&C section and the Redundancy/On-Board Checkout section. Table 1.7-1 summarizes key functions. Some primary points are as follows.

In the new baseline, in-flight redundancy management is performed on-board without ground support. The new baseline has simplified the redundancy management task; it does not create a requirement for ground support.

The Display Processor and alphanumeric CRT on the subsystem management panel provide the capability for performance monitoring of selected subsystem parameters, signaling anomalies to the crew and displaying relevant data. This greatly reduces the crew workload in monitoring, and in fact allows simultaneous sleep periods. In considering subsystem management, note that the indicated systems have built-in automatic safing



Table 1.7-1. Redundancy Management

-
- PERFORMED ON-BOARD WITHOUT GND SUPPORT
 - GN&C
 - FAILURE DETECTION & RECOVERY AUTOMATIC
 - PROVISIONS FOR MANUAL TAKE-OVER
 - SUBSYSTEM MANAGEMENT
 - PERFORMANCE MONITORING
 - AUTOMATIC SAFING
 - FUEL CELLS
 - APU
 - PROPELLANT TANKS
 - EPD&C
 - PARALLEL OPERATION
 - HYDRAULIC ACTUATORS
 - ENGINE BALL VALVES
 - MANUAL RECONFIGURATION
-

provisions, which means no action is required either from the crew or the Avionics system to restore a safe condition. In certain other systems, notably the hydraulic actuators, parallel operation is employed. For example, multiple actuators acting together are used to position aerosurfaces. If one fails, it is automatically deactivated by built-in equipment, and the redundant actuators continue to move the surfaces (with decreased but sufficient load capability). In such systems, no action is required by crew or Avionics to maintain operation.

In other cases, non-time critical after a failure is detected with the aid of the performance monitoring capability, the crew must reconfigure the subsystem manually.

1.8 NEW BASELINE IMPACT

Now that the new baseline has been described, consider the impact on other systems: ground, structures - installation and weight; and electrical load.

As summarized on Table 1.8-1, the new Avionics baseline still permits a two-week turnaround, and in fact slightly reduces the total program costs for pre-flight and post-flight ground support. When compared to the



Table 1.8-1. Avionics Configuration Effects On Ground

- PERMITS USE OF EXISTING GROUND SUPPORT SYSTEMS - ACE, SATURN PROPELLANT LOADING SYSTEM & BME

 - REQUIRES ADDITIONAL GROUND DATA PROCESSING & ANALYSIS
 - LRU ISOLATION IN 80% OF FUNCTIONAL PATHS VIA ON-BOARD DATA ACQUISITION

 - APPROXIMATELY 40 OF THE REMAINING 80 F.P.S HAVE 2 LRU'S

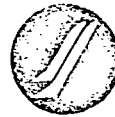
 - PERMITS OPERATIONAL TWO WEEK TURNAROUND
 - REDUCES PROGRAM COSTS - 17 MILLION
 - REDUCES DDT&E
 - INCREASE OPERATING COST
-

original Phase B baseline, some additional ground data processing and analysis is required for fault isolation. This is inevitable due to the elimination of the on-board diagnostic programs. However, it is significant that the data generated on-board during flight, supplemented by ground analysis will provide fault isolation to the LRU level in 80% of all functional paths.

The installation of Avionics equipment in the orbiter is illustrated in Figures 1.8-1 and 1.8-2.

Most of the avionics is located in the Forward Avionics Bay. In keeping with the objective of using off-the-shelf hardware, both air and liquid cooling are provided. Actually, to preserve spatial diversity of the three strings, the forward bay contains three isolated compartments, one for each string, each compartment having its own cooling system. Each compartment is maintained at a pressure differential of 0.35 psi delta p to prevent outgassing into the crew area.

Access to the avionics equipment when the vehicle is vertical is provided primarily by ladder rungs which are an integral part of the access panels. The access panels themselves open to form standing platforms. Access to the forward side of the main display console and to the balance of avionics equipment located behind the payload monitor console is provided as shown.



- INSIDE PRESSURE COMPARTMENT
- SEPARATE ECS FOR AVIONICS
- SEALED FROM CREW COMPARTMENT
- AIR ATMOSPHERE
- 0.36 PSI ΔP
- ACCESSIBLE
- 400 FT³ REOD

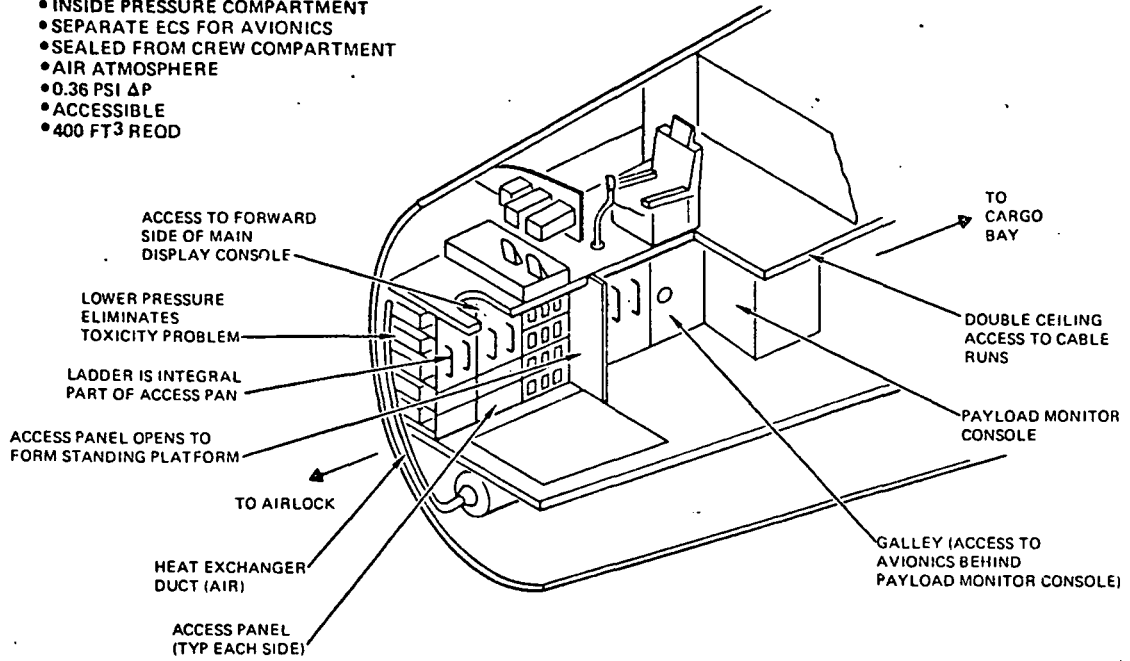


Figure 1.8-1. Orbiter Forward Avionics Bay

TYPICAL AFT BAY EQUIP:

- PWR DISTRIBUTION BOXES
- PCM REMOTE UNITS
- SIGNAL CONDITIONERS

- NEAR APU'S & MPS
- SKIN TEMP RANGE -225 F TO +350 F
- HEAT SINKS, HEATERS & INSULATION REQD
- NO ACTIVE ESC COOLING

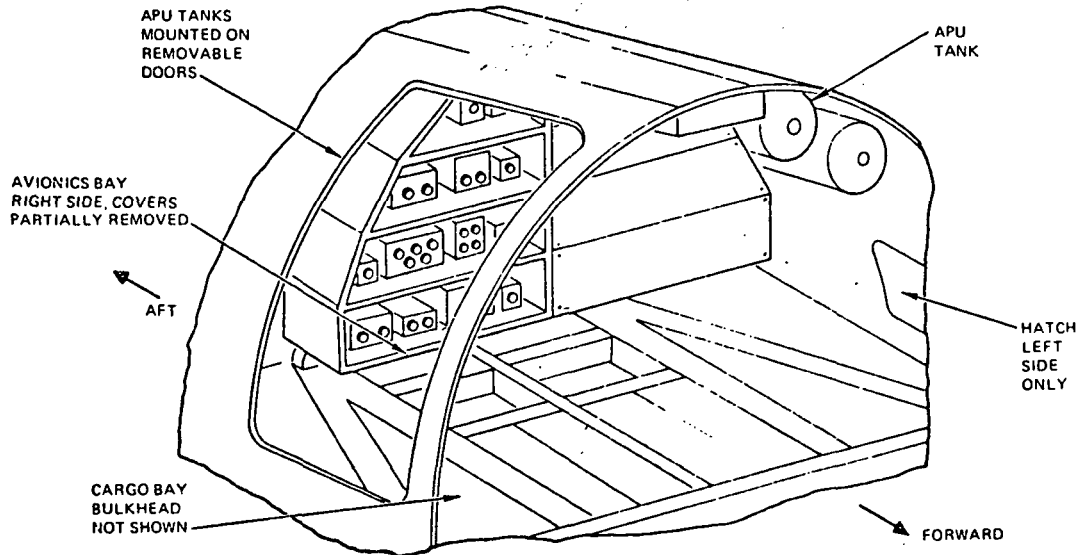


Figure 1.8-2. Aft Avionics Bays - Left & Right Sides



The aft equipment bay, shown in the second chart, contains equipment such as power distribution boxes, PCM remote units, and signal conditioners. Locating the PCM and signal conditioner units here shortens wire runs from sensors to the units.

It appears that no avionics equipment in the aft bay will require active ECS cooling. The expected skin temperature range and the use of insulation around the aft equipment bay make it possible to use passive cooling by heat sinks to protect the equipment at the high end of the temperature range. At the low end, it is expected that some of the aft bay equipment will require heaters; final analysis of this requirement is not complete.

Figure 1.8-3 shows a comparison between the new and the old baseline avionics weight and a tabulation of the weight impact on the vehicle. The operational weight of 7038 lbs for the Phase B baseline is consistent with previous reports. The new baseline weight is slightly less - 6915 lbs. This is not really a proper comparison because the new baseline reflects the results of "scrubbing" the measurement list, resulting in a significant weight decrease. If the same treatment had been given to the old Phase B baseline, it would be approximately 1200 lbs lighter than the new baseline. Weights for the flight test and operational configurations are shown. The weight of the DFI is subtracted for the operational configuration as shown. The overall vehicle impact of the new baseline is tabulated in Figure 1.8-3. Slight decreases occurred in avionics, power generation, and environmental conditioning. A 600 lb increase is incurred by the addition of the mechanical flight control equipment.

Figure 1.8-4 shows the electrical load requirement for the orbiter avionics by flight phase for the new and old baselines. It can be seen that the new loads are generally slightly reduced except for entry through landing, where the new load is slightly higher. No significant impact is seen.

1.9 DEVELOPMENT PLAN

Table 1.9-1 summarizes the Development Plan. No schedule problems have been identified for the avionics subsystems. In fact, authority to proceed (ATP) for the major avionics subcontractors can be deferred for 4 to 8 months from Prime Contractor ATP. Secondly, the program schedule indicates about a 5-year interval between Kit 1 and Kit 2 usage. This raises the question of whether it is better to phase the procurement (and encounter potential problems of production restarts), or to procure equipment early and store for 5 years (which leads to potential shelf-life problems). Such evaluations should be made for each item individually (this is not limited to avionics equipment), and this remains to be done.

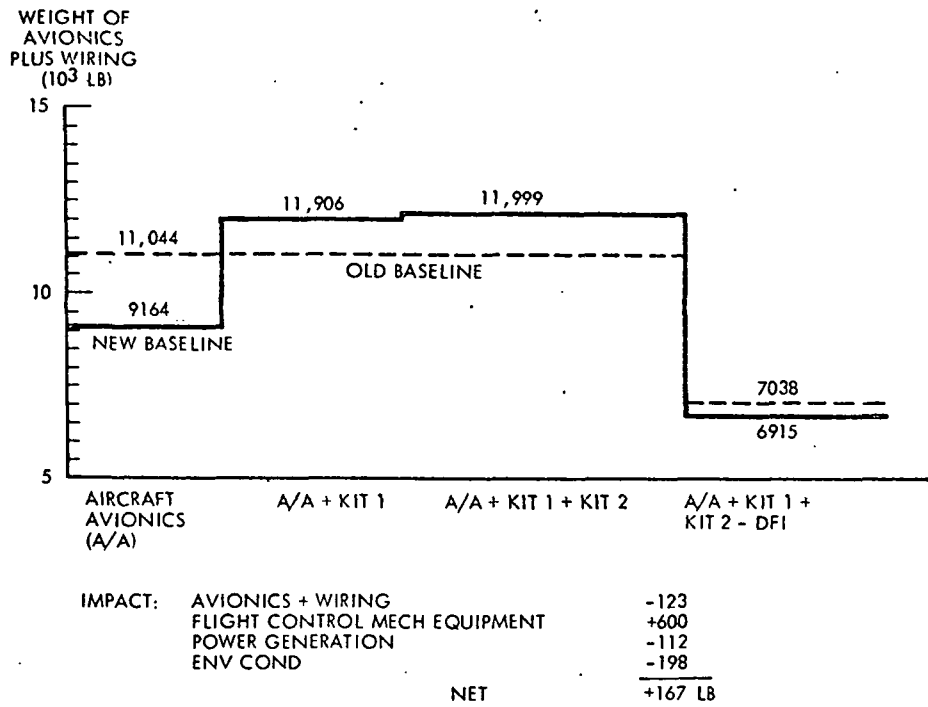


Figure 1.8-3. Orbiter Weight Impact of Selected Configuration

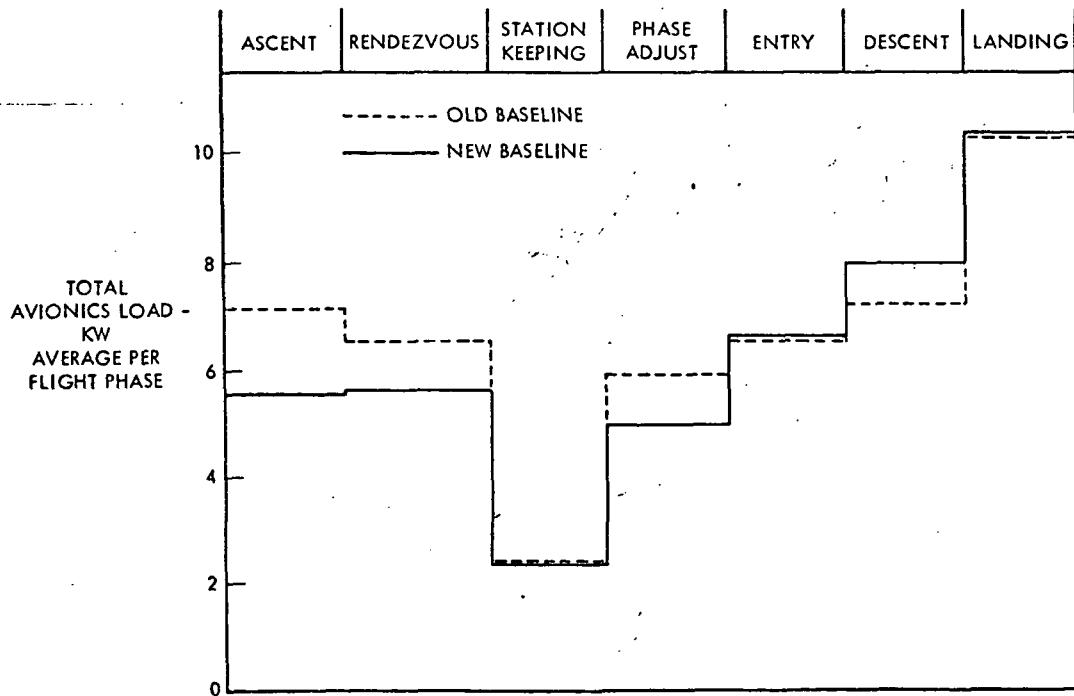


Figure 1.8-4. Avionics Electrical Load - Orbiter

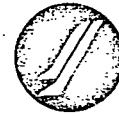


Table 1.9-1. Development Plan Summary

-
- APPROACH
 - PHASED (AA → KIT NO. 1 → KIT NO. 2) DEVELOPMENT
 - INDEPENDENT SUBSYSTEM DEVELOPMENT
 - MINIMAL COMPONENT DEVELOPMENT & QUALIFICATION
 - DEDICATED SUBSYSTEM LABS + INTEGRATED/HWWR EVALUATION FACILITY FOR GN&C/D&C REPLACES SIL
 - CONVENTIONAL TOTAL AVIONICS/NON AVIONICS INTEGRATION IN THE FLIGHT VEHICLE

 - NO SCHEDULE PROBLEMS IDENTIFIED WITH MAJOR AVIONICS SUBSYSTEMS
 - LOW RISK - SERIAL DEVELOPMENT/DESIGN/PRODUCTION
 - MAJOR SUBCONTRACTOR ATP 4 TO 8 MO AFTER PRIME ATP

 - SCHEDULE EFFECTS
 - KIT NO. 1 → KIT NO. 2 ≈ 5 YEARS
 - REQUIRES PHASED PROCUREMENT VS STORAGE EVALUATIONS
-

1.10 BOOSTER AVIONICS

This section will summarize the booster studies. Details are found in the Booster section. On Table 1.10-1, a major conclusion is that the LOX/RP Flyback version of the boosters requires its own GN&C for entry, which would also be used for the ascent flight phase. For the two non-flyback, pressure-fed boosters, no GN&C equipment is required on the booster, and ascent guidance would be accomplished by the orbiter GN&C. (The impact on orbiter GN&C software is discussed in the GN&C section; no hardware impact occurs other than signal wiring.)

Table 1.10-2 shows the functional differences between the orbiter and the fly-back booster. An S-band transmitter is added (and the S-band transponder is deleted) for DFI telemetry since the booster does not require the transponder for ground tracking during orbital operation. Similarly, no horizon scanner or up data link is required since the booster has no requirement for on-orbit state vector updates. The booster IMU does not need any alignment checks in its short flight; hence, no star tracker is required. The Translation Controller and Docking and Manipulator Station are also obvious deletions from the orbiter configuration. Next, the booster has no chutes, and the booster baseline does not provide a loop recorder because the short flight duration makes it possible to record all necessary data on the Maintenance Recorder.



Table 1.10-1. Booster Avionic Configurations

- COMPATIBLE WITH ORBITER INTERIM AVIONICS

- 3 VERSIONS
 - 1) LOX/RP - F-1 FLYBACK
 - 2) PRESSURE FED RETRIEVABLE
 - 3) PRESSURE FED EXPENDABLE

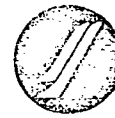
- LOX/RP FLYBACK VS ORBITER DIFFERENCES
 - SHORT MISSION - 9 MIN ASCENT, 30 MIN FLYBACK

- LOX/RP FLYBACK GUIDANCE -
 - IMU/COMPUTER REQUIRED FOR ENTRY BOOSTER
 - GUIDES MATED ASCENT

Table 1.10-2. Functional Differences

- ADDED FOR BOOSTER
 - S-BAND DFI TRANSMITTER

 - OMITTED FOR BOOSTER
 - STAR TRACKER
 - HORIZON SCANNER
 - DOCKING & MANIPULATOR STATION
 - TRANSLATION CONTROLLER
 - CHUTE CONTROLS
 - UP-DATA LINK
 - S-BAND TRANSPONDER
 - LOOP RECORDER
-



Finally, one open item in the booster configuration remains for further study. The new booster baseline at present does not have a mechanical flight control backup because feasibility for the booster has not been proved or disproved. In the interim, the booster continues to show a redundant fly-by-wire flight control system.

1.11 ORBITER HO TANK AVIONICS SYSTEM DESCRIPTION

Although the avionic system of the external hydrogen-oxygen tank were not significantly affected by the alternate avionics study, a brief description is given below to provide a complete vehicle avionics summary within this report.

1.11.1 Requirements

The definition of the External Tank Avionics System was driven by a number of key ground rules: (1) minimum cost; (2) minimum external tank hardware; (3) hardwire interface with orbiter; and (4) orbiter provision for command control prior to orbiter/external tank separation. Based upon these ground rules, analysis of the external tank mission requirements resulted in the minimum avionics system illustrated in Figure 1.11-1. The following avionics systems are required: propellant gauging, instrumentation, and separation and de-orbit.

The Propellant Gauging System, which is used for control of propellant loading and determination of propellant depletion, consists of only point sensors hardwired to the orbiter. During the development flight program, twenty-two level monitoring point sensors and five engine cutoff point sensors are located in the LH₂ tank, and twenty-two level monitoring point sensors are located in the LOX tank. The five LOX engine cutoff point sensors will be installed in the propellant feedline of the orbiter. All point sensors will be similar to those presently used on the Saturn S-II, and the orbiter will provide signal conditioning, timing, loading logic, and engine cutoff logic.

As a minimum, the Instrumentation System will consist of thirty-six transducers hardwired to the orbiter during the development flight program. These measurements are tabulated in Figure 1.11-1. The orbiter will provide excitation voltages, reference voltages, signal conditioning, multiplexing, recording, and transmission (RF link) to ground stations for all external tank measurements. Based upon early program testing, the number of measurements will be reduced so that the indicated interface (see Figure 1.11-1, approximately half of the wiring is twisted-shielded pairs) will be less for operational flights. No new hardware requirements have been identified since the proposed orbiter type transducers can be used in the external tank.

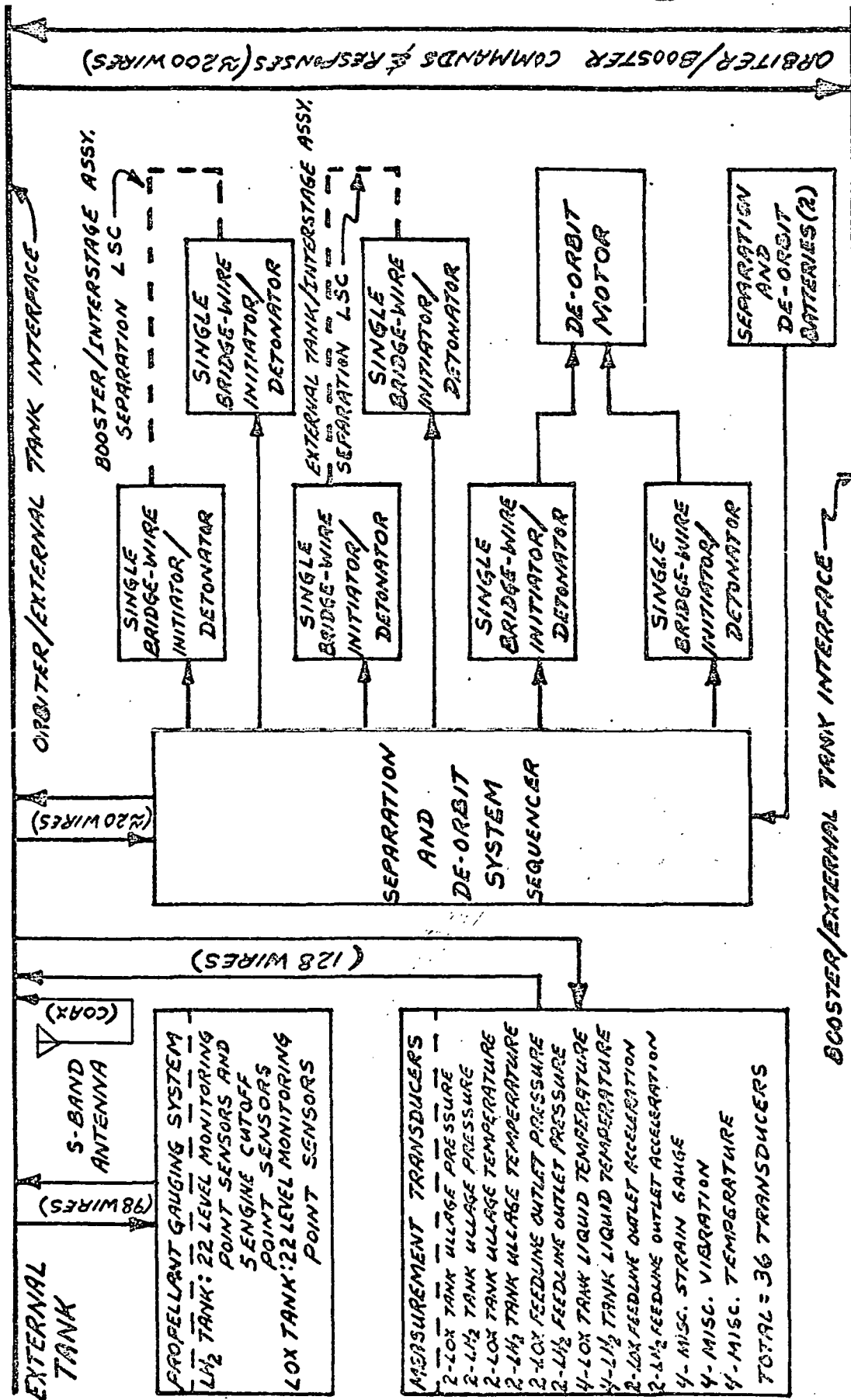


Figure 1.4-1. External Tank Avionics Block Diagram



The Separation and De-Orbit System will control the following ordnance systems: (1) booster/interstage assembly separation; (2) interstage assembly/external tank separation; and (3) de-orbit motor ignition. The system will consist of a sequencer for timing, inhibit logic, and permit logic; two (2) batteries; and six (6) single bridge-wire initiator/detonators which are similar to the electrically initiated, hotwire, igniter cartridges used on Apollo. During ground operations, the orbiter will provide electrical power to the external tank. During flight, electrical power will be furnished to the Separation and De-orbit System by the two (2) onboard batteries.

The above described system meets the established requirements and ground rules for a low-cost avionics system using proven off-the-shelf hardware. Only the minimum number of avionic components necessary to meet operational requirements were included on the external tank in order to reduce the amount of throw-away hardware.

1.12 ORBITER/TANK/BOOSTER INTERFACE

The Avionics interfaces between the orbiter and various booster configurations were defined in the study to permit sizing and physical arrangement of the interface cables and connectors. See Figure 1.12-1 for the wiring interface involving a pressure fed booster, both expendable and reusable. Table 1.12-1 illustrates the difference in number of interface wires based upon the type of booster and its role during boost flight. As shown in the Table, the TVC and command and control interface wiring is greatly reduced for a LOX/RP flyback booster, which has control of boost flight, since most of these commands are generated by the booster GN&C system. Also, since a flyback booster would have a communications system, the requirement for transmitting instrumentation data to the orbiter is eliminated. However, there would be an increase in Emergency Detection System and Separation System interface wiring because of crew safety considerations which require additional inhibit and permit logic signals. This interface wiring will pass from the booster, through the interstage assembly and external tank, and into the orbiter. The design of the connector disengagement mechanism used at each separation plane will be similar to the connector carrier plate pull-away system used on the Saturn S-II. The location of the carrier plates and the number of connectors required to meet the estimated interface wiring requirements are discussed in report section 2.9.

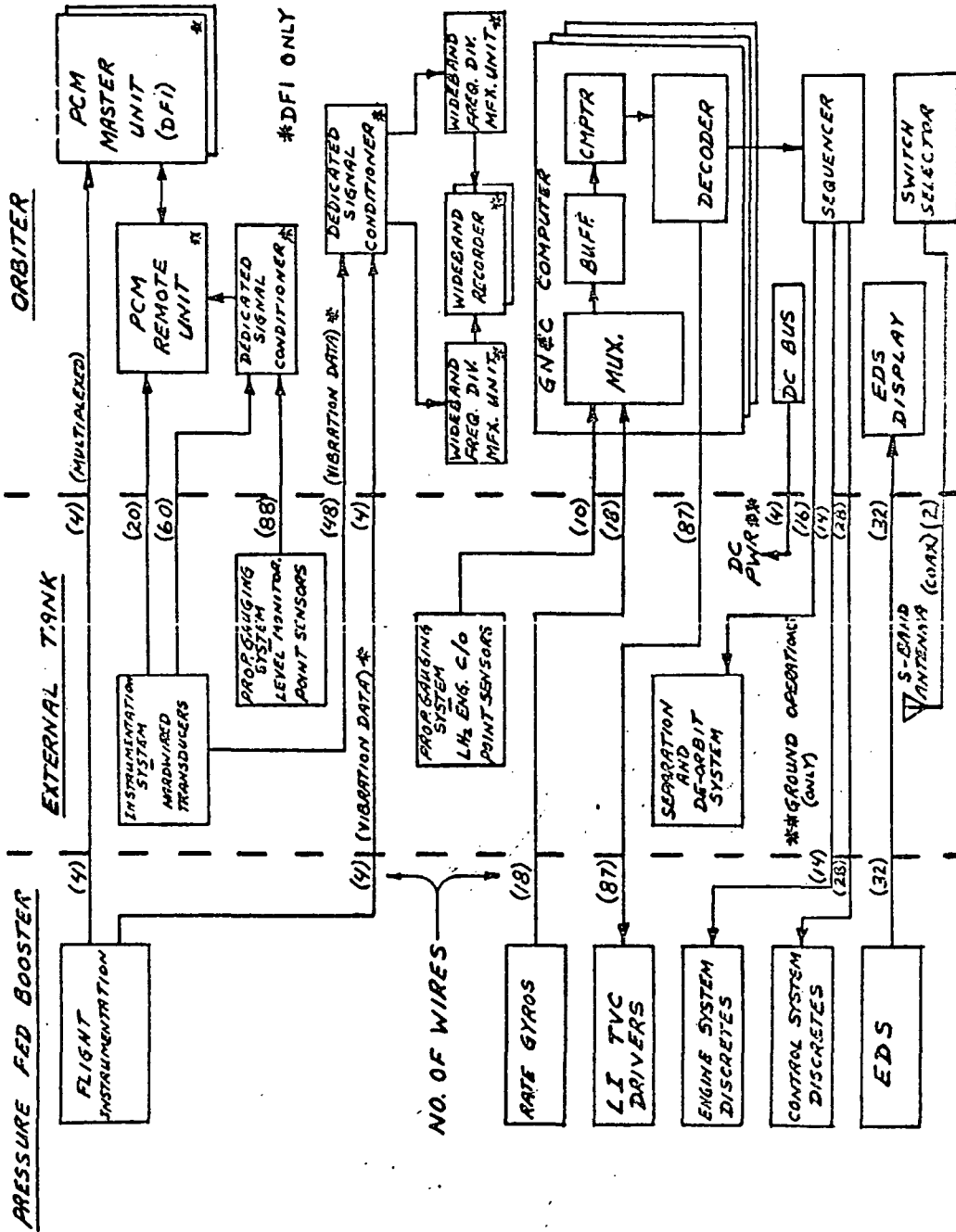
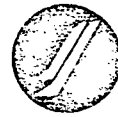


Figure 1.12-1. Booster/Tank/Orbiter Avionics Interface



Table 1.4-1. Orbiter/Booster Interface Wiring Matrix

ORBITER/BOOSTER INTERFACE FUNCTIONS	BOOSTER TYPE		
	PRESSURE FED EXPENDABLE	PRESSURE FED REUSABLE	LOX/RP FLYBACK
TVC	87	87	---
Rate Gyros	18	18	18
Emergency Detection System	32	32	40
Separation (Normal & Abort)	4	4	12
Instrumentation	8	8	---
Command and Control	38	38	16
Voice Communication	---	---	<u>12</u>
TOTAL WIRING	187	187	98



2.0 ORBITER

2.1 REQUIREMENTS

Report SD 71-342, titled "Phase B Extension Final Technical Report" is the primary document summarizing the Phase B Extension contract effort. That document is vehicle oriented and contains only a brief section on Avionics. This report supplements SD 71-342 for work done under the Phase B Extension and includes the effort performed for the Alternate Avionics System Study, a special task under the Phase B Extension contract. The program requirements, performance criteria, and vehicle configurations used in the avionics studies are defined in SD 71-342 and will not be repeated in this report. Mission requirements for ascent, separation, orbit insertion, rendezvous, docking, de-orbit, entry, and landing are included in the subsequent section 2.3, GN&C, which also describes orbiter and booster vehicle aerodynamic characteristics and mass properties. Additional requirements are found in each subsystem section.

2.2 CONFIGURATIONS

The new baseline Avionics Subsystem for the orbiter has been defined in Section 1.0, Introduction and Summary, of this report. A detailed description of the current Avionics Subsystem configurations and supporting rationale is given in the remainder of this section.



2.3 GN&C SUMMARY

This section describes the work accomplished during the Phase B' program within the GN&C subsystem. The descriptions are divided into the areas of requirements, analysis, hardware configuration development, and software configuration development. The descriptions are supported by rationale comprised of trade study results, design study results, and analyses/simulation data. This rationale is consistent with the Phase B' program objective of defining a lower cost GN&C system than that defined in the Phase B program. Major GN&C preliminary design tasks performed were as follows:

1. Modify functional requirements based upon relaxed mission objectives, less stringent avionics requirements, and modified GN&C subsystem performance requirements which better match off-the-shelf systems and equipment.
2. Update analytical results based upon new vehicle parametric data covering ascent, entry, approach, landing, and abort mission phases.
3. Provide low-cost hardware configuration including mechanization, interface definition, LRU weight/power, volume, and schedule data.
4. Modify the software configuration to meet the new baseline requirements and be compatible with the hardware configuration.
5. Optimize booster/orbiter interface to minimize data interchange consistent with the low-cost hardware objectives.

2.3.1 GN&C Requirements

The GN&C requirements are those summarized in Figure 2.3.1-1. The mission profile was analyzed considering the new vehicle configurations to determine areas where requirements could be relaxed to simplify system development.

During the mated ascent phase, basic guidance and navigation requirements remain the same as defined in Phase B. Load relief and body bending suppression are still considered cost effective due to the decrease in structural weight which they allow. While the tandem orbiter/booster configuration changes the separation procedure, the same degree and type of control during this phase are required. The expendable orbiter tanks impose an additional control requirement during separation. Booster flyback requirements remain the same for the LOX/RP flyback booster.

On-orbit operation requirements remain unchanged in that rendezvous with a cooperative target is required. The presence of a radio aid and light source on the target is baselined.

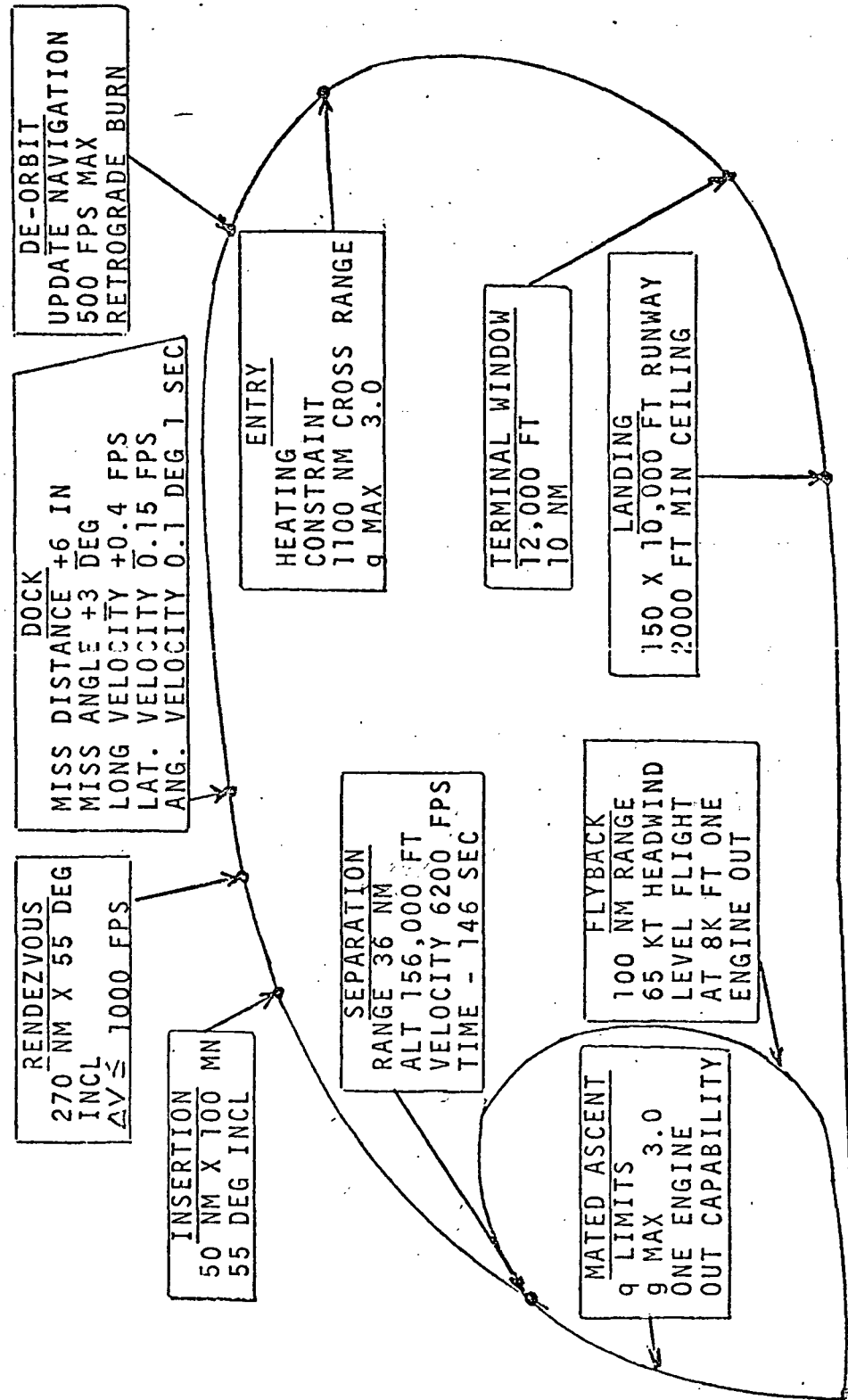


Figure 2.3.1-1 Space Shuttle GN&C Mission Requirements



The entry cross range has been established at 1100 n. mi with the guidance and control system controlling heating as well as targeting.

For the final descent and landing, it has been assumed that visual flight rules will apply with a ceiling of no less than 2500 feet.

The most significant change of the above requirements is the last, which eliminates the need for precision landing aids and automatic landing control. These simplifications are reflected in the configuration description.

2.3.2 GN&C Analysis

The major impact of the Phase B' studies has been on the physical characteristics of both the orbiter and the booster. The effort to develop a lower-cost system has resulted in a heat sink booster with a lower staging velocity than the Phase B baseline system, a smaller orbiter, and the orbiter propellants carried in an external oxygen and hydrogen tank. This tank is mounted under the orbiter fuselage and mated to the booster in a tandem arrangement.

The major portion of the GN&C analysis results reported herein primarily reflect the differences in aerodynamic characteristics and mass properties from the Phase B system. In general, the approach is similar. The ascent guidance and control concept is similar. Hence, the results show the flight control optimization results utilizing a load relief mechanization. The orbiter-booster separation concept is new, reflecting the tandem configuration. Preliminary results are favorable, although additional effort is required to optimize the concept.

On-orbit control of the orbiter reflects a pod-mounted ACPS configuration. Although this system is less efficient due to cross-axis coupling of certain thrusters, the physical design advantages of the pod concept are considered to outweigh the inefficiencies. The OMS engines are fixed, again reflecting less efficiency than the Phase B design. However, the concept results in lower cost because the actuators are eliminated. On-orbit navigation is accomplished using star trackers and MSFN rather than the precision ranging system with Kit 1, again a cost reduction item. With Kit 2, a horizon sensor will be added for more flexible navigation capability.

Orbiter entry control is accomplished through ACPS control through the alpha transition point, utilizing the elevons in a pitch trim capacity.

During the atmospheric descent part of the mission, primary control is through the stability augmentation system (SAS). Manual control is also provided with a mechanical system coupled with the SAS.



LOX/RP Integrated Vehicle Ascent Control Analyses

Early in Phase 2, a LOX/RP booster system was baselined incorporating four F-1 engines (configuration VC70-3058). Analyses to optimize the ascent trajectory and flight control system were initiated for this system. At a later date, the baseline configuration was updated to a five engine propulsion system. Due to time constraints, however, the flight control system optimization continued on the four engine version. The trends and results are applicable, however, because of the trajectory and thrust level shaping to adhere to the groundrule constraint on the maximum dynamic pressure of 650 psf. To meet this constraint, the fifth engine is ignited at launch (to provide one engine out capability), but is burned for only 30 seconds during boost. Therefore, the stabilization and control requirements of both vehicles are similar.

Booster control requirements were determined for this system. Two groundrule restrictions were imposed; namely, to use the F-1 engine and the F-1 engine actuator without modification. A third guideline imposed was to design the integrated vehicle to a structural loading criteria characterized by the shuttle Phase B criteria of maximum \bar{q}_α and \bar{q}_β of 2800 and 2400 psf-deg, respectively.

Five conditions were evaluated to determine the gimbals requirements for thrust vector control. They are to: (1) track the c.g. of both the integrated boost vehicle and the booster alone (in case of abort); (2) trim the vehicle in the event of an engine failure; (3) trim the vehicle in the event of an actuator failure; (4) an allowance for control of dynamic overshoots, bending, and slosh response; and (5) vehicle trim requirements at maximum \bar{q}_α and \bar{q}_β . For preliminary evaluations, a dynamic allowance of ± 2 degrees was used based on Phase B study results (item 4).

The gimbals requirements to satisfy items (1) through (4) above are summarized in Table 2.3.2-1. As seen, the total required travel for the integrated vehicle is within the F-1 actuator capability of ± 6 degrees (12 degrees total throw). For the booster alone, it would require a deflection of -13 degrees relative to the centerline to handle an engine out capability (including the dynamic allowance), or -15 degrees to trim an actuator failure. The total range would be +2.5 degrees (integrated vehicle) to -15 degrees (booster alone), a total throw of 17.5 degrees. This clearly exceeds the F-1 actuator capability. However, the chance of an engine or actuator failure in addition to prior failures which precipitated an abort is very remote. Hence, these failures are not imposed as requirements. The booster-alone requirement thus becomes -9.5 degrees maximum deflection. If the null position is set at -3.5 degrees, the capability of ± 6 degrees yields deflections from +2.5 to -9.5 degrees, which meets the criteria.



Table 2.3.2-1. Pitch Gimbal Requirements

	C. G. Tracking (1)	Eng. Out (2)	ACT Out (3) (Hard Over)	Dynamic (4)	Total
Integrated Vehicle	-1°/-1.5°	+0.5°/ -2.5°	0°/-4°	<u>+2</u>	+2.5°/ -6°
Booster Alone*	-7.5°	-11.0°**	-13°**	<u>+2</u>	-5.5°/ -9.5°

NOTE: Values show range of travel required.

*At Burnout

**Not a Requirement

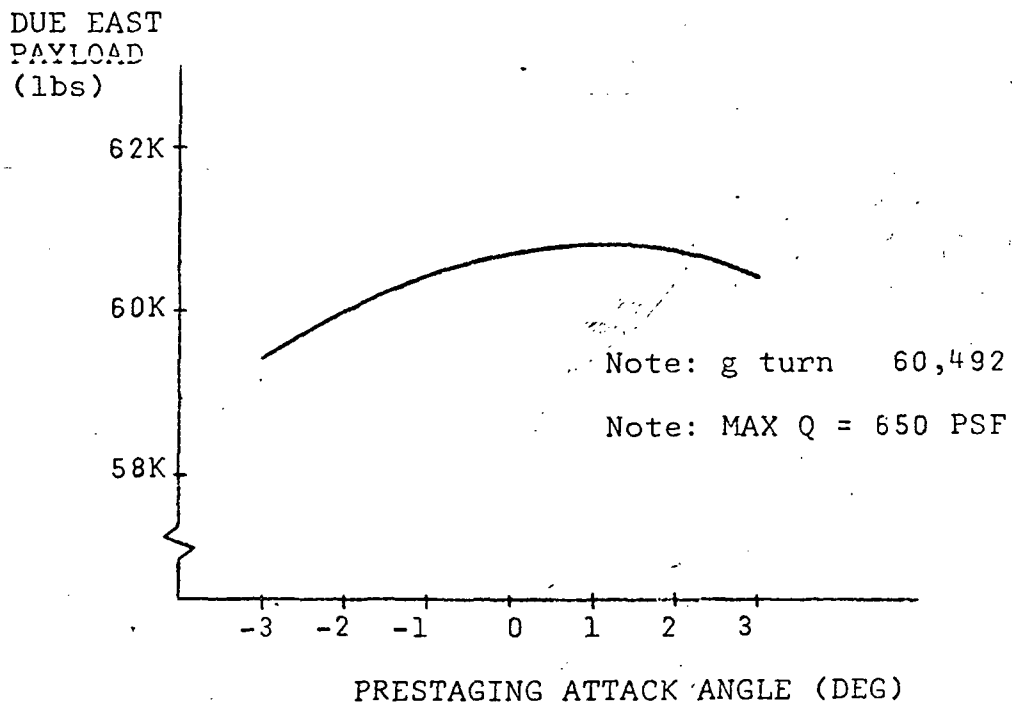
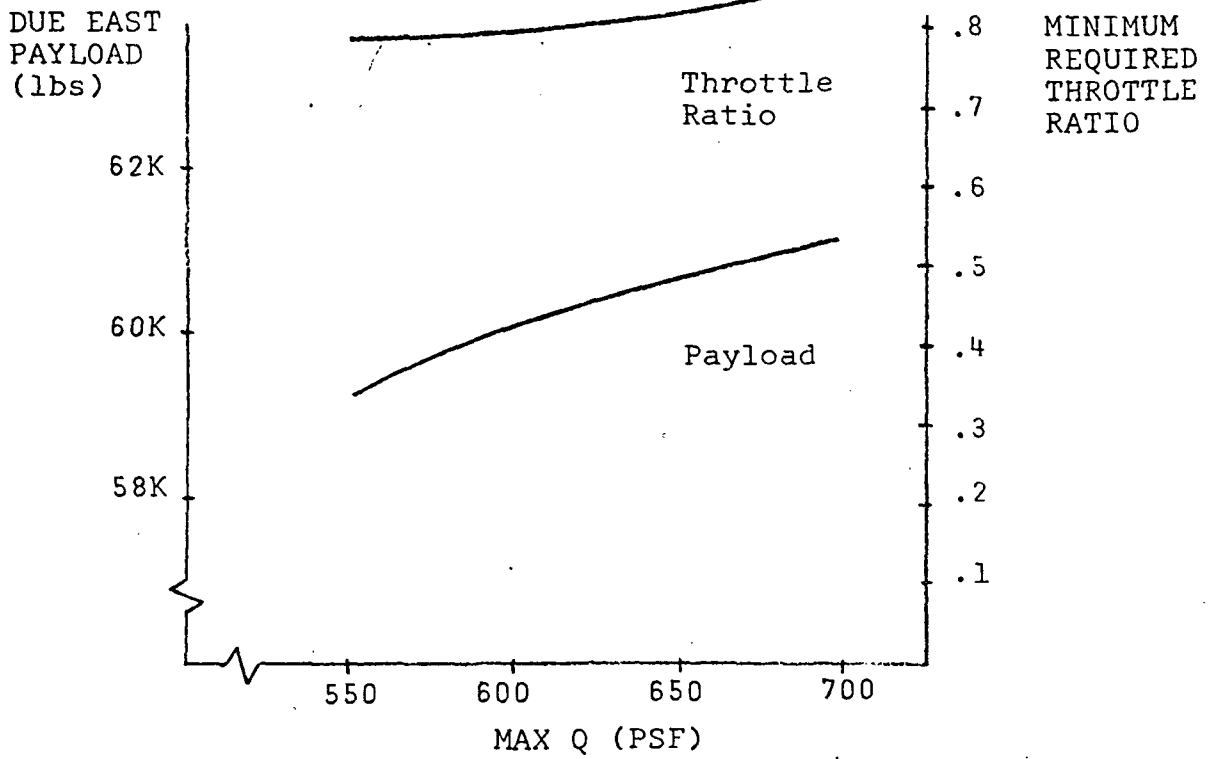
The gimbal authority in yaw and roll meets all requirements, again assuming no engine or actuator failures during booster-alone flight.

Trajectory and flight control optimization studies were conducted to evaluate the requirement to trim the vehicle at maximum $\bar{q}\alpha$ (item 5). Trajectory optimization preceded flight control optimization. Hence, these results are presented first.

Preliminary trajectory optimization results are shown in Figure 2.3.2-1. Payload delivered to orbit in a due-east mission is shown as a function of the dynamic pressure limit. For the groundrule limit of 650 psf, a payload capability of approximately 60,700 lbs is shown. These results do not consider booster flyback propellant or an OMS abort to orbit requirement. Also shown is the minimum throttle ratio necessary to limit the dynamic pressure to the abscissa value.

Payload capability as a function of the ascent angle of attack policy is shown in Figure 2.3.2-1. Whereas the Phase B baseline vehicle showed significant payload improvements for a lifting trajectory, this vehicle is essentially insensitive to angle of attack over the range of nominal interest. This characteristic will be exploited in planned follow-on studies as a tool to balance positive and negative aerodynamic loads induced by wind shears and gusts.

Flight control studies were initiated to establish booster control laws to minimize structural loads induced by wind shears and gusts, and the propellant penalties associated with flight path dispersions at staging. The control system performance was evaluated using the synthetic wind profiles found to be most critical during previous Phase B studies. Two wind profiles were



TRAJECTORY RESULTS FOR VC70-3058
FIGURE 2.3.2-1



used, one having a superimposed gust occurring at 12,600 feet altitude and one having a gust at 32,800 feet. Results were obtained using these profiles to simulate either a head wind, a tail wind, or a side wind. A pitch program for a zero angle-of-attack nominal trajectory was used in the boost simulation. The design objectives, based on Phase B study results, are as follows:

Maintain aerodynamic loads within the following limits

$$\bar{q}_\alpha \leq 2800 \text{ psf-deg}$$

$$\bar{q}_\beta \leq 2400 \text{ psf-deg}$$

Propellant weight penalty due to dispersions at burnout should be less than 2000 lbs (flight performance reserve).

The study approach taken was to assume first a simple attitude control system with fixed gains, compute dispersions and loads, and then modify the system to satisfy the design objectives. The initial performance results, obtained with a fixed gain attitude control system are shown in Table 2.3.2-2.

Table 2.3.2-2. Pure Attitude Control

Wind	Max. Gimbal Rate(deg/sec)	Max. Gimbal Deflect(deg)	Max. \bar{q}_α psf-deg	Max. \bar{q}_β psf-deg	Weight Penalty (propellant lb)
None	.3	4.82	962	84	347
Side	2.4	6.93	813	4126	210
Head	1.1	3.84	2607	74	2464
Tail	13.0	6.64	3672	99	-2370

These results pointed out both a gimbal authority problem and a load relief problem in both the pitch and yaw axes. The gimbal deflection limit of the F-1 engine is ± 6 degrees, and the gimbal rate limit is ± 5 degrees per second. Clearly, the ± 6 degree deflection limit was exceeded; the allowed \bar{q}_α limit was exceeded for a tail wind; and the allowed \bar{q}_β was exceeded for a side wind. The weight penalty objective, also exceeded for a head wind, is not considered to be a serious problem. It was concluded from these results that a load relief type control system is necessary during boost.



Design of a load relief control system was initiated to satisfy the $\bar{q}\alpha$ and $\bar{q}\beta$ design limits. The load relief system in both axes consisted of a blend of lagged acceleration and attitude feedbacks with a rate inner loop (see Figure 2.3.2-2). The gains on acceleration and attitude were varied as a function of time. Although the control effectiveness does vary over the flight, the rate gain was held constant during these preliminary analyses. The value of rate gain selected was chosen on the basis of anticipated coupling with structural modes. Further analysis will be required to optimize the gains and filtering when the vehicle becomes better defined. A simple fixed gain attitude control system was retained in the roll axis.

The results obtained with the load relief control system are summarized in Table 2.3.2-3. They show that acceptable load relief is obtained. The gimbals requirements are within the F-1 actuator capability. The $\bar{q}\alpha$ is within the groundrule design limit. The $\bar{q}\beta$ results indicate a potential reduction in this limit. Although the propellant weight penalty (2287 lb) exceeds the guideline limit of 2000 lb, the result is considered satisfactory.

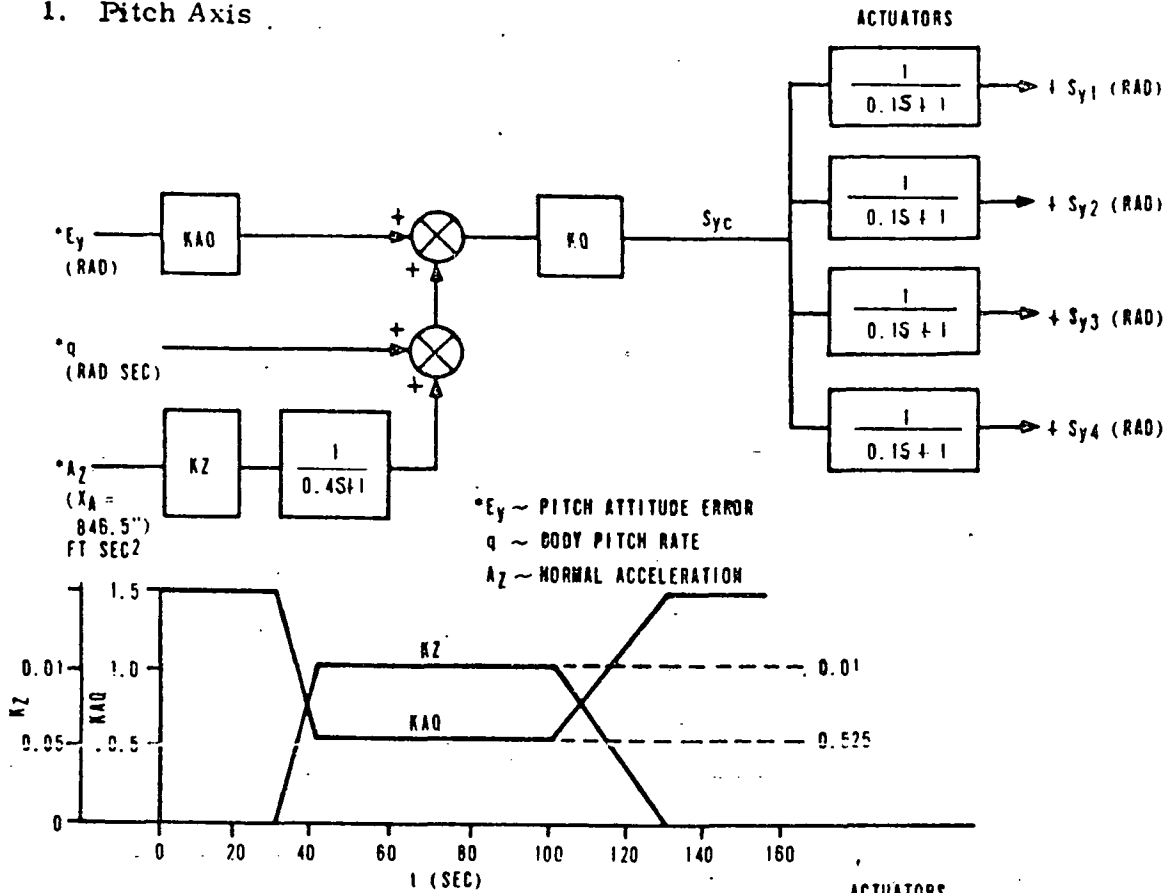
A significant point which should be mentioned is that a low gain roll attitude control was used to obtain these results. It was found in the study that the allowed $\bar{q}\alpha$ and $\bar{q}\beta$ constraints could not be satisfied with reasonable weight penalties unless the vehicle was allowed to roll. If the vehicle cannot be allowed to roll, then it is doubtful the aerodynamic load constraints can be satisfied using load relief control. Of particular interest is the case of a tailwind. In this situation, the vehicle will roll (assuming a low gain roll attitude control) automatically due to the fact the aerodynamic center lies above the center of gravity in the body Z axis. Because of this natural weather cocking tendency, the aerodynamic load is automatically shared between the pitch and yaw axes, thereby keeping the loads within their allowed limits.

Orbiter/Booster Separation

The nominal separation sequence is shown in Figure 2.3.2-3. Prior to separation, the orbiter engines are pre-aligned to provide a pitch down moment on the orbiter. The sequence begins at booster engine cutoff. When the sensed vehicle acceleration decays to 1.0 g, the orbiter engines are ignited. As the orbiter thrust builds up, the engine pre-alignment creates a pitch down moment on the mated vehicle. The booster engine is commanded to hold a zero pitch rate. Separation is initiated based on a timing sequence which is set such that the orbiter T/W exceeds the booster T/W. The pitch down engine moment on the orbiter causes the overhang of the orbiter tail over the booster to pitch up and away from the booster. Adequate clearance is thus assured. The engine pre-alignment angle is sized so as to provide equal vertical accelerations of the orbiter and booster separation planes. The combination of (nearly) zero pitch rate of the mated vehicle and the



1. Pitch Axis



2. Roll Axis

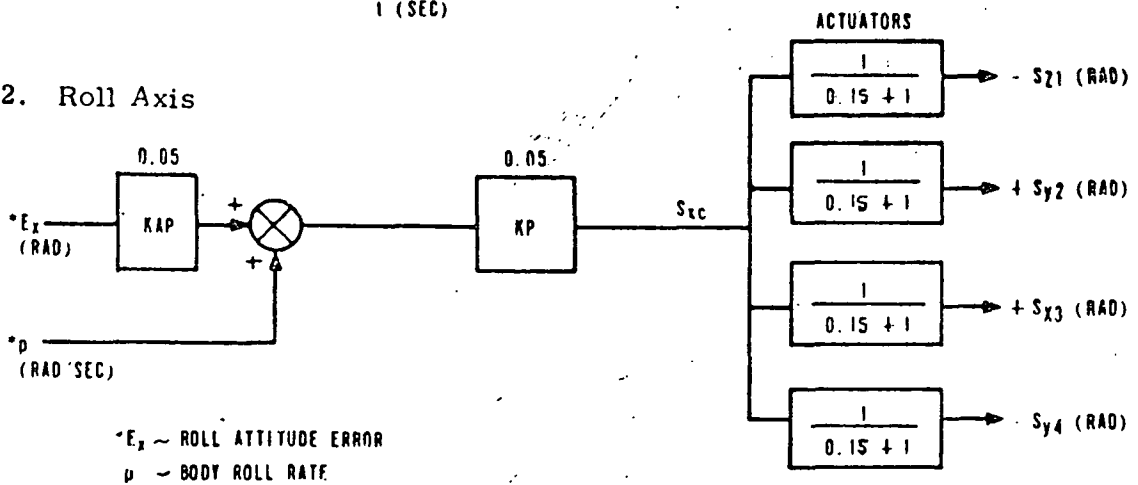


Figure 2.3.2-2. Mated Boost Flight Control Analytical Block Diagram



3. Yaw Axis

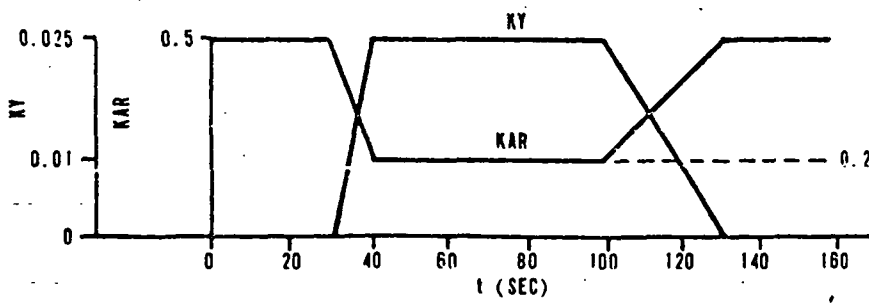
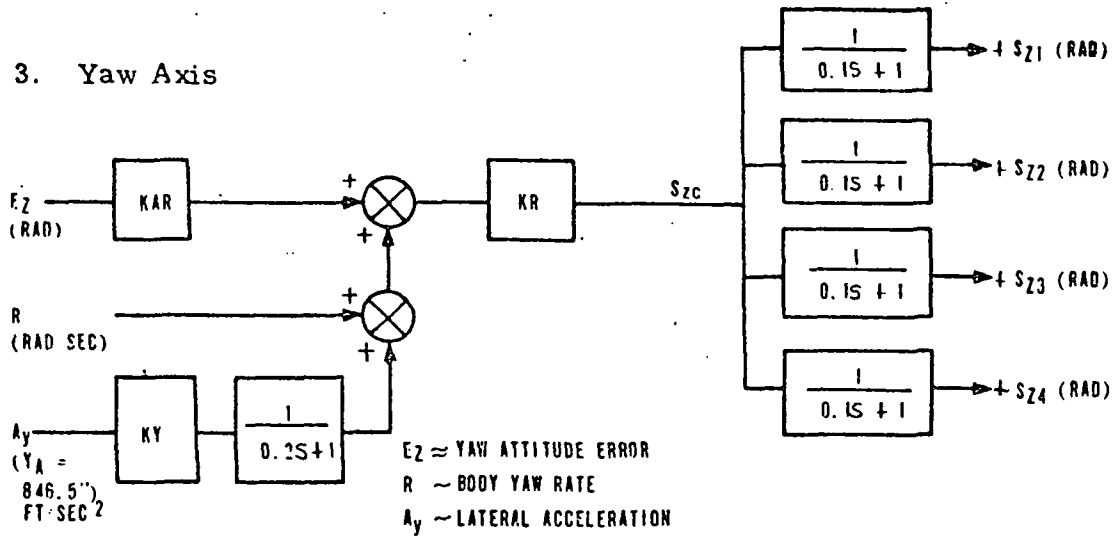


Figure 2.3.2-2. (Continued)

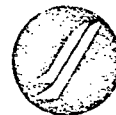


Table 2.3.2-3. Load Relief Control

Wind	Max. Gibal Rate (deg/sec)	Max. Gibal Deflect (deg)	Max. $\bar{q} \alpha$ psf-deg	Max. $\bar{q} \beta$ psf-deg	Propellant Weight Penalty, Lb.
12,600 ft Side	1.9	2.74	1891	2073	1110
12,600 ft Head	1.7	3.71	2146	44	1613
12,600 ft Tail	1.4	5.53	2772	1396	752
32,800 ft Side	1.3	3.16	2745	595	1497
32,800 ft Head	1.2	4.7	2212	72	1709
32,800 ft	2.4	5.77	2091	1991	2287

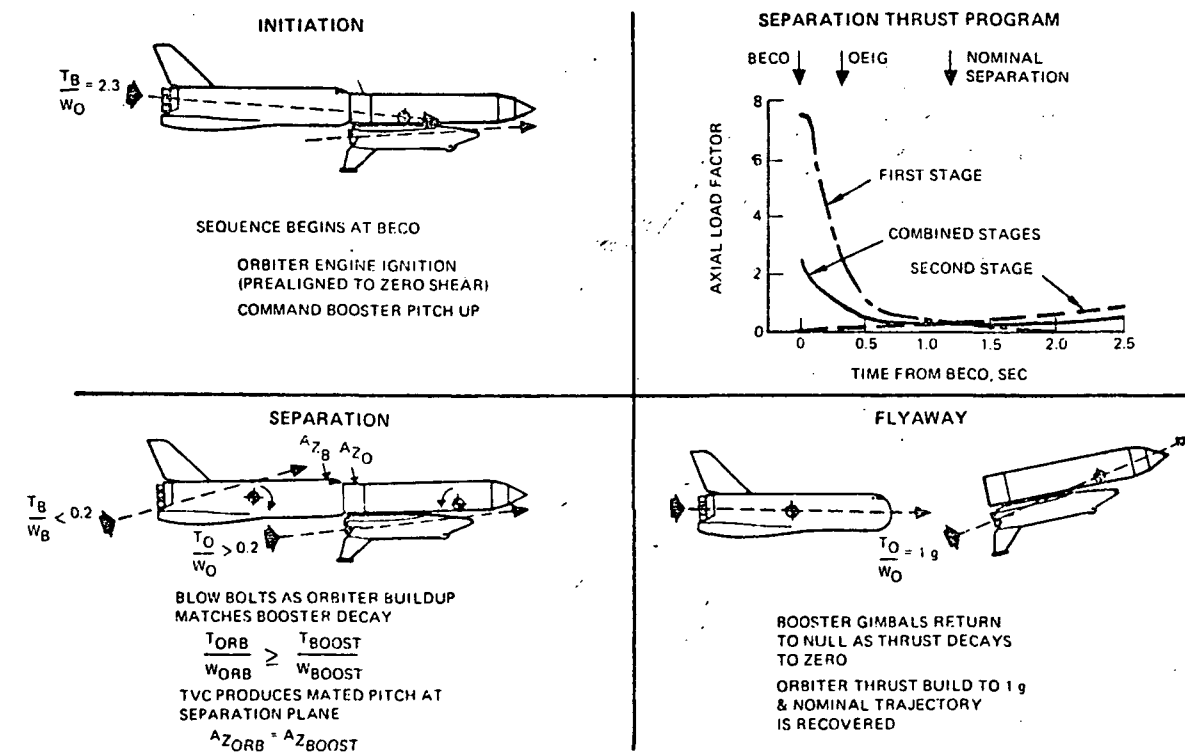


Figure 2.3.2-3. Orbiter/Booster Separation



(nearly) equal accelerations minimizes the tendency of interference transients at the separation plane. At separation, the booster engines return to null. After a short time delay to allow a proper separation distance, the orbiter pitch control system is activated to recover from the separation transient.

In preliminary studies, the sequence described above was found to provide adequate tail clearance both in and out of the atmosphere. Nominal staging, high \bar{q} abort, and pad abort were investigated. It also is relatively insensitive to timing errors, thrust time history uncertainties, wind and gusts, c.g. location, and variations in the thrust vector control system. Some pre-staging angle of attack control may be necessary at high \bar{q} .

The orbiter/booster separation is not without problems. The sequence theoretically provides zero vertical acceleration between the interstage and booster at the separation plane. However, the thrust buildup of the orbiter will void this equality as a function of time. The result is that there will be some interference at the separation plane (see Figure 2.3.2-4). Further effort is in process to assess sequence variations and configuration design solutions to eliminate the potential problem.

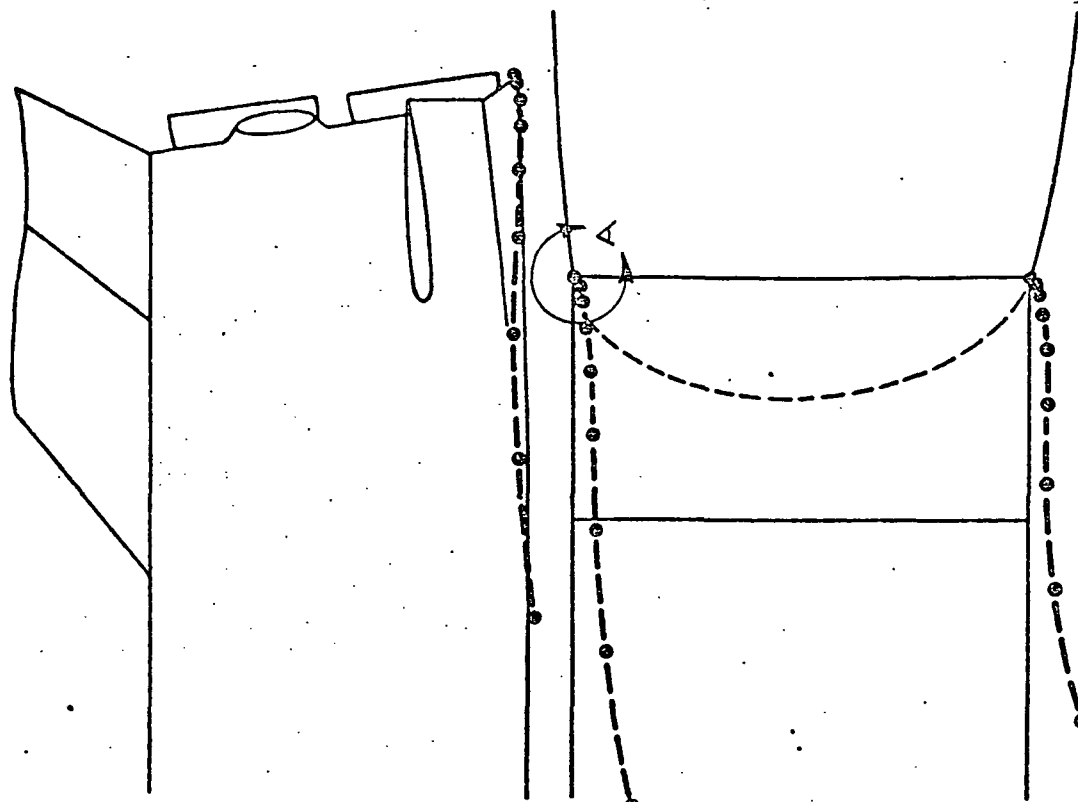
Orbiter Ascent Control Requirements

The orbiter MPS consists of four J-2S engines and utilizes off-the-shelf J-2 actuators. The total gimbale throw available is 14.6° peak-to-peak. The gimbale requirement is defined by the c.g. travel during boost, the capability to trim engine or actuator failures and an allowance to account for dynamic transients and bending and slosh suppression. The nominal c.g. travel requires thrust vector angles in pitch from -6.3° to -13° relative to the fuselage centerline. To additionally handle one engine out, this range increases to -4.25° to -14.25° . An actuator failure hard over exceeds the J-2 actuator capability. An actuator failed to null requires -5° to -14.75° . If a dynamic allowance of 1.5° (estimated from Phase B studies) is superimposed, the total range becomes -2.25° to -16.75° , a total range of 14.5° . The J-2 actuator can thus control the orbiter, but requires a fail to null actuator design.

Tank Separation

During the ascent coast, the external tank is separated from the orbiter and given a retro impulse so that the tank will enter and impact at the azimuth nodal point (-28.6° lat.) in the Indian Ocean. A study of the projected shipping densities in this area resulted in the following requirements for tank deorbit:

1. 300 ft/sec retro velocity
2. Thrusting attitude in plane with and 40° above velocity vector



- EQUAL VERTICAL ACCELERATION AT THE SEPARATION PLANE
- TWO F-1 ENGINES OPERATING

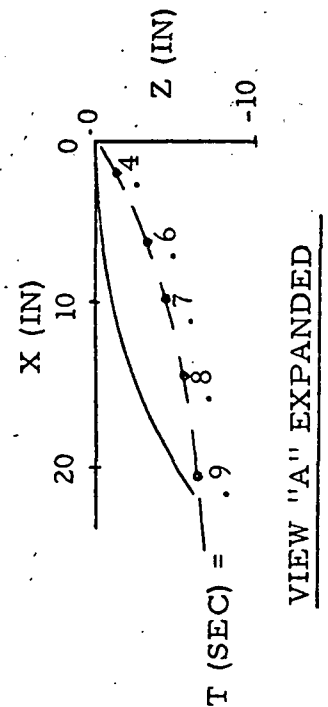


Figure 2.3.2-4. Separation Dynamics



3. Allowable attitude error during the retro burn = 30° , about the desired thrusting attitude.

4. No orientation or stabilization requirement during entry.

These requirements account for additional uncertainties such as injection errors, timing errors, retro impulse errors, and aerodynamic variations due to tumbling or non-tumbling entry.

Four separation and de-orbit concepts were identified. They are: (1) eject the tank from the orbiter and fire the deorbit motor on a time delay signal; (2) release the tank and translate the orbiter (fly-away), and fire the deorbit motor on a time delay; (3) install an active attitude control system on the tank; and (4) install the tank on rails and thrust it off the orbiter. The first two concepts were both proven feasible; therefore, an active control system control system concept was discarded. The rail concept in comparison imposes too high a weight penalty. It, too, was discarded.

The tank ejection concept is sensitive primarily to uncertainties in the delivered impulse of the ejection devices (best estimate is a $\pm 10\%$ uncertainty in each), the tank c.g. (estimated at 1-foot), and thrust vector misalignment of the deorbit motor (estimated as $1/4$ degree). These sensitivities are displayed in Figure 2.3.2-5, which shows their effect on attitude rate at separation, and the attitude error as a function of the tank clearance distance from the orbiter. Since the latter is independent of separation velocity, a nominal value of 5 ft/sec was selected. A clearance distance of 50-feet before deorbit motor ignition was selected to minimize plume impingement on the orbiter. For the estimated impulse and c.g. uncertainties, the attitude rate at separation is approximately 0.75 deg/sec. At a clearance distance of 50 feet, the attitude error is 7.5 degrees. The thrust vector misalignment will cause further attitude error during the retro burn, the magnitude of which is shown in Figure 2.3.2-5 as a function of the thrust level. To achieve an average error less than the allowable 30 degrees will require thrust level of at least 33,000 lb. The associated burn time is 17 seconds to achieve a 300 ft/sec ΔV .

An alternate to this concept is to spin stabilize the tank prior to deorbit motor ignition. For the current tank configuration and mass properties, however, it did not offer sufficient benefit to warrant the addition of a spin system.

The fly-away concept, shown in Figure 2.3.2-6, is less sensitive to uncertainties than the ejection concept. Attitude rate errors at separation are established by the capability of the orbiter flight control system. The only other significant error source is the deorbit motor thrust misalignment. Since the attitude rate errors are smaller than for ejection devices, a larger

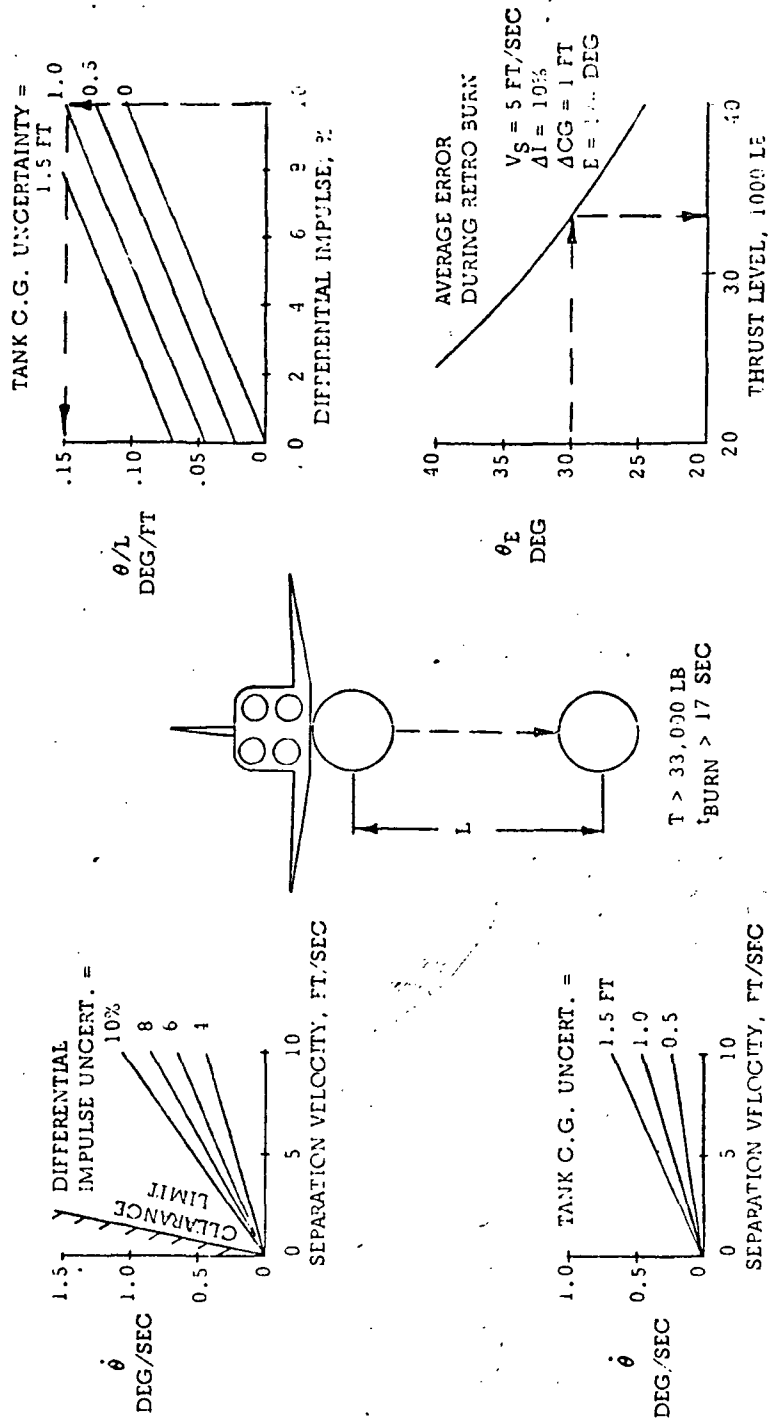


Figure 2.3.2-5. Tank Separation and Retro Errors - Tank Ejection Concept

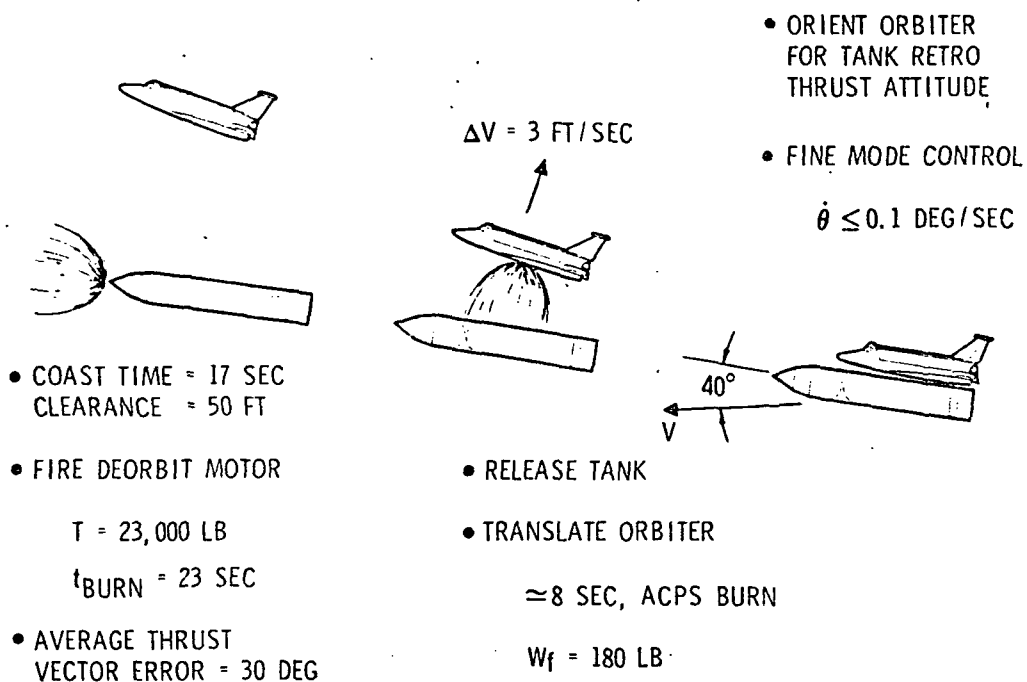


Figure 2.3.2-6. Nominal Tank Separation Concept

attitude error due to thrust misalignment is permissible. To maintain an average error no greater than 30 degrees dictates a retro thrust level of 23,000 lbs or greater. The corresponding turn time for 300 ft/sec is 23 sec.

The fly-away concept does not require ejection devices, and requires a smaller thrust deorbit motor. It was thus selected as the baseline concept.

During ascent coast, the mission sequence will first position the orbiter so that the retro thrust vector will be oriented to the 40 degree thrusting attitude. Tank release will be timed so the tank is targeted to the impact point. After release, the orbiter will translate away from the tank in the z-axis direction. A nominal velocity of 3 ft/sec has been selected. After a 17 second delay to achieve the 50 foot clearance, the deorbit motor is ignited.

The z-axis translation at 3 ft/sec requires an 8 second burn of the ACPS thrusters, expending 180 lb of propellant. Further effort is planned to trade this weight penalty against a lower velocity, a longer coast delay time, and the resultant higher deorbit motor thrust level to maintain the allowable 30 degree attitude error.



Orbiter On-Orbit Control

The orbiter is equipped with 32 ACPS thrusters contained in two pods located at the wing tips and a third pod mounted on the vertical tail (see Figure 2.3.2-7). The thrusters develop 1050 lbs of thrust and have an Isp of 290 seconds. Maneuver propellants, shown in Table 2.3.2-4, represent requirements for 1 deg/sec rotational maneuvers, 1 ft/sec translation, and +1 deg limit cycles.

Table 2.3.2-4. On-Orbit Maneuver Propellant Consumption

Rotation		<u>On-Orbit</u>	Limit Cycle	Rotation	<u>*Docking</u>	Translation
ϕ	2.7		.00244	2.7	X	42
θ	12.1		.00543	54.4	Y	84
ψ	22.2		.00294	22.2	Z	147

*Includes propellant required for cross-coupling compensation.

The roll and yaw rotation ACPS control are pure couples but pitch rotation results in both X and Z translation. For translation control, plus "X" translation is accomplished without cross-coupling, plus "Y" translation produces 0.325 deg/sec² yaw angular acceleration, and "Z" translation results in 0.675 deg/sec² in pitch. The latter results in large propellant penalties during rendezvous and docking.

OMS Burn

The OMS engines are fixed at a cant angle of 6.2 degrees in pitch and 12.2 degrees in yaw to point the thrust vector at the centroid of the center of gravity travel during OMS burn. The ACPS propellant consumption required to offset the OMS moments was derived for a total ΔV burn of 723 ft/sec. The results of the study are summarized in Figure 2.3.2-8 for two and one OMS engines with and without misalignments. A total misalignment budget of 0.2 degree was used in this study resulting in 34 pounds of ACPS propellant required in pitch and 40 pounds in yaw with one OMS engine failed. The study was performed using the 110E orbiter mass and inertia data and two 3500 pounds thrust OMS engines with an ISP of 309 seconds.

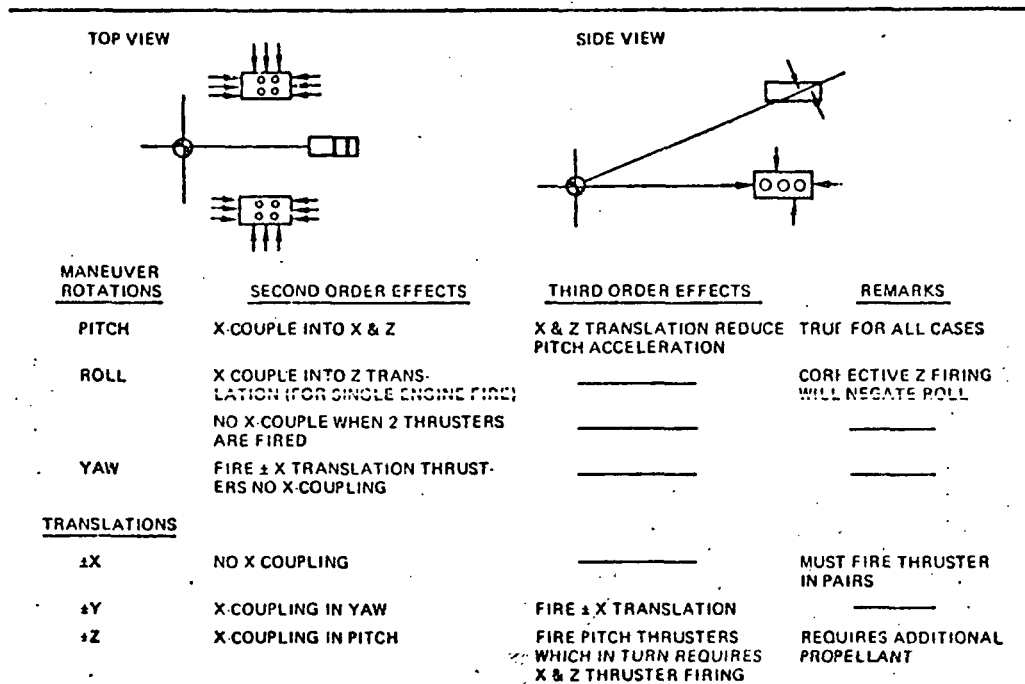
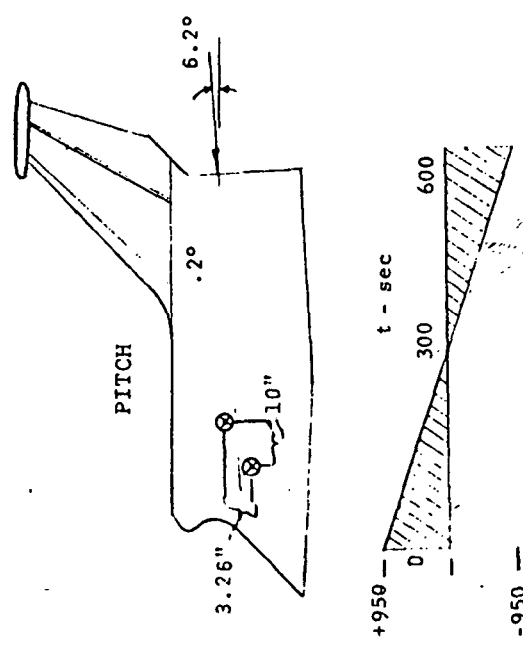
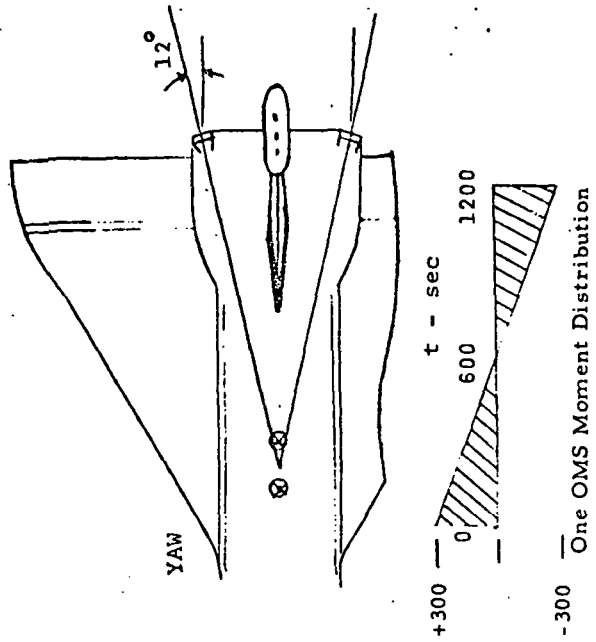


Figure 2.3.2-7. POD Installation Cross Coupling



OMS Moment Distribution

	ACPS Propellant	
	Pitch	Yaw Total
2 OMS Engines 0 Mis.	17	0
2 OMS Engines 0.2° Pitch Mis.	34	0
1 OMS Engine 0 Mis.	17	20
1 OMS Engine 0.3 Yaw Mis.	17	40
		57

Conditions	
Total ΔV	723 Ft/Sec
OMS Thrust	3500 LB
OMS ISP	309 SEC
ACPS Thrust	1050 LB
ACPS ISP	290 SEC

Figure 2.3.2-8. ACPS Propellant Consumption During OMS Burn



On-Orbit Navigation and Guidance

During this phase of the study, the technique for determining the state vector on-orbit was changed from the Phase B baseline (PRS) to MSFN for Mark I vehicles and a horizon scanner for Mark II vehicles. Although this change was promoted primarily from cost considerations, it represents a sound technical approach. MSFN accuracy for state vector update is well within the shuttle requirements and it is reasonable to assume that enough stations would be operational during the Mark I orbital flights even though on-board orbit determination were available.

For the Mark II vehicles, the horizon scanning technique of orbit determination is cost effective. Complete systems such as Barnes and Quantic are available off-the-shelf, space qualified, and of proven reliability. Horizon scanning techniques were evaluated and traded off during the Phase B study program and were found to be an acceptable solution. The PRS was chosen at that time because it satisfied several functions with one piece of equipment with a significant reduction in the number of antennas required.

The horizon scanner system has two technical drawbacks. The ultimate accuracy is not quite up to that of PRS or MSFN, and at least a half orbit is required to bring initial state vector errors down to an acceptable level. It is not felt that the 30-40 minute settling time will present an operational problem. The slightly larger residual error will cause some ΔV fuel penalty; however, preliminary estimates indicate that this penalty is negligible. Further analyses are planned which will refine the estimates and lend confidence in the horizon scanner technique.

Although horizon scanners have been used extensively for spacecraft attitude determination, there is not much real data on their orbit determination ability. There have been many detailed simulations and analyses performed, however, which prove without a doubt the validity of the concept. The combination of space qualified sensors and extensive study of the orbit determination problem indicate the soundness of the approach.

Orbiter Entry Control

The control of the orbiter during entry was evaluated using an all ACPS control mechanization with aero surface trim assist. The analyses were performed using the 110E aerodynamics and mass inertia properties, and the high cross range entry profile.

The baseline entry control mechanization consists of an angle of attack command with pitch rate feedback in pitch, roll attitude and roll rate in roll and lateral acceleration and yaw rate in yaw. The ACPS acceleration



requirements during entry are summarized in Figure 2.3.2-9. The most stringent requirement is that of yaw with 1.30 deg/sec^2 of yaw acceleration being required. This requirement was derived based on the entry profile of Figure 2.3.2-10 and a side slip budget consisting of 0.86 degrees of side slip occurring with 2.0 deg/sec roll reversals and 0.3 degrees associated with a typical lateral acceleration bias displacement of 0.2 degrees. A wind gust of 30 feet/sec at an altitude of 70K feet was also included in the budget, raising the side slip angle safety margin to 1.75 degrees. The yaw acceleration requirements sized the ACPS jet thrust level at 1050 pounds.

The entry ACPS propellant consumption requirement is 750 pounds. Trim control of the orbiter is achieved by use of the elevons. The elevon hinge moment requirement of $1.92 (10^6)$ inch-pound for entry (Figure 2.3.2-11) is no longer the determining factor in sizing the hydraulic subsystem. This is because of the low dynamic pressure (120.psf) associated with the 110E orbiter during entry. The elevon hinge moment (at maximum rate) required during approach to meet the 30 degrees of bank angle in 2.5 second requirement equals that required during entry.

An alternate control mechanization was evaluated, consisting of full elevon pitch control with ACPS control in roll and yaw. This concept does not increase the elevon hinge moment requirement and results in an ACPS propellant consumption saving of 300 pounds or a 40% reduction. This concept has not been incorporated in the baseline control because the increase in system complexity incurred with blended aero surface and ACPS control has not been fully evaluated.

The switch over from ACPS to aero surface control is scheduled to take place subsequent to the termination of the angle of attack transition from 34.5 degrees to 15 degrees. The switch over is to occur on all three axes simultaneously at the end of the angle of attack transition because the rudder does not become effective until the angle of attack has been reduced below 20 degrees. The aero surface control switch over will occur without severe transients since the aero surfaces have been utilized for trim control during the earlier phase of entry.

Manual Flight Control System

The manual flight control system consists of direct cables to the surfaces augmented by a 3-axis stability augmentation system. The pitch and yaw systems will be normal acceleration command systems with augmented damping. The roll system will command roll rate. In all axes, the resulting feel and response to the pilot will be that of an adequately damped airplane with relatively constant airplane response per unit of stick or pedal motion.

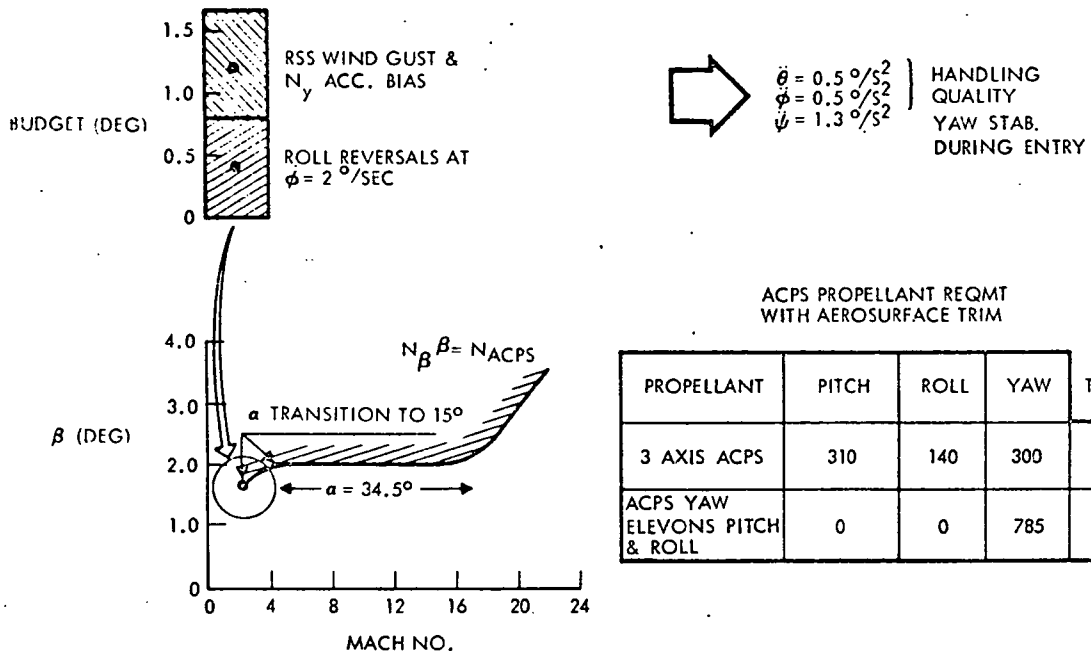


Figure 2.3.2-9. 110E Entry ACPS Requirement Summary

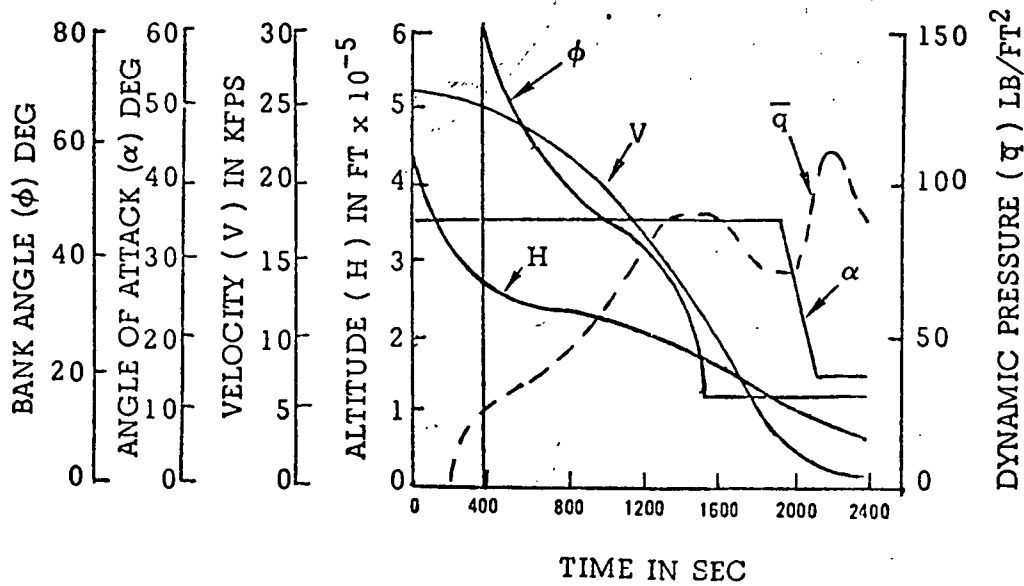


Figure 2.3.2-10. 110E (MK II) Entry Profile



AERO-SURFACE	DUTY CYCLE		MAX HINGE MOMENT AT ZERO RATE (IN.-LB)	HINGE MOMENT AT MAX RATE		MANEUVERS						MAX DEFLECTION (DEG)
						ENTRY			APPROACH & LANDING			
	POS (DEG)	RATE DEG/SEC		HM (IN.-LB)	RATE DEG/SEC	POS (DEG)	RATE DEG/SEC	HM (IN.-LB)	POS (DEG)	RATE DEG/SEC	HM (IN.-LB)	
ELEVON (PER PANEL)	0.5	3.0	1.92X10 ⁶	1.75 X10 ⁶	15	4	5	1.9X10 ⁶	10	15	1.75X10 ⁶	+15, 45
RUDDER	0.4	3.0	0.146X10 ⁶	0.14 X10 ⁶	10				10	10	0.14X10 ⁶	+10, 10
SPEED BRAKE (PER PANEL)			0.335X10 ⁶	0.32 X10 ⁶	10				35	10	0.32X10 ⁶	45, 15

- RUDDER REQUIREMENT GIVEN FOR ZERO SPEED BRAKE DEFLECTION
- SPEED BRAKE REQUIREMENT INCLUDES 10 DEG OF RUDDER DEFLECTION
- 110E AERO SURFACE HINGE MOMENT 47 PERCENT LOWER THAN 161C
- 110E ELEVON HINGE MOMENT REQUIREMENT SIMILAR DURING ENTRY & APPROACH & LANDING

Figure 2.3.2-11. 110E Aero Surface Hinge Moment Minimum Requirements

The elevon system, as illustrated in Figure 2.3.2-12, provides parallel control of the elevons through the mechanical direct link and the stability augmentation system. Movement of the Pilot's Control Stick accomplishes two actions: (1) the elevon is displaced through the mechanical linkage, and (2) an additional displacement is commanded through the SAS. The SAS is a normal acceleration (N_z) command system in pitch and the command is derived from the position pick-off at the pilot's stick. The SAS actuator will control elevon displacement to achieve and maintain commanded N_z .

Referring to Figure 2.3.2-12, a mechanical mixing link, near the stick, sums the single elevon inputs from fore and aft (pitch) and lateral (roll) stick motion. Motion is then carried through the cables, with tension regulators, to the aft section of the vehicle. From the aft sector, the linkage actuates the valve of a master cylinder and is connected to an artificial feel bungee. The bungee provides stick force proportional to stick displacement. The master cylinder provides mechanical isolation from the SAS actuator to prevent it from "kicking back" into the control stick. The master cylinder and the SAS servo displacement are summed in a walking beam and the resultant motion is transmitted to the elevon hydraulic power actuators. An electric trim actuator allows the pilot to trim the elevon.

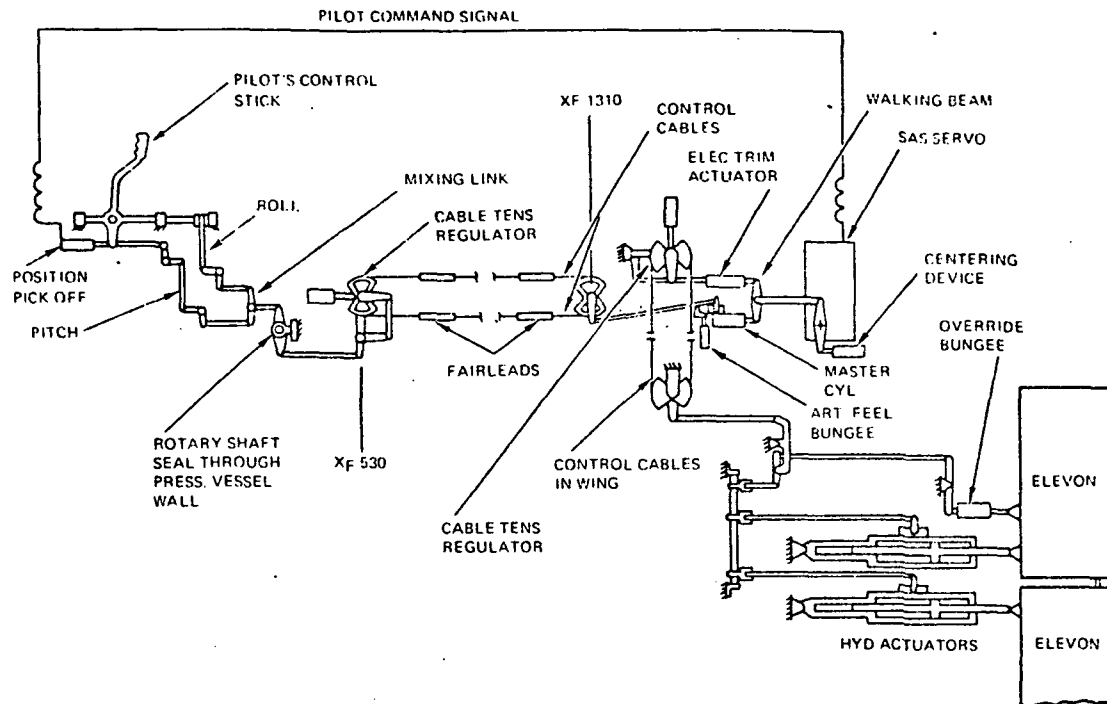
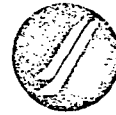


Figure 2.3.2-12. Elevon Aerosurface Flight Control System Schematic

Normal operation is with both mechanical system and SAS. In the event of a mechanical control system failure, the master cylinder will be used to anchor the bottom end of the walking beam and a fly-by-wire system through the SAS actuator will result. In the event of a failure of the SAS servo, the centering device will lock the center of the walking beam and control will be through the mechanical system via the master cylinder.

The basic system, described above, is similar to those used on the B-70, F-111, and F14 and no abnormal development problems are expected. Similarly, the fly-by-wire backup is not expected to present problems. The mechanical backup configuration appears to present controllability problems, however, because of narrow longitudinal aerodynamic stability margins. This results in over control problems because of lack of pitch aerodynamic stiffness.

As shown in Figure 2.3.2-13, two approaches are being pursued to solve this problem. The most effective change would be a modification of aerodynamic characteristics, for example a change in the wing planform or relocation of the c.g. by ballasting. These will be discussed in more detail in a later section.

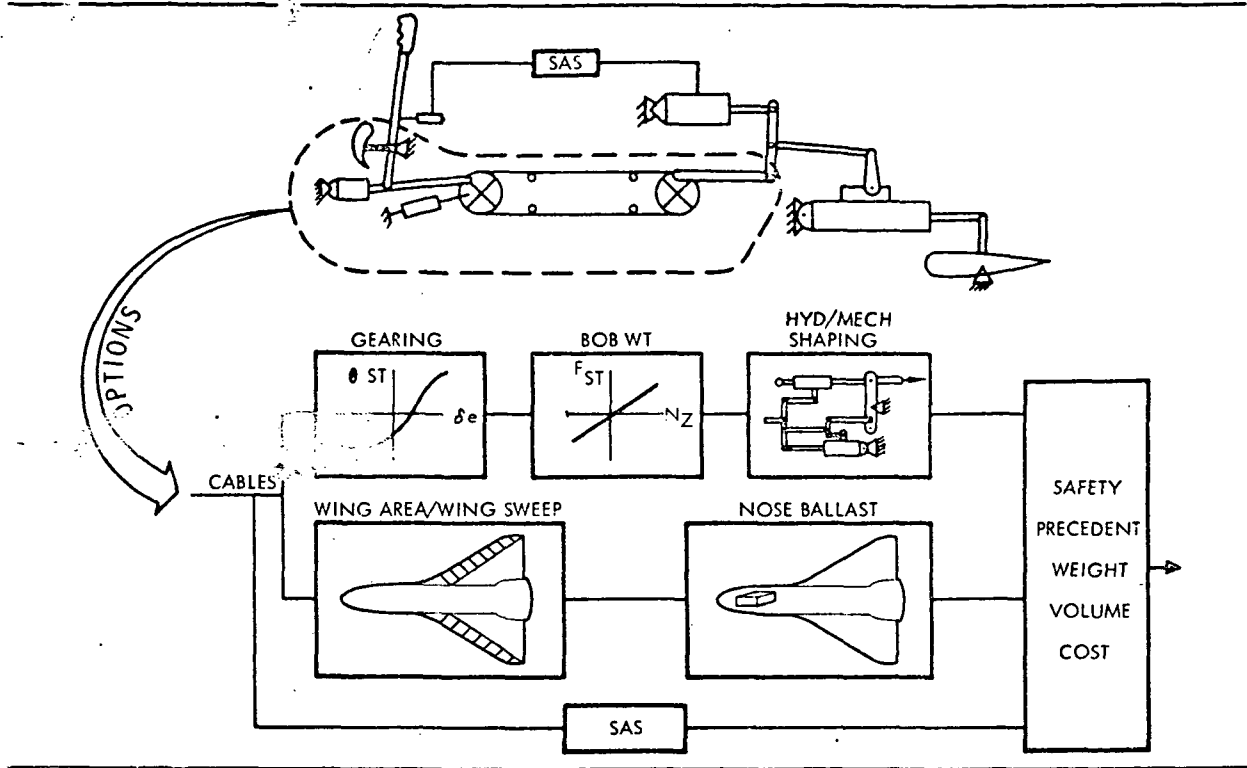


Figure 2.3.2-13. Mechanical Flight Control System Definition

A less technically effective but perhaps more cost effective approach may be modification of stick to elevon gearing; use of bobweights; use of hydraulic-mechanical "shaping networks" or a combination of the three. Some work has been completed in the area of gearing - see Figure 2.3.2-14. Correct choice of a gearing ratio can significantly improve controllability.

A second approach is the use of a bobweight on the control stick to produce force feedback to the pilot. This also increases the stick free frequency but may tend to decrease dynamic stability and, hence, adversely affect pilot inclined oscillations.

A third approach is the use of a hydraulic-mechanical shaping to produce a lead-lag between the stick and the elevon. While this could improve controllability through stick motions, it would not improve stability for gust inputs.

During the next phase, the above approaches will be studied singly and in combination. The most sophisticated approach would be a combination of the three providing an isolated stability augmentation system similar to the electronic one. The most cost effective approach will be established.

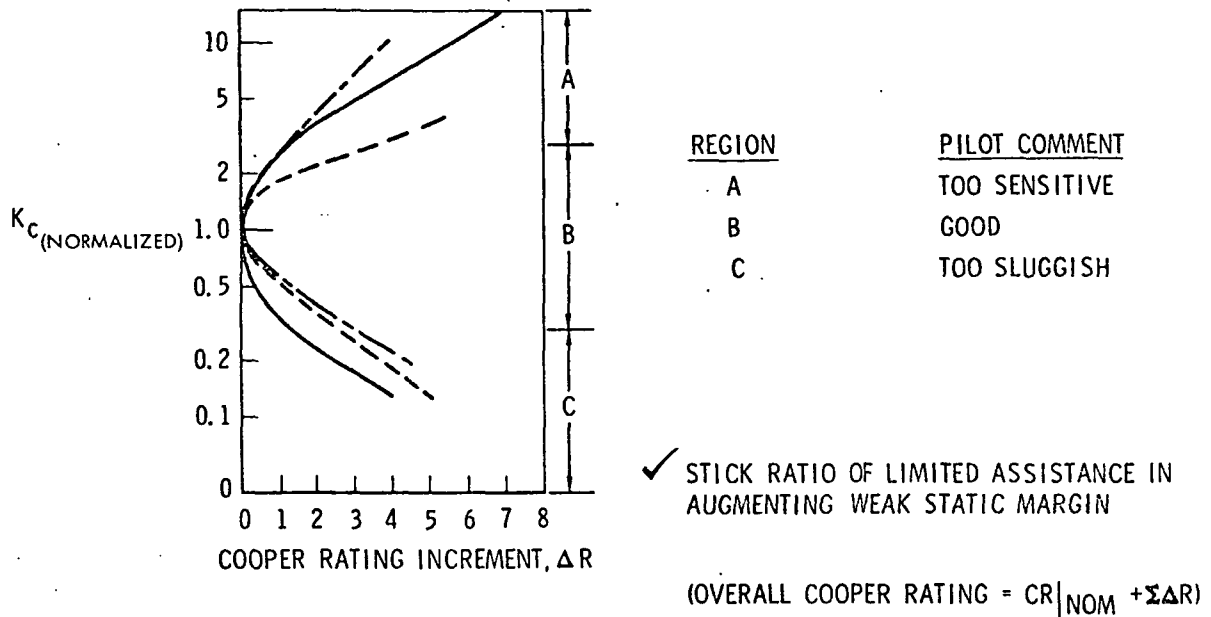
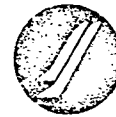


Figure 2.3.2-14. Influence of Stick Gear Ratio/Pilot Gain On Handling Qualities

Subsonic Longitudinal Stability

The decision to provide for direct pilot control via mechanical links places emphasis on the handling qualities of the unaugmented airframe. Figure 2.3.2-15 presents the percent static margin as a function of c. g. for the baseline 040-A orbiter vehicle. Also shown is the projected increase in static margin when the trailing edge of the wing is moved aft such that it is aligned with the MPS nozzles. Note that the baseline vehicle has negative static margin for the most aft c. g. and only 2.0 percent for the forward c. g. Satisfactory static margin for only the forward c. g. is achieved by moving the wing aft. The relation of Cooper-Harper ratings to static margin as shown on Figures 2.3.2-15 and 2.3.2-16 are approximations made using MIL 8785-B as a guideline.

A preliminary analysis of variations in wing sweep indicates that a large gain in static margin can be acquired by reducing the sweep to 50 deg from the baseline 60-degree sweep angle. Figure 2.3.2-16 shows the effect of this approach on the basic airframe handling qualities. For the 50 deg \curvearrowright , the satisfactory results are achieved without changing the location of the trailing edge. It is estimated that the reduced sweep will increase the leading

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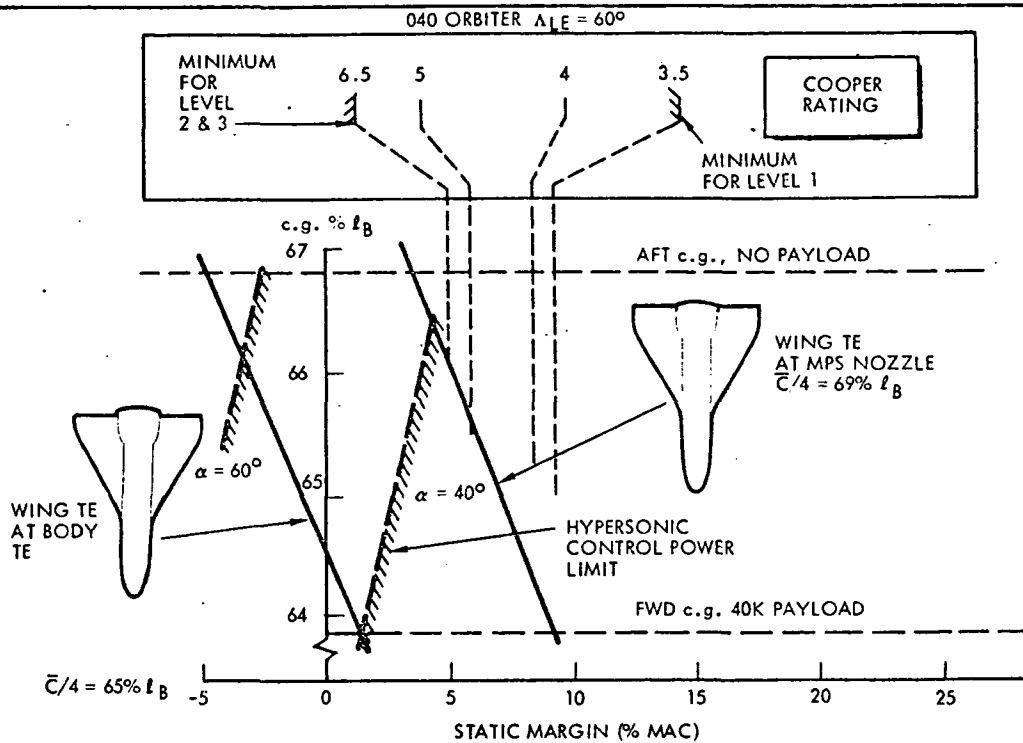


Figure 2.3.2-15. Variation of Low Speed Static Margin W/c.g. & Wing Location

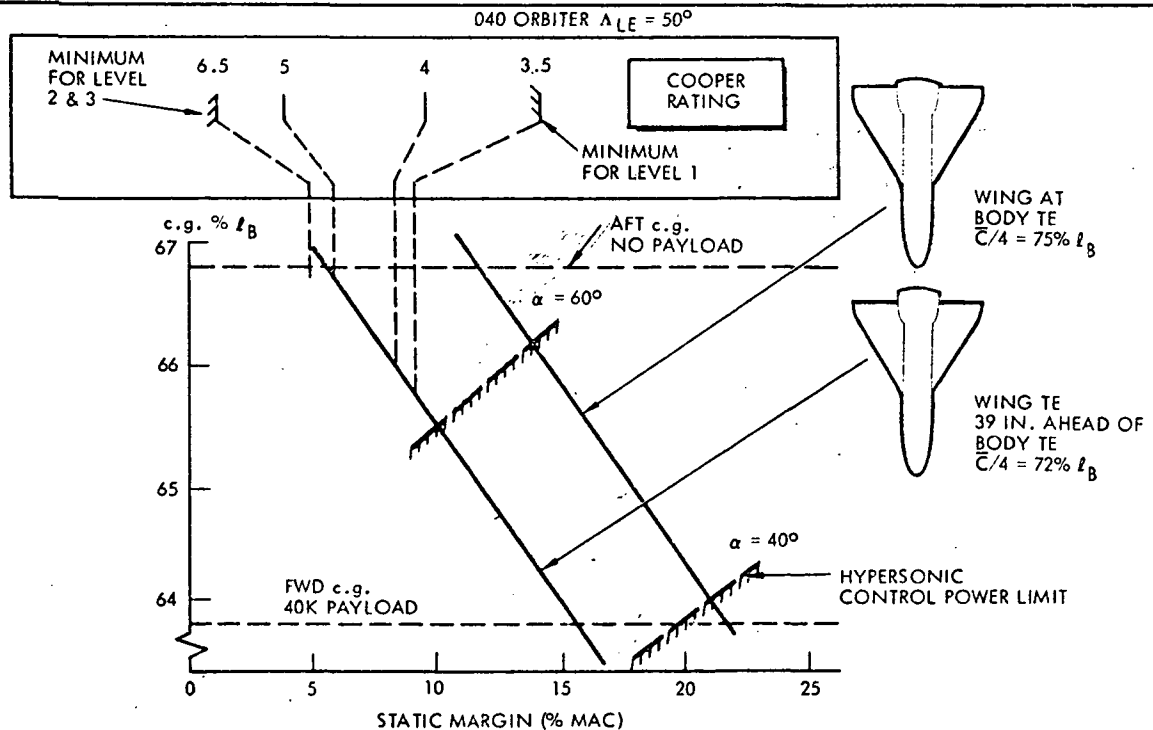


Figure 2.3.2-16. Variation of Low Speed Static Margin W/c.g. & Wing Location



edge temperature by approximately 150 deg - 200 degrees. Figure 2.3.2-17 shows a comparison of the present 040-A baseline with the modified 50-degree wing sweep. Preliminary estimates indicate a net weight savings of approximately 600 lbs for the modified configuration.

Although all the ramifications of reduced sweep angle may not be identified at this time, it appears to be a promising design approach in achieving acceptable vehicle handling qualities when flying a direct mechanical system.

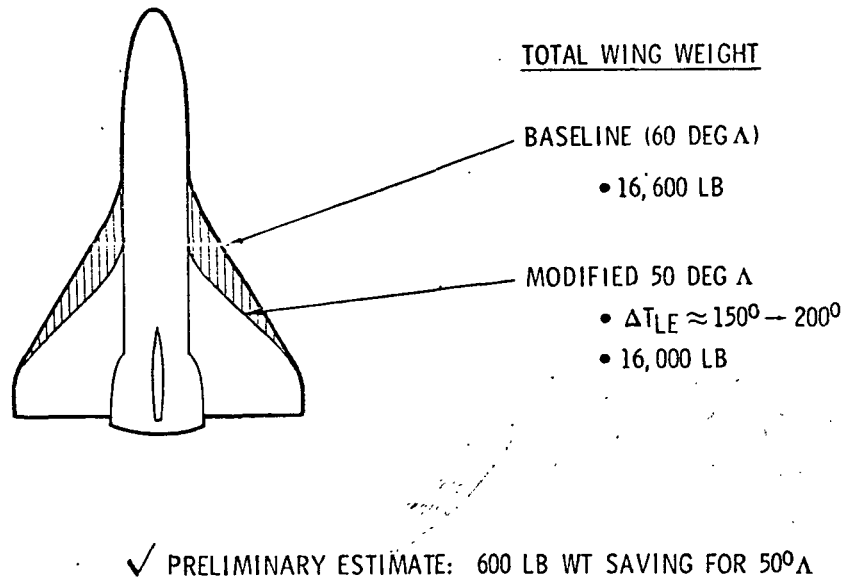


Figure 2.3.2-17. Effect of Wing Sweep On Vehicle Weight



2.3.3 GN&C Configuration Description

Summary of Hardware Changes

The primary driver on hardware changes from the Phase B' GN&C system mechanization was cost reduction, influencing hardware definition from two major standpoints: Adaptability to a phased orbiter development program and utilization of existing equipment designs wherever feasible. The resultant system is illustrated in the GN&C Equipment schematic, Figure 2.3.3-1.

Table 2.3.3-1 summarizes the effects of changes to the GN&C system on overall characteristics. The total number of GN&C LRU's, their weight, and electrical power requirements are all seen to have increased. Major factors influencing these parameters include:

1. Absorption of GN&C computation from the DCM (adding 9 LRU's, 315 lb, 105 wt).
2. Replacement of multiformat CRT displays with conventional instrument array for both aerodynamic and space flight.
3. Addition of analog inner loop aero flight control and manual TVC electronics.
4. Replacement of multilateration (PRS) navigation equipment with horizon sensors and TACAN. Addition of updata link in Kit I.

Other system changes influencing the GN&C hardware definition:

1. Utilization of a conventional center stick rather than side stick for manual aero flight control.
2. Multiplexed and hardwired interconnections replacing the digital data bus for a more conventional system approach.
3. The adoption of the three-single-string system redundancy mechanization with minimized cross-strapping and physical separability for simplicity and safety.

General Functional Description

The guidance, navigation, and control system consists of three identical strings. The baseline does not have cross-strapping of GN&C strings. The redundancy management of the strings is external to the

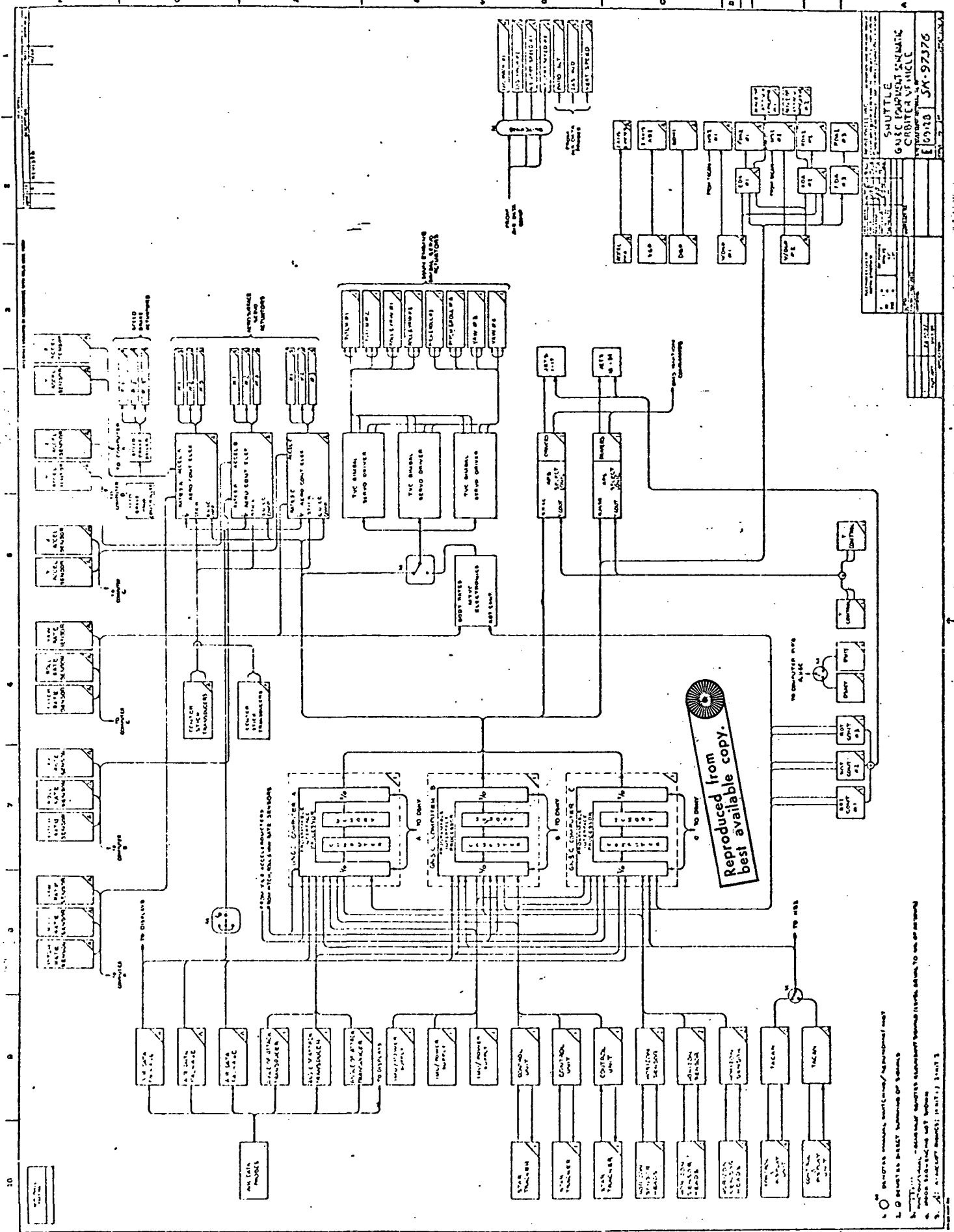


Figure 2.3.3-1. GN&C Equipment Schematic

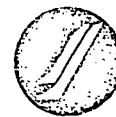


Table 2.3.3-1. Hardware Characteristics Changes:
Present System Vs. Phase B System

	ØB	ØB'	Increase
Number of LRU's - Orbiter	59	84	25 LRU's (42%)
Booster	50	60	10 LRU's (20%)
System Weight - Orbiter	609.2 LB	1015 LB	406 LB (67%)
Booster	528.6	831.5	
Peak Power Drain -Orbiter	1170 W	2537 W	1367 W. (117%)
Booster	1241.7	2325	1083 W (88%)

computer. The computer and software provide the entire GN&C computations from prelaunch through entry or the point where aerosurfaces are in control and the ACPS discontinued. The aerodynamic control is performed by an analog SAS electronics with guidance and navigation commands from the computer to appropriate analog systems until the pilot initiates manual control to land the orbiter. The computer and software provides the display processing and interfacing with the DSKY/CRT for all digital data input and output.

The GN&C system consists of the computer, sensors, interface drivers and driver electronics, and software required to perform the GN&C functions for the orbiter. The IMU provides the attitude reference and navigation data to the computer. As shown in Figure 2.3.3-2, the star tracker will be used for on-orbit state vector update in conjunction with the ground during Mark I and a horizon sensor during Mark II. This data will update the attitude due to error caused by IMU drift. The star tracker will also provide the target line of sight during rendezvous. The target's range will be provided by an onboard TACAN in conjunction with a TACAN located on the target.

Accelerometers and rate gyros provide the input to the computer for load relief and body bending during boost. SAS electronics receive guidance and navigation commands from the computer to fly the aerodynamic phase (aero surface control).

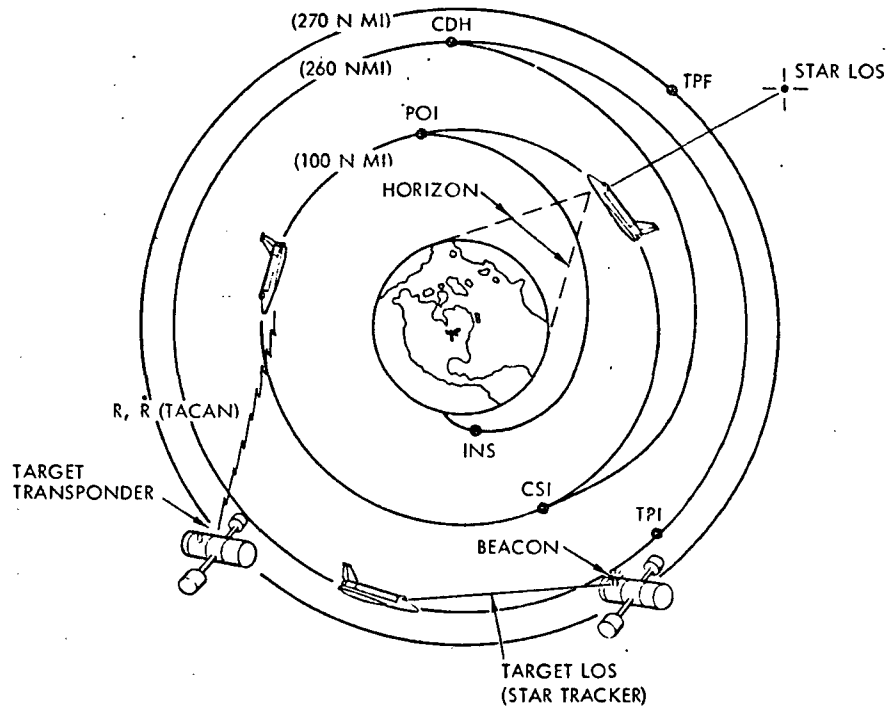


Figure 2.3.3-2. Operational Profile - Orbiter (Ascent/Rendezvous)

The air data computer interfaces with the SAS. The computer provides control signals to the jet select logic. The rotation controller provides the manual control of the orbiter. The DSKY/CRT provides the communication link between pilot and computer.

The ADI is supplied with attitude error and rates from the computer. Also, the computer supplies the angle of attack information to the cross pointer.

GN&C Configuration

The objectives of the Phase B' shuttle hardware studies were (1) to further develop the Phase B GN&C system to meet system performance objectives of low cost, simplicity, and low risk, and (2) to determine a technically feasible GN&C program plan which minimized peak annual funding. The system hardware defined here achieves the stated objectives. Development costs have been minimized by the application of existing technology and/or the use of existing hardware elements wherever feasible.

Of the the total of 28 different LRU types occurring in the orbiter system, 17 existing designs are directly applicable or require only minor



modifications for use; 6 involve modification and repackaging of existing subassemblies. Only 5 (2 TVC LRU's, Star Tracker Sensor/Shutter and its control unit, and speed brake driver unit) are viewed as new designs but make use of existing conventional technology in their development.

A detailed breakdown by LRU, of the degree of development required for orbiter hardware elements, is presented in Table 2.3.3-2.

Supporting material identifying the program, model identification, and original vendor of each hardware element on which the shuttle orbiter LRU's are based, are tabulated in Table 2.3.3-3.

The size and weight of the individual LRU's are included in Table 2.3.3-4. This table does not include the various displays, controls, and navigational aids which interface directly with the primary GN&C equipment. Table 2.3.3-5 contains a tabulation of the individual LRU power requirements by mission phase. Individual LRU detailed descriptions follow.

Air Data Equipment. The Air Data Equipment for shuttle has been selected for maximum use of existing equipment and permits a phased development program.

For the Mark I Orbiter, the horizontal flight control system requires only the computation of dynamic pressure for gain scheduling of the stability augmentation system; displayed parameters include altitude, altitude rate, mach number, and airspeed. These parameters are computed by the Air Data Package from input probe pressure signals corresponding to static and total pressures.

The interim selection for Air Data Package is a modified version of the Honeywell Digital Air Data System designed for the DC-10. Utilizing a computer (7 CPU cards, 1 memory card), this device converts pressure signals to digital format, computes flight parameters, and produces both analog and parallel digital output to displays and to the aerodynamic control electronics.

Updating the Air Data System for the Mark II vehicle requires the addition of an Angle of Attack Transducer package, which produces digital outputs permitting computation of angle of attack by the GN&C computer. This device consists of two differential pressure-to-electrical signal transducers and utilizes module building blocks in converting to the required digital outputs.

Three each of the Air Data and Angle Transducer units are required in each system, for single-string redundancy in a spatially diverse vehicle layout.

Table 2.3.3-2. Degree of Development Required for Orbiter LRU's

LRU	Existing Equipment Minor Mods	Modified Off-Shelf Equipment	Repackaged- Existing Building Blocks	New Develop- ment, Existing Technology & Circuits
Air Data Package	X			
Attack Angle Transducer			X	
IMU	X			
IMU Power Supply	X			
Star Tracker		X		
Sensor/Shutter Assembly				X
S/T Control Unit				X
Horizon Sensor Electronics	X			
Horizon Sensor Head	X			
Body Rate Sensor			X	
Accelerometer			X	
GN&C Computer	X			
Program I/O Processor			X	
Memory Unit		X		
TVC Driver				X
MTVC Electronics				X
Aero Control Electronics			X	
Speed Brake Driver				X
APS Logic/Driver		X		
Rotation Controller		X		
Translation Controller		X		
Center Stick Grip		X		
FDAI	X			
EDA		X		
Attack Angle Indicator		X		
FCS Control Panel			X	
TACAN	X			
TACAN Control Unit	X			
Totals: 28 LRU types	<u>9</u>	<u>8</u>	<u>6</u>	<u>5</u>

Table 2.3.3-3. LRU Background/Development Sources

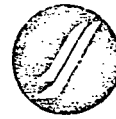
LRU	Program/Vehicle	Hardware Ident.	Source
Air Data Package	DC-10	HG280	HI
Attack Angle Transducer	DC-10	HG280	HI
IMU	A7D/E	KT70	Singer
IMU Power Supply	A7D/E	KT70	Singer
Star Tracker	Aero Bee	-	ITT
Sensor/Shutter Assy.	-	-	-
S/T Control Unit	-	-	-
Horizon Sensor Electronics	Classified	15-163	Barnes
Horizon Sensor Head			
Body Rate Sensor	F-14 AFCS	GG1027	HI
Accelerometer	F-14 AFCS	GG1026	HI
GN&C Computer	F-15	AP-1	IBM
Program I/O Processor	-	-	-
Memory Unit	F-15	AP-1	IBM
TVC Driver	-	-	-
MTVC Electronics	Apollo SCS	BG286	HI
Aero Control Electronics	F-14	AFCS	HI
Speed Brake Driver	-	Various	-
APS Logic/Driver	Apollo SCS	BG287	HI
Rotation Controller	Apollo LM	CG163	HI
Translation Controller	Apollo CSM	CG160	HI
Center Stick Grip	F-14	AFCS	HI
FDAI	Apollo SCS	JG264	HI
EDA	Apollo SCS	BG285	HI
Attack Angle Indicator	Apollo LM	X-Pointer	HI
FCS Control Panel	F-14		HI
TACAN	F-4K	AN/ARN-91	Hoffman
TACAN Control Unit	F-4K	AN/ARN-91	Hoffman

Table 2.3.3-4. System Weight and Volume of Orbiter GN&C LRU's

<u>LRU</u>	<u>Qty</u>	<u>Weight Lb/LRU</u>	<u>Volume Ft³/LRU</u>	<u>Total Wt.-Lbs.</u>	<u>Total Vol-Ft³</u>
Air Data Package	3	25	0.52	75	1.56
Attack Angle Transducers	3	10	0.15	30	0.45
IMU & Power Supply	3+3	47	0.80	141	2.40
Star Tracker and Control Unit	3+3	10	0.18	30	0.54
Horizon Sensor (1) & H/S Heads (2)	3+6	22	0.20	66	0.60
Rate Sensor	9	1.5	0.02	13.5	0.18
Accelerometer	6	0.8	0.01	4.8	0.06
GN&C Computer	3	40	1.03	120	3.09
Prog I/O Processor	3	35	1.03	105	3.09
Memory Unit (0)	3	30	0.70	90	2.10
TVC Driver Unit	3	8	0.12	24	0.36
MTVC Electronics	1	6	0.10	6	0.10
Aero Control Elect.	3	13	0.29	39	0.87
Space Brake Driver	1	6.4	0.10	6.4	0.10
APS Logic/Driver Unit	2	19.6	0.48	30.2	0.96
Totals, GN&C LRUs	55			789.9 Lbs.	16.46 Ft³

Table 2.3.3-5. GN&C Power Requirements by Mission Phase - Orbiter

GN&C Equipment	Qty/ System	Unit Power (DC/AC)	System Power (DC/AC)	Duty Factor In			Average Power in Watts						
				Ascent	Orbit	Entry	Ascent	Orbit	Entry	Aero			
Air Data Package	3	0/89	0/267	0	0	1	1	1	-	-	0/267	0/267	0/267
Attack Angle Transducer	3	0/10	0/30	0	0	1	1	1	-	-	0/30	0/30	0/30
IMU & Power Supply	3	50/70	150/210	1	1	1	1	1	150/210	150/210	150/210	150/210	150/210
Star Tracker Equipment	3	15/6	45/18	0	0.1	0	0	0	-	5/2	-	-	-
Horizon Sensor Equipment	3	22/0	66/0	0	0.1	0	0	0	-	7/0	-	-	-
Rate Sensor	9	0/2	0/18	1	0	1	1	1	0/18	-	0/18	0/18	0/18
Accelerometer	6	0/0.5	0/3	0	0	1	1	1	-	-	0/3	0/3	0/3
GN&C Computer Equipment	3	150/0	315/0	1	1	1	1	1	315/0	315/0	315/0	315/0	315/0
MTVC Electronics	1	12/8	12/8	1	0	0	0	0	12/8	-	-	-	-
Aero Control Elect.	3	42/29	126/87	0	0	1	1	1	-	-	126/87	126/87	126/87
APS Logic/Driver Unit	2	2/2	4/4	1	1	1	0	0	4/4	4/4	4/4	4/4	-
Speed Brake Driver Unit	1	5/10	5/10	0	0	1	1	1	-	-	5/10	5/10	5/10
Totals									511/271	481/216	600/629	596/625	



Rate and Accelerometer Sensors. Sensors for both the orbiter and booster vehicle GN&C systems include both body rate gyros and accelerometers for control loop stabilization.

The hardware elements for both rate and acceleration sensors are based on existing equipment in current production by Honeywell for the Grumman F-14 flight control system. Available configurations include both dual and triple redundant single-axis packages utilizing heaterless GNAT gyros, and triple accelerometer packages. In applying the single-string system criterion of spatial diversity to the redundant gyros and accelerometers, subassemblies and circuits of the F-14 units will be handled as building blocks in repackaged sensors of four types: roll, yaw, and pitch body rate packages, and a 2-axis accelerometer package; each LRU includes the inertial sensor, plus loop electronics and power supply. Separating rate sensors by axis is assumed to be required to permit locating the sensors at different airframe body stations because of vehicle bending mode variation by axis.

IMU. Each IMU consists of two LRU's; the platform and the power supply. Both are shown in Figure 2.3.3-3.

The platform has four gimbals with appropriate synchros, resolvers, and torque motors for each gimbal. The angular sequence starting with the inner gimbal is pitch, roll, and yaw with the fourth gimbal providing redundant roll. The stable element (inner member) contains two 2DOF gyros with SRA's directed along the pitch and yaw gimbal axes, respectively. One gyro controls the roll and yaw platform gimbals while the other controls the pitch gimbal axis. The platform baseline is typified by a Kearfott KT-70.

The accelerometers, also mounted on the stable member, consist of a two axis accelerometer, measuring accelerations in the X and Z axis and a single axis accelerometer measuring Y axis accelerations. This definition applies when the gimbal angles are driven to zero in all axis with respect to the vehicle body axis system. The coordinate system X, Y, Z, defined in the conventional sense, corresponds to the roll, pitch, and yaw axes of the vehicle. The gimbal torquing electronics and the accelerometer rebalance electronics are located in the platform assembly.

Star Tracker System. The star tracker is a strapped down optical sensor using electronic gimbaling to determine star positions within the eight degree diameter field-of-view (FOV). Usage of the tracker is extended to include acquiring and tracking a space station light beacon. The acquisition mode, results in a scan of the entire FOV after which the brightness object is selected. The tracker then enters a tracking mode in which the selected object is scanned over a very small FOV on the order of 16 arc-minutes. The position of the object is measured in two axes with respect to the bore-sight of the tracker.

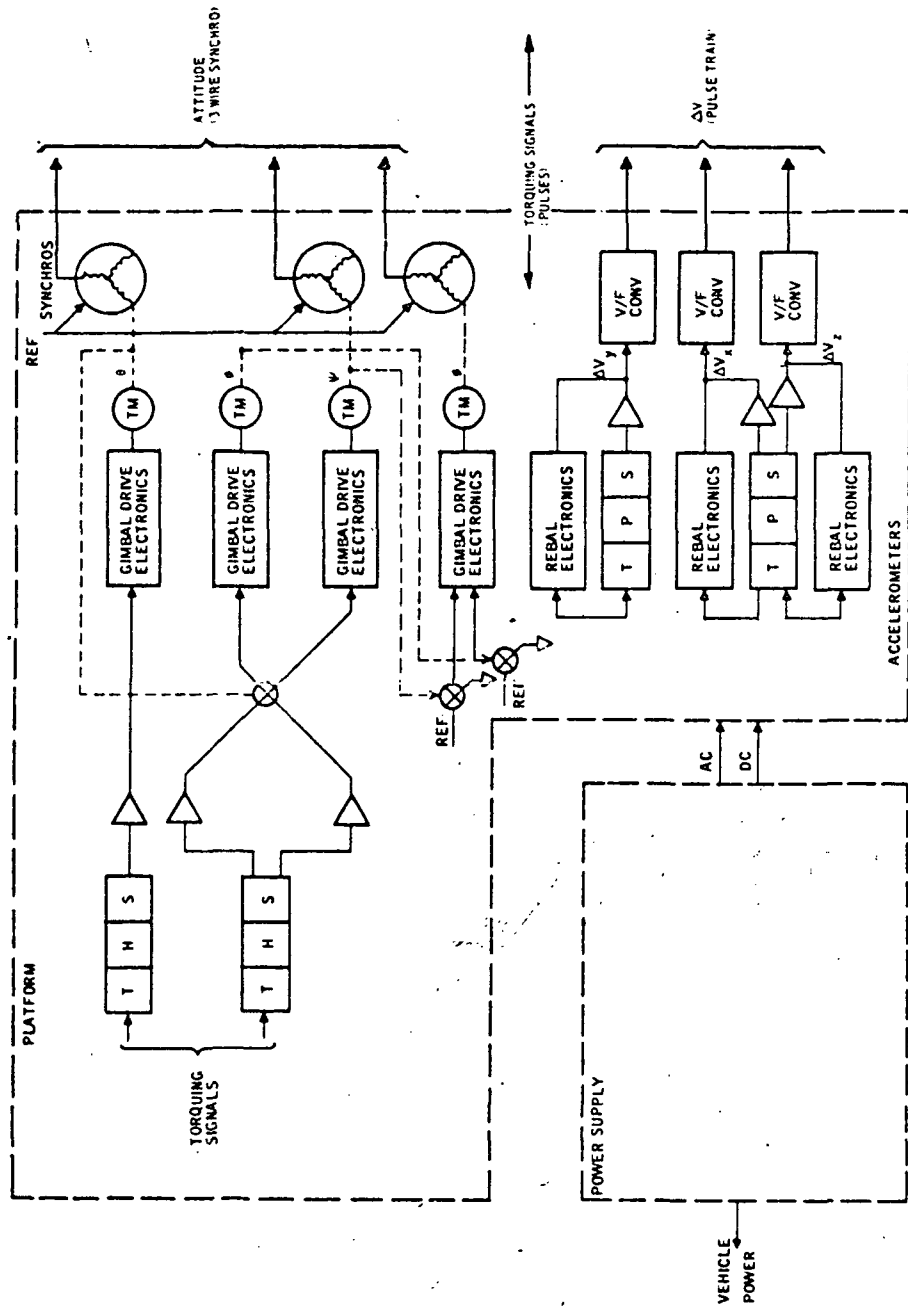


Figure 2.3.3-3. IMU Block Diagram



The star tracker (baseline is typified by the ITT Dual Mode Star Tracker) consists of an optical lens system, photosensor and electronic circuitry as shown in Figure 2.3.3-4. The lens gathers and brings to focus the radiant energy from the source at the photo cathode of the multiplier phototube. The photo cathode surface forms an electron image of the focused light source. An accelerating voltage applied between the photo cathode and a limiting aperture causes electrons from a particular area of the photo cathode (instantaneous photo cathode area) to pass through the aperture. This then defines an instantaneous FOV of a small region in space. A multiplier section behind the aperture amplifies the signal.

Deflection coils placed around the image section of the phototube provide a means of deflecting the electron image. A search sweep generator and a track sweep generator provide deflection signals to the coil to cause the electron image to sweep across the aperture during the acquisition and track modes, respectively. The search sweep generator is used to scan the entire tracker FOV whereas the track sweep generator scans a small pre-selected area.

The video amplifier and star selection circuits amplify the star presence signal, and set a reference level corresponding to the largest signal encountered in the FOV. On receipt of a tracking signal from mode control, the tracker scans the FOV and stops when the brightest star enters the instantaneous FOV. Once this happens, the tracker enters the tracking mode. The tracking loop circuits develop an analog error signal which is used to control the deflection-coil signals so that the fine scan is centered on the star image. The output error amplifiers sample the deflection coil signal and provide output error signals to the computer.

The power supply requires both plus and minus 27 vdc input power.

The star tracker system is comprised of 3 LRU's:

Star Tracker/Shutter
Sun Shield/Sun Sensor/Shutter Assembly
Control Unit

Horizon Sensor. The horizon sensor, used operationally during navigation is of the conical scan variety. The sensor approach uses two horizon sensing heads interfacing with a signal processing box. A block diagram of the sensing system is shown in Figure 2.3.3-5.

Each head contains an optics section with a motor driven rotating mirror. A bolometer/amplifier provides a signal indicating the level of radiance received through the rotating FOV. A marked change in radiance levels indicates a horizon crossing. Under normal operation, two horizon

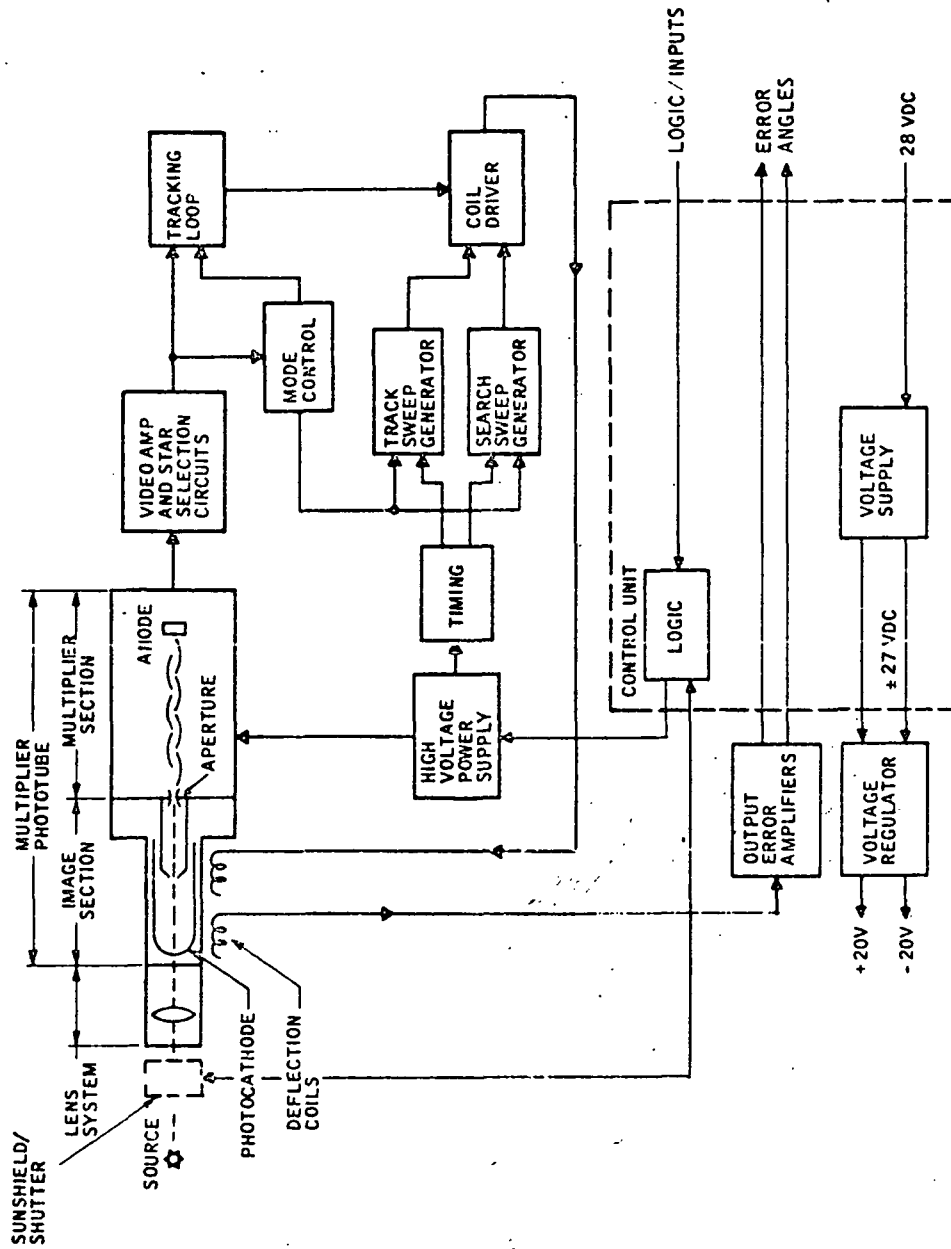


Figure 2.3.3-4. Star Tracker Block Diagram

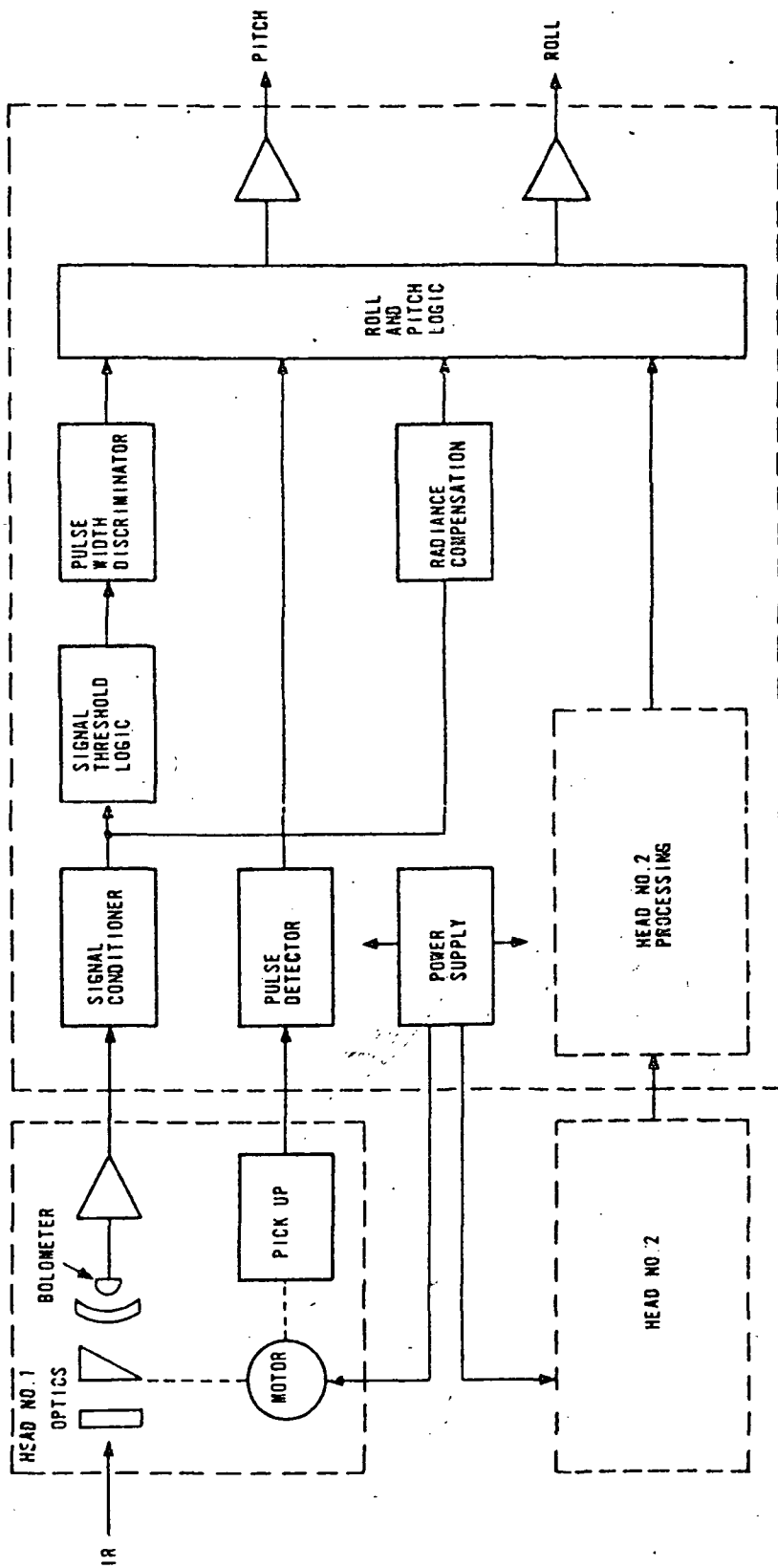


Figure 2.3.3-5. Horizon Sensor Block Diagram



sensor crossings per scan are received by each head. A motor packoff provides a reference timing signal which locates the center of each scan pattern.

Within the processor box, pitch information from each head is derived from the horizon crossing signals and the reference pulse. The pitch information from each head is averaged to obtain a pitch angle output signal. A roll output signal is obtained by comparing the period of "Earth crossing" pulse from each head.

Radiance compensation circuitry is provided to suppress the effects of horizon anomalies on the output performance of the horizon sensor.

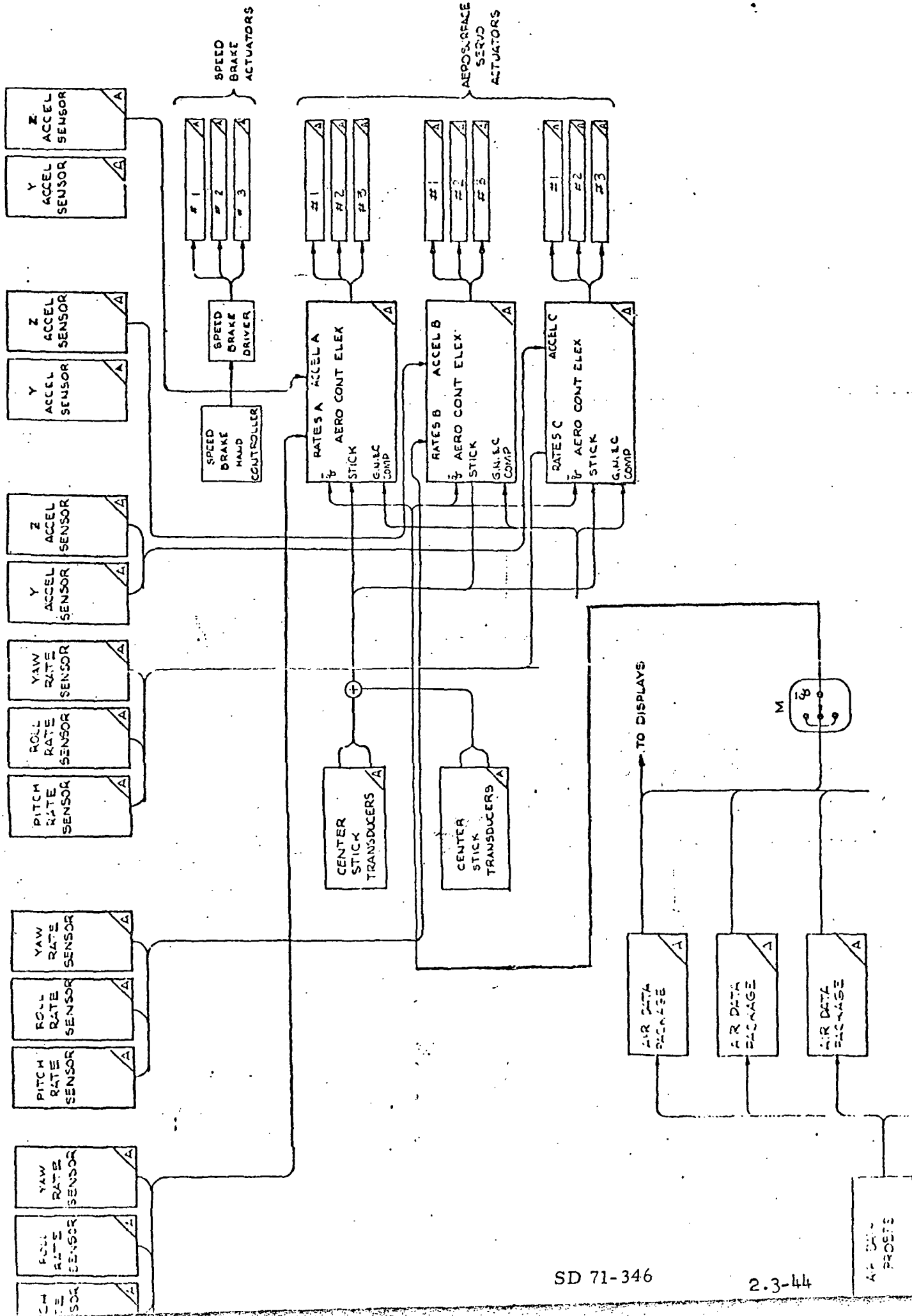
The horizon sensor, typified by the Barnes 13-156 Conical Scan Horizon Sensor System, provides a highly accurate, long life, local vertical reference over a wide range of altitudes (80 - 6000 nautical miles). The optics are designed to view the CO₂ special band (15 microns) which is the optimum spectral band for minimizing horizon variations. The output error angle signals are linear over a +5 degree region saturating at 10 degrees. A sun presence signal, detected in the short wavelength spectral band, inhibits the signal output thus preventing corruption of the navigation processing with erroneous updates.

Analog SAS. The selection of an analog SAS shown in Figure 2.3.3-6 as opposed to a digital SAS was based on the fact that there are several flight proven and operational fail operative and fail safe analog flight control systems in use today. All aircraft currently flying employ this method of control. There is no production digital stability augmentation other than Apollo. Therefore, analog is tentatively selected as the conventional off-the-shelf approach in order to minimize risk and be available for horizontal flight.

Analog aerodynamic flight control electronics are packaged in three identical 3-axis LRU's in adherence to the spatial diversity concept of redundant hardware independence. In each, the pitch, roll and yaw control channels receive control commands from the GN&C computer and provide crossed commands to the elevon control surface actuators, and in yaw channel from transducers in pedals. Capability also exists in the roll and pitch channels to provide surface actuation signals in response to analog manual input commands from transducers on the center control stick. Vehicle body rate and lateral and normal acceleration feedback signals are introduced for axis stabilization and a dynamic pressure signal from the air data system is used in each axis for gain scheduling.

In addition to the control electronics and actuator servo amplifiers, each LRU includes an independent set of middle-select comparators and the required power supplies. In packaging of the aero control electronics,

Figure 2.3.3-6. Analog SAS





extensive use will be made of existing aircraft flight control hardware and packaging techniques. Plug-in circuit boards will be modified from C-5 and F-14 hardware; wire-wrap interconnecting base plate and fabricated aluminum enclosure are adapted from the existing F-15 aircraft flight control electronics hardware.

TVC Electronics. Hardware for thrust vector control is packaged in four LRU's: 3 identical TVC gimbal servo driver units and one manual TVC electronics box.

Each of the TVC gimbal servo driver units contains 8 servo amplifier circuits, middle select level detectors for redundancy management, and bias supplies, and represents a single layer of redundant drivers for the 8 engine gimbal actuators. Gimbal command inputs are accepted from either the GN&C computer in automatic control mode, or upon manual switchover to manual TVC, from the MTVC electronics in a backup mode.

The MTVC electronics unit accepts 3-axis commands from the two rotation hand controllers and body rate signals. Stick and rate inputs are amplified, and summed in an integrating amplifier; the resultant command signal outputs are fed to the appropriate gimbal servo drivers in each of the three TVC driver LRU's. All of the TVC electronics hardware is new design, with circuits based on Apollo BG286 MTVC and BG288 servo amp background experience. Packaging will utilize plug-in circuit boards, wire wrap interconnecting matrix, and non-hermetic aircraft-grade enclosure design of fabricated aluminum sheet, with thermal design based on convection cooling.

APS Logic/Driver Unit. Control of the ACPS thruster valve solenoids and of OMS engine ignition is handled by two APS Logic/Driver Units. Based on the existing Apollo Reaction Jet/Engine on-off control, this device accepts ACPS thrust commands from either the GN&C computer or from the Translation Hand Controller; it contains the logic necessary to select the appropriate thruster in response to the commanded rotation or translation maneuver, and provides the switching circuits necessary for controlling current to the thruster valve solenoids. This device also includes the necessary transient suppression and bias supply circuits, and circuits for OMS engine ignition and cutoff control.

The unit will utilize the BG287 chassis as well as the basic existing welded matrix and cordwood module configuration. Growth space is adequate to accommodate revisions necessary to tailor the existing jet select logic to shuttle ACPS system requirements, add one 2-jet driver module to the existing 16-jet control capability, and modify the SPS engine ignition timing and logic to meet shuttle OMS needs.



GN&C Computers. The GN&C computational facilities consist of three, single thread, dedicated data acquisition strings and computers. At the present time, no provisions are being made to permit the transfer of data between computers. Figure 2.3.3-7 presents a simplified block diagram of the selected concept.

The selected concept consists of a standard remote multiplexer and input buffer, a 48K by 32-bit general purpose computer, and an output decoder.. Data collected via the multiplexer/buffer will be, with very limited exceptions, already conditioned to a specified DC level. The decoder will be custom designed to meet subsystem interface requirements.

The hybrid/display (DSKY) system consists of two DSKY units. The system is implemented such that the keyboard data is made accessible to all three computers simultaneously. The display source and format is selectable at each GN&C DSKY.

GN&C Software Summary. The Orbiter GN&C software has changed only slightly from the Phase B baseline. Perhaps the best measure of the degree of change is exemplified by the memory requirement shown in the following table:

FUNCTION	MEMORY REQUIRED - 32 BIT WORDS	
	PHASE B	PHASE B ¹
Guidance	9,500	11,100
Navigation	10,100	7,800
Control	3,200	2,900
TOTALS	22,800	21,800

An overall memory requirement decrease of 1000 words has resulted, primarily because of substitution of an analog control system in the aerodynamic or airplane modes. The decrease would have been somewhat greater except that further definition of the guidance functions results in a requirements increase.

Significant GN&C software changes were not anticipated because the GN&C system requirements are essentially the same. However, there were several changes which will impact detailed software design. These changes are summarized in Table 2.3.3-6. Guidance and control of the booster is shown as a possible requirement change, since the booster configuration has not been finalized. Table 2.3.3-7 indicates potential orbiter software impact resulting from use of the pressure fed booster. The numbers presented are

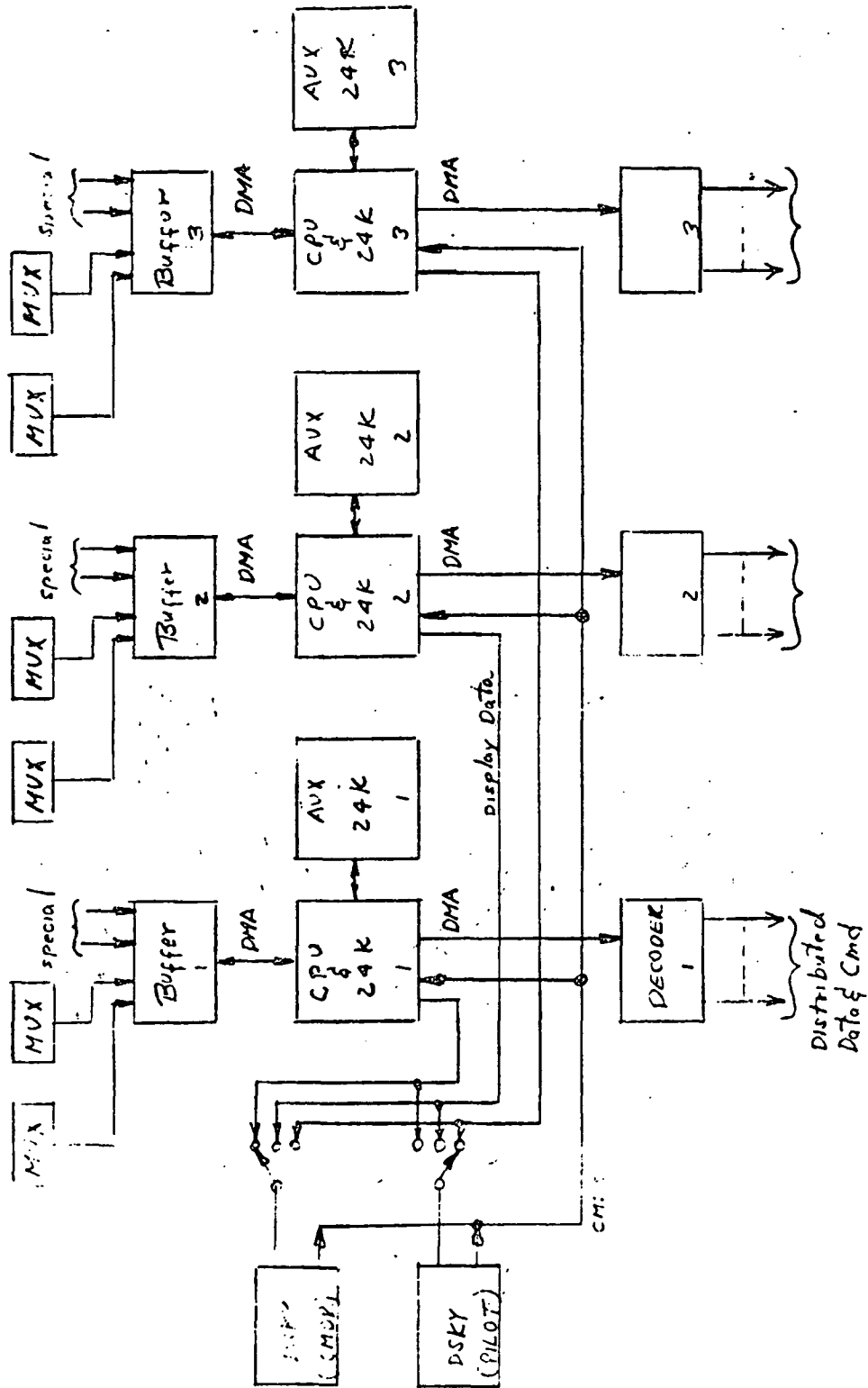


Figure 2.3.3-7. GN&C Computer Concept



Table 2.3.3-6. Orbiter GN&C Software Functional Requirements Changes

FUNCTION	FUNCTIONAL REQUIREMENT/SYSTEM CHANGE
GUIDANCE	<ul style="list-style-type: none">• NO ABORT TO ORBIT REQUIREMENT• NO AUTOLAND - MANUAL VFR• BOOSTER GUIDANCE
NAVIGATION	<ul style="list-style-type: none">• PRS, RADAR ALTIMETER DELETED• AUTONOMOUS NAV IMPLEMENTED WITH HORIZON SENSOR• TACAN EMPLOYED FOR ATMOSPHERIC NAV UPDATES, RENDEZVOUS• DEDICATED AIR DATA COMPUTER
CONTROL	<ul style="list-style-type: none">• NO DIGITAL AERO SURFACE CONTROL• HARDWARE JET LOGIC• BOOSTER TVC?
GN&C REDUNDANCY MANAGEMENT	<ul style="list-style-type: none">• SINGLE STRING• MANUAL CONTROL - NO SOFTWARE CONTROLLED SWITCHING• IN-LINE MONITORING REQUIRED
D&C	<ul style="list-style-type: none">• DEDICATED ATTITUDE/RATE DISPLAY• NO MULTI-FORMAT VECTOR TYPE DISPLAYS

Table 2.3.3-7. Software Requirements For Throw Away Vs. Flyback Booster

	ORBITER MEMORY (32K BIT)	BOOSTER MEMORY (32K BIT)
L02/RP F1 FLYBACK BOOSTER	30,300	16,400
PRESSURE FED BOOSTER	32,640	0

ASSUMPTION ON THROW AWAY BOOSTER

- . BOOSTER ALLOWED TO TUMBLE AFTER SEPARATION
- . BOOSTER CONTROLLED BY TVC ONLY
- . TVC INJECTOR LOGIC IN GN&C SOFTWARE
- . BODY RATES, ENGINE THRUST SIGNALS, PROPELLANT DEPLETION SIGNALS AND DC POWER STATUS SIGNALS ARE TRANSMITTED TO ORBITER

ASSUMPTION ON ORBITER

- . INCLUDES NAVIGATION ON-ORBIT UP-DATE (KIT 2)





predicated upon the assumption that there would be no GN&C computer in the pressure fed booster.

Orbiter/Booster GN&C Interface. As illustrated in Figure 2.3.3-8, the Pressure Fed Booster TVC control is achieved via 12 injectors on each of the 7 pressure-fed engines. No booster ACPS or surface control are used. Failure of any two injectors is regarded as yielding a "safe" condition.

For load alleviation and relative motion data, body rate sensors are required on the booster; the vehicle is assumed to provide local 28 VDC power sources.

The mated vehicles are controlled by orbiter with TVC injector logic handled in orbiter GN&C software. Additional booster data required by orbiter includes fuel depletion and DC power status signals.

As illustrated in Figure 2.3.3-9, the LOX/RP Flyback Booster controls mated vehicle during mated boost, with the following required data from orbiter:

- GN&C System Status
- Abort Command
- Separation Command
- Event Sequence Timing
- Orbiter Body Rates

In the mated boost phase, orbiter will require the following booster GN&C data as a minimum:

- GN&C System Status
- Load Relief Commands, for alleviation via orbiter aero surface control
- Event Sequence Timing
- Abort Command
- Propellant Depletion Data

Based on the foregoing interface data requirements, a qualitative study of the GN&C intra vehicle interface yielded the following results, applicable to either of the combined vehicles:

1. A hardwired interface is recommended for simplicity and lowest cost.
2. Three identical, physically separated umbilicals are required for redundancy and spatial diversity.
3. In consideration of the large number and relatively low rates of change of the interfaced signals, multiplexing is recommended.

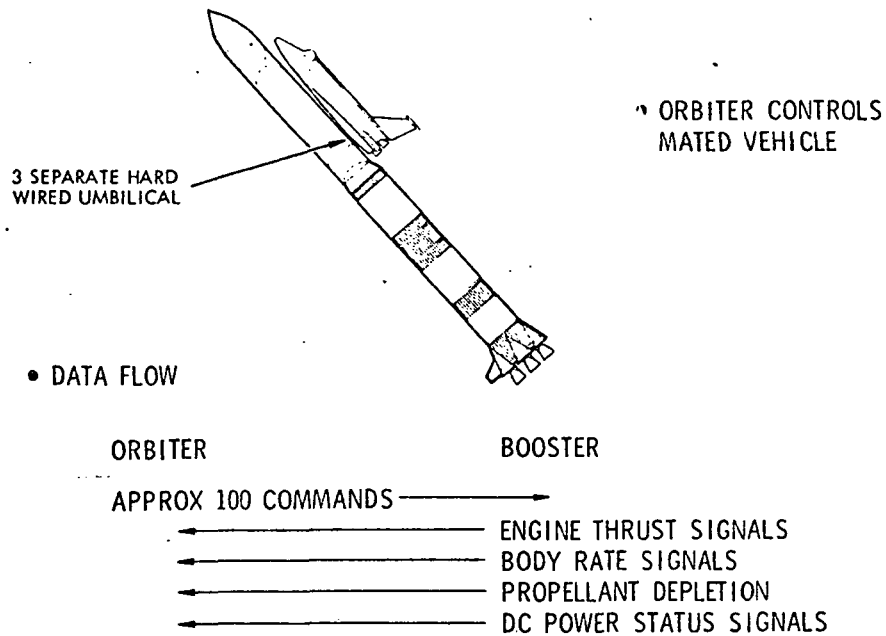


Figure 2.3.3-8. Pressure Fed Booster - Orbiter Interfaces

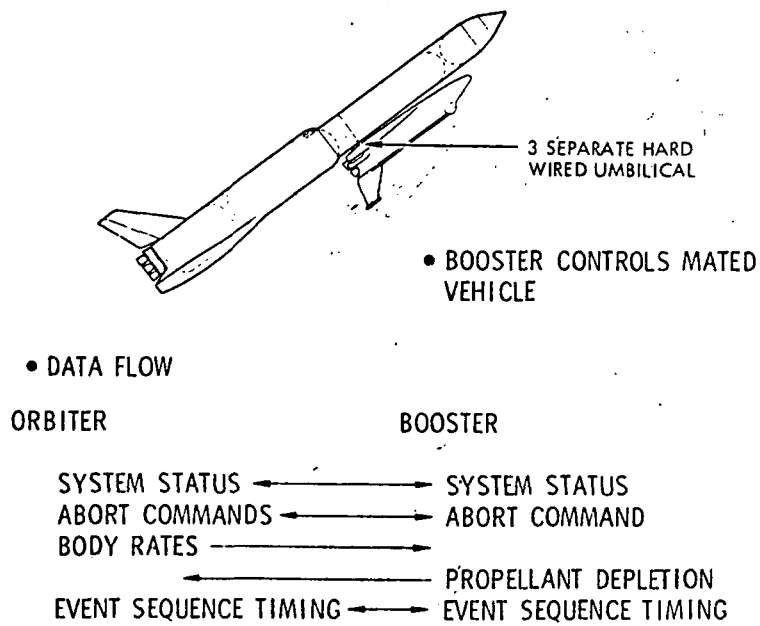
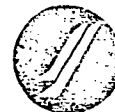


Figure 2.3.3-9. LO₂/RP Flyback Booster - Orbiter Interfaces



2.4 REDUNDANCY MANAGEMENT AND ONBOARD CHECKOUT

2.4.1 Redundancy Level Selections

The recommended degree of fault tolerance in the Avionics is generally fail operational-fail safe for flight safety and mission essential functions and fail safe for non essential items such as certain instrumentation. This recommendation is unchanged from the original Phase B trade study results and could have been predicted since the original trade study considered both state-of-the-art systems and conventional systems with the same results for both.

Some of the factors involved in the final level selections are shown in Figure 2.4.1-1. A trade study was conducted for some of the key GN&C components and the results are shown in Figure 2.4.1-2. It should be noted that weight, power, and cooling were not quantized in terms of cost due to vehicle cost model not being complete when the study was performed. It is estimated, however, that the value of the 147 pounds, 700 watts peak power, and 550 watts forced air cooling seen in the GN&C curve is over \$1M, thus providing the increased cost for the higher level of fault tolerance.

The breakdown by subsystem is summarized below.

<u>System</u>	<u>Reqmt.</u>	<u>Levels</u>	<u>Rationale</u>
GN&C	FO/FS	3	Criticality I - Safety
COMM	FO/FS	2	Criticality II - Mission
PDS	FO/FS	3	Criticality I, Interfaces
INSTR	FS	1	Criticality III, Precedence
D&C	FO/FS	*	*As required by interfaces

FMEAs and Logic Diagrams. In order to assure that safety and reliability were not degraded, a Failure Mode Effects Analysis was performed for each subsystem. In addition, logic diagrams were drawn. The analyses utilized a short form version of the FMEA and were performed to the depth necessary to identify single failure points and criticalities. The results of these analyses indicate that there are no critical single failure points in the Avionics subsystems. The logic diagrams and analyses are summarized on the following charts.

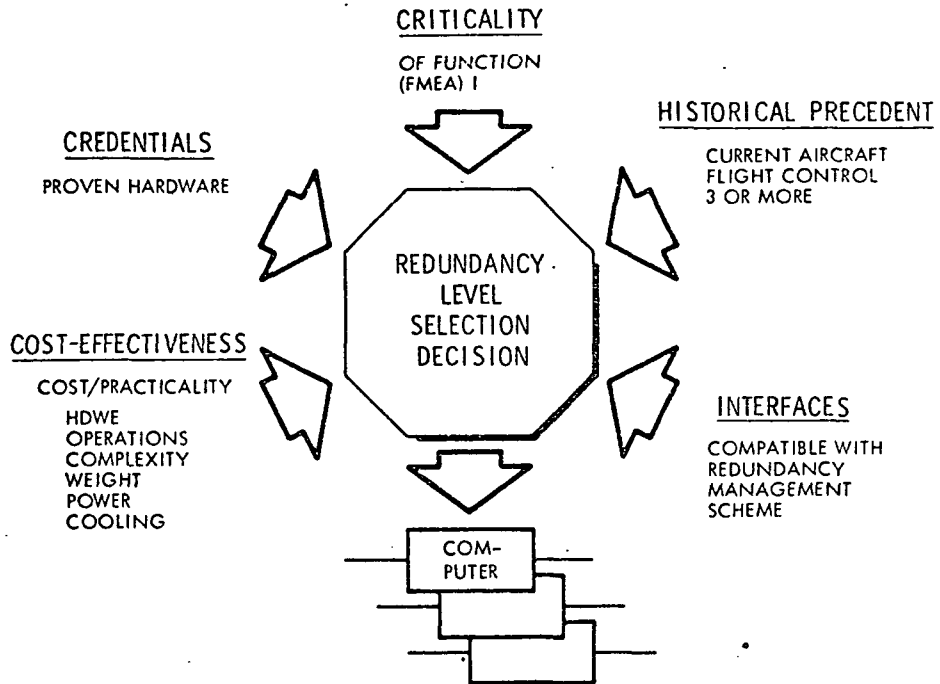
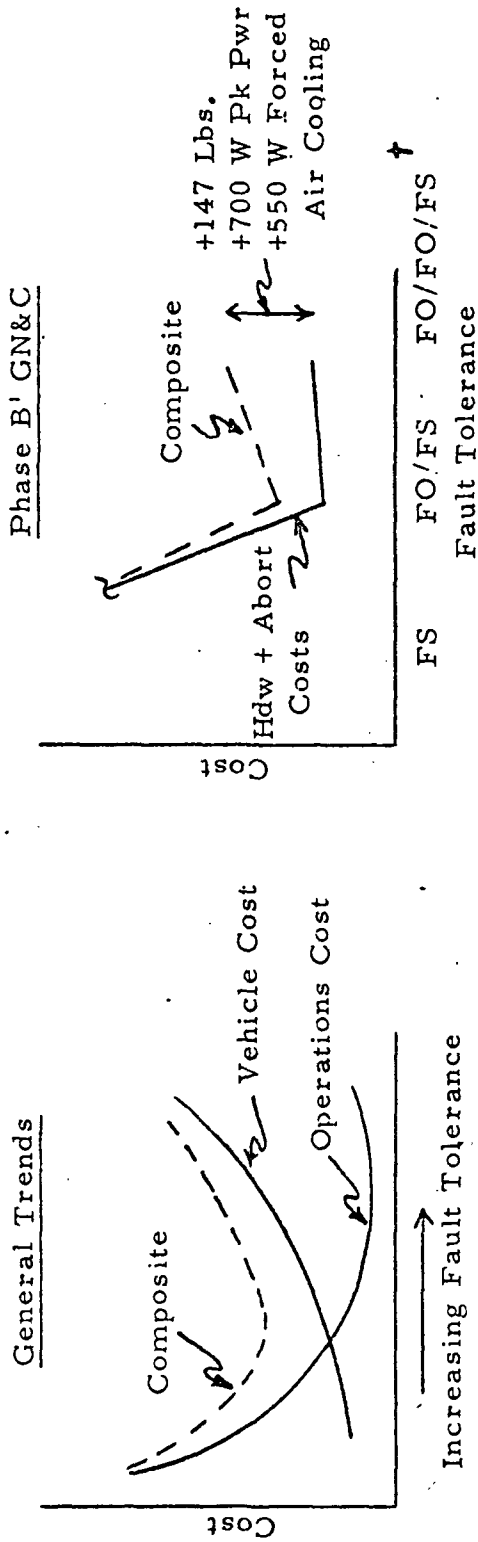


Figure 2.4.1-1. Redundancy Level Selection - GN&C Computers



Figure 2.4.1-2

GN&C Redundancy Trade Study



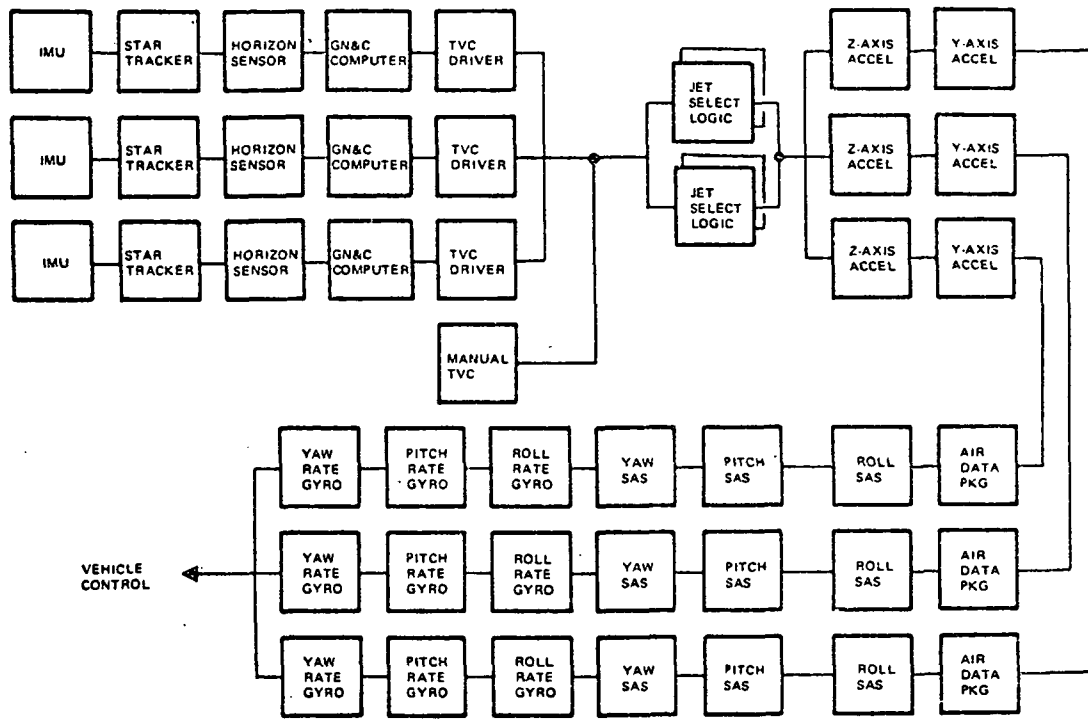
Hardware	Aborts/445 Missions		*Cost, \$M		FO/FO/FS	
	FO	FO/FS	FS	FO/FS	FO/FS	FO/FS
Rate Sensors	0.207	0.000	1.05	0.26	0.34	
IMU	10.638	0.191	46.77	3.15	3.13	
Air Data	0.132	0.000	1.35	1.25	1.53	
Star Tracker	3.877	0.025	17.17	1.55	2.16	
Computer	22.549	0.864	101.43	12.07	11.22	

*GN&C Hardware Recurring Cost Plus Abort Cost (At \$4.25M Each)

†Additional IMU and Computer Only



GUIDANCE NAVIGATION & CONTROL SUBSYSTEM RELIABILITY SUCCESS LOGIC



FAILURE MODE EFFECTS ANALYSIS

SUBSYSTEM GN&C		SHUTTLE PHASE B EXTENSION	SCHEMATIC DATE
ITEM IDENTIFICATION & QUANTITY USED	SCHEMATIC REF NO.	FAILURE MODE & EFFECT (IDENTIFY POTENTIAL HAZARD)	CRIT
INERTIAL MEASUREMENT UNIT 3 USED		NO OUTPUT, DEGRADED OR ERRONEOUS OUTPUT. LOSS OF VEHICLE CONTROL IN WORST CASE, LOSS OF ENTRY ATTITUDE & LOCATION. THREE PATHS	3
STAR TRACKER 3 USED		NO OUTPUT OR ERRONEOUS OUTPUT TO UPDATE IMU. INCORRECT BURN RESULTING IN EXCESSIVE FUEL USAGE & PREMATURE ABORT, IMPROPER ENTRY ATTITUDE. THREE PATHS	3
HORIZON SENSOR 3 USED		NO OUTPUT, ERRONEOUS OUTPUT. LOSS OF OR DEGRADATION OF STATE VECTOR UPDATE FOR POSITION RESULTING IN IMPROPER ENTRY. THREE PATHS	3
GUIDANCE NAVIGATION & CONTROL COMPUTER 3 USED		NO OUTPUT, ERRONEOUS OUTPUT, DEGRADED OUTPUT. LOSS OF ABILITY TO CONTROL AND/OR POSITION INFORMATION; POSSIBLE LOSS OF VEHICLE. THREE PATHS	3
JET SELECT LOGIC 2 USED		NO OUTPUT, ERRONEOUS OUTPUT. FAILURE TO FIRE, PREMATURE FIRE, FAIL TO CUTOFF, PREMATURE CUTOFF. RESULTS ARE FUEL DEPLETION OR LOSS OF VEHICLE CONTROL. TWO PATHS (IN ADDITION, UNITS HAVE DUAL POWER SUPPLIES)	3
AIR DATA PACKAGE 3 USED		NO OUTPUT, ERRONEOUS OUTPUT. COULD RESULT IN INCORRECT COMMAND TO AERO SURFACES & LOSS OF VEHICLE. THREE PATHS	3



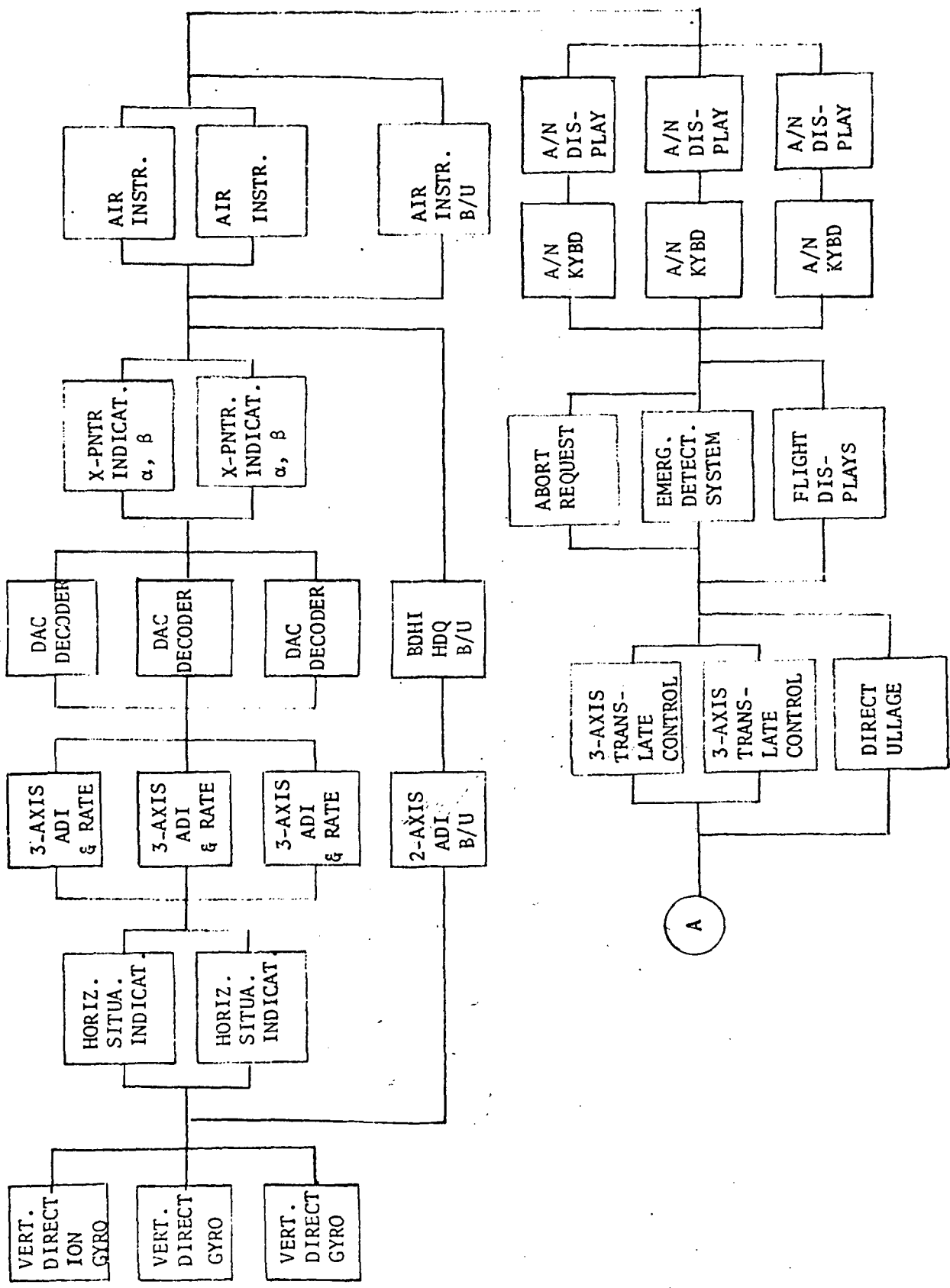
FAILURE MODE EFFECTS ANALYSIS

SUBSYSTEM GN&C SHUTTLE PHASE B EXTENSION SCHEMATIC DATE _____

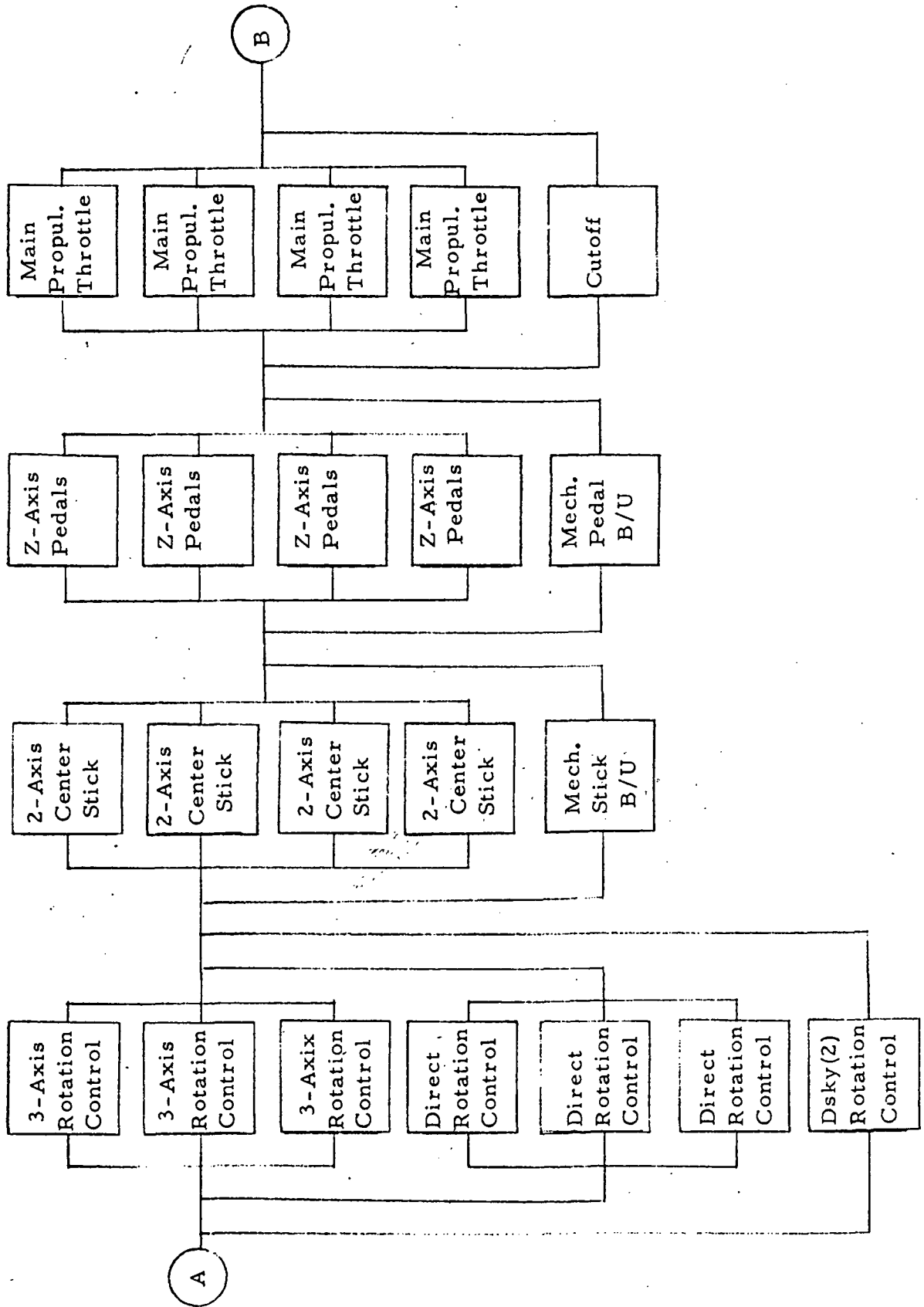
ITEM IDENTIFICATION & QUANTITY USED	SCHEMATIC REF NO.	FAILURE MODE & EFFECT (IDENTIFY POTENTIAL HAZARD)	CRIT
ACCELEROMETER PACKAGE 3 USED		NO OUTPUT, ERRONEOUS OUTPUT. ERRONEOUS ACCELERATION DATA RESULTING IN INCORRECT COMMAND TO AERO SURFACES & POSSIBLE LOSS OF VEHICLE. THREE PATHS	3
RATE GYRO 3 USED		NO OUTPUT, ERRONEOUS OUTPUT. INCORRECT DATA, RESULTING IN INCORRECT COMMAND TO AERO SURFACES & POSSIBLE LOSS OF VEHICLE. THREE PATHS	3
STABILITY AUGMENTATION SYSTEM 3 USED		NO OUTPUT, ERRONEOUS OUTPUT. DEGRADED OUTPUT. RESULTS IN UNWANTED OR ERRONEOUS POSITION OF AERO SURFACES WITH POSSIBLE LOSS OF VEHICLE. THREE PATHS - PLUS MECHANICAL SYSTEM	3
THRUST VECTOR DRIVER 3 USED		NO OUTPUT, ERRONEOUS OUTPUT. UNABLE TO ALIGN THRUST VECTOR THROUGH THE c.g. LOSS OF CONTROL OF VEHICLE DURING BOOST. THREE PATHS	3
THRUST VECTOR CONTROL 1 USED		NO OUTPUT, ERRONEOUS OUTPUT. LOSS OF MANUAL BACKUP CAPABILITY FOR THRUST VECTOR ALIGNMENT. 4TH ORDER FAILURE. ONE PATH	3

DISPLAYS & CONTROLS SUBSYSTEM

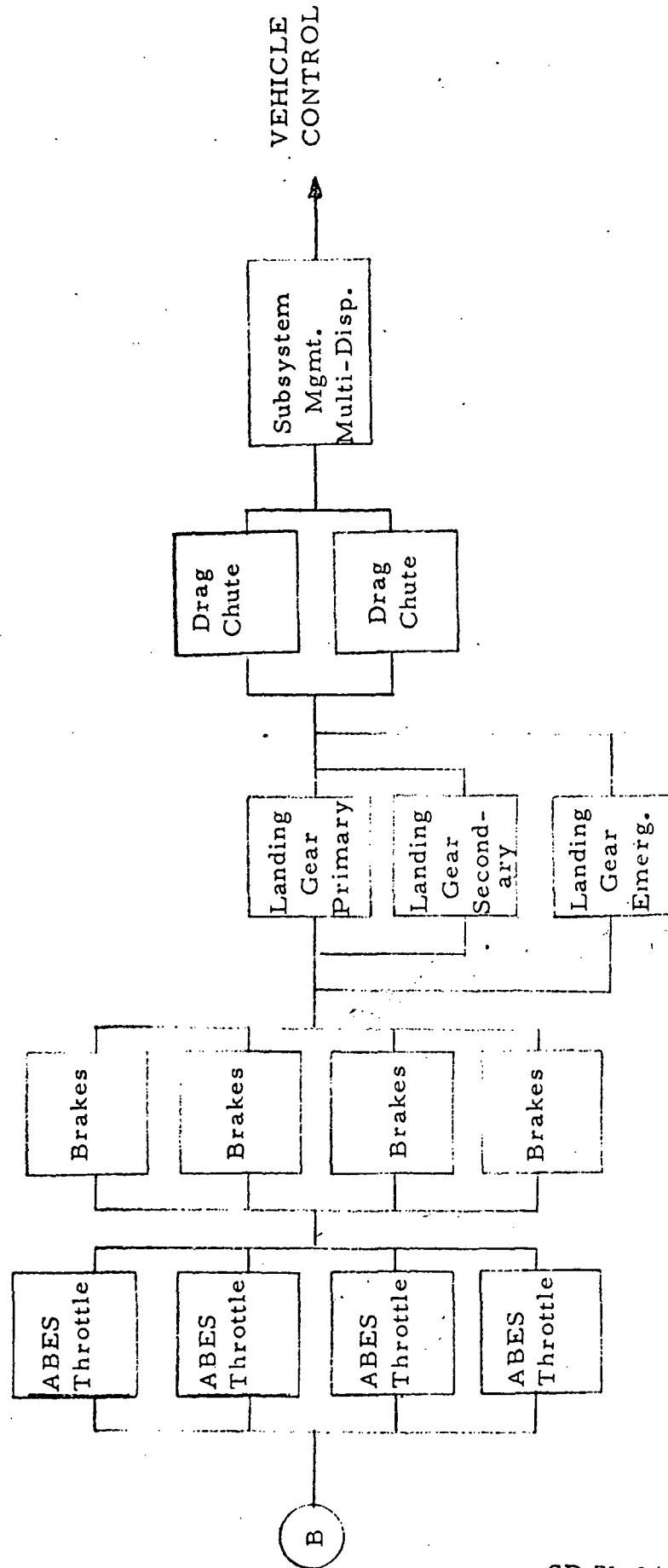
RELIABILITY SUCCESS LOGIC



DISPLAYS & CONTROLS SUBSYSTEM
RELIABILITY SUCCESS LOGIC



DISPLAYS & CONTROLS SUBSYSTEM
RELIABILITY SUCCESS LOGIC



FAILURE MODE EFFECTS ANALYSIS

SUBSYSTEM DISPLAYS & CONTROLS

SHUTTLE PHASE B EXTENSION

SCHEMATIC DATE 10-28-71

1 of 3

ITEM IDENTIFICATION AND QUANTITY USED	SCHEMATIC REF. NO.	FAILURE MODE & EFFECT (IDENTIFY POTENTIAL HAZARD)	CRIT.
Vertical Direction Gyro 3 Required	--	No output, erroneous output. Loss of horizontal reference plane and heading reference for location display. Units are redundant. Further backup is provided by the IMU.	3
Horizontal Situation Indicator 2 Required	--	No output, erroneous output. Loss of bearing and distance information from TACAN in aero-mode. Units are redundant and have further backup from the IMU.	3
3-Axis Attitude Direction Indicator 3 Required	--	No output, erroneous output. Loss or degradation of the Orbiter inertial orientation monitor. Units are redundant with backup by the 2-Axis ADI and BDHI and the IMU and computer.	3
DAC Decoder 3 Required	--	No output, erroneous output. Loss or degradation of the Orbiter inertial orientation monitor. Units are redundant with backup by the 2-Axis ADI and BDHI and the IMU and computer.	3
Air Instrumentation Direct Probe B/U 2 Required	--	No output, erroneous output. Loss of air data display (air speed, altitude, etc.). Possible disorientation and loss of vehicle control. System is redundant with direct probe backup.	3
A/N Display 3 Required	--	No output, erroneous output. Loss of ability to monitor computer activity. System has 3 success paths.	3
A/N Keyboard 3 Required	--	No output, erroneous output. Loss of display and update capability. Loss of vehicle and crew. System has 3 success paths.	3
Emergency Detection System 1 Required	--	No output, erroneous output. Loss of ability to detect a critical failure or an unwanted reaction to a false alarm. System is backed up by abort request and flight displays.	3
3-Axis Translation Control 2 Required	--	No output, erroneous output. Loss of translation ability or inadvertent translation. Loss of mission objective. System is redundant with backup by direct ullage. System will have an arming circuit.	3

FAILURE MODE EFFECTS A LYSIS

SUBSYSTEM DISPLAYS & CONTROLS SHUTTLE PHASE B EXTENSION SCHEMATIC DATE 10-28-71

2 of 3

ITEM IDENTIFICATION AND QUANTITY USED	SCHEMATIC REF. NO.	FAILURE MODE & EFFECT (IDENTIFY POTENTIAL HAZARD)	CRIT.
Rotation Control 3 Required	--	No output, erroneous output. Loss of ability to orient the orbiter for burn. Loss of vehicle and crew. System is backed up by 3 direct controllers and a computer (DSKY) mode, thus having 7 success paths.	3
2-Axis Center Stick 2 Required	--	No output, erroneous output. Loss of aero-dynamic surfaces (elevons), loss of vehicle and crew. System has 4 success paths and a mechanical backup.	3
2-Axis Pedals 4 Required	--	No output, erroneous output. Loss of rudder control in atmospheric flight. Possible loss of vehicle and crew. System has 4 success paths and a mechanical backup.	3
Main Propulsion Throttles 4 Required	--	No output, erroneous output. Loss of accurate vehicle acceleration "g" control. Possible to exceed vehicle "g" limits. System has 4 success paths with individual engine cutoff as a backup mode.	3
Air Breathing Engine System Throttles 4 Required	--	No output, erroneous output. Loss of atmospheric propulsion, possible loss of mission objectives. System has 4 success paths and will have an arming circuit.	3
Braking System 4 Required	--	No output, erroneous output. Possible to overrun landing strip or land with brakes locked. Possible loss of vehicle and crew. System has 4 success paths.	3
Landing Gear Control Primary Secondary Emergency (gravity drop)	--	No output, erroneous output. Loss of ability to lower landing gear, loss of vehicle and crew. Inadvertent lowering of landing gear with possible loss of vehicle and crew. System has 3 success paths and an arming circuit.	3
Drag Chute Control 2 Required	--	No output, erroneous output. Degradation of stopping ability, possible landing strip overrun. Inadvertent deployment with possible loss of vehicle and crew. System is redundant and employs an arming circuit.	3

SD 71-346

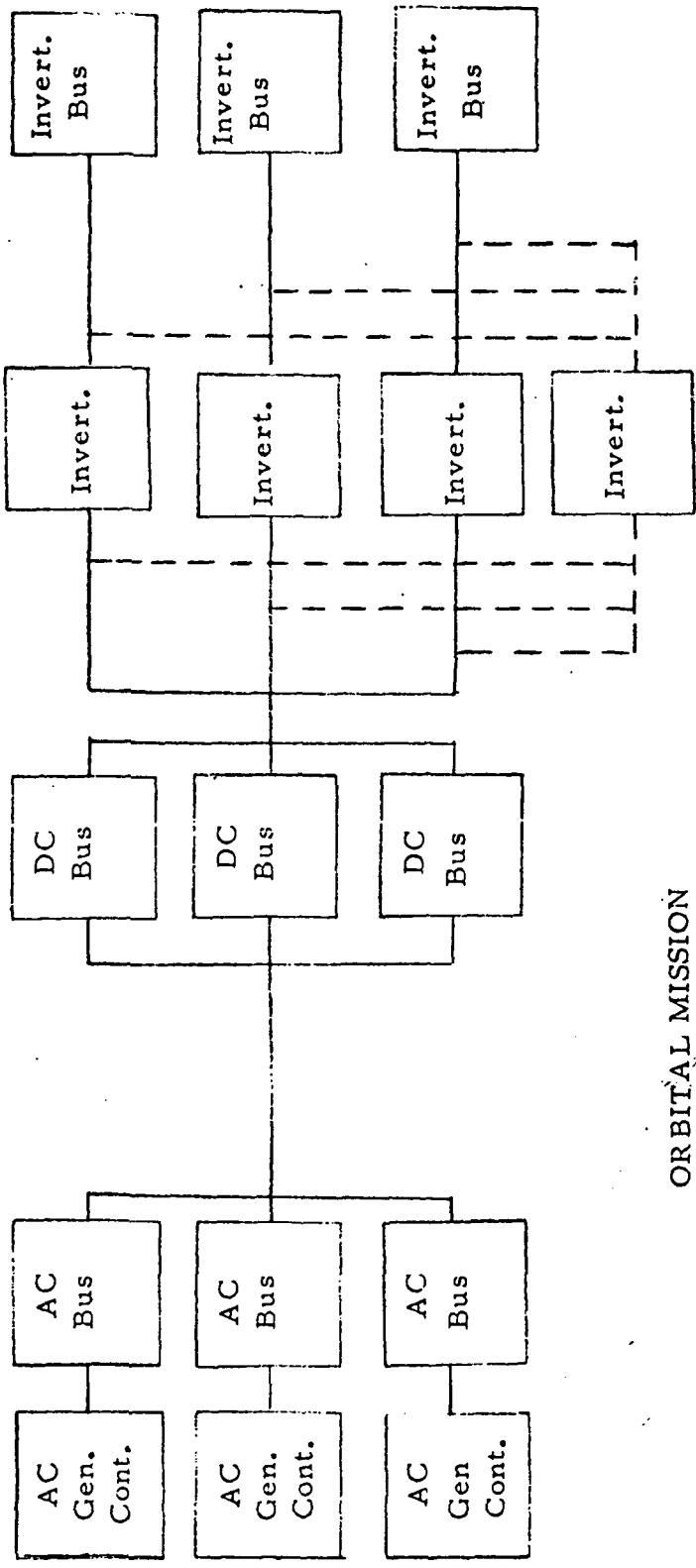
FAILURE MODE EFFECTS ANALYSIS

SUBSYSTEM DISPLAYS & CONTROLS SHUTTLE PHASE B EXTENSION SCHEMATIC DATE 10-28-71

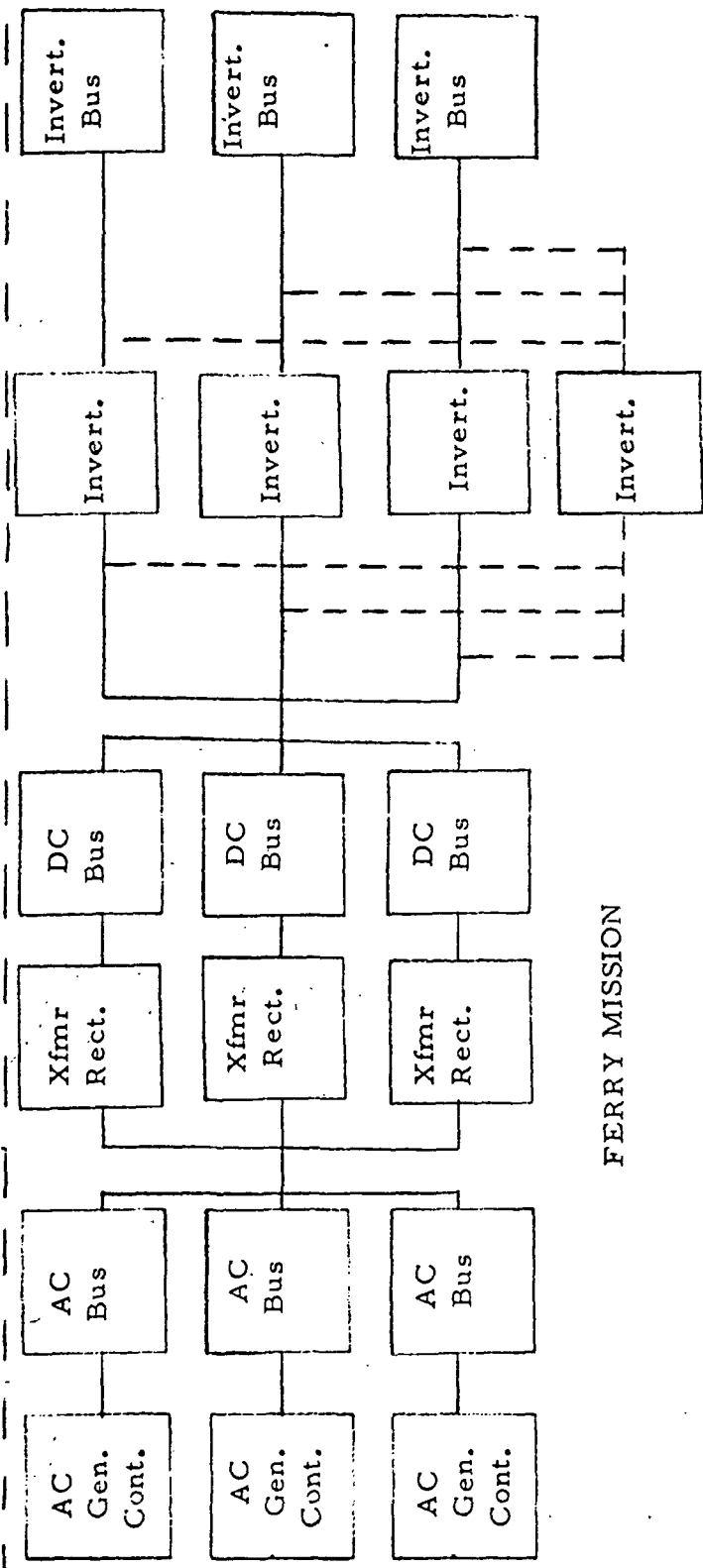
3 OF 3

ITEM IDENTIFICATION AND QUANTITY USED	SCHEMATIC REF. NO.	FAILURE MODE & EFFECT (IDENTIFY POTENTIAL HAZARD)	CRIT.
2-Axis ADI 1 Required	--	No output, erroneous output. Loss of reference plane and steering information. IMU is a backup. This system is a backup system to the 3-Axis ADI.	3
Bearing Direction Heading Indicator 1 Required	--	No output, erroneous output. Loss of heading information. IMU and Communications are backup. This system is a backup system to the 3-Axis ADI.	3
Cross Pointer Indicator 2 Required	--	No output, erroneous output. Loss of flight path error information. System is redundant and backed up by the ADI and DSKY.	3
Subsystem Management 1 Required	--	No output, erroneous output. Station is a collection of data display. Various data can be combined to calculate mission or in-error measurement. Subsystem measurement data presented are not critical to crew safety. Possible loss of mission objectives.	3

ELECTRICAL POWER DISTRIBUTION SUBSYSTEM
RELIABILITY SUCCESS LOGIC



ORBITAL MISSION



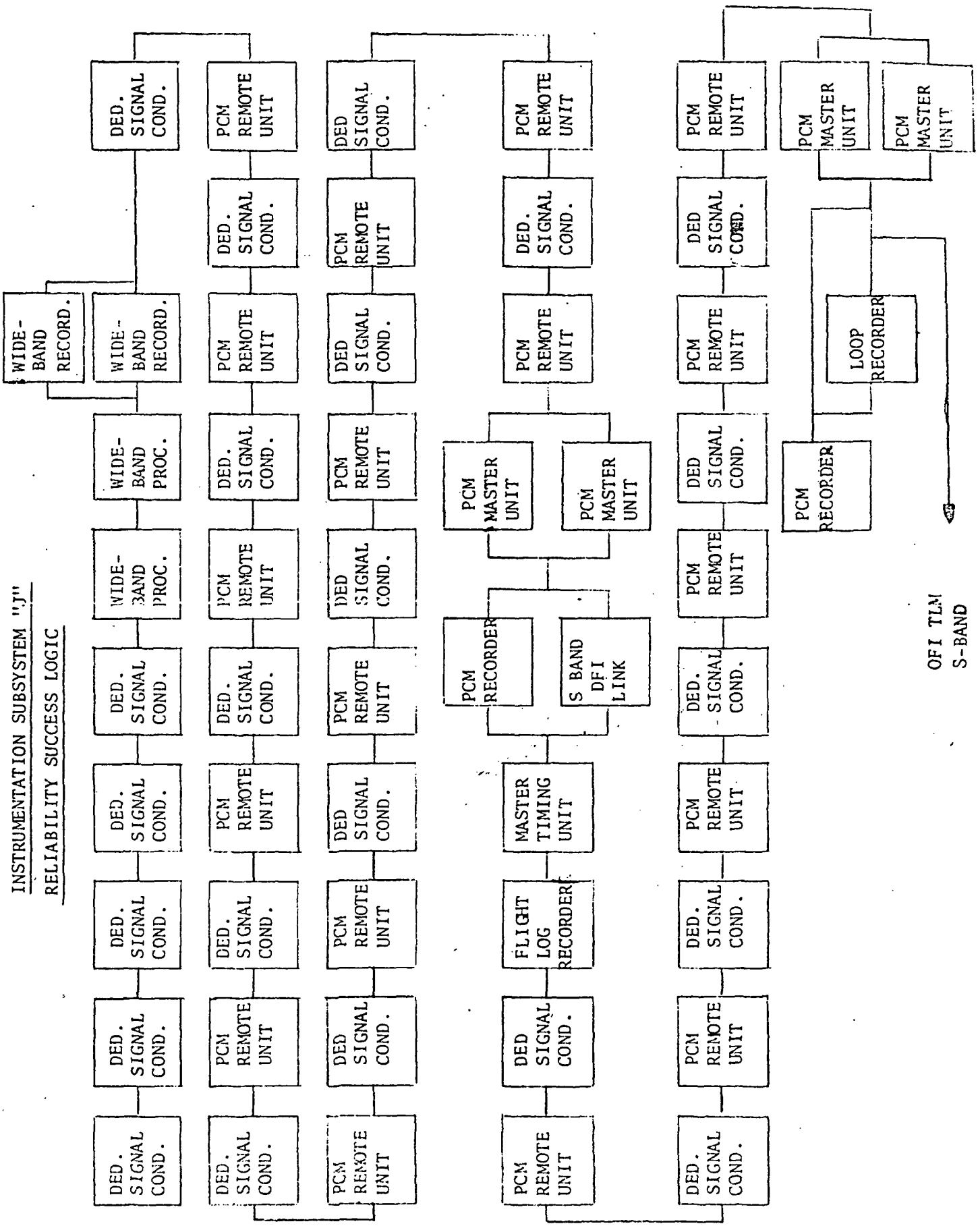
FERRY MISSION

FAILURE MODE EFFECTS ANALYSIS

SUBSYSTEM Electrical Power Distribution SHUTTLE PHASE B EXTENSION SCHEMATIC DATE 10-22-71

ITEM IDENTIFICATION AND QUANTITY USED	SCHEMATIC REF. NO.	FAILURE MODE & EFFECT (IDENTIFY POTENTIAL HAZARD)	CRIT.
AC Generator Control 3 Used	--	Loss of voltage sense or erroneous sense will result in generator being disconnected from AC bus.	3
AC Bus 3 Used	--	No output, short to ground. An open circuit will have no effect as AC buses are cross-strapped thru contactors. Since a short circuit may cause unit damage due to current transients and arcing, circuit breakers and fuses provide safety.	3
Transformer Rectifiers (Ferry) 3 Used	--	No output, short circuit. No effect as the other two units can power all DC buses. Since a short may allow AC or high current transients to damage units, circuit breakers and fuses provide protection.	3
DC Bus 3 Used	--	No output, short to ground. No effect - other two buses can carry load. Since a short may introduce high current transients that could damage units, circuit breakers and fuses provide safety.	3
Inverters 4 Used	--	No output. Loss of inverter bus and load. Inverters are connected to provide redundancy to each bus.	3
Inverter Bus 3 Used	--	No output. Loss of bus and load. Redundancy is provided by tying the load to two buses.	3
Emergency Control Bus 2 Used	--	No output, short to ground. Loss of ability to return power source back on line in the event of total loss of normal power. Redundant buses.	3
Battery 2 Used	--	No output. Loss of ability to power emergency control buses. Redundant batteries.	3
Battery Charger 1 Used	--	No output. Loss of ability to charge batteries.	3

INSTRUMENTATION SUBSYSTEM "J"
 RELIABILITY SUCCESS LOGIC



OFI TLM
 S-BAND

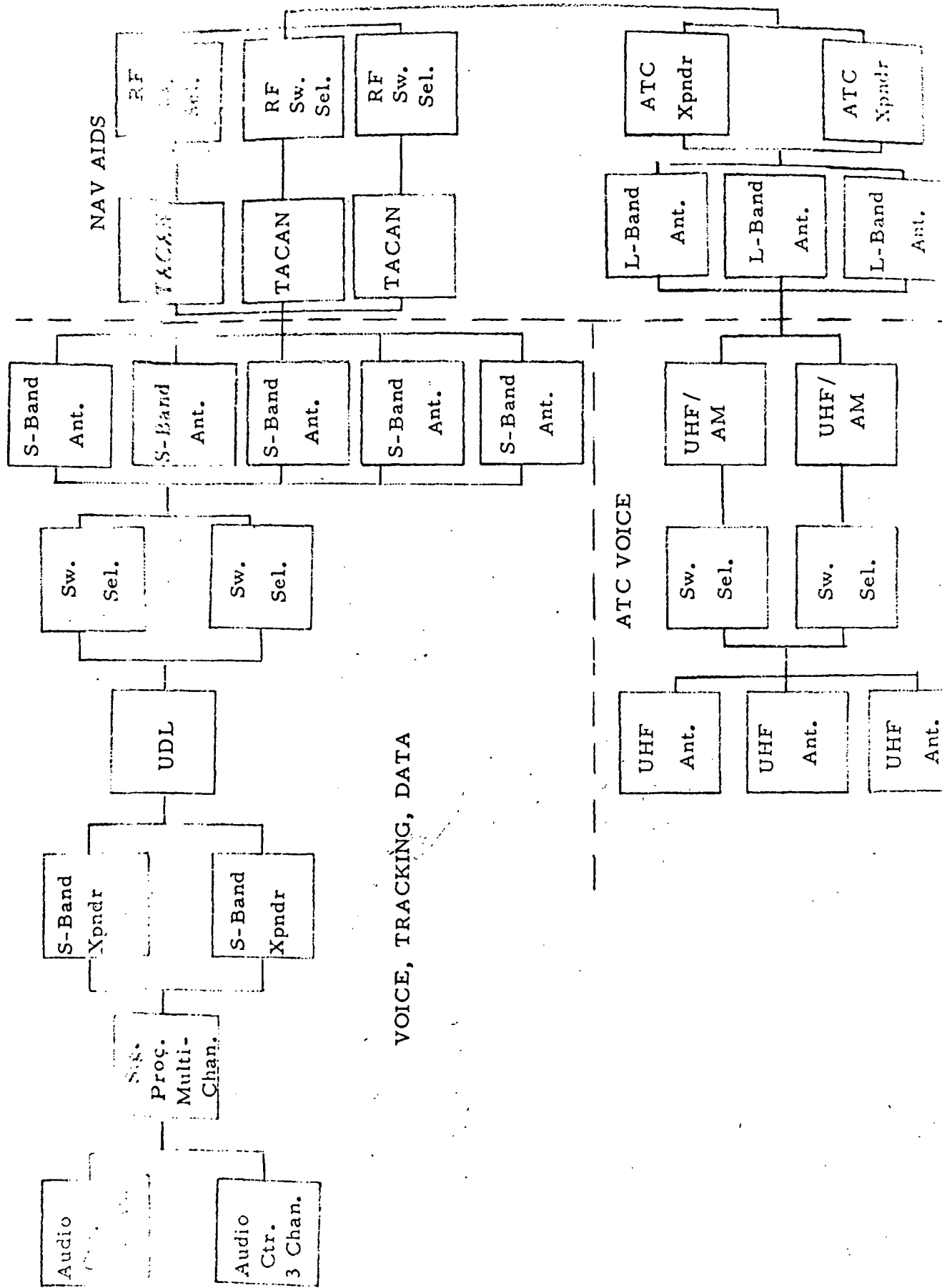
FAILURE MODE EFFECTS ANALYSIS

SUBSYSTEM Instrumentation "J" SHUTTLE PHASE B EXTENSION SCHEMATIC DATE 10-15-71

ITEM IDENTIFICATION AND QUANTITY USED	SCHEMATIC REF. NO.	FAILURE MODE & EFFECT (IDENTIFY POTENTIAL HAZARD)	CRIT.
Dedicated Signal Conditioners 23 Used	--	No output, erroneous output. Loss of a portion of the measurement data.	3
PCM Remote Unit 18 Used	--	No output, erroneous output. Loss of a portion of the measurement data.	3
Wide Band Processor 2 Used	--	No output, erroneous output. Loss of a portion of the DFI measurement data.	3
Wide Band Recorder 2 Used	--	No output, noise. Loss of a portion of the DFI measurement data. Units are redundant.	3
PCM Master Unit 4 Used	--	No output, erroneous output. Two units are redundant for DFI data and two units are redundant for OFI data. Loss of both units in either channel will lose all data.	3
PCM Recorder 1 Used	--	No output, noise. Loss of a portion of the OFI data. Backup (redundancy) exists in ground station recorders.	3
PCM Redorder 1 Used	--	No output, noise. Loss of all DFI data. Backup (redundancy) exists in ground station recorders.	3
Loop Recorder 1 Used	--	No output, noise. Loss of ability to look back at data.	3
Flight Log Recorder 1 Used	--	No output, noise. Loss of flight data for crash analysis.	3
Master Timing Unit 1 Used	--	No output, erroneous output, loss of synch. Loss of time base reference for data analysis. Ground station backup exists.	3

NOTE: Valuable and critical individual measurement data will be provided with redundancy by being routed through two or more physically separate instrumentation channels.

COMMUNICATION SUBSYSTEM
RELIABILITY SUCCESS LOGIC



FAILURE MODE EFFECTS ANALYSIS

SHUTTLE PHASE B EXTENSION SCHEMATIC DATE 10-20-71

SUBSYSTEM COMMUNICATIONS

ITEM IDENTIFICATION AND QUANTITY USED	SCHEMATIC REF. NO.	FAILURE MODE & EFFECT (IDENTIFY POTENTIAL HAZARD)	CRIT.
Audio Center 2 Used, 3 channels each	--	No output; loss of voice communication (on S-Band). Units are redundant. UHF communication is direct.	3
Signal Processor 1 Used	--	No output, or erroneous output. Loss of data to ground. Voice communication capability thru UHF/AM. Unit has multiple channels with switching capabilities.	3
S-Band Transponder 1 (dual) Used	--	No output, or degradation of output. Loss of DFI and OFI data. Loss of one link of voice communication. UHF/AM voice link still available. Transponders are in a redundant configuration.	3
Switch Selector	--	No output, no transfer. Loss of data or voice communication. Units are in a redundant configuration (2, 2, 3).	3
UHF/AM 2 Used	--	No output, degraded output. Loss of voice communication to Air Traffic Control. Units are redundant.	3
UDL 1 Used	--	No output. Loss of state vector data. Backup provided by voice capability on ground thru USB Transponder.	3
ATC Transponder 2 Used	--	No output, or degraded output. Loss of identification and altitude signals to Air Traffic Control facilities. Units are redundant.	3
S-Band Antenna 5 Used	--	No output. Loss of S-Band data. Units are redundant through partial pattern overlap for given attitude.	3
L-Band Antenna 3 Used	--	No output. Loss of navigation input. Units are redundant with partial coverage overlap.	3
UHF Antenna 3 Used	--	No output. Loss of Air Traffic Control voice capability. Units are redundant with partial coverage overlap.	3
TACAN 3 Used	--	No output. Loss of navigation input. Three paths.	3



2.4.2 GN&C Redundancy Management

The general topic of redundancy management is covered in the Onboard Checkout discussions (Section 2.4.3). Specific efforts in the area of GN&C redundancy management are unique, however, and merit separate discussion.

A special study addressing the problem of GN&C redundancy management was initiated. The study approach, guidelines, and some of the candidate techniques are presented in the following discussions. Preliminary results indicate that the FO/FS requirement can be implemented with three levels of redundancy without any significant sacrifices in the degree of coverage (detectability and recoverability) that would be afforded by a fourth string.

The time critical nature of the boost, entry, and approach/landing phases makes it necessary to use automatic redundancy management. The rationale for this is made clear by examining Figures 2.4.2-1, 2.4.2-2, and 2.4.2-3. The figures illustrate that failure conditions must be detected and corrected in less than 1-second in order to avoid catastrophic results. It is felt that a crew member cannot do this manually in the allotted 1-second time period.

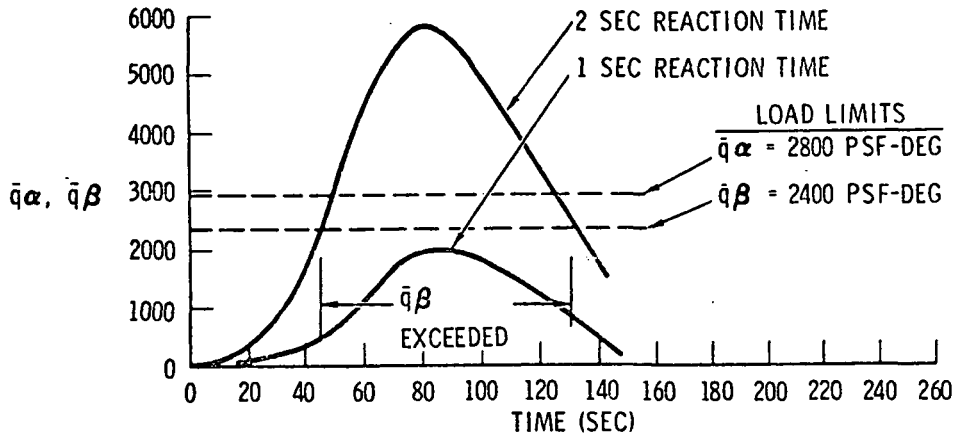
Figure 2.4.2-1 shows the effects of a hard-over gimbal (5-degree) on the structural loading during boost. A 2-second delay in failure detection and correction will result in exceeding the $q\beta$ load limits over 60% of the boost flight. This class of failure must be detected within one second in order to avoid exceeding the load limit, as shown in the figure.

Figure 2.4.2-2 illustrates that during re-entry, the sideslip angle β must not exceed approximately 2-degrees over the range of MACH numbers from 2 to 4. It is seen from the figure that an undetected yaw moment due to sideslip can cause the sideslip angle to diverge to the limit in 1.5 seconds. It is necessary to take corrective action (turn on opposing jets) within one second to avoid the uncontrollable region.

Figure 2.4.2-3 illustrates the effects of elevator deflection on normal acceleration. The chart on the left hand side shows that the load factor at maximum coefficient of lift increases with time. It is necessary to control C_L using elevator deflection to maintain the normal acceleration less than the limit of 2.5 g's. This chart shows that at about MACH 0.4, a sudden deflection of the elevator causing a maximum C_L could cause the loading to exceed 6 g's. The right hand chart illustrates the effect a ten degree elevator deflection at MACH 0.4 has on load factor. The deflection, lasting only one second, causes the load factor to peak at about 7 g's; clearly a rudder deflection lasting only one second causes excessive loading. Again, this figure illustrates the need to detect and correct failures in less than 1-second.



• STRUCTURAL LOADS DUE TO HARDOVER GIMBAL (5 DEG)



• CONCLUSION—LOADS COULD BE EXCEEDED OVER 60% OF BOOST FLIGHT

Figure 2.4.2-1. Boost Loads Due to Pilot Detection & Switching Lag

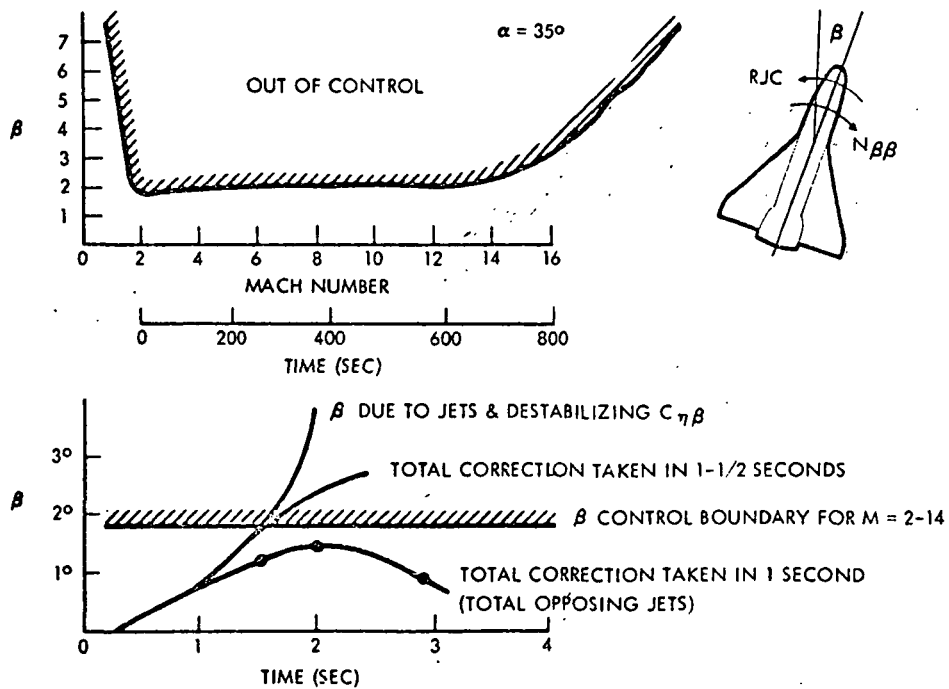
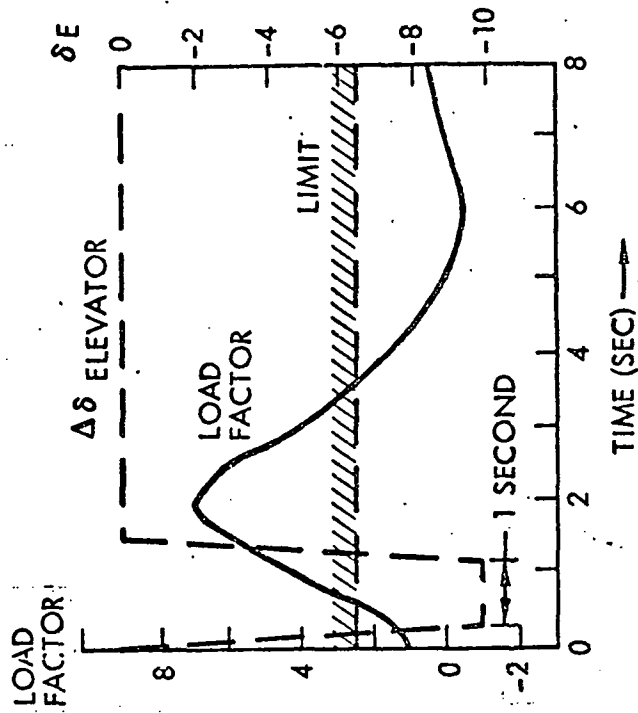


Figure 2.4.2-2. Entry ACPS Control

FIGURE 2.4.2-3
AEROSURFACE CONTROL

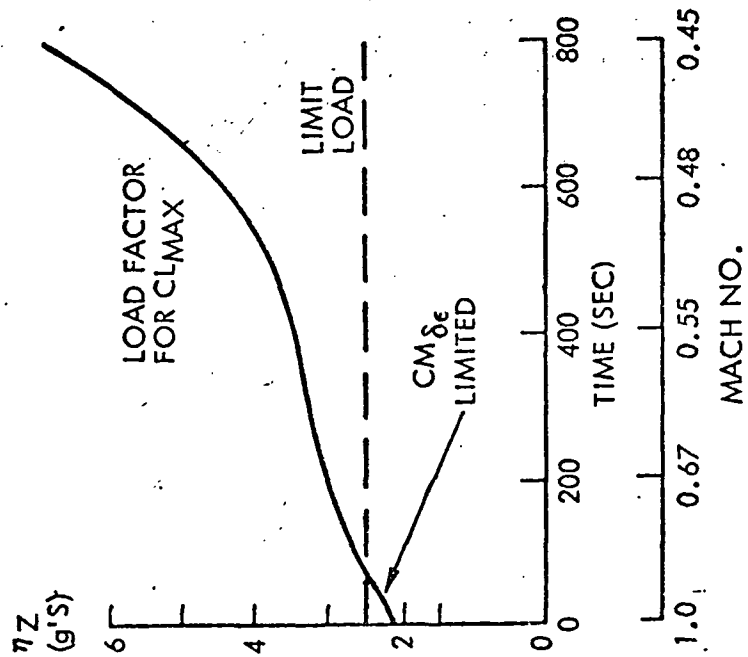


$M = 0.4$
 $h = S.L.$



$\delta e_{MAX} = -45^\circ \text{ TO } +15^\circ$

$\delta e_{MAX} = \text{UP TO } 20^\circ/\text{SEC}$





The GN&C computer is the source of the information used by the crew to determine the general health of each redundant string. Visual examination of the side-by-side computer operational outputs of each redundant string on the alphanumeric displays is one method of detecting and isolating failures to a given string. This method, however, does not give sufficient information about the status of each sensor during a system checkout prior to mission critical phases when the sensors are not being exercised. Additional hardware (BITE) is needed to provide the necessary status information. The BITE provides the secondary signal interface between the LRU's and the GN&C computer units. The computer, after monitoring or converting these signals, transmits the additional status information to the caution and warning display to complement the operational signals as an aid in the redundancy management and fault isolation task.

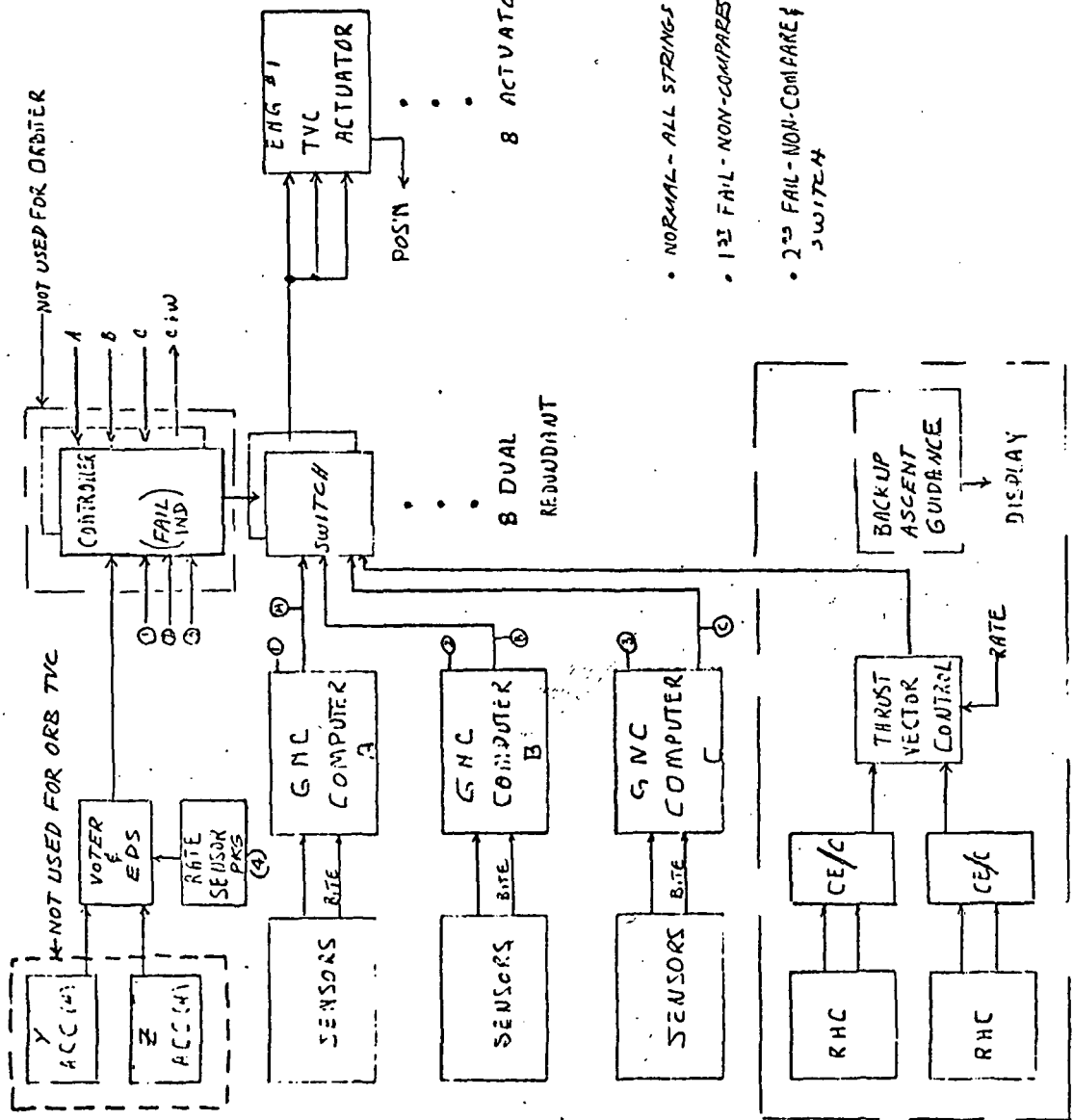
GN&C Redundancy Management Follow-On Study. The new baseline uses a masking technique (middle select or majority voting) for redundancy management of the first failure. Techniques for the second failure require further study; also, there are alternates for the first failure that should be considered further. Figures 2.4.2-4 through 2.4.2-7 represent some promising techniques.

Additional techniques under consideration are:

1. Four strings. A FO/FS system utilizing a masking technique for the first and second failures. The fourth string is used as a standard and is not available as a control string. This technique has the advantages of transient-free operations and having the same recovery strategy for both failures.
2. Dissimilar Backup (fourth) string for TVC and ACPS. A FO/FS system using a comparison technique for the first and second failures. If a second failure occurs, control is switched over to the backup string. Again, one string is used as a standard and is not available as a control string.
3. Establish redundant ACPS groups. Functionally separates the ACPS engines to create a triple redundant "muscle" which can be managed appropriately. Since there are insufficient jets for three complete groups (42 needed), about 1/3 of the jets must be shared by redundant functional strings.

Some of the specific problem areas to be addressed are:

1. Use of EDS type tests to detect the second failure and initiate switching.
2. Setting a workable threshold for "non-compare" may require time syncing computers and/or cross strapping sensors.
3. The redundancy management controller could easily become complex.
4. Comparisons between ACPS strings represents a potentially serious timing problem.
5. It is essential that the system be designed to allow the control operation to recover from an unstable yaw attitude during entry.



- NORMAL - ALL STRINGS UP, ONE IN CONTROL
- 1st FAIL - NON-COMPARE OR FAIL IND = SWITCH
- 2nd FAIL - NON-COMPARE EDS IND, OR FAIL IND = SWITCH



Figure 2.4.2-4. TVC Redundancy Management Candidate

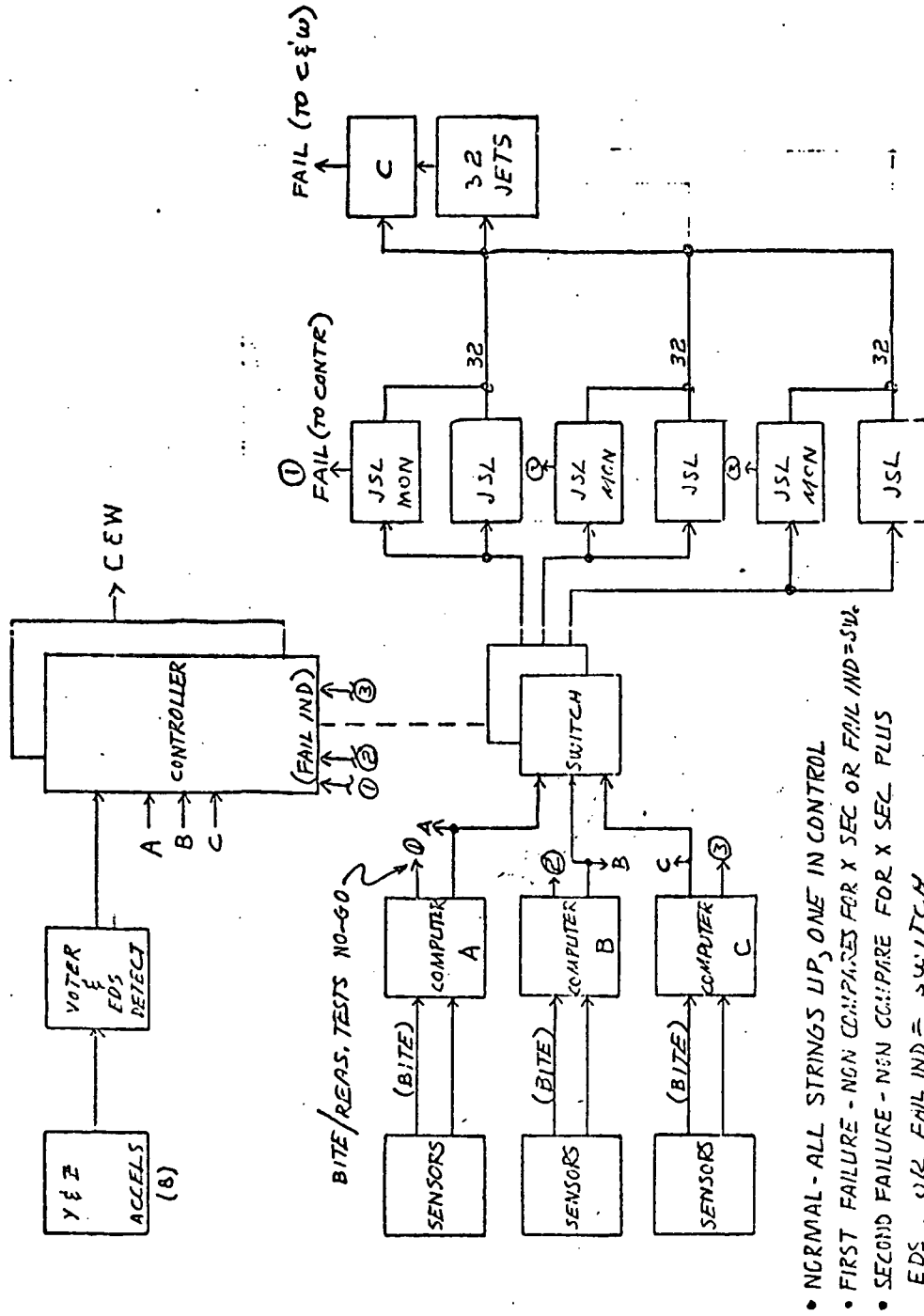


Figure 2.4.2-5. GN&C ACPS Management - Entry

- NORMAL - ALL OPERATE
- FIRST FAILURE - VOTE, DISABLE FAILED STRING
- SECOND FAILURE - LOCK SERVOS - USE STICK
- FLIGHT MAY RE-ENGAGE GOOD CHANNEL

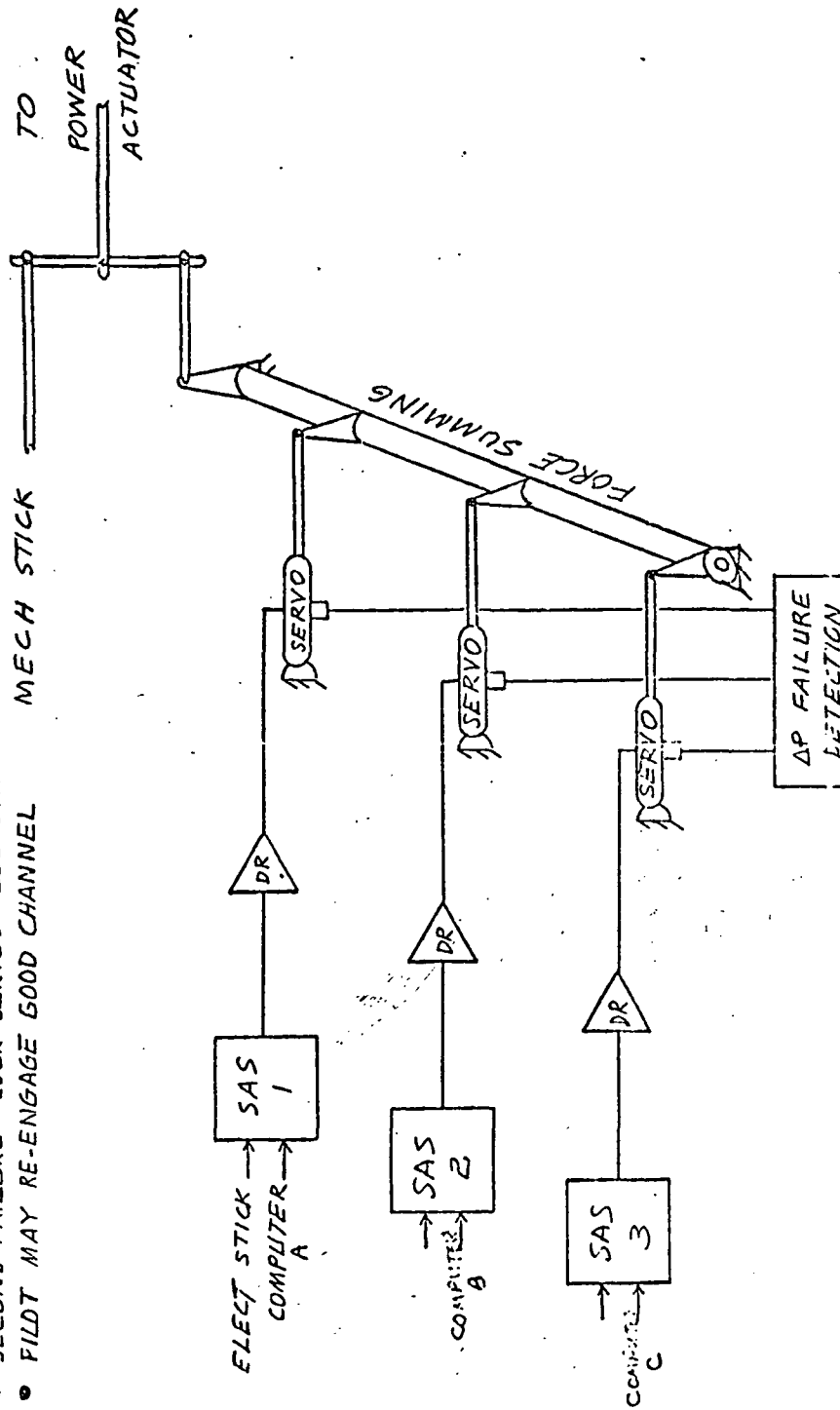


Figure 2.4.2-6. Aero Surface Control

- NORMAL - ALL OPERATE
- FIRST FAILURE - VOTE, DISABLE FAILED STRING
- SECOND FAILURE - LOCK SERVOS - USE STICK
- PILOT MAY RE-ENGAGE GOOD CHANNEL

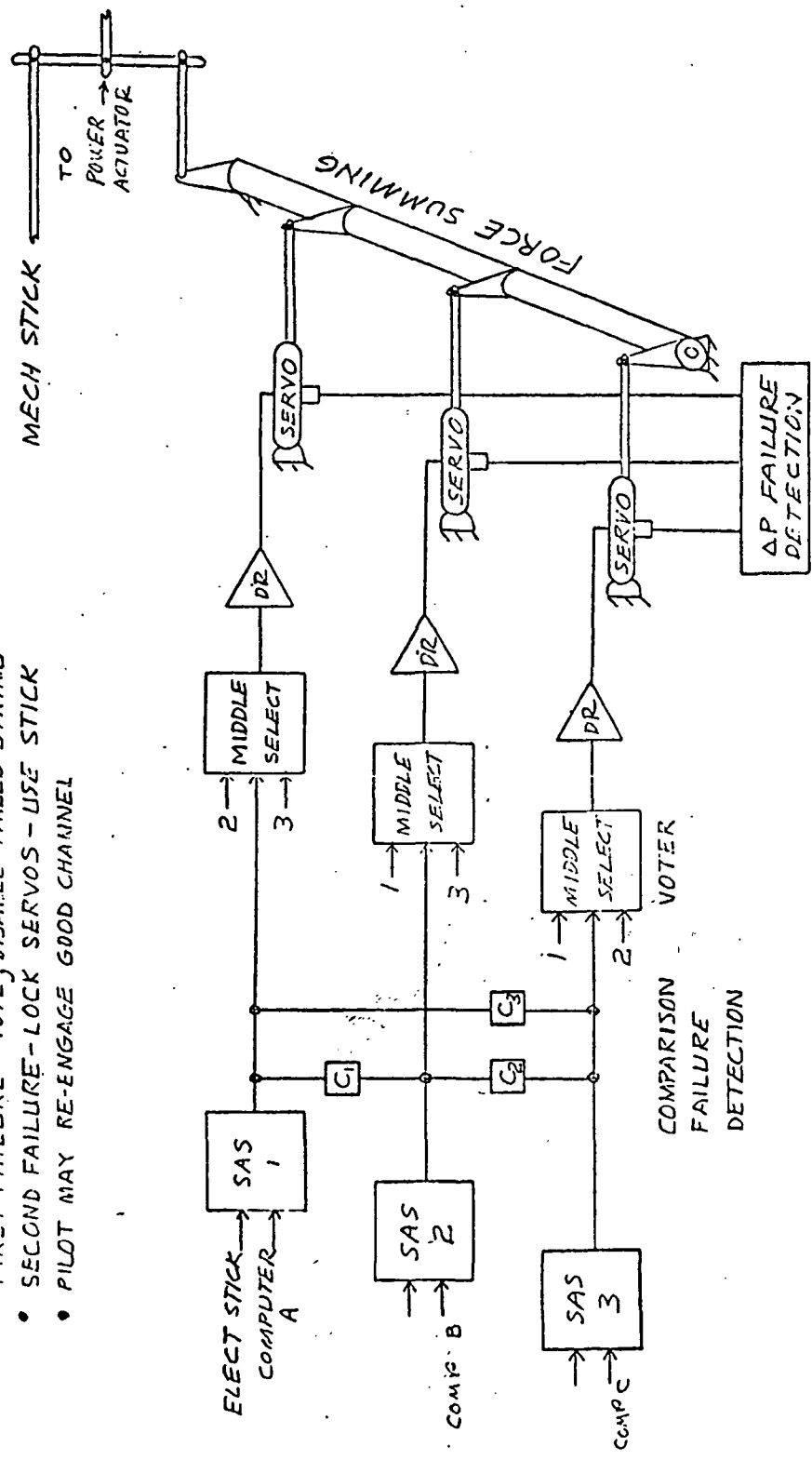
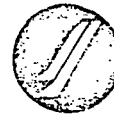


Figure 2.4.2-7. Aero Surface Control



2.4.3 Onboard Checkout and Redundancy Management

The two primary issues relative to onboard checkout are autonomy and turnaround. The autonomy objective is to eliminate manpower required to support each flight, thereby reducing operational costs. The primary method of achieving this objective is to provide crew aids in the form of onboard performance monitoring to the functional path level. The turnaround objective is to minimize the checkout and test required during the operational phase. This can be accomplished by providing sufficient data acquisition capabilities onboard including permanent storage of information just prior, during, and just after the occurrence of a malfunction.

PRIMARY ISSUES

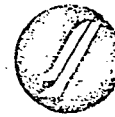
- AUTONOMY - ACHIEVE WITH CONVENTIONAL, OFF-THE-SHELF EQUIPMENT—ELIMINATE GROUND BASED FLIGHT SUPPORT

ONBOARD PERFORMANCE MONITOR TO FUNCTIONAL PATH LEVEL

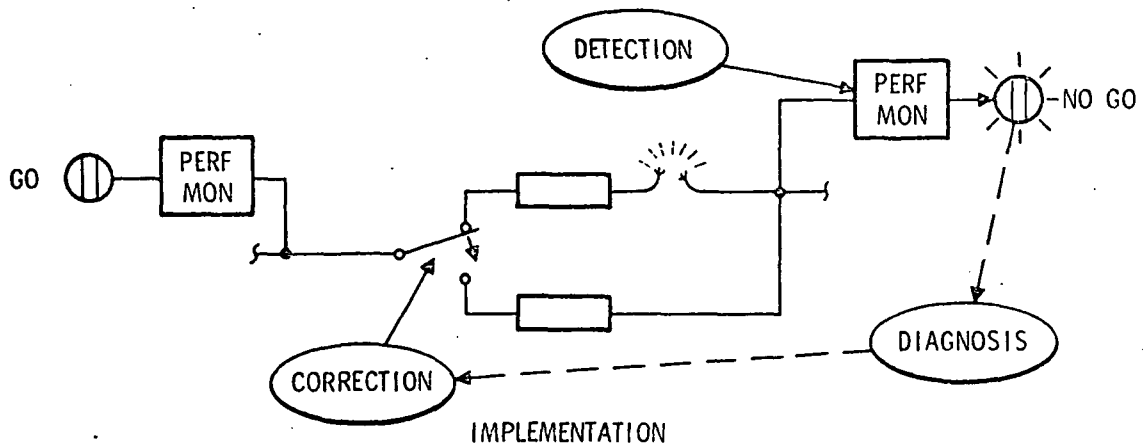
- TURNAROUND - ACHIEVE WITH MINIMUM GROUND CREW & EQUIPMENT

IDENTIFY PROBLEM TO REPLACEABLE UNIT(S)

Figure 2.4.3-1. Onboard Checkout & Redundancy Management



Redundancy Management. Redundancy management is defined to be the inflight management of identical or non-identical redundant functional paths. Three distinct processes are required to manage these functional paths in the presence of malfunctions: (1) Detection of a fault, (2) Diagnostic action to determine the fault source, (3) Corrective action to disable the faulty functional path and enable a properly operating redundant path. The comparison of how these three processes are accommodated in Phase B' vs. Phase B is as shown.



IMPLEMENTATION

FUNCTION	PHASE B	PHASE B'
DETECTION	SOFTWARE ORIENTED	HARDWARE & SOFTWARE
DIAGNOSTICS	SOFTWARE ORIENTED	CREW ORIENTED
CORRECTION	CREW ORIENTED	CREW ORIENTED

Figure 2.4.3-2. Redundancy Management

Performance Monitoring. Performance Monitoring (PM) is used for the detection process of redundancy management. PM can take several forms and can be accomplished by a variety of mechanizations ranging from completely automatic to completely manual. Analysis of subsystem measurement requirements indicates that the subsystems can generally be managed manually except for specific functions in time-critical situations (ascent, reentry, landing). However, completely manual subsystem management would make large time demands on the crew for essentially routine tasks. Restrictions on crew activities such as rest periods, eating, payload functions would be incurred or crew size would be increased. Automatic fault detection will provide a reasonable off-loading of routine tasks and allow attention to be directed where attention is due.



High-low limit testing and BITE types lend themselves to automation; while tracking and consumables management are more typically manual operations. Agree-disagree and position types occupy a middle ground. The numbers in the left hand margin of Figure 2.4.3-2 represent the current estimate for types of PM on the orbiter vehicle.

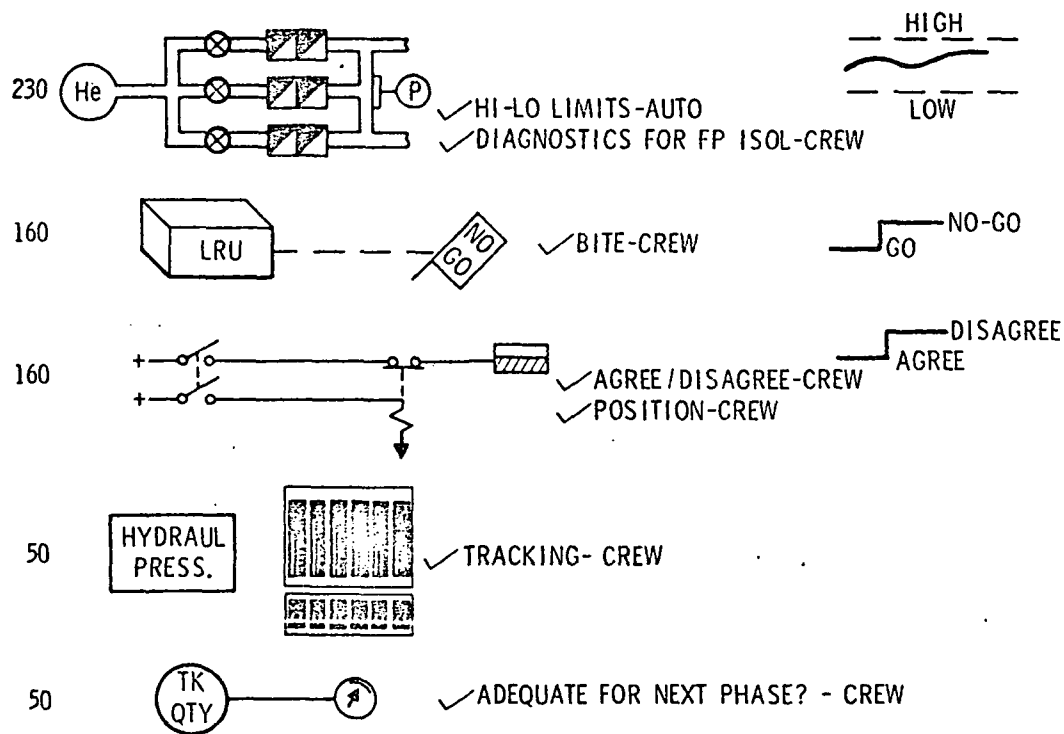


Figure 2.4.3-3. Performance Monitoring

The following requirements were established for performance monitoring:

- (1) Time Critical functions must be accommodated within subsystems as BITE so that the PM function can be mechanized as a non-redundant function. This does not exclude critical parameter monitoring by PM.
- (2) PM must be capable of evaluating 4 different limit sets/parameter (mission phase dependent).
- (3) PM must contain false alarm (transient) avoidance provisions.
- (4) PM must include capability for enabling external recording devices when any parameter exceeds limits.
- (5) PM must have capacity for 600 measurement (50% Analog).
- (6) PM must be able to control 150 external annunciators.

The PM mechanization selected for the new baseline is described in Section 2.5.



Non-Avionic Subsystems. A definition of the major non-avionics subsystems and how they operate normally is necessary in order to determine PM requirements. Table 2.4.3-1 is a summary of the redundancy management technique for non-avionics subsystems. Two significant points should be emphasized: (1) Most of the subsystems are of the load-sharing type; (2) most are highly mission-phase oriented. There are exceptions in the prime services area. Fuel cells may be reconfigured for quiescent on-orbit phases by placing 2 in standby to conserve consumables. The ECLSS will be configured "all up" during critical maneuvers (i.e., ascent, reentry) to reduce the complexity of manual recovery from a prime loop failure.

Table 2.4.3-1

NON-AVIONIC SUBSYSTEM DESCRIPTIONS

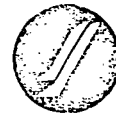
SUBSYSTEM	CONFIGURATION	PRIME PHASES	FUNTIONAL PATH OPERATION
MAIN PROPULSION SYSTEM (MPS)	(4) J2-S ENGINES EXTERNAL DROP TANKS (H_2/O_2)	ASCENT	ALL PARALLEL
ORBIT MAN. SYSTEM (OMS)	2 LM ASCENT ENGINES 2 REMOVABLE PODS (HYPERGOLIC)	ON-ORBIT	ALL PARALLEL
ATTITUDE CONTROL (ACPS)	34 ENGINES 3 REMOVABLE PODS (HYPERGOLIC)	ON-ORBIT/ REENTRY	ALL AVAILABLE/ SELECTED
AIR BREATHING ENGINES (ABES)	4 ENGINES (B-1) 2 REMOVABLE PODS	FERRY	ALL PARALLEL
HYDRAULICS (HYD)	3 INDEPENDENT SYSTEMS + ACCUMULATORS	ASCENT/ AERO	ALL PARALLEL
FUEL CELLS (FC)	3 FUEL CELLS 2 H_2/LO_2 TANKS BALANCED DEPLETION	ALL VERTICAL	ALL PARALLEL
AUXILIARY POWER UNITS (APU)	3 N_2H_4 APU's 3 REMOVABLE FUEL PODS	ASCENT/ REENTRY AERO	ALL PARALLEL
ENVIRONMENTAL CONTROL & LIFE SUPPORT (ECLSS)	3 ATMOSPHERIC CONTROL LOOPS 2+ COOLANT LOOPS 2 WASTE/WATER LOOPS	ALL	1 UP/1-2 STANDBY



Avionic Subsystems. The emphasis in avionic subsystems is on the use of the conventional hardware and single strings. This is summarized in Table 2.4.3-2. As in the non-avionic subsystems case, most of the paths are operated in parallel - the communications subsystem is the exception. The cockpit displays and controls are not listed as a separate subsystem; they are considered (for checkout purposes) to be a part of the subsystem they service. All of the subsystems are heavily BITE oriented.

Table 2.4.3-2. AVIONIC SUBSYSTEM DESCRIPTIONS

SUBSYSTEM	CONFIGURATION	PRIME PHASE	FUNCTIONAL PATH OPERATION
COMMUNICATIONS	(4) VOICE (UHF + USB) (2) DATA (USB) 1 UDL 3 TACAN 2 ATC XPNDR	ALL ASCENT ALL ATMOS/REND ATMOS	1 UP/STANDBY 1 UP/STANDBY 1 UP ALL UP - SEQUENCED 1 UP + STANDBY
ELECTRICAL POWER DISTRIBUTION & CONTROL	3 MAIN AC BUSES 3 MAIN DC BUSES 3 INVERTER BUSES 3 TR's 3 CONTROL BUSES 3 GCU's	ASCENT/ATMOS ALL ALL FERRY ALL ASCENT/ATMOS	ALL PARALLEL ALL PARALLEL ALL PARALLEL ALL PARALLEL ALL PARALLEL ALL PARALLEL
GUIDANCE, NAVIGATION, & CONTROL	3 ANALOG SAS + SENSORS 3 DIGITAL GN&C COMPUTERS + SENSORS 1 MANUAL CONTROL	AERO ALL ALL	ALL PARALLEL ALL PARALLEL STANDBY
INSTRUMENTATION & DATA	1 PCM REMOTE SET 2 MASTER PCM 1 LOOP RECORDER 1 PCM RECORDER	ALL ALL ALL ALL	1 UP 1 UP/1 STANDBY 1 UP ON DEMAND



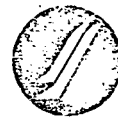
Operational Measurements. The number of operational measurements required for the orbiter was determined by analysis of each subsystem on an avionics user basis and is detailed in Table 2.4.3-3. The "cockpit display" column and the "Record/GND USE" column are independent in that the same measurement is accounted for in only 1 of these two columns; therefore, the "Orbiter Total" column is the sum of the first two. The "Cockpit Display" total represents all of the measurements that the crew would have access to during a mission. This does not imply that all 595 measurements are simultaneously displayed since most of the analog displays are time shared types (switchable meters, page format CRT, etc.). As shown on the summary, the large majority of measurements originate in non-avionics subsystems.

Table 2.4.3-3. Operational Measurements

• AVIONICS USER ESTIMATE									
SUBSYS	COCKPIT DISPLAY			RECORD/GND USE			ORBITER TOTAL		
	ANALOG	DISCR	TOTAL	ANALOG	DISCR	TOTAL	ANALOG	DISC	TOTAL
MPS	7	26	33	145	100	245	152	126	278
OMS	26	26	52	14	10	24	40	36	76
ACPS	33	48	81	48	110	158	81	158	239
ABES	29	54	83	20	5	25	49	59	108
HYD	25	10	35	13	1	14	38	11	49
FC	18	38	56	25	12	37	43	50	93
APU	18	24	42	11	1	12	29	25	54
ECLSS	62	0	62	42	5	47	104	5	109
MISC	1	0	1	0	0	0	1	0	1
NON AVIONICS	219	226	445	318	244	562	537	470	1007
INSTR/DATA	0	15	15	0	0	0	0	15	15
EPDC	24	31	55	17	2	19	41	33	74
COMM	0	10	10	2	6	8	2	16	18
GN&C	30	40	70	26	34	60	56	74	130
AVIONICS	54	96	150	45	42	87	99	138	237
GRAND TOTAL	273	322	595	363	286	649	636	608	1244

80%

20%



Apollo Inflight Measurements. In order to gain a perspective on the number of inflight measurements required for both cockpit display and performance monitoring, Apollo CSM measurement lists were reviewed for comparison. Referring to Figure 2.4.3-4, one observes that operational CSM's required a total of 455 inflight measurements, 257 of these were available to the crew and 311 were available to ground controllers. This means that 70% of the measurements were available for ground controller performance monitoring. It also indicates that 144 measurements were exclusively available to the crew and 198 were exclusively available to the ground.

Table 2.4.3-4. Apollo Inflight Measurements

	<u>IN FLT MEAS</u>	<u>AVAIL TO CREW</u>	<u>PCM TO GND</u>
SPS	36	19	24
COMM/INST	32	11	22
RCS	61	54	36
GN&C - SCS	115	39	78
LVEDS	32	30	5
ELEC PWR/SEQ	116	72	88
ECS	38	26	38
MISC	26	6	20
TOTAL	<u>455</u>	<u>257</u>	<u>311</u>



Checkout Data Flow. The Instrumentation subsystem illustrated in Figure 2.4.3-4 provides the data acquisition services through 6 remote PCM units. The 6 remote PCM output trains are combined into a single composite PCM train of < 100 KBS. The composite PCM is made available to 4 separate users: (1) communications subsystem for ascent telemetry; (2) ground umbilical connector for ground operations; (3) onboard recording system consisting of a continuous loop recorder and a permanent, on demand, maintenance recorder; (4) performance monitor unit. The PM unit, GN&C, and Caution and Warning can enable transfer of the previous 5-minutes of data from the loop recorder to the maintenance recorder. This would occur when a fault is detected in any of these areas. The transfer can also be manually enabled by the crew.

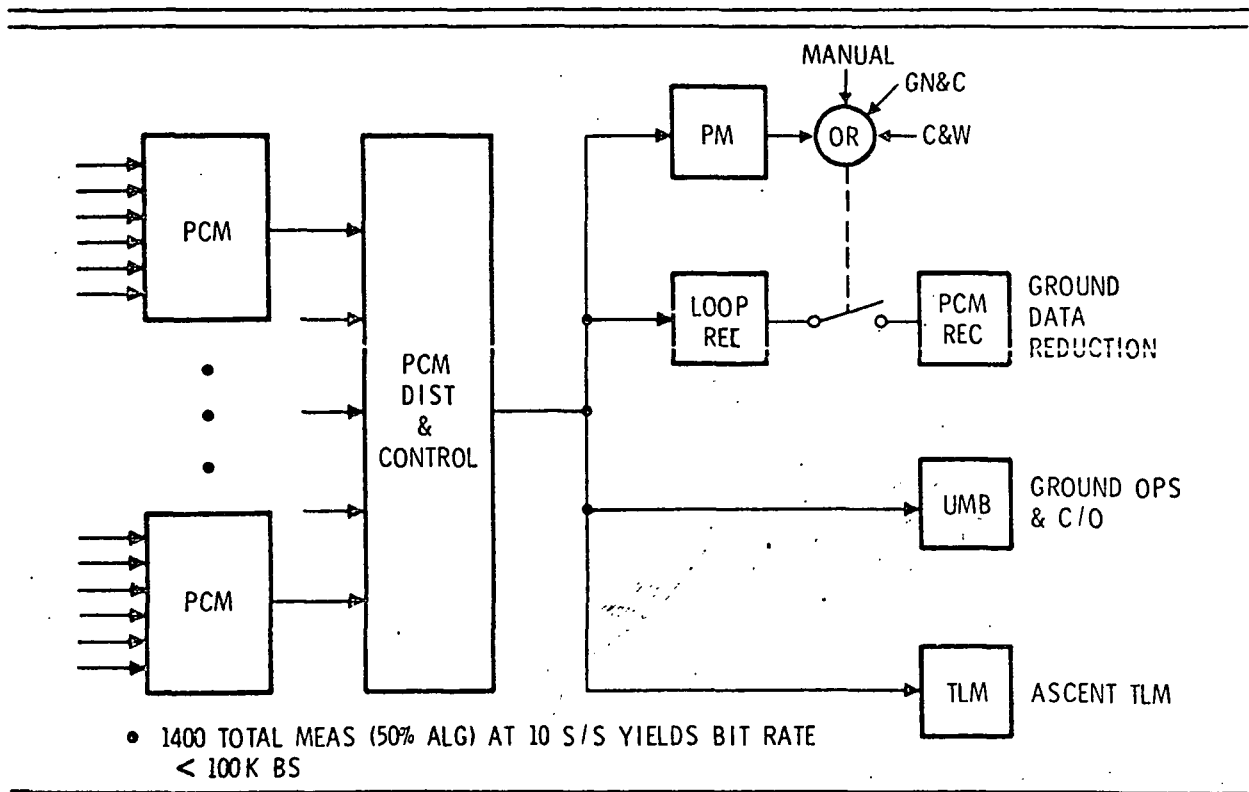


Figure 2.4.3-4, Checkout Data Flow



LRU Isolation Capability. Estimates of LRU Isolation capability obtained in the new baseline avionics approach have been made for all sub-system functional paths. Figure 2.4.3-5 shows that approximately 80% of the functional paths can be isolated to the LRU level for the most prevalent failure modes. This estimate is based on both the astronauts actions required for functional path management and the lower level data that is recorded whenever a fault occurs. It is also based on an operational vehicle that has flown a number of flights from which failure tendencies are historically based.

DERIVED FROM FP MGMT

	<u>FUNCT PATHS</u>	<u>LRU ISO AVAILABLE</u>	<u>PERCENT</u>
AVIONICS	105	82	80
NON-AVIONICS	<u>265</u>	<u>128</u>	<u>48</u>
SUBTOTAL	370	210	56

Δ DERIVED FROM DATA RECORDING/REDUCTION + NONSUPPORTED GND OPS + VISUAL INSP

AVIONICS		+ 8	
NON-AVIONICS		<u>+73</u>	
SUBTOTAL		<u>+81</u>	<u>+24</u>
TOTAL	370	291	80

REMAINING 20% }
 10% 2 LRU'S
 5% 3 LRU'S
 5% 4 OR MORE

Figure 2.4.3-5. LRU Isolation Capability

3 ⊕



Operational Phase Telemetry. Operational telemetry is used during the ascent phase to minimize the onboard Main Propulsion System (MPS) management task. This approach was taken because of the short duration of MPS usage (<10 minutes) and the fact that most of the MPS data must be supplied to the ground system for propellant loading and engine conditioning prior to launch. A fallout of this approach is that MPS performance data will be available immediately for turnaround planning for the next flight.

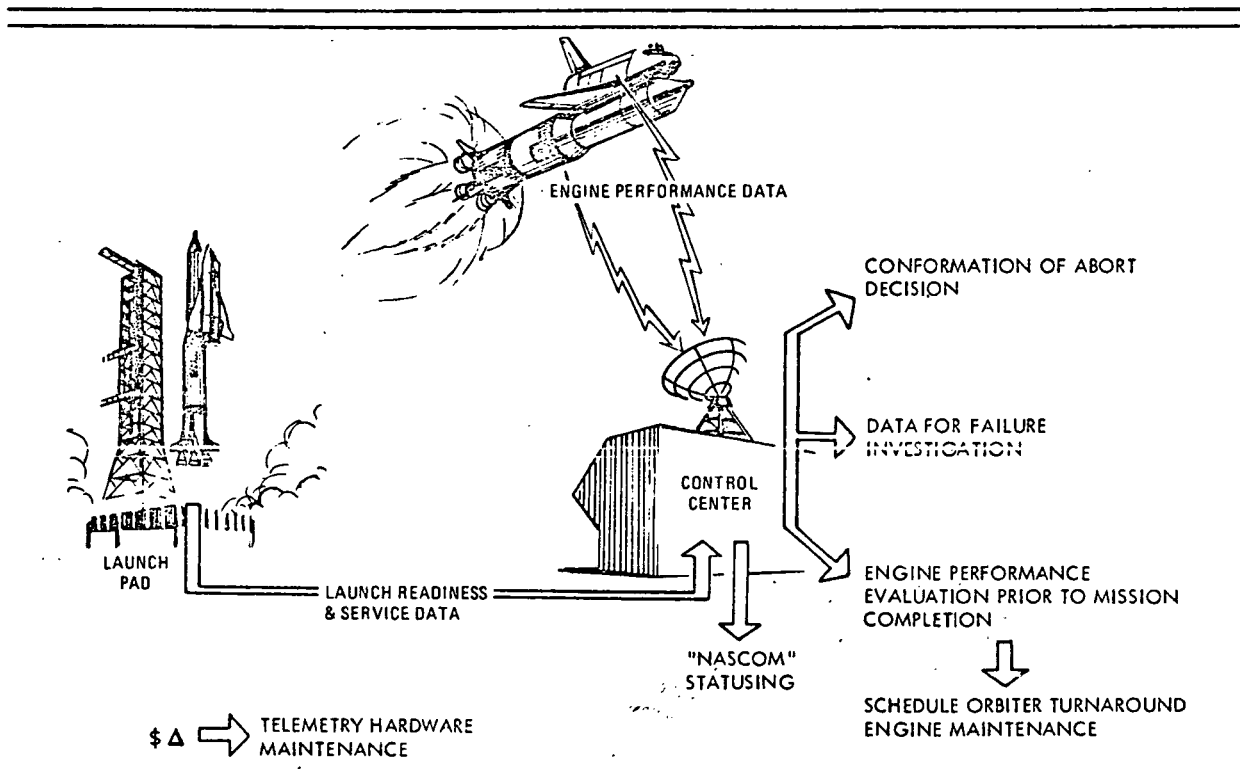


Figure 2.4.3-6. Operational Phase Telemetry



Summary. Operational autonomy from a checkout viewpoint can be achieved using conventional approaches including automatic fault detection performance monitoring. Sufficient data acquisition and recording capability are included to permit the anatomy of a failure to be determined by ground data reduction.

The keys to maintaining a two-week turnaround capability are (1) relying on the previous flights performance data to ascertain vehicle condition thereby reducing ground testing, and (2) providing sufficient but not excessive instrumentation to obtain the performance data.

Table 2.4.3-5. Summary

OPERATIONAL AUTONOMY

- . CAN BE ACHIEVED USING CONVENTIONAL APPROACHES INCLUDING PERFORMANCE MONITORING

OPERATIONAL TURNAROUND

- . 2 WEEK TURNAROUND CAN BE ACCOMPLISHED WITH INFLIGHT DATA RECORDING & ON-CONDITION MAINTENANCE



2.5 DISPLAYS & CONTROLS

2.5.1 Scope

The following tasks were addressed in the subject study:

1. Define a minimum display/control configuration consistent with the low-cost avionics concept, ground rules, and subsystem/vehicle interfaces.
2. Evaluate low-cost mechanization approaches for flight and subsystems display hardware, and define cost and phasing implications of such approaches.
3. Explore display techniques which could be used to augment or upgrade minimum configurations for added capability.

Configuration Selection

A number of alternate D&C approaches were evaluated for consistency with program/mission concepts, interfaces, and cost/phasing objectives. The major drivers in the areas of development cost and interface complexity were identified and the effects of the various approaches assessed. The resulting configuration is a compromise in terms of development cost, interface complexity, crew size requirements and vehicle development phasing considerations.

The selected configuration for the Orbiter employs a 2-man flight crew in a 3+ station flight deck arrangement. Physical and operational aspects of flight deck stationing are shown in Figures 2.5.1-1 and 2.5.1-2. The purpose of employing 2 rather than 3 or more flight crewmen is to minimize vehicle impact. The considerations involved in this selection are discussed under Crew Size Considerations, below.

The selected D&C hardware approach emphasizes conventional, dedicated hardware in the interest of minimizing hardware and software development costs, which contributed a substantial portion of total subsystem cost in earlier integrated configurations, and improving availability of the hardware needed for early vehicles. The majority of the principal instruments are used either as-is from past aircraft/spacecraft programs, or with the minimum modification to be usable in the Shuttle cockpit. In all D&C hardware, the credential is historically traceable to similar or parent models which have prior flight experience. In addition to reducing cost and improving availability, a further byproduct of the configuration selection has been to improve the credibility of hardware and software cost and schedule estimates, which should contribute to reduced program development risk.

The nature of the vehicle development phasing necessitates an evolution from an airplane-type cockpit for horizontal flight test to a full combined aircraft/spacecraft arrangement for the operational Orbiter, maintaining in the process an efficient cockpit arrangement and avoiding



"block-change" type impact of abandoning one hardware type and substituting another. This has been done in the selected configuration by using a building-block approach in which the items needed early are readily available, but replacement of these items in later kits is not mandatory.

2.5.2 Flight Deck Panels

The forward-facing portion of the cockpit is organized in a basically airplane-like arrangement, as shown in Figure 2.5.2-1. The forward console, center console and side panels contain those instruments and controls which must be accessible to the crewmen while they are restrained in the seats. This category includes all D&C used to operate the vehicle in ascent, thrusting, entry and aero flight modes, together with the provisions to effect failure switch-over and contingency reconfiguration for critical subsystems. Directly aft of the forward stations are vertical side bays whose panels are generally not accessible from the seats. These are suitable locations for lower-level and non-critical D&C that do not require immediate access or need not be operated when the crewmen are constrained to the seats. This category includes D&C used in post-insertion reconfiguration for on-orbit operations; reconfiguration to prepare for deorbit; low-level redundancy switching and troubleshooting; on-orbit housekeeping functions for the vehicle subsystems; and individual load circuit breakers. Further aft, space is provided for a docking/cargo handling station, where a single crewman can operate the manipulators using a combination of direct viewing through aft and overhead windows, and TV viewing. Provisions are included at the aft station to perform vehicle maneuvers in rotation and translation using visual and optical alignment.

Figure 2.5.2-2 shows the basic arrangement of one flight station at the forward console. The primary flight displays are organized in the conventional aircraft basic "T" located directly in front of the pilot. A 3-axis, all-attitude, flight director type ADI is employed for both aero and space flight modes. On either side of the ADI are the primary air data instruments which present barometric altitude, indicated airspeed, mach number and vertical speed information obtained via the central air data computers. Below the ADI is a conventional HSI which can present gyro heading, TACAN bearing and/or heading error, and DME distance readout. Supplementing the basic group are a cross-pointer indicator for two-axis flight path error information and an alphanumeric CRT display and keyboard for intercommunication between the crew and the GN&C computers. The primary display set is duplicated at the other forward station. The right-hand station only is additionally equipped with EDS displays, main engine gimbal position displays, and a two-axis g meter. Both stations have center sticks and pedals for aero surface control, as well as 3-axis side-arm controllers for space ACPS and TVC modes. In the center of the forward console and the center console are located backup flight instruments, which are driven from sources independent of the primary instruments; prime displays for the propulsion and hydraulic systems and power and environmental services; flight guidance and navigation mode controls; secondary flight controls for engine throttling, gear, brakes, drag chute, etc.; and the caution and warning readouts.

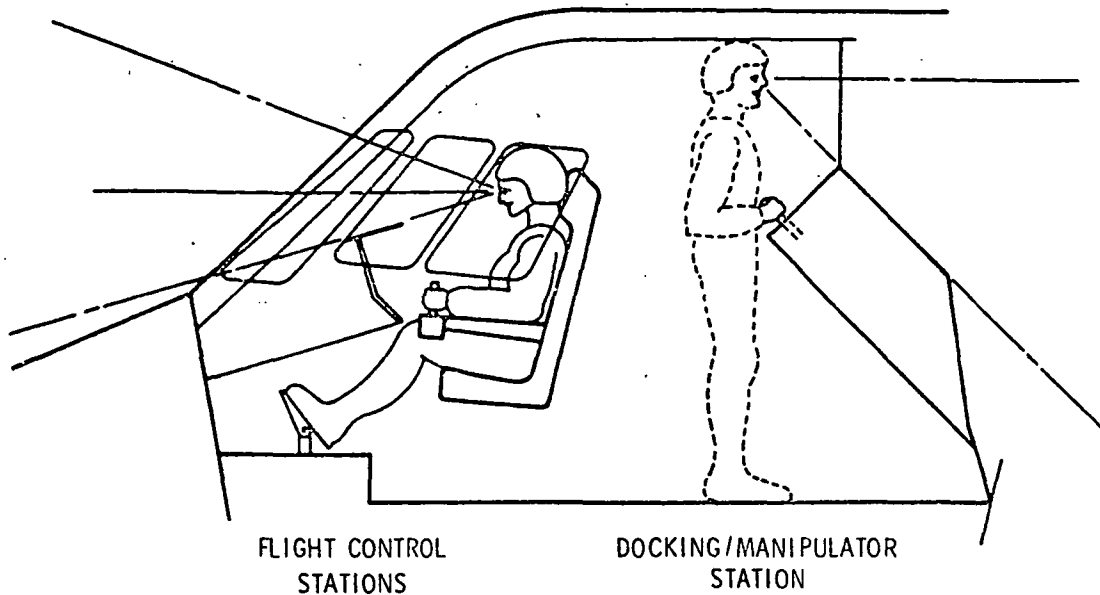


FIGURE 2.5.1-1 ORBITER FLIGHT DECK - SECTION

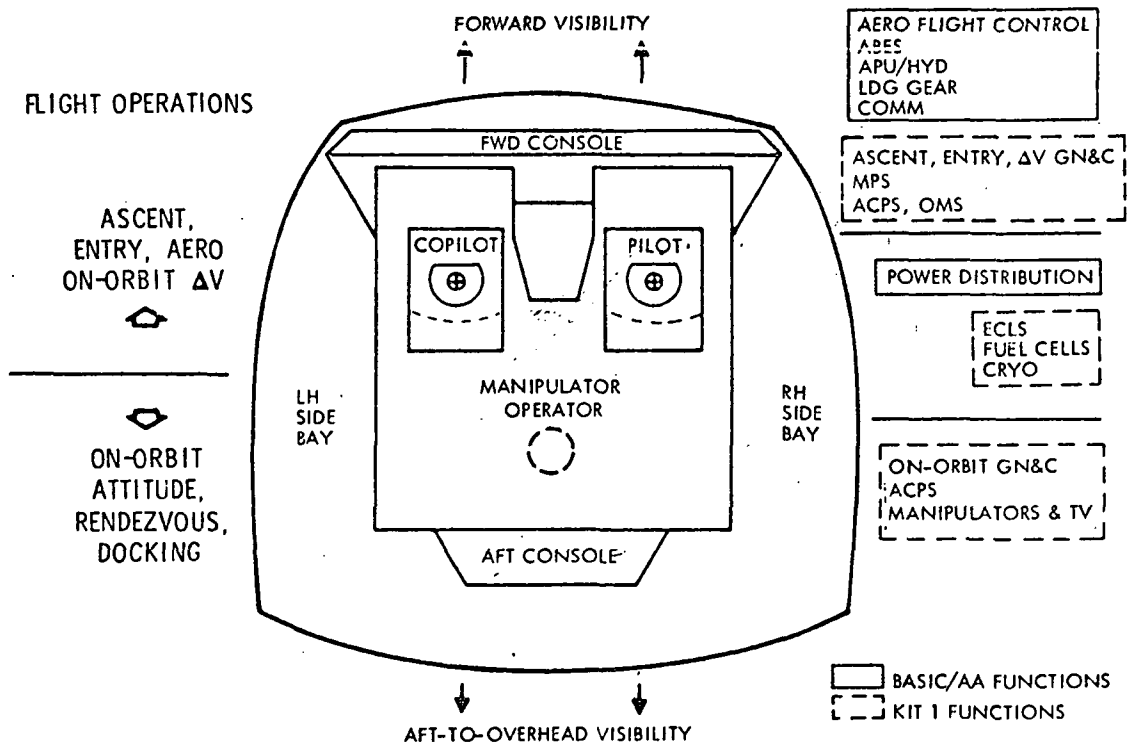


FIGURE 2.5.1-2 ORBITER FLIGHT DECK - PLAN

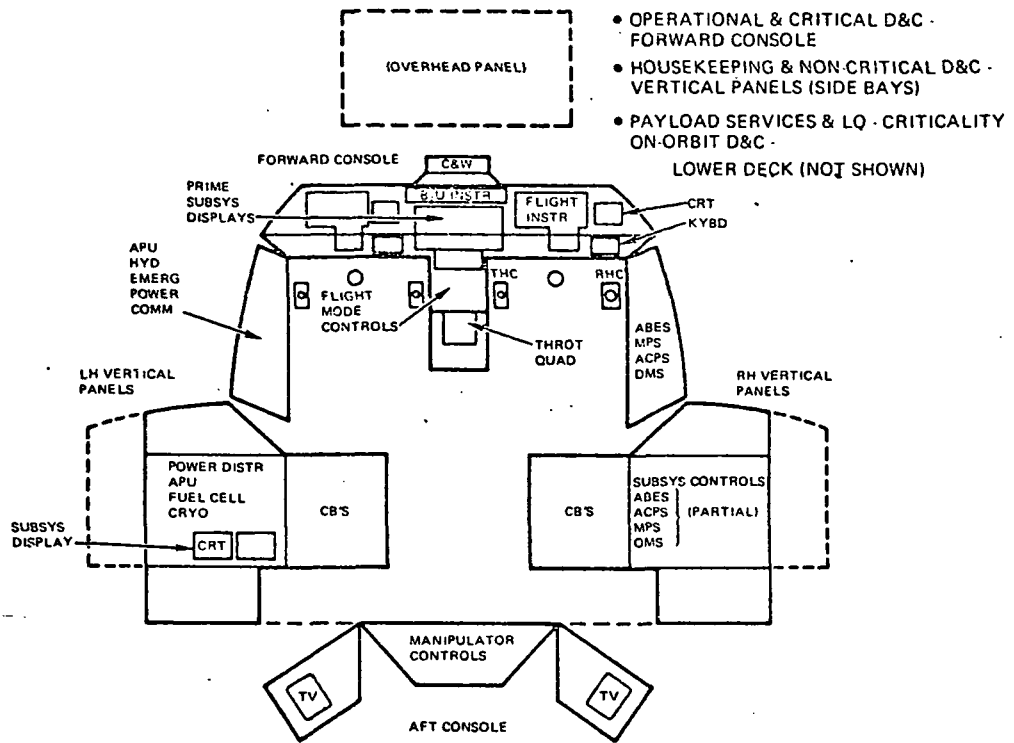


FIGURE 2.5.2-1 ORBITER PANEL ARRANGEMENT

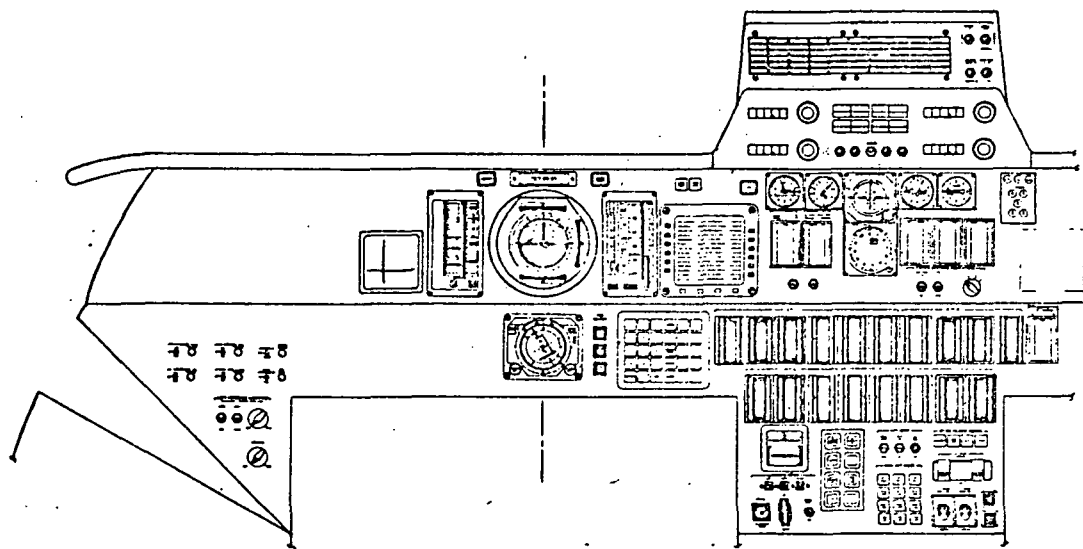


FIGURE 2.5.2-2 FORWARD DISPLAY CONSOLE



2.5.3 Crew Size Considerations

It has been a Shuttle goal to employ a flight crew not exceeding two men, and to have the flight controls and the cockpit organized such that the vehicle is flyable by one man from either prime station. Earlier studies have shown that this goal is achievable with a configuration employing integrated displays and providing automatic control for routine, repetitive functions.

With any significant configuration change, especially in the direction of increased use of conventional equipment or increased crew participation in vehicle/subsystem operation, this conclusion must be re-examined and revalidated. Both of these tendencies do, in fact, appear in the avionics configuration presented in this report. To assess their effect on crew size requirements, the basic necessities of vehicle and subsystem operation must be reconsidered by category and any significant differences identified. There are two basic issues that must be resolved: (1) access to required instruments and controls, and (2) crew workload. It has been established that, considering the combination of aircraft and spacecraft operations and subsystems involved, a fully conventional mechanization would require more panel space than could be seen or reached from the two prime stations. It is also established that with conventional instruments, monitoring and troubleshooting requires more scanning and switching and thus contributes to increased crew workload. The question to be resolved is whether these effects exceed tolerable levels or can be ameliorated through an appropriate cockpit design approach.

Considering first the vehicle control problem, precedent tells us that no significant access or workload problems should be encountered in flight control, guidance or navigation using a crew of two in a pilot/copilot relationship. Both prior aircraft and spacecraft have employed such an arrangement and have used conventional cockpit hardware extensively. The most significant change affecting the crew in the present configuration has been the deletion of closed-loop automatic guidance for the approach and landing operation. This effects normal-mission workload only, since in contingency the crewman had to be available to back up the autoland operation by manual takeover. Thus, no significant increase in peak worst-case workload has occurred.

The central issue, then, is whether (a) a third full-time crewman must be devoted to subsystem management, as is done with the Flight Engineer in transport aircraft, or (b) these duties can be time-shared by the same crewman who serves as pilot and copilot for vehicle control. Resolution of this issue involves the trade of the vehicle impact to add the third man (principally habitation, support and escape provisions) versus the avionics impact to support the two-man type of operation. Considering the vehicle cost involved in adding the third man, which is on the order of several tens of millions of dollars, one could afford a substantial avionics delta expenditure for crew aids, etc., to support two-man operation, and still realize a net overall program saving.



A preliminary assessment has been made of the potential access and workload problems that could occur with the present configuration in the two-man type of operation, with the following results:

1. The use of conventional instruments for subsystem display creates a potential access problem if all functions of all subsystems are considered. It is felt, however, that this problem can be averted by efficient arrangement of the cockpit, using an operational priority arrangement concept. In such a concept, information and control access that is needed for real-time support of vehicle control and maneuvering, for time-critical reconfiguration, and for major-functional-path-level failure switchover, is located in prime space, so as to be accessible from the seat. Non-critical monitoring and switching for lower-level redundancy management, detailed troubleshooting and on-orbit housekeeping type operations may then be relegated to less prime space where the crewman may be required to leave the seat and move about the flight deck to gain access to the D&C. A gross area evaluation has been made for this type of arrangement, and it is found that the approximately 150 analog and 160 discrete display functions and 250 control discrettes which must be accessible from the seat can be accommodated within the available area at the front stations, provided that a moderate amount of display time-sharing (approximately 2:1) is practiced. The remaining subsystem functions, consisting of some 150 additional analog and 160 discrete display functions, 300 control discrettes and 500 individual load-point circuit breakers, may be located elsewhere on the flight deck.

2. The significant changes that have occurred in crew workload in the subsystem management area are principally due to the increased scanning and switching operations needed to configure the subsystems, verify their operation and support decisions as to how to reconfigure for failure correction (troubleshooting). The magnitude of the workload increase has been assessed for the delta operations between the Phase B and the present configurations. Table 2.5.3-1 summarizes the results of this assessment; it appears that for a typical normal mission "busy" phase, from deorbit preparation through entry interface (4,00 K ft), the task load per crewman has increased from about 1/3 that of the Skylab CSM to roughly 1/2 that value. This is not considered to be excessive for the mission phase considered, since most of the effort falls in the deorbit preparation phase where the task scheduling can be adjusted to avoid time constraints. It is thus concluded that the workload increase is tolerable for the nominal mission case. For the contingency case, the workload is actually indeterminate for either configuration, since a great many combinations of failures and times of occurrence may be postulated. Therefore, one is not free to establish a specific sizing of contingency workload on a task basis, for feasibility evaluation. About the best that can be done is to minimize contingency workload as much as possible by mechanizing crew aids to simplify the process of finding and fixing troubles. It is essential that the requirement be avoided for constant attention or repetitive scanning to determine the existence of faults, since such an operation would necessarily divert the crewman's attention at times when it must be focussed specifically on vehicle operations. Basically, what is needed to avoid this problem is a device that will continuously monitor and check the condition and

COMPARISON CREW TASK - CRITICAL PHASE - APOLLO VS SHUTTLE
(PREPARING FOR AND PERFORMING THE DEORBIT MANEUVER - TO ENTRY)

APOLLO VS SHUTTLE

NOMINAL CREW WORKLOAD SUMMARY	# PHYSICAL TASKS	
	TOTAL	AVG/CREWMAN
APOLLO (SKYLAB)	606	202
SHUTTLE - PHASE B CONFIG.	132	66
SHUTTLE - PHASE B' INTERIM CONFIG.	204	102

*PHYSICAL TASKS - DISCRETE CONTROL MOVEMENTS (DID NOT CONSIDER
EQUIPMENT STOWAGE, ETC.)

TABLE 2.5.3-1 CREW WORKLOAD COMPARISONS

WHAT	HOW MECHANIZED	WHY
FLIGHT INSTRUMENTS & MANUAL FLIGHT CONTROLS	CONVENTIONAL, DEDICATED AIRCRAFT/APOLLO DERIVED	LOW DEVELOPMENT COST/ RISK MINIMUM SOFTWARE COMMIT. EARLY AVAILABILITY
GUIDANCE/NAV COMPUTER DATA READOUT	CRT ALPHANUMERIC DISPLAY	COMPACT, FLEXIBLE ENGLISH LANGUAGE COMMUNICATION LOW COST/CHARACTER TYPE IN PRODUCTION FOR COMMERCIAL A/C
PRIME SUBSYSTEM DISPLAYS	CONVENTIONAL, DEDICATED AIRCRAFT/APOLLO DERIVED IMPROVEMENT OPTION (MK II): REPLACE WITH INTEGRATED CRT GRAPHIC/ALPHA TYPES	LOW DEVELOPMENT COST/ RISK MINIMUM SOFTWARE COMMIT. IMPROVED FLEXIBILITY REDUCED PANEL SPACE, SCAN COST DELTA NOT SIGNIFICANT
NON-CRITICAL SUBSYSTEM DISPLAYS	CRT ALPHANUMERIC DISPLAY + PCM DATA SELECTOR/ PROCESSOR	COST, WEIGHT, PANEL SPACE MINIMUM SIGNAL WIRING LOW CHANGE COST
SUBSYSTEM CONTROLS	CONVENTIONAL, DEDICATED	IMMEDIATE ACCESS NO DEVELOPMENT

TABLE 2.5.4-1 D&C MECHANIZATION RATIONALE



performance of the subsystems, attract the crewman's attention if and only if there are identified off-nominal conditions, and identify the type and location of the problem for further assessment. In the integrated configuration this was done by a software package in a central computer which had access to all subsystem measurements and could verify that key measurements were within predetermined allowable limits. In the present configuration, dedicated hardware is employed to accomplish the function. Its mechanization and operation are described later in this section.

In summary, NR believes that the employment of a third full-time flight crewman for subsystem management is not mandatory, provided that (1) the cockpit is efficiently arranged by function priority according to access criticality; and (2) fault detection and isolation aids are mechanized to minimize both routine operations and contingency workload related to subsystem management.

2.5.4 D&C Mechanization

Table 2.5.4-1 summarizes, by category, the D&C mechanization concept and rationale utilized in the selected configuration. Certain of the displays selections resulted from detailed hardware trades which are summarized later in this section.

In the categories where conventional, dedicated hardware has been selected, the strong drivers are the early availability and low development cost of such hardware, together with the freedom from the substantial initial software investment which must be made prior to first flight. It is this initial software commitment factor which, together with hardware maturity considerations, leads us to select an initial configuration which uses dedicated conventional prime subsystem displays, with CRT graphic/alpha-numeric integrated types recommended as a later improvement option for downstream vehicles of the Mark II (operational) configuration.

Flight Displays and Manual Flight Controls

During this study, the use of conventional flight instruments has been evaluated both for cost impact and effects on crew interface. Where possible, existing displays from Apollo, IM, F-111, DC10 and other current-generation vehicles have been adapted for Shuttle. Surveys were made of existing display and interface equipment to compare physical characteristics and cost and evaluate the capability to convey the information required for manual vehicle control and monitoring of automatic vehicle control, as defined for the present configuration. Consideration was also given to using the crew for more vehicle control operations and reducing the number of normally automatic modes.

Typical candidate instruments from those surveyed were selected and were laid out in a nearly conventional manner for the two primary flight stations. Estimates were made of the impact of using conventional instruments on pilot operations and on the crewmen's capability to perform the piloting tasks. No quantitative study of pilot timelines using conventional



instruments has yet been done; however, estimates made during earlier studies had assured that the impact would be minor.

Control and display parameter requirements have been identified by mission phase and function assignments made to specific hardware.

Several alternative configurations have been examined which completely separate the aircraft and spacecraft modes of vehicle operation in separate flight stations. It was concluded from this examination that it is unwise to go to extremes in separating or dedicating aircraft and spacecraft operations, as unwarranted duplication of similar equipment must occur, and redundancy management becomes more complex when two crewmen cannot back each other up in a pilot/copilot relationship.

The specific mechanization selected for flight displays and manual flight controls is depicted in Figure 2.5.4-1, which shows hardware complement, interfaces with the flight subsystems, and functional redundancy. Items shown in solid boundaries are required for the basic vehicle (Aircraft Avionics), while those shown dashed are added in the kit for vertical/orbital flight. It will be noted that (1) no items are deleted or exchanged, and (2) all basic vehicle control and display hardware items are available in nearly off-the-shelf hardware.

The configuration and mode of employment of the hardware follows conventional practice, except for the ADI's and the computer keyboard/readout. A 3-axis, all-attitude ADI is used from the start to avoid the block change impact which would occur if a 2-axis aircraft unit were used initially and then later replaced with the all-attitude type for space modes. The third (yaw) total attitude axis of the ball can be locked up for aircraft modes to avoid confusion, since course/heading information is available separately via the HSI. The ADI and HSI gain a new digital interface when the GN&C computers are installed for vertical flight, to display inertial attitude, errors, course and heading. The total attitude information from the IMU's is obtained for display via the computers as shown, to avoid the complexities encountered in interfacing the 3-gimbal ball with 4-gimbal platforms via analog circuitry. The interface with the vertical/directional gyro packages for aircraft flight, and with the rate gyros, is analog.

The selected method of GN&C computer data readout is an interactive CRT alphanumeric display which functions in a manner similar to the unit defined for Phase B. A unit of this restricted capability is readily realized by modification of functionally similar commercial aircraft hardware or by build up from available modules employed in CRT graphic/alphanumeric displays presently used in the F-14 airplane. The latter approach appears more attractive both for minimizing cost and for retaining adequate flexibility to interface with presently undefined computer operating modes. Cost trades comparing the CRT and dedicated-individual-character approaches are shown later in this section. A full alphanumeric readout of about 200 to 250 character capacity is required for English-language readout of computer data. This form of data readout and entry is strongly recommended

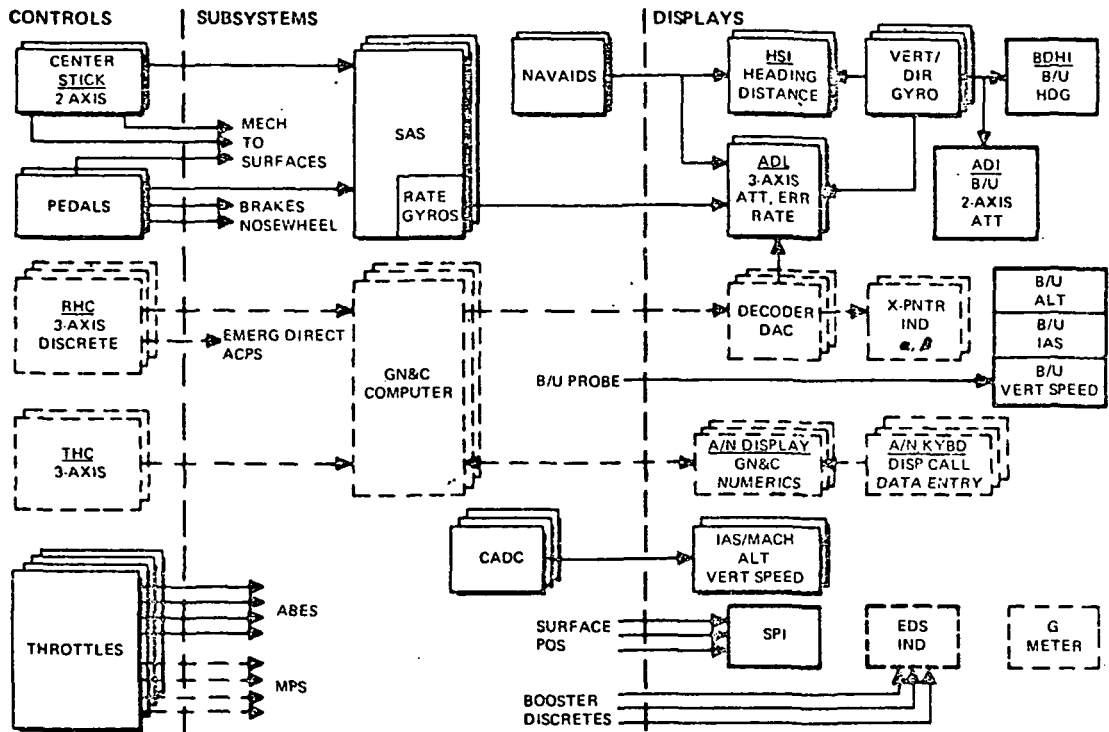


FIGURE 2.5.4-1

MECHANIZATION-FLIGHT DISPLAYS & MANUAL FLIGHT CONTROLS



in place of the program-verb-noun coded address technique used in prior spacecraft, to avoid the expenditure of operational time and training effort incurred by the complex numeric code language, and generally simplify the crew/computer interface. With present-generation digital techniques, the full alphanumeric type readout can actually be realized at a lower per-character cost than could the Apollo DSKY.

Display Hardware Trades

Table 2.5.4-2 summarizes display hardware selections and decision factors for the new baseline configuration and, for comparison, the old Phase B baseline. The selection rationale, by category, is as follows:

1. Flight Instruments. The integrated multifunction flight display approach employed in the Phase B baseline configuration represented one of the largest hardware and software development commitments in the cockpit, together with substantial cost/schedule risk. When realized, however, it promised reduced panel space requirements, low change cost, and the ability to back up other similar displays. The selected dedicated instruments, predominantly of conventional electromechanical types, definitely cost less and are easier to realize; in fact, all the principal types already exist and are flying. There is not presently a panel area constraint where these displays are located, though future growth opportunities are of course somewhat inhibited. Nor does potential change cost appear to be a problem; prime flight displays are historically an area of low change traffic. Also, interface redundancy requirements are presently such that two prime displays and a single independent backup will suffice. Thus, the use of conventional primary flight displays appears generally advantageous, and was selected. The remaining alternate, the dedicated electronic type, was rejected because the delta development cost to adapt present aircraft types to an effective all-attitude presentation for space use, operationally equivalent to a 3-axis all-attitude ball display, would be substantial.

	MULTIPURPOSE ELECTRONIC	DEDICATED ELECTRONIC	DEDICATED CONVENTIONAL
FLIGHT INSTRUMENTS	<p><u>FULL MULTIFORMAT (OLD BASELINE)</u></p> <ul style="list-style-type: none"> • SINGLE DESIGN-FLEXIBLE • MINIMUM CHANGE COST • REDUCED PANEL SPACE, SCAN 	<p><u>EADI, VSD, HSD</u></p> <ul style="list-style-type: none"> • IN CURRENT AIRCRAFT • LOW SOFTWARE COST • DELTA DEVELOPMENT FOR SPACE USE 	<p>✓ <u>ADI, HSI, ETC</u></p> <ul style="list-style-type: none"> • AVAILABLE-NO DEVELOP. • AVOIDS SOFTWARE • FLIGHT EXPERIENCE • PROVEN RELIABILITY
SUBSYSTEM DISPLAYS	<p><u>FULL MULTIFORMAT (OLD BASELINE)</u></p> <ul style="list-style-type: none"> • SINGLE DESIGN-FLEXIBLE • MORE MEASUREMENTS ACCESSIBLE • CAN TIME-SHARE WITH NAV <p>(OPTION)</p>	<p>✓ <u>LIMITED CRT</u></p> <ul style="list-style-type: none"> • IN PRODUCTION FOR AIRCRAFT • LOW INITIAL & CHANGE COSTS • REQUIRES SUPPORT SOFTWARE ONLY • EASY PCM INTERFACE • COMPACT 	<p>✓ <u>METERS, TAPES, ETC</u></p> <ul style="list-style-type: none"> • AVAILABLE • AVOIDS SOFTWARE • FLIGHT EXPERIENCE • PROVEN RELIABILITY
COMPUTER DATA READOUT	(NOT CONSIDERED)	<p>✓ <u>INTERACTIVE ALPHANUMERIC CRT (OLD BASELINE)</u></p> <ul style="list-style-type: none"> • IN PRODUCTION FOR AIRCRAFT • ENGLISH LANGUAGE • COMPACT • LOW PER-CHARACTER COST • REQUIRES SUPPORT SOFTWARE ONLY 	<p><u>INDIVIDUAL-CHARACTER ALPHANUMERIC (EL, INC, MECH)</u></p> <ul style="list-style-type: none"> • AVAILABLE • FLIGHT EXPERIENCE

TABLE 2.5.4-2 DISPLAY HARDWARE COMPARISON SUMMARY

CRT VS CONVENTIONAL DISPLAY HARDWARE

- ① ALPHANUMERIC READOUT OF COMPUTER DATA
- ② ANALOG BAR (GRAPHIC) OR ALPHANUMERIC PAGE PRESENTATION OF SUBSYSTEM DATA

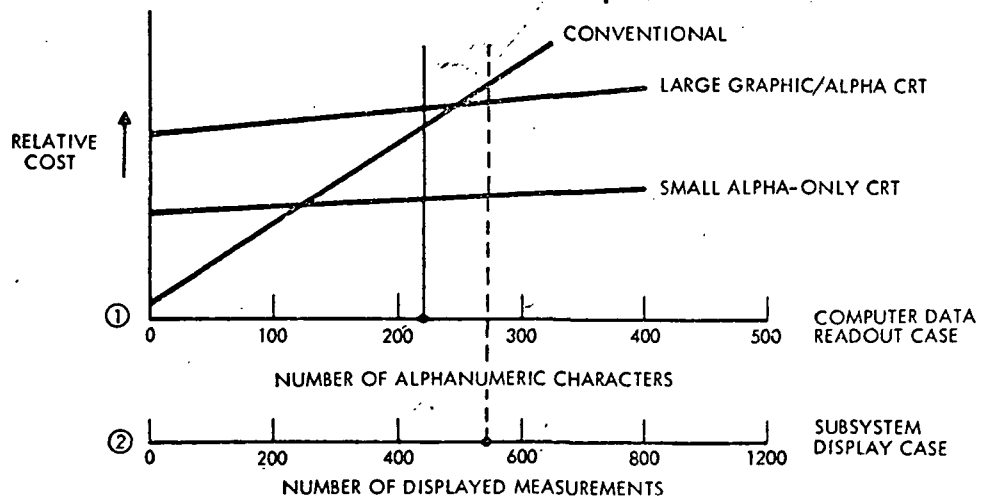


FIGURE 2.5.4-2 DISPLAY COST COMPARISON



2. Subsystem Displays. The integrated multiformat display approach employed in the Phase B baseline configuration used the same hardware for subsystem display as was discussed above for flight display; consequently, there are the same disadvantages. By contrast, a small, limited-capability alpha-only CRT is readily mechanized at low cost from available modules and avoids most of these disadvantages. However, without a graphic (multiple-bar or equivalent) format capability it is unsuitable for rapid-scan applications in the front cockpit, such as engine displays. It is also intrinsically simplex, and an attempt to make it redundant would propagate a comparable redundancy requirement back up the data stream into the PCM equipment. The limited CRT was therefore selected for presentation of detailed troubleshooting and housekeeping data, where scan time is not a constraint and the display function is not mission-critical, because it is backed up by the prime subsystem displays for the first failure. Conventional meter and servo tape displays were selected for prime display of operational and critical subsystem information, as these are the only readily available instrument types which offer an acceptable presentation for rapid-scan data.

3. Computer Data Readout. Cost, weight, and power trades conducted during Phase B verified that CRT and LED technologies have a competitive edge on arrays of individual-character readouts for computer data displays above 150 characters capacity. Of these technologies, the CRT is mature and flying while the LED is not. The CRT is thus a clear choice for minimum cost/risk and early availability, is flexible, and imposes no constraints on reading out data in easily understood English-language format. The selected mechanization, which is described in detail later in this section, avoids repetitive software programming and verification costs by using a one-time-prepared interpreter/assembler support software package on the ground, through which a keyboard operator may communicate new formats or format changes to a flight loader tape in a one-pass operation, without manual preparation of revisions to the flight software package.

4. Display Cost Comparison. Cost trade data compiled for both the computer readout and subsystem display cases are presented in Figure 2.5.4-2. The cost advantage of the limited CRT for the shuttle requirement is evident. Even the large graphic/alpha CRT (top line) is competitive for subsystem data display; it is consequently recommended as a subsystem display option. Since it not flying in similar service at present, however, it is considered insufficiently mature for early use and consequently the recommendation is that it be considered for later (Mark II) incorporation as an improvement option.

5. Subsystem Management Display Mechanization. The necessity for off-nominal annunciation for lower-level subsystem measurements was discussed in conjunction with crew workload, earlier in this section. This function, designated Performance Monitor, must be accompanied by display of the actual values of the same measurements in order to support reconfiguration decisions which are essential to management of available subsystem redundancy to achieve the design fault tolerance.



mechanization approaches for the processing and display of data to accomplish this task are shown in Figure 2.5.4-3. Approach #1 is the obvious brute-force solution of using more hardware of the type employed for prime subsystem analog displays and processing electronics. This is the most conventional approach; however, the disadvantage is that cost, weight, and panel space increase with the number of inputs. Approaches #2 and #3 consider that the data exists on board a CRT type display, used for computer data processing, and a digital PCM data stream containing all operational measurements, used for maintenance recording. These approaches then consider the forms of processing electronics that could be used to accomplish the remaining required operations. Approach #2 has the relative advantage that the digital hardware is customized; Approach #3 thus offers a closer to off-the-shelf configuration, as there are several low-cost microprocessors available that are adequate for the task.

Table 2.5.4-3 shows a comparison and ranking of the three approaches. One of the major reasons for rejecting Approach #1 is its low flexibility and consequent high change cost in a traditionally high change traffic environment. Of the remaining two, Approach #3 is preferred because it is more readily mechanized. This approach is described in further detail later in this section.

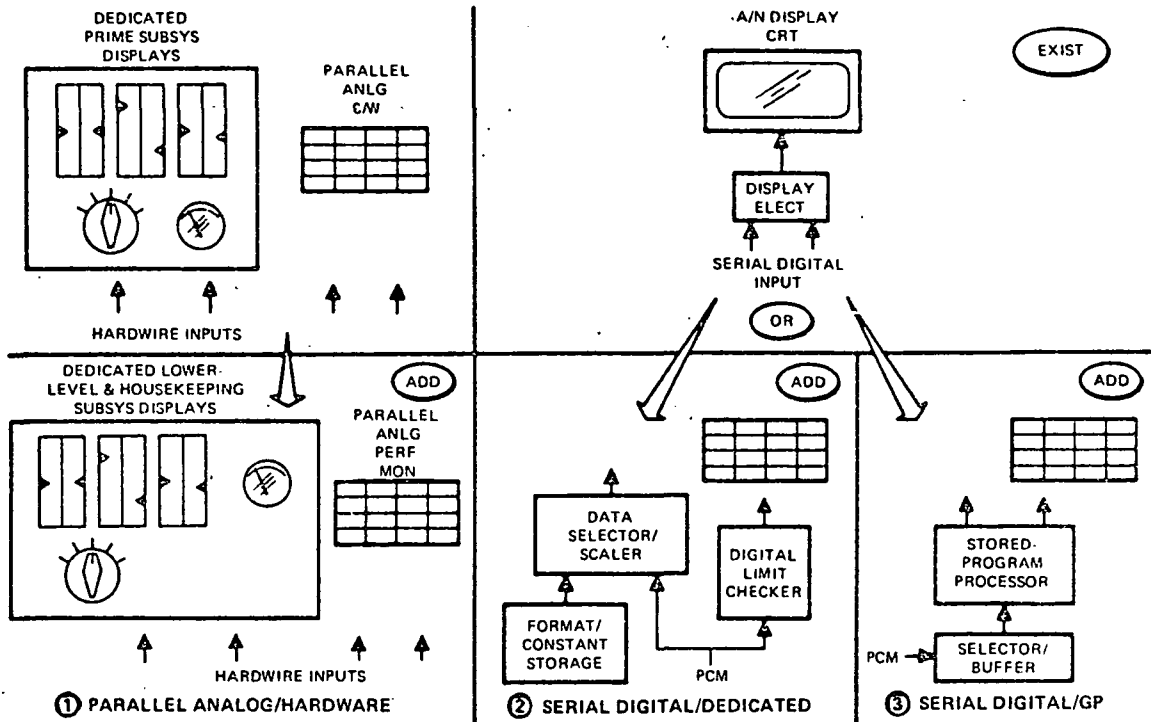


FIGURE 2.5.4-3 ALTERNATE SUBSYSTEM DISPLAY MECHANIZATIONS

	① PARALLEL ANLG/ HARDWARE	② SERIAL DIGITAL/ DEDICATED	③ SERIAL DIGITAL/ GP
INITIAL COST	COMPARABLE		
AVAILABILITY	BEST	GOOD	VERY GOOD
CHANGE COST	HIGHEST	LOW	LOW
FLEXIBILITY	POOR	GOOD	BEST
PANEL AREA	HIGHEST	LOW	LOW
WIRE WEIGHT	HIGHEST	LOW	LOW
POWER DEMAND	VERY LOW	LOW	MODERATE

TABLE 2.5.4-3 SUBSYSTEM DISPLAY COMPARISON



Alphanumeric Display/Keyboard Set

The Display/Keyboard set is made up of three units: a CRT display unit (DU), a companion display electronics unit (DEU), and a reformattable keyboard. The display function provides a page-organized, tabular presentation of stroke-written alphanumeric characters under control of an interfacing computer. Growth capability is provided for later implementation of vector graphic formats. The keyboard function provides display selection, display mode control, computer data entry and limited computer program control. The set can be interfaced with up to three GN&C computers or, by reformatting of the keyboard, with a subsystem display processor. The DEU provides the computer interface logic; up to 16 K - 16 bit words of reloadable display format storage; electronics for refresh buffering, symbol generation, positioning and writing for the CRT; and encoding/decoding of keyboard data and control discretes for the display and computer interfaces.

Display Unit (DU)

The unit employs a small (4 x 4 inches usable), high-brightness CRT to present up to 14 lines of 24 alphanumeric characters each. Located on the DU bezel are 12 line designator pushbuttons used in conjunction with the keyboard for display selection and data entry. The face of the DU, with a typical GN&C format displayed, is shown in Figure 2.5.4-4. The top line of the format is normally used for page titles and numbers; lines 1 through 12 for data names, data and units; and the bottom line is reserved for keyboard data entry and verification, and for computer error messages.

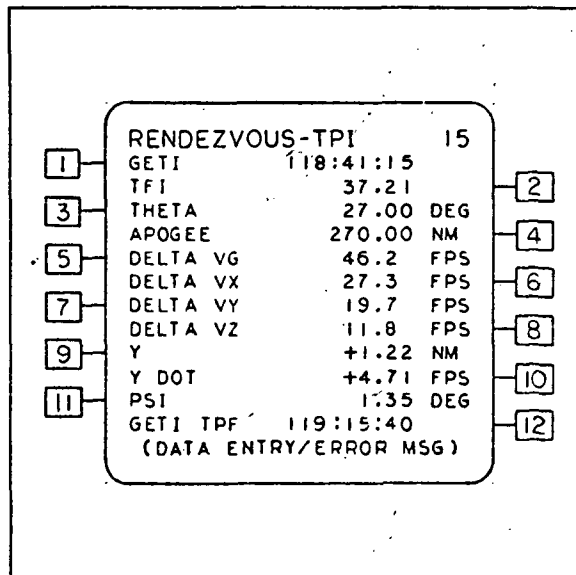


Figure 2.5.4-4. Alphanumeric Display CRT



Keyboard

A 6 x 5 key matrix is used to control the CRT display, select index and data pages, and communicate with the interfacing computer. The standard keys are zero through 9, +, -, and decimal point. The remaining function keys are reformattable for flight and subsystem display applications.

DEU

The DEU interfaces with the Keyboard, the CRT Display Unit, and from one to three computers, one at a time. It has its own core storage memory unit which acts mainly as a storage bin and refresh buffer for 35 or more tabular display format skeletons. All three interfacing computers can read from or write into the DEU memory, which is the method used for updating variable data in the display formats.

The DEU is capable of generating the entire alphanumeric set of characters plus special symbols. It can operate the display in single-character mode or in vertical or horizontal typewriter modes.

System Operation

The Display/Keyboard set operates in two basic modes, an index mode and a data mode. Index mode operation is initiated through actuation of an INDEX key on the keyboard; initially the top index page is called from a fixed location in DEU memory and displayed. Each line on this page represents either a subindex page or a data page. A one-of-twelve selection may then be made by pushing the line selector push-button opposite the desired page title. Alternatively, data pages may be called directly by page number from the keyboard by actuating a PAGE SELECT key, or may be paged through consecutively. When data page selection is made via the index tree, the display shifts to data mode as soon as a specific data page is selected. In the data mode, the DEU assembles a data page from a fixed skeleton called from a selected DEU memory location and variable data requested from the selected interfacing computer. The request to the computer is identified by data page number, which is decoded in the computer to initiate routines which select and convert constants from memory and results of on-going computations, and assemble this data in line order for transmission to the DEU. This cycle is repeated twice per second for update. The page number is sent with each update block and compared with the called page number in the DEU to assure data/format skeleton compatibility. By a combination of field masking techniques and special instructions, the average alphanumeric data page skeleton occupies less than 120 16-bit words in DEU memory and requires approximately 30 16-bit words of data from the computer for a full page update.

Actuation of the numeric and sign keys on the keyboard when in the data mode is interpreted as a data entry to the computer. The data goes into a buffer line in DEU memory and comes up on the bottom scratchpad line on the display as it is entered; the interfacing computer does not take cognizance of the data until it has been inspected, its disposition identified



by a discrete from one of the display line selector buttons, and the ENTER key actuated. The computer then evaluates the entered data for acceptability. If the data is acceptable, the computer clears the scratchpad line and substitutes the entered value for the value in its memory, causing the same number to appear in the selected data line on the next update. If the data is unacceptable (not a program constant, wrong value, no disposition, etc.) the computer sends a discrete to light an OPERATOR ERROR annunciator and an error code to initiate display of an error message on the scratchpad line. If during the execution of programs the computer encounters problems, it can light a PROGRAM ALARM annunciator and send an error code to initiate display of an error message on the scratchpad line. The keyboard has priority in that error messages will not be forced-displayed while keyboard entry is in process, but will come up if the operator clears the scratchpad line. Error messages are stored in the DEU memory and addressed by conversion of error codes into memory locations.

Hardware Functional Description

Logic organization of the system electronics, together with computer, keyboard and display interfaces is shown in Figure 2.5.4-5.

Computer Interface

The DEU is capable of interfacing with any of three computers, one at a time. The DEU, therefore, contains the driver and receiver circuits required to send and accept signals from 3 computers. The selection of the one active computer interface is based upon discrete signals received in the DEU from a panel control switch. The computer selection logic samples the selection control discrettes and electrically ties the active computer interface to the remaining DEU logic.

Operation of a function key on the keyboard, such as enter, end, proceed, clear, etc., causes the DEU to request the computer to initiate a command or a data transfer from the DEU to the computer. The computer will first issue a read-I/O-address command which causes the DEU to transfer the memory address where the last encoded key action is stored. (Keyboard entries, including line/column select, are stored sequentially in a fixed portion of the DEU core memory.) By performing a subtraction, the computer then initiates read commands to the DEU beginning at the start of keyboard entry storage up to the last entry which initiated the transfer. The computer then issues a write command to reset the keyboard entry table to blanks. This has the effect of clearing line 13. The computer then operates upon the operator request.

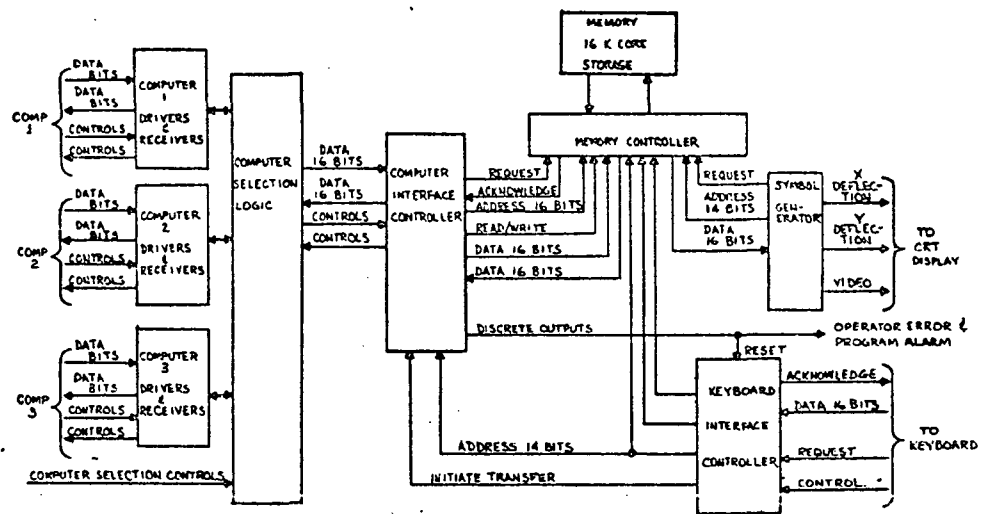


FIGURE 2.5.4-5 DISPLAY ELECTRONICS



The Computer Interface Controller (CIC) contains the logic required to communicate with the selected computer. It is capable of responding to read or write commands issued by the computer. It is also capable of generating an interrupt to the computer when commanded by the Keyboard Interface Controller (KIC) to initiate a transfer. The CIC also contains a discrete register which is settable and resettable under computer control. Two of the discretely light indicator lamps called Operator Error and Program Alarm, another discrete is used to reset the KIC address counter.

Keyboard Interface

The Keyboard Interface Controller (KIC) receives parallel data from the keyboard which is encoded to define which of the 42 keys on the keyboard or display unit has been depressed. The keyboard initiates data transfers by raising a request line to indicate that a key has been pressed. The KIC acknowledges the keyboard input and then stores the data in the DEU memory. The KIC discerns two types of data, that which is to go to the computer and that which is to be displayed first on the scratch pad line on the CRT display. The KIC processes these data types differently including routing them to different buffer areas in memory.

For computer-destined data, the KIC notifies the CIC which interrupts the computer and requests a read of the desired data. An area in the DEU memory is reserved for keyboard service data. For this data, the KIC does a special formatting operation for processing and display on the data entry (scratchpad) line. If the Enter key is depressed, indicating that the scratchpad line should be sent to the computer, the computer reads the scratchpad locations in DEU memory and then clears them. This has the effect of transferring the data from the scratchpad line to the selected data line. The scratchpad line can also be cleared from the keyboard.

Memory Controller

The Memory Controller controls the data flow to and from memory. It receives requests to perform memory access operations from three internal sub-units -- the Computer Interface Controller, the Keyboard Interface Controller, and the Symbol Generator. Since the three sub-units compete for memory usage, the Memory Controller resolves the conflicts on a priority basis as follows:

- . Symbol Generator - Highest Priority
- . Keyboard - Second Priority
- . Computer - Third Priority

At any given time, the Memory Controller selects the highest priority request and gates it through to the memory while it "holds-off" any other requests.



Memory Unit

The DEU Memory Unit is composed of 16K 16-bit words of core storage. This storage is partitioned into four areas - Display Refresh Buffer, Variable Data Storage, Scratchpad Storage and Keyboard Storage.

The Display Refresh Buffer contains, in sequence, the stored display formats. These are stored in the form of program lists composed of 16 bit words which define symbol type and placement. The selected format is displayed on the CRT. The selection of format for display purposes is accomplished by a branch instruction stored in location zero which points to the proper format. All formats end with a branch instruction that points to the scratch pad memory locations so that the scratch pad line is always interrogated regardless of which format is being displayed.

One section of memory is used to store variable data parameters that are part of the display formats. These parameters are stored in a present sequence by the computer. These locations can be made an integral part of the refresh buffer for constructing the display. This is accomplished by branching from the main refresh program to the variable data area for a preset number of words and then branching back to the main program.

The scratchpad area of memory contains the starting X and Y positions of the scratchpad line on the display plus storage locations for character codes to be written on the scratchpad line.

A buffer storage area is provided for keyboard data which is to be stored until interrogated by the computer.

Symbol Generator

The Symbol Generator is comprised of digital and analog circuitry which controls the presentation of display formats. It is used to generate alphanumeric characters. The refresh and generation of the symbols are controlled by a list of instruction words which are stored in the DEU core storage. The digital instruction words are processed by converting them to the x-deflection, y-deflection, and z-axis video signals required to drive the CRT display.

The Symbol Generator provides the following display capabilities:

- . Generate alphanumeric characters from stroke information stored in the DEU memory.
- . Format characters in type mode or random positions.
- . Growth capability for drawing vectors of any length.
- . Position characters at any location on a 512 by 512 matrix.



- Present a uniform display by controlling the intensity of the characters (and vectors).
- Operate in a typewriter mode associated with either the X or Y direction.

Subsystem Display Processing

Processing of data for display on the subsystem CRT and for lower-level subsystem performance and operation monitoring is accomplished by a Subsystem Display Processor set which includes a modular, serial-output, address-programmable PCM data selector/buffer, a small stored-program processor CPU-I/O, and lamp driving logic. The equipment interfaces with the composite PCM stream, with an alphanumeric display/keyboard set which has been loaded with format skeletons for subsystem display, and with a number of fault status and agree/disagree annunciators located on the subsystem panels. There is also an optional interface with the vehicle Caution & Warning system for use in sleep periods or at times when the subsystem panels are unattended. Functional organization of the equipment is shown in Figure 2.5.4-6.

PCM Select/Buffer. This element accepts a Manchester biphasic coded data stream at rates up to 100 KB/S, strips sync from the stream to provide frame and word sync and bit clock, and sequentially selects, buffers and transmits RZ-coded logic-level data to the processor. Both the incoming PCM address and the outgoing data are buffered for at least one PCM word interval in order that contiguous words in the stream may be selected.

Processor CPU - I/O. The processor is a low-cost general-purpose, high-speed, core-memory machine which basically executes four tasks: data selection, display scaling, data limit checking and system self-test. The memory is modular up to 16K, 16-bit words. The processor retains in its reloadable memory the PCM address tables for selection of up to 1,000 measurements; the denormalizing constant tables for scaling of up to 1,000 measurements for display, together with the display address tables for up to 50 12-line tabular alphanumeric CRT display pages; the pre-determined operating limits for limit-checking and logic comparison of up to 600 analog and discrete measurements, together with annunciator address tables for up to 150 fault status annunciators; and the program to accomplish the four tasks defined above. The limit-checking operation may apply up to three selectable sets of limits to monitored measurements, depending on subsystem configuration and/or mission phase.

Fault Status Annunciator Logic. This electronics accepts and decodes outputs words from the CPU to determine which of the 150 fault status annunciators should be activated. A provision is incorporated to hold annunciators activated until acknowledged by the crew. The decoded and processed discrettes are outputted to the annunciators via lamp drivers to provide the required power switching.

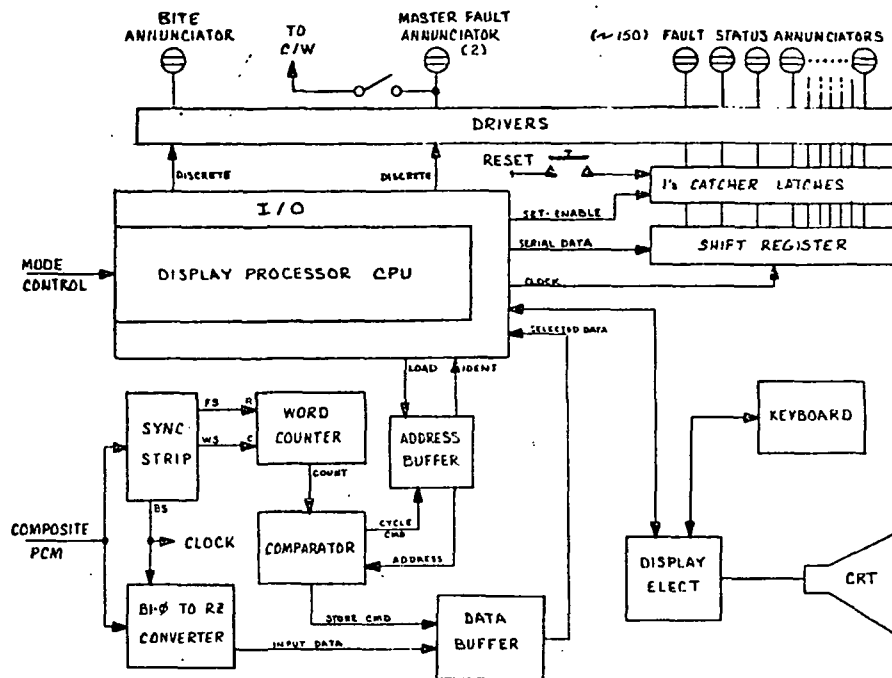


Figure 2.5.4-6. Subsystem Display Processing

System Operation: CRT displays are requested by index tree or page number via the keyboard, as described in the previous section. In either case the page number code is decoded by the processor to access the appropriate denormalizing constant and display address tables for the selected page. The processor then addresses and selects the requested data, converts the data and the PCM frame time into BCD-coded engineering units format, assembles it in display line order and transmits it to the display set as described previously. This operation is repeated an integral number of PCM frames later for update; the number is selectable to achieve a nominal twice-per-second update with either real-time or tape-delayed non-real-time input data.

Measurement selection for limit-checking will be established by a predefined PCM address table. The processor will select from its memory the appropriate limit and status tables for the selected operational phase and subsystem configuration, and will ordinarily make a limit check or a state comparison for all selected measurements. However, if a known off-nominal measurement repeatedly triggers the alarm, the processor can be instructed via the keyboard to disregard the condition of that measurement until it is notified to the contrary. The processor will maintain a table of inhibited measurements for display upon request.

Limit checking can be done at repetition rates up to about 10 per second. However, the processor will normally be instructed to verify that an off-nominal condition persists for a number of consecutive data samples



(2 to 5 for most measurements) before outputting an alarm discrete. When an off-nominal condition is verified, the processor will output discrettes to the annunciator logic identifying the appropriate individual fault status annunciator, and to the master fault annunciator, and will write the number of the display page containing the offending measurement(s) on the scratch-pad line. If the appropriate display page is up, or when it is brought up, the processor will identify the data it has evaluated as off-nominal by requesting a "bug" symbol on the appropriate display line which will flag the measurement(s) as out-of-tolerance high or low. A performance monitor status table will be maintained, to be displayed in page format on request, listing in chronological order the measurements which have been abnormal since the table was last cleared and indicating what that status was (high, low, off, failed, etc.).

In each major cycle of processing for display and performance monitor, the Processor executes a routine which checks the data, the system hardware, and itself. A BITE annunciator is provided which is normally held off by a pulse train from the Processor I/O. However, should the PCM data fail, or a check of calibration words or PCM BITE discrettes from the data stream indicate "no-go", or the processor self-test routine indicate "no-go", or the processor power fail, the output pulse train will cease and the BITE annunciator will illuminate. If the processor itself is still healthy, it will send an error code to the display set to initiate display of an error message identifying the problem.



2.6 COMMUNICATIONS & TRACKING

The orbiter Communications and Tracking Subsystem defined in the earlier Phase B study was reevaluated under the Alternate Avionics System study, considering the phased development program and emphasizing maximum use of unmodified off-the-shelf equipment.

2.6.1 Requirements

To satisfy mission objectives, the Space Shuttle Communications subsystem will provide:

1. Voice intercom between crew, ground, space station, payload, and booster.
2. RF two-way duplex voice between orbiter, MSFN, and space station.
3. Simplex voice between orbiter and ATC stations.
4. Data between orbiter and MSFN, booster, payload, and space station.
5. Capability of receiving and retransmitting ranging signals to MSFN.
6. Range between orbiter and cooperative targets (space station, payload).
7. Identification and altitude information to ATC ground tracking stations.
8. RF atmospheric nav aids for range and bearing determination.

These requirements and the overall communication and tracking approach are characterized in Figure 2.6.1-1.

2.6.2 Study Results - Summary

The major study results discussed in Section 2.6.4, are summarized in Table 2.6.2-1, which identifies the individual functions studied and shows the alternates considered. The selected configurations shown were chosen from studies considering the following factors:

1. Voice and data service requirements of basic operations do not exceed the contact afforded by the NASA ground tracking network as currently planned. A relay satellite link is not required.
2. Limitation of new development, and the adequacy of MSFN tracking, preclude need for a long range PRS system. The use of the horizon sensor provides independence from global ground networks without dependence upon relay satellites.

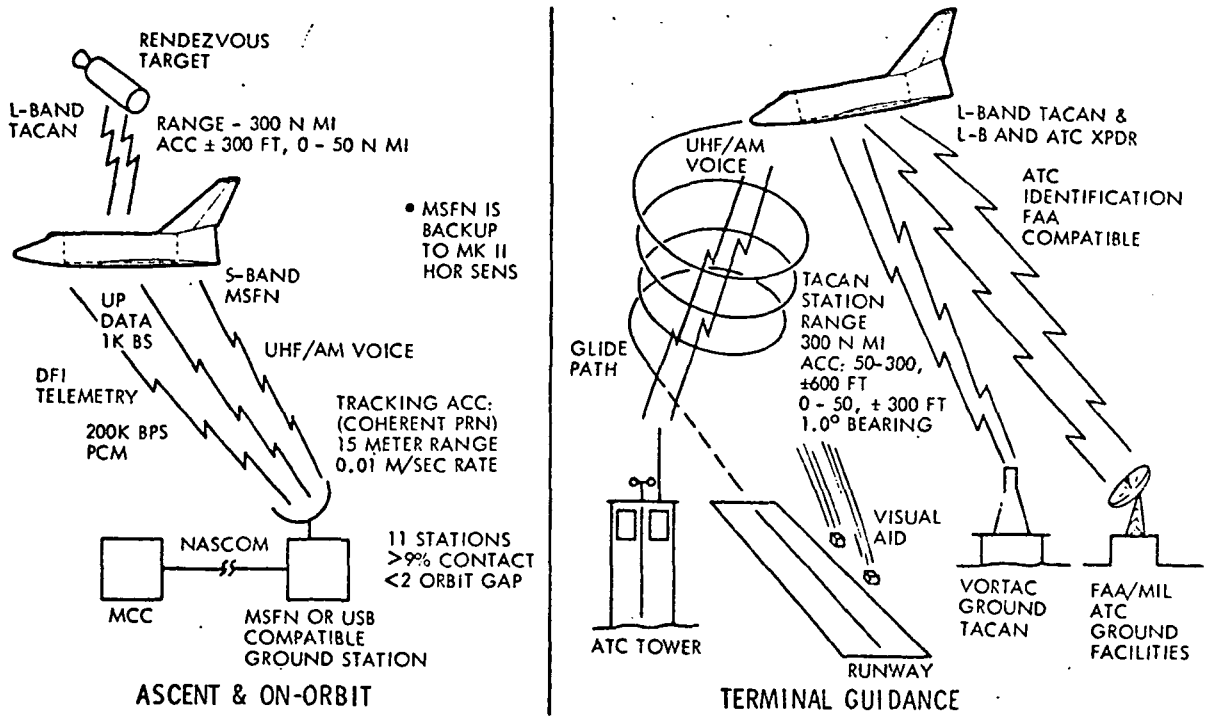


Figure 2.6.1-1. Communication & Tracking Summary

Table 2.6.2-1. Communications & Tracking - Study Summary

FUNCTION	DESCRIPTION	SELECTED	B ¹ ALTERNATES
VOICE CONTACT	MONITOR - LAUNCH - CONSTANT ON-ORBIT - 9% TIME ATMOS - 90% TIME	MSFN - USB MSFN - USB UHF/AM ATC	RELAY SATELLITE VHF ATC
STATE VECTOR DETERMINATION MK I MK II	$R, \dot{R} \approx 1/\text{ORBIT}$ R, \dot{R} AS REQUIRED	MSFN HORIZON SENSOR MSFN BACKUP	RELAY SATELLITE PREC RANGE SYSTEM
STATE VECTOR TRANSMISSION	112 DECIMAL CHARACTERS OR 448 BINARY BITS	UDL DIGITAL DECODER (15 SEC) (VOICE BACKUP)	VOICE (3.5 MIN. NOM)
APPROACH & LANDING G&N UPDATE	POST-BLACKOUT TO 2000 FT ON-BOARD DERIVED	TACAN	VOR/DME, MSFN, PRS
AIR SPACE CONTROL	POSITIVE AIR TRAFFIC CONTROL	ATC TRANSPONDER	CLEARED AIRWAY
LANDING	VFR, 2000 FT CEILING MANUAL	VISUAL AIDS FOR GLIDESLOPE	ILS, GCA, PRS
RENDEZVOUS	RANGE 300 N MI ACQUISITION	TACAN-AIR/AIR MODE	VHF/AM, USB, PRS



3. The updata link serves digital data delivery during flight since the crew and onboard capabilities preclude the need for continuous ground command. Use of uplink command is planned for ground safing area operations when the orbiter is unmanned.

4. Standard military TACAN provides required accuracies to update the inertial navigator without MSFN or PRS installations at landing areas.

5. Provision of the ATC transponder supports horizontal test, ferry operations, and positive air traffic control during operational landings.

6. The vehicle is designed for manual landing under 2000 ft ceiling conditions. No IFR aids are required for landing pending further mission definition. VASI glideslope lights aid VFR landing.

7. The use of TACAN for cooperative rendezvous provides required accuracies without added equipment.

The Communications and Tracking will support the two phase, MARK I and MARK II, program approach in the following manner:

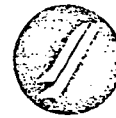
MARK I Implementation. For ascent and orbital operations, an 11-station MSFN service provides communication and state vector update. For descent and landing approach, TACAN updates the inertial navigator to within 300 feet of the desired position at 2000 feet breakout to VFR terminal approach. Visual glideslope lights (VASI) and pilot observation are employed in manual landing. UHF voice communications are provided to MSFN stations and to landing sites during the atmospheric flight modes. Cooperative rendezvous employs the standard air-to-air mode of TACAN, calling for an 11-pound TACAN transponder on the target vehicle. The L-band ATC transponder identifies the orbiter to Air Traffic Control to assist in air space control.

MARK II Implementation. During MARK II flights, the horizon-sensor will have been added for use with the star tracker to derive autonomous navigation data. MSFN will be used for communication and for emergency tracking. Antennas will be modified to accommodate the MARK II thermal protection change from ablative to reuseable covering.

2.6.3 Selected Communications and Tracking Subsystem Configuration

The derived configuration, as shown in Figure 2.6.3-1, employs equipment used on space program and/or on commercial and military airplanes, and will support the requirements defined for the Communications and Tracking subsystem.

Voice intercom between the crew, ground personnel, space station, payload, and space shuttle booster originates and terminates in the crewmen's personal communications assemblies. These assemblies are connected to the Audio Center which provides amplification, switching, and routing of voice for intravehicular use, and transmission and reception via USB and UHF/AM transceivers.



In addition to the two-way duplex voice, the USBE provides data (DFI, OFI, payload status - 5Kbs) and turnaround ranging for state vector update.

The UHF/AM provides two-way simplex voice communication with the air traffic control stations and also serves as a backup mode with the MSFN stations.

The Signal Processor provides the necessary interface connection between the airborne data gathering equipment and the RF electronics. It also accomplishes signal modulation, demodulation, mixing, routing, switching and matching.

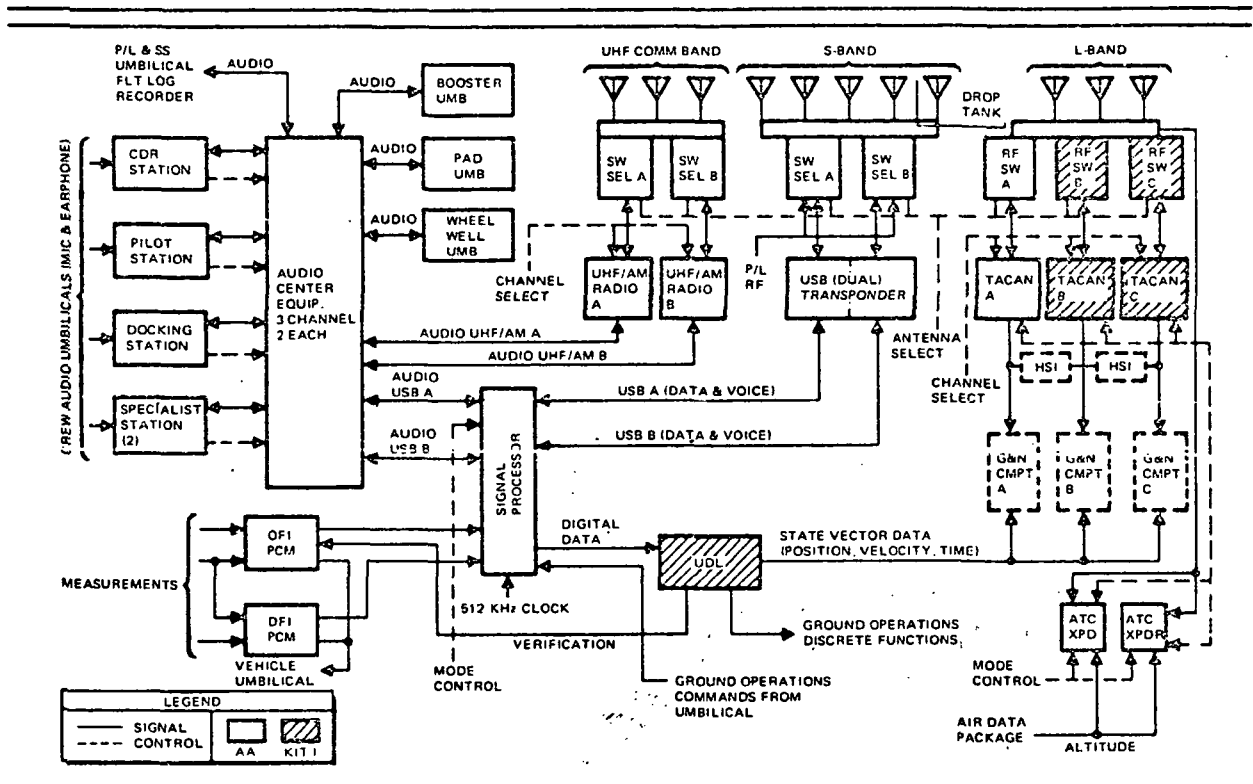


Figure 2.6.3-1. Communications & Tracking Subsystem Configuration

A saving in the subsystem cost and complexity was achieved through the utilization of the TACAN equipment for multi-functional purposes. First, the subsystem is used in an "Air-to-Ground" mode as an atmospheric nav aid for bearing and range determination from preselected ground stations. Second, in an "Air-to-Air" mode acquisition, tracking, and ranging will be accomplished with cooperative targets equipped with a compatible transponder.

An Air Traffic Control transponder is implemented for transmission of identification and altitude information, upon interrogation by the ATC ground tracking facilities.



Maximum operational autonomy of the communication equipment is assured, through the capability of manual control from the Display and Control stations. To minimize crew activities, however, the subsystems are equipped with RF switch selectors for automatic antenna transfer. Manual override capabilities of the RF antenna selectors is also incorporated.

The TACAN triple redundancy is in support of the 3-string GN&C subsystem for which it is a critical element during the descent and landing phase. The updata link and signal processor are single units as in Apollo, since their functions are backed up by voice transmission of data and direct carrier modulation of the USBE respectively.

The L-band antenna serves the TACAN and the ATC transponder, each of which provide inhibit pulses to preclude simultaneous transmissions. A fifth S-band antenna is mounted on the drop tank for coverage prior to separation.

The following section describes the subsystem studies conducted to establish the configuration described above. These studies include: On-orbit Communications, Ranging, Navigation Aids, and Antennas.

2.6.4 Subsystem Studies

On Orbit Communication

Figure 2.6.4-1 presents communications coverage for a 100 n.mi. circular orbit at 28.5° inclination using eleven MSFN stations. Overall communication coverage by mission is depicted showing the longest gaps and percent earth coverage with a typical days contact time at a 100 n.mi, 28.5° orbit. The data shown is based on eleven MSFN stations ground net, similar to Skylab, minus the tracking ship anchored off Buenos-Aires. Satellite communications by TDRS is not shown as broader coverage is not currently required by the shuttle on-orbit. To determine the need for a digital data link (up-data link or UDL) voice transcripts from Apollo missions were reviewed and it was determined that a nominal time of 3.5 minutes was required for read-up and read-down of state vector data by voice. This causes a consequently great reduction in available voice contact time to perform other services. With the UDL, voice service is concurrent with state vector transmission. The UDL is utilized for state vector transmission on-orbit and offers remote command capability during unmanned ground operations (safing, propellant loading).

Eleven station MSFN coverage will support the mission requirements for state vector update (three revs prior to reentry on MARK I and MSFN back-up for MARK II) and voice and data contact.

On Orbit Ranging

The results of a trade study for the selection of an on-orbit ranging subsystem are summarized in Table 2.6.4-1. In addition to the weight and significant cost savings, subsystem complexity reduction is achieved through the use of TACAN.



MODE	ORBIT	FULL MSFN SERVICE % EARTH COVERAGE		LONGEST GAP		MSFN STNS	
				REVS	TIME (HR)		
PRIMARY UPDATA BY UDL	100 N MI, 90°	9	UPDATA	2.0	2.9	HSK	MIL
	100 N MI, 28°	9	VOICE	1.5	2.2	CRO	BDA
	270 N MI, 57°	30	SIMUL	1.2	2.0	GWM	ACN
BACKUP UPDATA BY VOICE	100 N MI, 90°	4	UPDATA	5.0	7.3	HAW	CYI
	100 N MI, 28°	5	SHARES VOICE	3.0	4.4	GDS	MAD
	270 N MI, 57°	28		1.2	2.0	TEX	

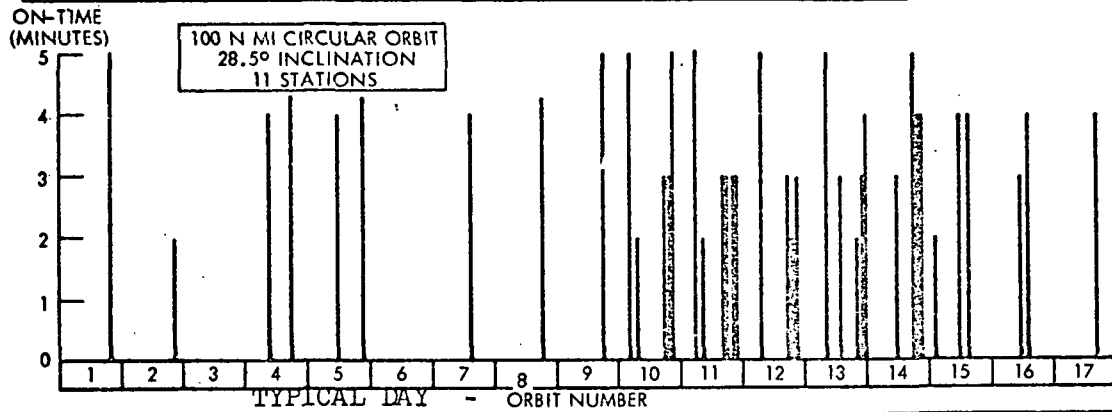


Figure 2.6.4-1. Communication Coverage

Table 2.6.4-1. Rendezvous Ranging

	VHF/AM + DRG (APOLLO)	PRS CR 100 - 1 (USAF)	✓ TACAN (MILITARY)	REMARKS
NO. LRU/SHIPSET	6	2		TACAN ON-BOARD FOR ATM NAVIGATION
WEIGHT/SHIPSET	41.7 LB	50 LB		TRANSMISSION LINES, D&C, C/O NOT INCLUDED
COST/SHIPSET	\$600K	\$200K		COST OF EQUIPMENT ONLY
RANGE, MAXIMUM	327 N MI	200 N MI	300 N MI	ASSUMED REQU RMAX = 250 N MI MANDATORY RSTART = 30 N MI
RANGE, MINIMUM	1600 FT	50 FT	1000 FT	REQ RMIN = 2000 FT
RANGE ACCURACY 1 σ AT 30 N MI	+300 FT	= 15 ft	±600 FT ANAL ± 300 FT TO G&N	REQ ACCURACY +600 FT UP TO 0.33 N MI



The use of military qualified TACAN for atmospheric navigation (RF Aids for Reentry and Approach) offers the advantage of a built-in air-to-air mode providing + 300 ft range accuracies for rendezvous ranging comparable to the Apollo VHF/AM unit and without the additional equipment or antennas. Ground interference is automatically precluded since the mode employs different frequency channels and pulse separations. An 11-pound transponder (available military equipment) aboard the target permits ranging at over 300 n.mi range. A full TACAN (25-pounds) aboard the target (or another orbiter) permits simultaneous mutual ranging with display in both vehicles. TACAN/star tracker performance supports ΔV conservation requirements.

RF Aids For Reentry and Approach

An analysis was conducted to evaluate RF aids for entry and approach. The study was based on the defined requirements for entry updates at a minimum altitude of 100K feet, and bearing and range determination from preselected ground stations during atmospheric flight.

The TACAN with its multifunctional capability, combined with its established performance and operational history was found to be superior to other subsystems under consideration. The primary candidates for atmospheric navigation (terminal guidance) update sensors are the current widely used VOR/DME of commercial and general aviation, the Cubic Corp. CR100-1 used in the military CIRIS implementation and the widely used military TACAN (see Table 2.6.4-2). TACAN was chosen as the terminal guidance nav aid because of its small antenna size (an L-band antenna which is shared with ATC transponder), probable production base in shuttle time frame, wide dispersion of compatible ground stations (CONUS and NATO) and additional performance features offered such as air-to-air ranging, digital output and current usage with military aircraft inertial navigator sets.

An investigation was conducted to determine the probability of simultaneous interrogation of multiple TACAN ground stations during the terminal guidance phase. Assuming "blackout" terminates between 150K ft and 120K ft altitude, the vehicle is approximately 300 n.mi from the landing sites. In this area of 283,000 square n.mi, only two stations approximately 550 n.mi apart use the same vortac frequencies (Charleston and Key West on Channel 82). Twenty-five other stations are available with dedicated frequencies. No interference from ground TACAN stations is expected during the terminal guidance phase.

The terminal guidance characteristics of the vehicle were examined in terms of altitude, velocity, and range requirements. The MARK II vehicle characteristics were utilized as a worse case condition (MARK I velocities and ranges are not as severe due to shorter cross range). Based on the terminal guidance simulations (Honeywell Trade Study #11) updates are desired to commence starting at 100K ft altitude or above. This dictates a minimum nav aid capability of 210 n.mi slant range and 3000 ft/sec range with accuracies at least as good as VOR/DME. Comparisons of available nav aids resulted in selection of TACAN as the update sensor. The model for



TACAN selected for this study was the Hoffman RT-870/ARN 91 currently in production and selected as the RT for the U. S. Army AN/ARN-103. Performance characteristics of the selected model for TACAN are depicted on the descent and landing profile as shown in Figure 2.6.4-2.

Two curves are plotted. One velocity, the second altitude, as a function of distance to the landing site. The TACAN acquisition slant range of 300 n.mi exceeds the required 210 n.mi short range.

Table 2.6.4-2. Atmospheric Navigation Equipment

CANDIDATES:			
SELECTED →	VOR-DME TACAN PRS (CR 100-1)	COMMERCIAL/FAA MILITARY (VORTAC) MILITARY	L-BAND & VHF L-BAND ONLY S-BAND
CRITERION	VOR/DME	PRS	TACAN
USE STATUS	OFF-THE-SHELF RCVR DEVELOP VHF ANT.	NEW (1971)	OFF-THE-SHELF MILITARY STDS
GROUND EQUIPMENT	GLOBAL	REQUIRES TRANSPONDERS	CONUS + MILITARY
RANGE	380 N MI	200 N MI	300 N MI
ACCURACY 1σ	±600 FT	±60 FT	±300 FT
WEIGHT EACH	37 LB	25 LB	25 LB
COST Δ	+\$150K	+1.0M	--

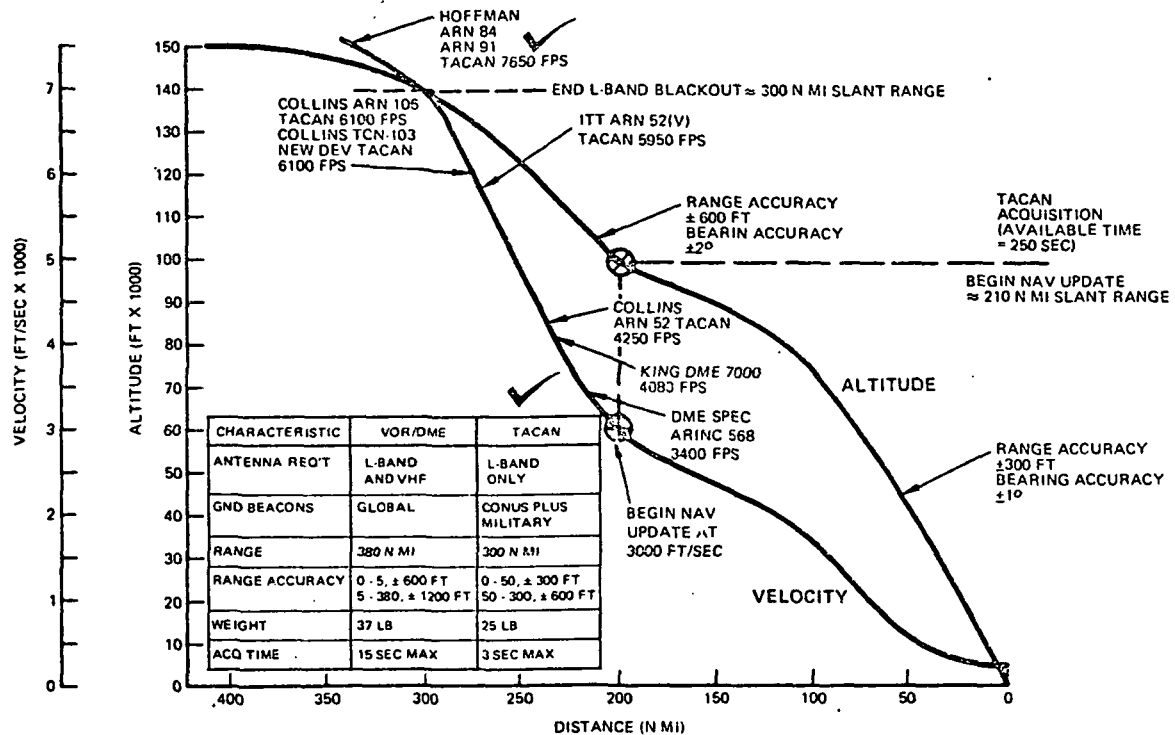


Figure 2.6.4-2. TACAN Characteristics - Descent & Landing

Antennas - Type and Coverage

Figure 2.6.4-3 shows the orbiter antenna locations selected to satisfy the required radiation coverage. The expected temperatures and vehicle structure compatibility were major drivers in the determination of antenna locations.

Typical Antenna Installation

The preliminary configurations of the antennas are based upon the most stringent transmission problem, that of operating with an ablative shield as is currently considered for the MARK I orbiter. Limited data now available indicates that SLA-220 material can be molded with the prime ablator, SLA-561, forming an ablating window with satisfactory loss tangents even when hot. Further studies and tests were planned as further data is released by the government on these particular materials. Figure 2.6.4-4 depicts a typical installation of the flush mounted antennas and lists the preliminary selection of antennas for the orbiter.

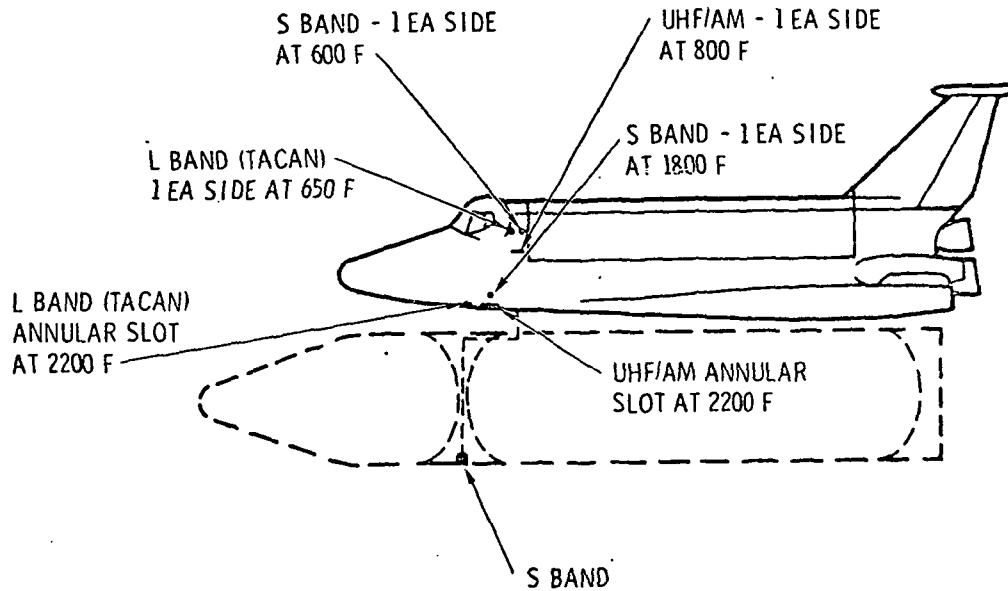
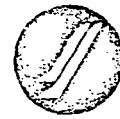
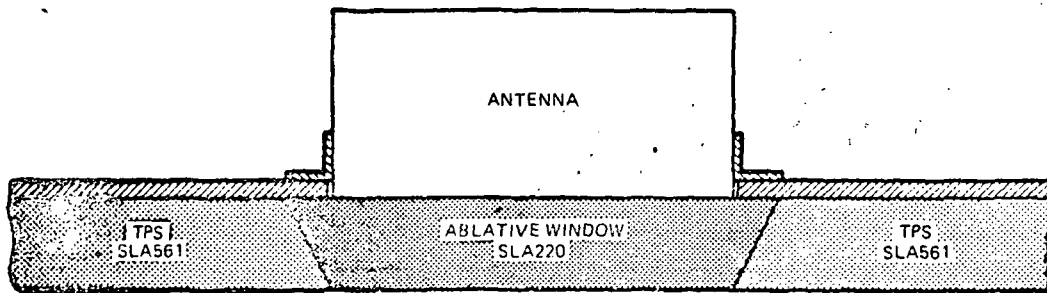


Figure 2.6.4-3. Orbiter Antenna Locations



PRELIMINARY SELECTIONS

ANTENNA	FREQ	QUAN	TYPE	POLAR	PATTERN COVERAGE	DIEL IN CAVITY	DIEL WINDOW
UHF/AM	225 TO 400 MHz	2	HORIZ SLOT WITH CAVITY	VERT	120° SOLID ANGLE LEFT & RIGHT SIDES	HD-0092 BORON NITRIDE	SLA220
		1	ANNULAR SLOT (BOTTOM)	VERT	OMNI DIR IN AZIMUTH FOR NEG ELEV ANGLES 0° TO -80°	HD-0092 BORON NITRIDE	SLA220
L BAND (TACAN)	960 TO 1215 MHz	2	HELIX IN CAVITY	RHCP	140° SOLID ANGLE, L&R SIDES, UPPER FWD	SINTERED QUARTZ	SLA220
		1	ANNULAR SLOT (BOTTOM)	VERT	OMNI DIR IN AZIMUTH FOR NEG ELEV ANGLES 0° TO -80°	HD-0092 BORON NITRIDE	SLA220
S BAND (USBE)	2.0-2.3 GHz	4	HELIX IN CAVITY	RHCP	OMNI DIR, WITH REDUCED GAIN ±20° FORE & AFT	SINTERED QUARTZ	SLA220

Figure 2.6.4-4. Typical Installation



UHF Annular Slot Antenna

The annular slot antenna provides the same (vertical) polarization and pattern coverage as the vertical monopole or "blade" antennas installed on the top or bottom fuselage surfaces of commercial or military aircraft. This antenna has become the "basic vertically polarized element" for installations requiring flush mounting.

In recent years, ablative materials have been developed for efficient operation under reentry conditions while providing excellent r-f transmission for microwave frequencies. Silicone ablaters, such as the Martin-Marietta type SLA-220, form predominately siliceous chars; whereas, ordinary ablaters based on epoxy and phenolic resins form carbon chars that preclude signal transmission. SLA-220 provides a high ablative efficiency that compares favorably with the type SLA-561 material currently specified for the adjacent TPS covering for the orbiter. The main problem that can be foreseen for ablative materials is the current lack of test data to support qualification for the relatively long thermal cycle for orbiter reentry.

UHF Horizontal Slot Antenna

Voice communication or orbit will normally be conducted via the unified S-band equipment (USBE), but the UHF/AM equipment will be available on a back-up basis. This added capability will be provided by a pair of vertically-polarized cavity-backed slot antennas on the left and right sides of the fuselage. The main function of these antennas will be to provide coverage above the orbiter, thereby complementing the annular slot, which will cover only the lower hemisphere. The estimated radiation coverage for the UHF antenna subsystem is shown in Figure 2.6.4-5.

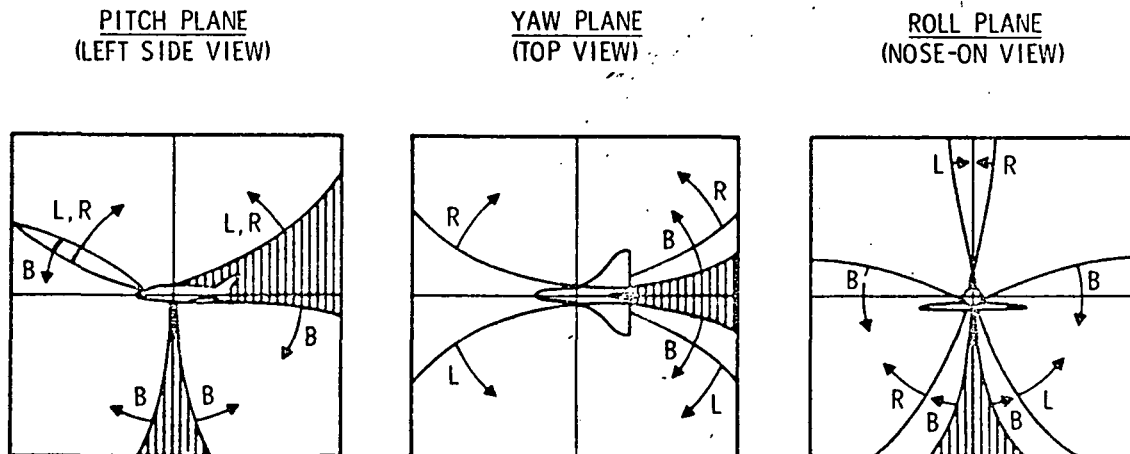


Figure 2.6.4-5. Estimated Antenna Pattern Coverage - UHF Band - 3 Reqd



L-Band Annular Slot Antenna

TACAN navigation during approach and landing requires vertical polarization with omnidirectional coverage in azimuth, so that most of the discussion under "UHF" will also apply to this case. Again, the annular slot is the only flush-mounted type antenna capable of providing this coverage. The L-band high-temperature window design will be much less difficult, however, due to the reduced diameter of seven inches. The lack of coverage directly to the rear is not a serious matter for TACAN operation.

L-Band Helix in Cavity Antenna

TACAN is used for ranging during rendezvous in orbit. The required pattern coverage above the orbiter will be obtained by means of a pair of antennas on the left and right sides of the fuselage. The location will be above the UHF antenna on each side.

Circular polarization has been specified to permit operation at any relative orientation of the TACAN-equipped vehicles. The antenna selected for this application will consist of a scaled-up version of the Apollo S-band helix in a quartz-filled cavity. The cost of scaling a proven design will be much less than the development cost of any other type antenna.

The rendezvous operation will require navigation toward another vehicle located in the forward quadrant above the orbiter. The corresponding pattern coverage, labelled "L" and "R" on Figure 2.6.4-6 will be obtained by a tilted mounting arrangement for the cavities, with the axis of each helix inclined forward about 30-degrees from the perpendicular to the fuselage surface. This technique has already been employed successfully with S-band antennas. The estimated L-band radiation coverage is shown in Figure 2.6.4-6.

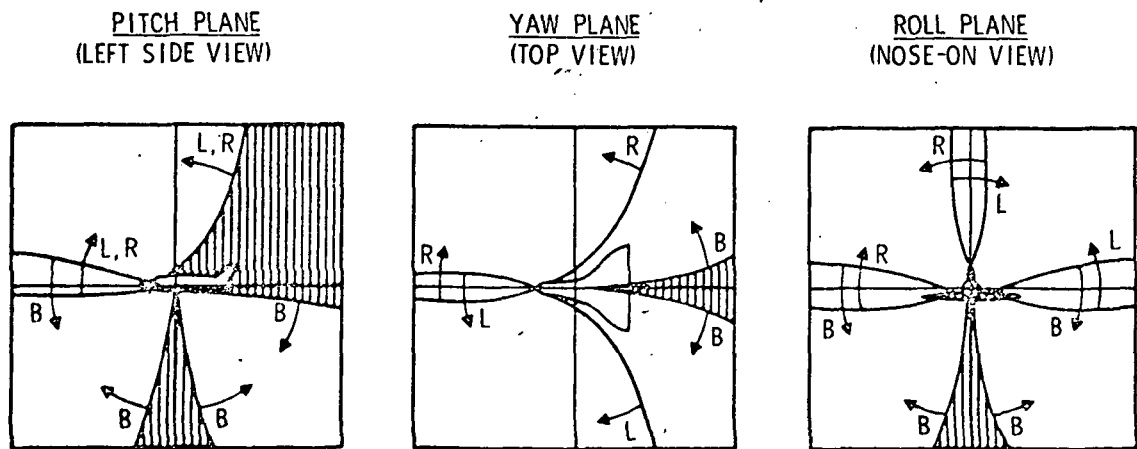


Figure 2.6.4-6. Estimated Antenna Pattern Coverage - L Band 3 Req



S-Band Helix in Cavity Antenna

Four Apollo S-band antennas will be located at 90-degree increments around the orbiter fuselage for communication on orbit as shown in Figure 2.6.4-7. The regions within 20 degrees of the roll axis, both fore and aft, will experience reduced gain with some nulls. If further studies indicate the need for improvement of the forward coverage, this can be readily achieved by tilting the antennas forward.

Because of the requirement for USBE operation during the launch and boost phases, a study was conducted to determine the pattern blockage effect of the external tank attached to the orbiter. It was clear that the 30-degree sector directly below the orbiter would be illuminated only by energy diffracted past the tank. Since resulting reduced gain and nulls in the pattern could not be tolerated prior to tank separation, a fifth S-band antenna will be located on the bottom of the tank housing, fed by coaxial line as part of the forward orbiter/tank umbilical.

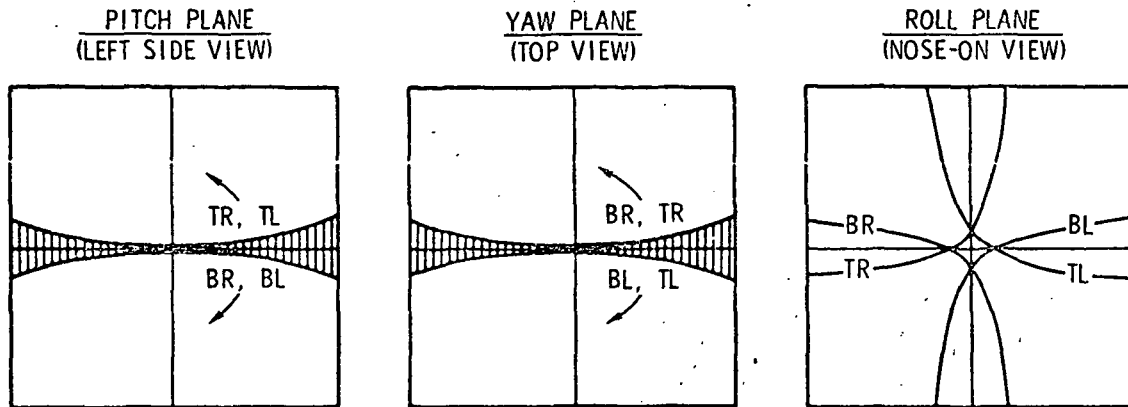


Figure 2.6.4-7. Estimated Antenna Pattern Coverage - S Band - 4 Req'd After Tank Separation



2.7 INSTRUMENTATION

2.7.1 Instrumentation Requirements

The instrumentation subsystem is required to sense and acquire data to be displayed, telemetered, or recorded for vehicle checkout, in-flight performance monitoring, analysis of subsystem malfunctions or failures, and vehicle qualification.

Real time data is required for display and/or telemetry. Transmission of PCM data is required on both horizontal and vertical flights and must be compatible with the Manned Space Flight Network (MSFN). Critical data for display will be hardwired to the display panel. Other display data is derived from the PCM system.

Data acquisition is required for two hours on ferry flight or horizontal test flights. On orbit flights, acquisition of data is required for 7 days. Telemetry and recording full coverage capability is required during boost, de-orbit, and landing. During orbit, telemetry and recorded coverage will be required only on a "snapshot" basis such as during docking or engine burns.

Transducers required for data acquisition will be distributed throughout the vehicle. All other instrumentation equipment (signal conditioning, PCM equipment, recorders, frequency multiplexers, and timing equipment will be located in either the aft cargo bay (aft of Aft Cargo Bay bulkhead), or avionics bay. The equipment must be capable of withstanding, or be protected from, the following estimated environments.

	<u>Temperature</u>	<u>Vibration</u>
Avionics Bay	30°F to 175°F	1g ² /Hz
Aft Cargo Bay	30°F to 350°F	2.6 g ² /Hz
Fuselage (Interior)	-240°F to 350°F	2.6 g ² /Hz
Surface	3440° F	

Redundancy will be provided on a black box level to prevent catastrophic loss of data. Some critical measurements will be taken with redundant sensors and routed through separate signal conditioning packages (if dedicated conditioning is required) and through separate PCM remote units.

Checkout of instrumentation will be accomplished by carry-on test equipment, built-in test equipment (BITE), ground readout of PCM, and playback of onboard recordings.



All measurement systems fall into one of the following three categories:

1. Measurement systems in which sensors and signal conditioning is an integral part of that subsystem. In such cases, that subsystem will directly interface with the instrumentation subsystem PCM remote unit, a recorder, or be directly hardwired to the display panel.
2. Measurement systems in which the transducer, only, is an integral part of the subsystem. Interface with the instrumentation subsystem will be either with the dedicated signal conditioning units or the PCM remote units.
3. Measurement systems in which transducers and signal conditioning is not a part of the using subsystem.

2.7.2 Alternate Concepts

In the selection of a system concept, commonality of hardware was given maximum consideration. By standardization of ranges, mounting provisions, etc., installation design is minimized, checkout and documentation is simplified and logistic spares requirements are reduced. This becomes cost effective in both manpower and hardware costs. By using the same components in both OFI and DFI, standardization is further enhanced.

Use of identical hardware for phased buildup of horizontal and vertical flight instrumentation requirements helps reduce overall costs as hardware used in horizontal need not be removed. Additional components are installed for the increased measurement requirements of vertical flights. There is no need for new horizontal flight instrumentation and therefore additional installation drawings for horizontal flight are not required.

Maximum usage of off-the-shelf hardware was considered in selecting a system concept. Minor modifications to developed hardware such as expanded capacity and repackaging of developed circuitry to conform to specific shuttle requirements will be necessary in signal conditioning and PCM equipment.

Of the several instrumentation subsystems considered, two approaches were selected for final consideration, a dedicated system in which the OFI and DFI are separate functions, and an integrated system in which PCM and recording functions are shared by OFI and DFI.

In the dedicated system approach there is a total separation of OFI and DFI function. Separate PCM data streams are generated for OFI and DFI, and separate recorders are used for PCM recording. Common



hardware is used to the maximum extent possible. Signal conditioning and PCM equipment is the same for both OFI and DFI. There is additional redundancy of data. A catastrophic failure of either OFI or DFI will not affect the other system.

A block diagram of the dedicated instrumentation subsystem is shown in Figure 2.7.2-1.

In the integrated approach, the signal conditioning and PCM remote units are separated for the OFI and DFI, with the PCM master unit and PCM recorder common. The block diagram of the integrated approach is shown in Figure 2.7.2-2. The system is similar to the dedicated system with the deletion of the two DFI PCM master units and the DFI PCM recorder. The OFI master unit now programs and formats all PCM data (OFI and DFI) into one serial digital stream.

Some advantages to the dedicated approach include lower data rates for each PCM stream, redundancy of key data is allowed through separate PCM systems, and the OFI PCM data format will be allowed to remain relatively constant for the entire program. Most measurement changes are anticipated in the DFI system. Disadvantages, of course, include the additional power, weight, and volume of the additional hardware.

Some advantages of the integrated approach are less hardware, with consequent reduction of power, weight, and volume, and a simplified interface with ground equipment (only one PCM data stream). Disadvantages are higher data rates (complicates data storage) and changes in DFI measurement requirements has the potential of affecting the OFI if the PCM is re-programmed. In addition, the data stream will be different during the operational phase after the DFI has been removed.

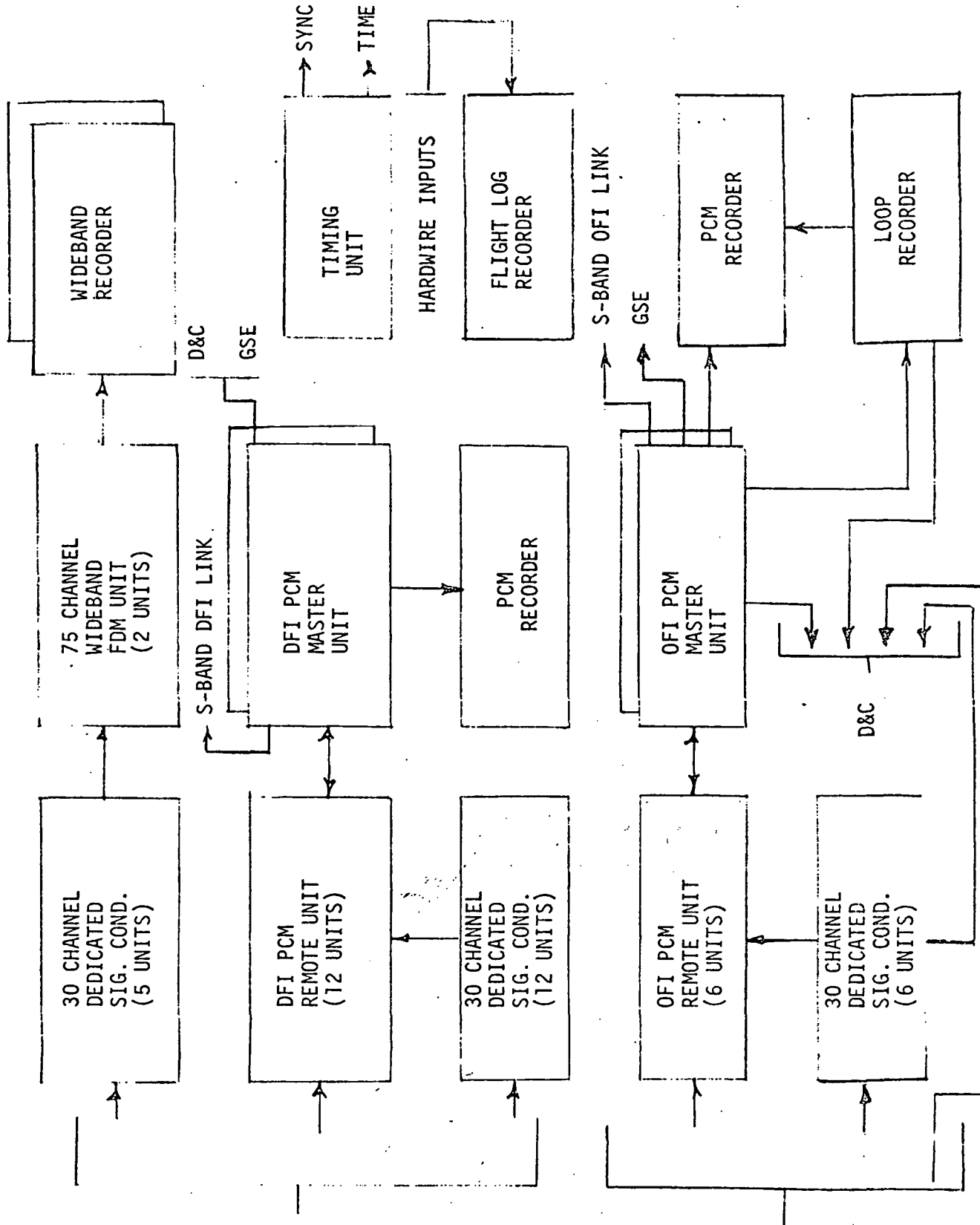
The dedicated OFI and DFI system was selected as the baseline system using the following rationale:

1. Since scar weight must be a minimum, only the PCM master units and PCM recorders could be integrated.
2. Development and operational data used somewhat differently. The separation of the two data streams allows them to be handled differently.
3. Lower data rates eases the recording requirements and allows the operational system sizing to be constant throughout the program.
4. Dedicated systems allows different power sources to be used in the OFI and DFI. Since DFI will be removed during the operational phase, it should not be a factor in sizing the operational power generation system.



DEDICATED OPERATIONAL &
DEVELOPMENT FLIGHT INSTRUMENTATION SUBSYSTEM

Figure 2.7.2-1



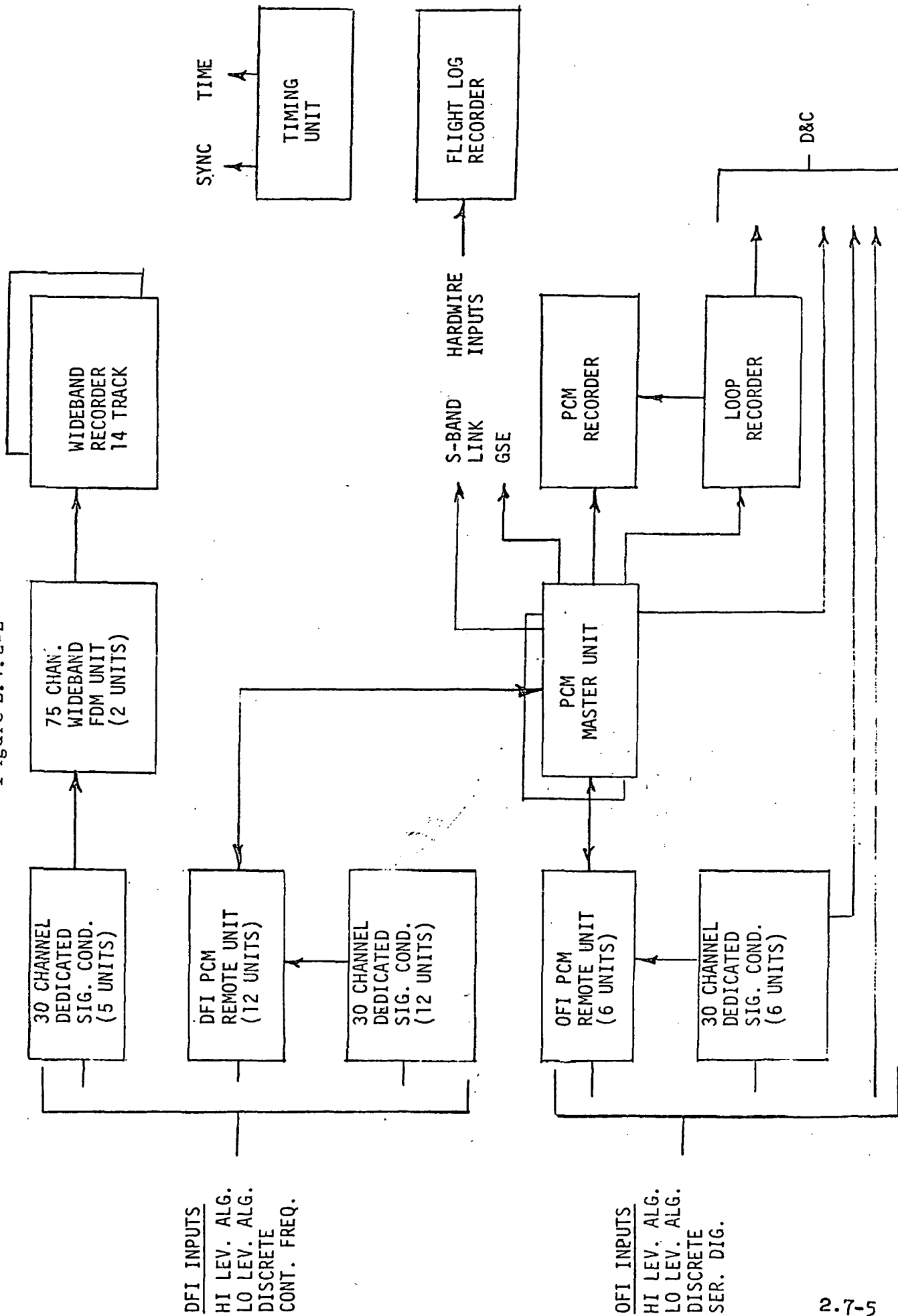
DFI INPUTS
LO LEV. ALG.
HI LEV. ALG.
DISCRETE
CONT. FREQ.

OFI INPUTS
HI LEV. ALG.
LO LEV. ALG.
DISCRETE
SER. DIG.



INTEGRATED OPERATIONAL &
DEVELOPMENT FLIGHT INSTRUMENTATION SUBSYSTEM

Figure 2.7.2-2





5. The additional DFI hardware is available for later vehicles and/or spares.
6. There is a greater operational flexibility if the OFI and DFI have no interface.
7. The DFI can provide a cross check on the OFI data.

The physical factors (i. e., weight, power, and volume) favor the use of an integrated system; however, management, change effects, and ground usage favor the dedicated approach. Change effects and data usage outweigh the physical factor gains of the integrated approach and, therefore, the dedicated OFI and DFI system, using common hardware to the maximum extent, has been selected as the baseline system.

2.7.3 Instrumentation Subsystem Description

The shuttle instrumentation subsystem consists of transducers, signal conditioning equipment, PCM encoding equipment, PCM recorders, wideband recorders, a flight log recorder and timing unit (see Figure 2.7.2-1). The instrumentation subsystem is made up of two separate functional systems, Development Flight Instrumentation (DFI) and Operational Flight Instrumentation (OFI). The responsibility for the selection, specification, and control of all sensors and associated signal conditioning, not an integral part of a subsystem control loop, is allocated to the instrumentation subsystem in order to achieve maximum compatibility and commonality of hardware.

All data used for telemetry, recording, or display is classified as a measurement and is processed by the instrumentation subsystem. Signals used for control functions only are not processed by the instrumentation subsystem. Tables 2.7.3-1 and 2.7.3-2 summarize an avionics users estimate of the operational and development flight measurements by type and subsystem, respectively.

All operational measurement data is either hardwired to the display panel, processed by the PCM equipment, or hardwired to a recorder (flt log meas.). PCM data will be recorded continuously on a loop recorder and also on a separate PCM recorder as required on a snapshot basis.

All quasi-static DFI data will be processed by PCM equipment and recorded on a snapshot basis on a PCM recorder. Continuous frequency data, such as vibration, acoustic, and flutter measurements will be frequency multiplexed and recorded on a wideband recorder.

Signal conditioning will be time shared as much as possible by preconditioning and low level multiplexing in the PCM remote units.



Table 2.7.3-1. Operational Flight Instrumentation

SUBSYSTEMS	T - TEMP	P - PRESSURE	U - UNIDENTIFIED	S - STRAIN, STRESS	V - VIBRATION	X - EVENT	Y - ACOUSTICAL	H - POSITION/ATTITUDE	A - ACCELEROMETER	Q - QUANTITY	R - RATE	E - AMPS/VOLTS/PWR	TOTAL
	AERO SURFACE												
VEHICLE STRUCT													
THERMO PROTEC													
MAIN PROP	30	33		69			24	1	8	8		173	
OMS		70		130			20		3	20		243	
ACPS		40		171								211	
ABES		20					12		2	12		46	
GN&C							70	20			5	95	
HYD PWR				45			50					95	
ECLSS				61								61	
FLT CREW SUP				1								1	
INSTR				15			2					17	
COMM				4			2					6	
ELECT PWR DIST		3		52							12	67	
ELECT PWR GEN				140								140	
MECH PWR GEN	6	10		70					3			89	
TOTAL	36	176		70	688		180	21	16	40	17	1244	



Table 2.7.3-2. Development Flight Test Instrumentation

SUBSYSTEMS	T - TEMP	P - PRESSURE	U - UNIDENTIFIED	S - STRAIN, STRESS	V - VIBRATION	X - EVENT	Y - ACOUSTICAL	H - POSITION/ATTITUDE	A - ACCELEROMETER	Q - QUANTITY	R - RATE	E - AMPS/VOLTS/PWR	TOTAL
AERO SURFACE	30	190	177										397
VEHICLE STRUCT	150	41		314	140	30	27	13	10				785
THERMO PROTEC	1016							4					1020
MAIN PROP	126	63			52			4	23		4		272
OMS		42	12		18			12		6			90
ACPS		30			41								71
ABES	65	78								1			144
GN&C			73										73
HYD PWR			50										50
ECLSS													
FLT CREW SUP	5			3									8
INSTR	5												5
COMM					28			11					39
ELECT PWR DIST	5			20						7			32
ELECT PWR GEN					54								54
MECH PWR GEN					51								51
TOTAL	1402	444	312	314	214	223	27	44	93	1	13	4	3091



Dedicated signal conditioning is provided to condition those special signals which cannot easily be accommodated by the pre-conditioning, continuous data such as vibration and acoustic measurements, and those continuous dedicated measurements required for display.

2.7.4 Instrumentation System Component Selection Criteria

Sensors - Low level sensors (10 to 20 MV output) will be used for temperatures, pressures, strains, etc. These types of sensors are well within the state-of-the-art and have been space proven in the Saturn and Apollo programs. Low level sensors lend themselves to low level multiplexing, thus saving weight, power, volume, and cost of signal conditioning. Temperature environment for the majority of sensors precludes the use of DC to DC types. Vibration environment and life requirements limits the use of potentiometer type sensors.

Signal Conditioning - Most signals for recording and telemetry will be pre-conditioned, low level multiplexed, and amplified on a time share basis by a common amplifier in each PCM remote unit. Special signals such as momentary discrettes which must be held until sampled by the PCM, will be conditioned by a dedicated signal conditioner. Dedicated signal conditioning will be used for continuous signals, not to be time shared, such as vibration, flutter, and acoustic measurements. Also, those critical hardwired display measurements which require conditioning will be conditioned by a dedicated signal conditioner.

Charge amplifiers will be used for piezoelectric measurements (acoustic and vibration) for minimum line capacity effect.

The signal conditioning packages will feature flexibility (types of conditioners interchangeable). Thirty channels of conditioning will be available in each unit, with each conditioner having its own isolated power supply. Excitation voltage supply for engine valve position potentiometer is located within the signal conditioning package.

PCM System - PCM was selected for the main data handling for the following reasons:

1. A large number of the measurements are quasi static and compatible with low PCM sampling rates.
2. PCM is an efficient method of handling and processing data.
3. Power consumption of PCM equipment is low compared to other data handling techniques (PDM, PAM, frequency multiplexing, oscillographs, etc).



4. PCM telemetry has no transmission error.
5. PCM data streams are easily adaptable to computer processing.

The PCM Master/Remote concept was selected because the large number of measurements creates wiring complexity if a single PCM unit is used. If numerous individual PCM units are used to reduce wiring complexity, a complexity for recording, telemetry, display and GSE interface is created because of the large number of PCM output data streams. The master/remote concept reduces wiring complexity as well as data handling complexity.

A PCM master unit was chosen which performs programming only was selected. The unit sequences and interleaves the remote unit outputs into one serial data stream, but does not perform arithmetic. The computation function is not warranted for the present data gathering requirements. A unit which is software programmable in a read only memory will be used. Several selectable formats and bit rates may be stored so that the equipment is adaptable to mission phase data requirements.

The PCM remote units will be limited to approximately 250 channels. Integral preconditioning (bridge completion, thermocouple reference, etc.) will be used to reduce signal to multiplexer wiring complexity and amount of dedicated signal conditioning. Integral preconditioning allows direct compatibility with low level sensors. Flexibility will exist in interchangeable plug-in preconditioning modules.

Built-in test equipment (BITE) checks multiplexer through analog to digital converter. Anomalous amplifier gain change may be identified by the BITE and allows for correction of all data passing through that amplifier. BITE also features a continuity test back through the low level multiplexer and sensor (reference Section 2.7.5). The BITE circuitry was used on the 747 Airborne Integrated Data System (AIDS).

Tape recorders are selected as the recording medium for stored data. In the present state of the art, tape recorders give the highest data capacity and rates per unit weight and volume and playback is compatible with ground processing equipment. PCM bi-phase recording techniques allow high bit packing density and has been used on Apollo J mission. FM recording will be used on the wideband recorder to achieve very low frequency response (down to DC) and greatest amplitude accuracy.

The Apollo Central Timing equipment has been chosen as it meets the shuttle requirements for providing mission elapsed time and synchronizing signals for communications and data systems.



The instrumentation subsystem form factors for the selected interim candidate system is shown in Table 2.7.4-1.

2.7.5 Equipment Descriptions

Sensors and Signal Conditioning - The severe environmental conditions for some of the transducers of the orbiter vehicle have in many cases limited the choice of transducer types. In some cases, further transducer development may be required to meet the environmental conditions.

For the measurement of cryogenic temperatures, platinum resistance probes will be used. For high temperatures, up to 2500°F, chromel constantan thermocouples may be employed. For intermediate temperature ranges, copper/constantan thermocouples or platinum resistance bulbs will be used.

Strain gauge pressure transducers will be used for pressure measurements. Temperature environment for these transducers will be in the range of approximately -65°F to +350°F.

Piezoelectric transducers are used for vibration and acoustic measurements in temperature environments of -250°F to +1200°F.

Resistance strain gauges will be used in measuring strains from cryogenic temperatures to 900°F.

Technology development is potentially required for the following types of measurements.

- Zero Gravity Gauging
- Fatigue
- Low Frequency Accelerometer (high vibration environment)
- Cryogenic Pressure
- Hydrogen Fire Detection

Signal conditioning, in general, will be accomplished in the avionics bay or aft cargo bay. The severe environments at the sensor locations preclude the use of remote signal conditioning or integral signal conditioning.

The major portion of signal conditioning will be accomplished on a time share basis in the PCM remote units. Dedicated conditioning for panel display measurements, continuous frequency measurements, and signals which cannot be handled by the preconditioners will be accomplished in the signal conditioning packages. These packages will each contain 30 dedicated conditioner modules, each having its own isolated power supply, and drawing approximately an average of one watt of power. The package will be flexible

Table 2.7.4-1

Instrumentation Form Factors - Interim Candidate System

Item	Unit	Unit Wt.	Unit Vol.	Unit CuFt	MK I Horiz						MK I & MK II Vertical					
					QTY			Total Pwr. Watts	Total Wt. Lbs.	Total Vol. CuFt	QTY			Total Pwr. Watts	Total Wt. Lbs.	Total Vol. CuFt
					A	V	O				KI					
											A	V	O			
DFI	---	0.15	---	---	1000	---	150	---	2000	3000	---	450	---	---	---	
Transducers	---	0.15	---	---	1000	---	150	---	2000	3000	---	450	---	---	---	
PCM Remote Unit	50	6	0.14	0.70	2	3	5	250	3	4	7	5	12	600	72	1.68
PCM Master Unit	60	20	0.52	1.04	2	2	40	*60	-	-	2	2	2	*60	40	1.04
Signal Cond. Unit	35	15	0.28	1.96	2	5	105	245	6	4	10	7	17	595	255	4.76
Wideband FDM	50	15	0.23	0.28	1	1	15	50	1	1	1	1	2	100	30	0.56
Wideband Recorder	60	50	1.85	1.85	1	1	50	60	1	1	2	2	2	*60	100	3.70
PCM Recorder	60	50	1.85	1.85	1	1	50	60	-	-	1	1	1	60	50	1.85
TOTALS				7.68			440	725				440	7.68	1475	997	13.59
OFI	---	0.15	---	---	500	---	75	---	500	1000	---	150	---	---	---	---
Transducers	---	0.15	---	---	500	---	75	---	500	1000	---	150	---	---	---	---
PCM Remote Unit	50	6	0.14	0.42	1	2	18	150	2	1	3	3	6	300	36	.84
PCM Master Unit	60	20	0.52	1.04	2	2	40	*60	-	-	2	2	2	*60	40	1.04
Signal Cond. Unit	35	15	0.28	0.84	1	2	45	105	2	1	3	3	6	210	90	1.68
PCM Recorder	60	50	1.85	---	-	-	---	---	1	1	1	1	1	60	50	1.85
Loop Recorder	40	20	0.29	---	-	-	---	---	1	1	1	1	1	40	20	0.29
Flt. Log Recorder	60	50	1.85	1.85	1	1	50	60	-	-	1	1	1	10	50	1.85
Timing Unit	30	20	0.40	0.40	1	1	20	30	-	-	1	1	1	30	20	0.40
TOTALS				4.55			248	405				248	4.55	760	456	7.95
*Total pwr. equal to unit				12.23	TOTALS			1130		TOTALS			2235	1453	21.54	
Pwr. - Redundant units not operated simultaneously					Totals OFI & DFI			688		Totals OFI & DFI			2235	1453	21.54	



in that any of the conditioner outputs will be normalized to zero to 5-volt output. The conditioners will consist of charge amplifiers, differential amplifiers, active bridges, momentary event holding circuits, ac to dc converters, frequency to dc converters phase sensitive demodulators, etc.

The signal conditioning package will contain provisions to plug in a 5-volt excitation power supply in lieu of one or more of the conditioner modules.

PCM Equipment - The PCM equipment will be a stored program, non destruct, read only memory type. The equipment will be capable of being programmed with four programs and 8 bit rates, selectable by external logic. The data rate baseline for OFI is 64 kilobits/second and for DFI, 128 kilobits/second.

Identical PCM equipment will be employed in both the OFI and DFI, and each system consists of two, switchably redundant PCM master units with the capability to address a number of PCM remote units. In the phased hardware approach, the DFI will have 5 remote units during the horizontal test phase, and 7 units will be added with Kit 1 to bring the total remote units to 12. The OFI will have 3 remote units during the horizontal test phase and 3 more will be added with Kit 1.

A simplified block diagram of the PCM equipment is shown in Figure 2.7.5-1. Each measurement channel is addressed according to the selected master unit program. Pre-conditioners in the remote units furnish excitation for sensors, provide bridge completion networks, compensate thermocouple secondary junctions, and normalize signals for the low level multiplexers. The multiplexers, operated by the master program, furnish outputs to an addressable switched gain amplifier. The amplifier output is digitized and processed by the digital data processor which outputs data to the master unit. The master unit obtains data from each remote unit addressing according to the stored program and interleaves all data into one serial PCM stream. Data is encoded into 10 bits, 8 of which contain measurement intelligence, plus 2 tag bits which contain BITE and overrange information.

The switchable gain amplifier has an overrange feature in addition to the addressable programmed ranging. If a measurement is over its normal range, the amplifier automatically reduces to half its programmed range. If the measurement is still over this range, the amplifier automatically reduces to one-fourth its programmed range. This greatly increased the dynamic range of the input signals. Three of the four possible states of the two tag bits indicate normal, half, or quarter gain of the amplifier. The fourth possible state of the tag bits is reserved for BITE and indicates amplifier gain out of tolerance condition and is discussed below.

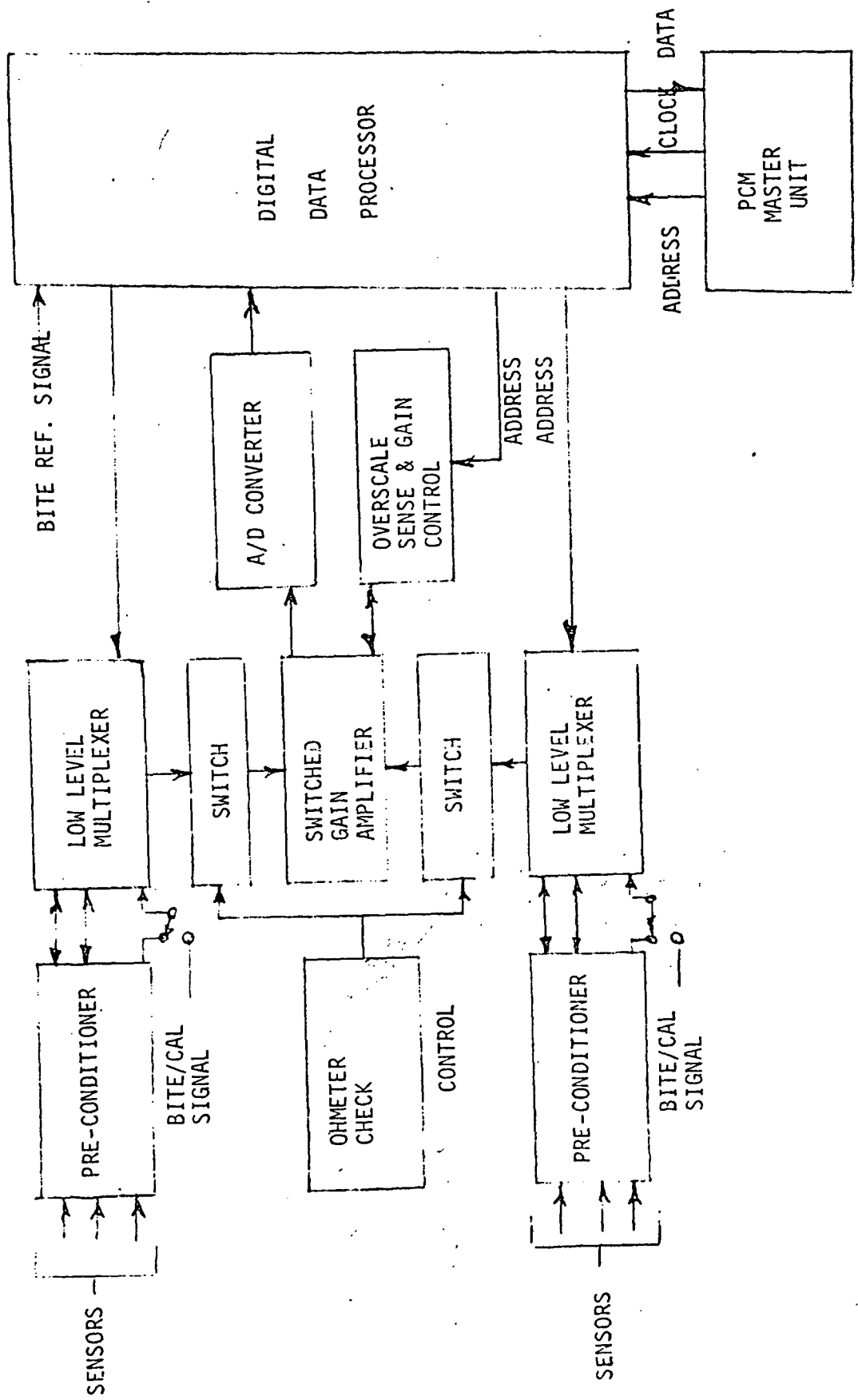


Figure 2.7.5-1. PCM System - Block Diagram



A calibration signal is switched to one channel of each multiplexer during the frame sync word. This calibration signal is digitized and compared to a reference word in the digital data processor. If the calibration signal is out of tolerance it is allowed to remain switched in during the normal data cycle and this is indicated in the state of the tag bits as discussed above. The percentage out of tolerance condition may be determined from the calibration signal and correction can be made to all data passing through the out of tolerance amplifier.

The ohmeter circuit is included which is actuated by external command. This circuit switches a precision ac voltage source through one side of the differential low level multiplexer, through the sensor source, back through the other side of the low level multiplexer and through a precision impedance. The voltage across the precision impedance is a function of multiplexer gate impedance and source impedance. This voltage is rectified and fed through the switchable gain amplifier, A/D converter, and digital processor. The PCM output data may then be analyzed for sensor circuit continuity. This function is used during ground checkout and anomaly investigation.

PCM Recorders - The PCM recorders for OFI and DFI are similar but are required to record different bit rates at different tape speeds. Each recorder will be capable of four hours recording time on the following basis.

<u>Event</u>	<u>Recording Time (Seconds)</u>
Boost to Insertion	421
OMS Burn	373
Docking	1,800
Re-docking	1,800
Pre-entry	1,140
Entry	3,600
Other Events (Estimated)	<u>5,266</u>
Total	14,400 = 4 hrs.

Other PCM recorder characteristics are as follows:

	<u>DFI Recorder</u>	<u>OFI Recorder</u>
Channels	1. 128 KBPS digital data from PCM Master unit	1. 64 KBPS digital data from PCM Master Unit
	2. IRIG "B" time code (100 PPS) from Timing Unit	2. IRIG "B" time code from Timing Unit
		3. 64 KBPS digital data from Loop Recorder
		4. IRIG "B" time code from Loop Recorder



	<u>DFI Recorder</u>	<u>OFI Recorder</u>
Speed	15 IPS	7-1/2 IPS
Bit Packing Density	8500 Bits/Inch	8500 Bits/Inch
Tape Length	9000 feet (tape reversed and track change at end of tape)	9000 feet
Control	Manual	Manual and/or automatic (from Performance Monitor)

Loop Recorder - A loop recorder is provided which records the OFI PCM data continuously on a 5-minute loop. The playback and record heads are located such that playback occurs 5-minutes after data is recorded. The loop recorder may be dumped onto the DFI PCM recorder to preserve data taken during an anomolous condition. A block diagram of the OFI recording system is shown in Figure 2.7.5-2. At any time the PCM recorder is actuated, data and time from the loop recorder is dumped into channels 3 and 4 of the PCM recorder. This data is that recorded the previous 5 minutes on the loop recorder.

Flight Log Recorder - The flight log recorder is a two track recorder based on the Aeronautical Radio Incorporated (AIRINC) 573 specification for a non-ejectable, crash protected recorder.

Flight log data is recorded on one track and voice on the other. Flight log data is multiplexed and digitized to a 768 BPS digital signal. Tape speed is 0.43 IPS, providing 6.25 hours of recording on 810 feet of tape. Data is hardwired to the recorder which contains electronics for multiplexing and digitizing flight data, and an amplifier for the voice channel. Flight log recordings will be made during the following periods.

- Boost to Insertion
- Pre-Entry
- Entry
- Horizontal Flight
- Landing

Wideband Data Acquisition and Recording - Provisions for acquiring and multiplexing 150 channel of wide band data (vibration, flutter, and acoustic) have been made. A block diagram of the wideband data acquisition and recording is shown in Figure 2.7.5-3. The wideband signals are amplified and normalized by dedicated signal conditioners, the outputs of which modulate

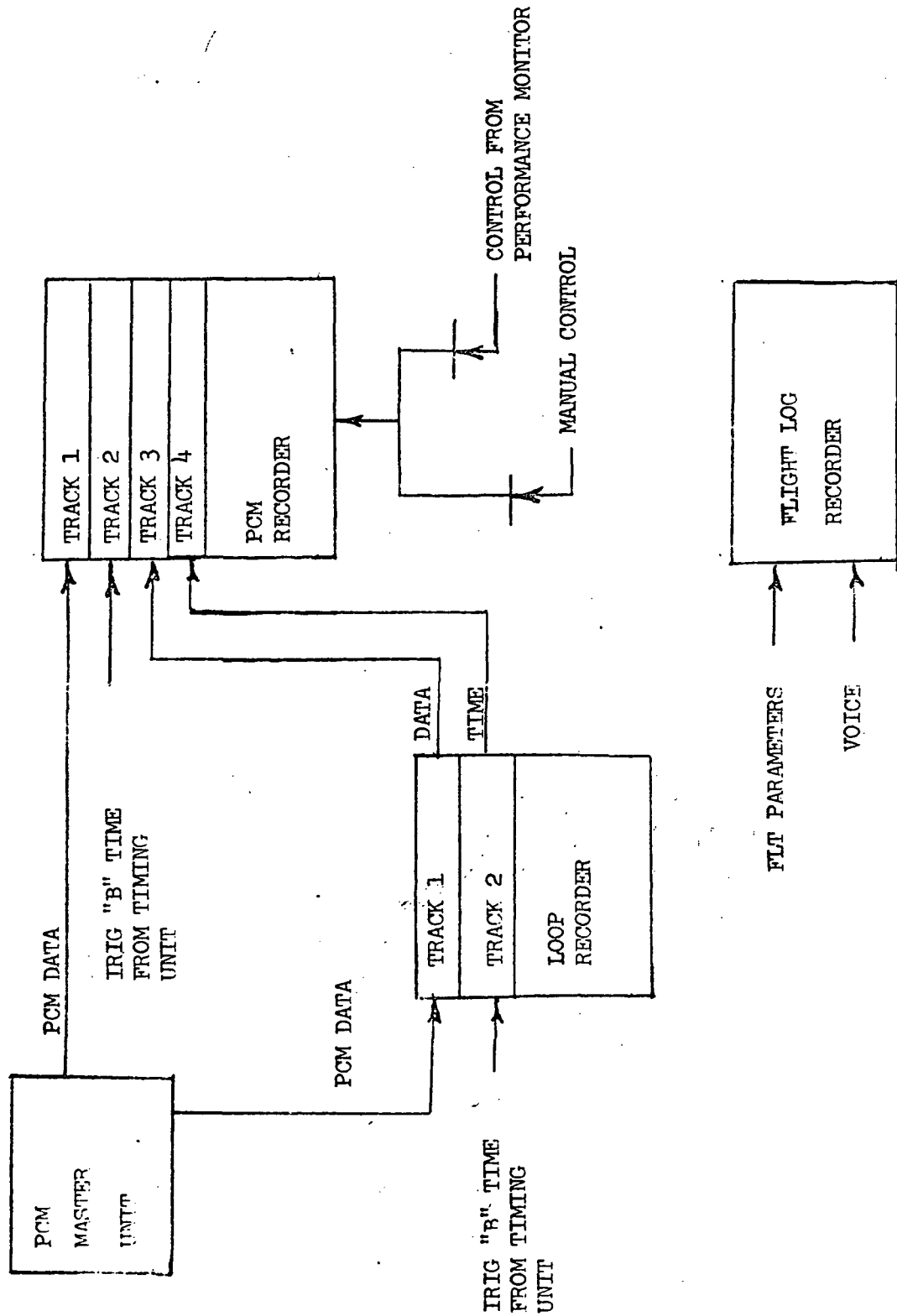


Figure 2.7.5-2. Operational Flight Instrumentation Recording

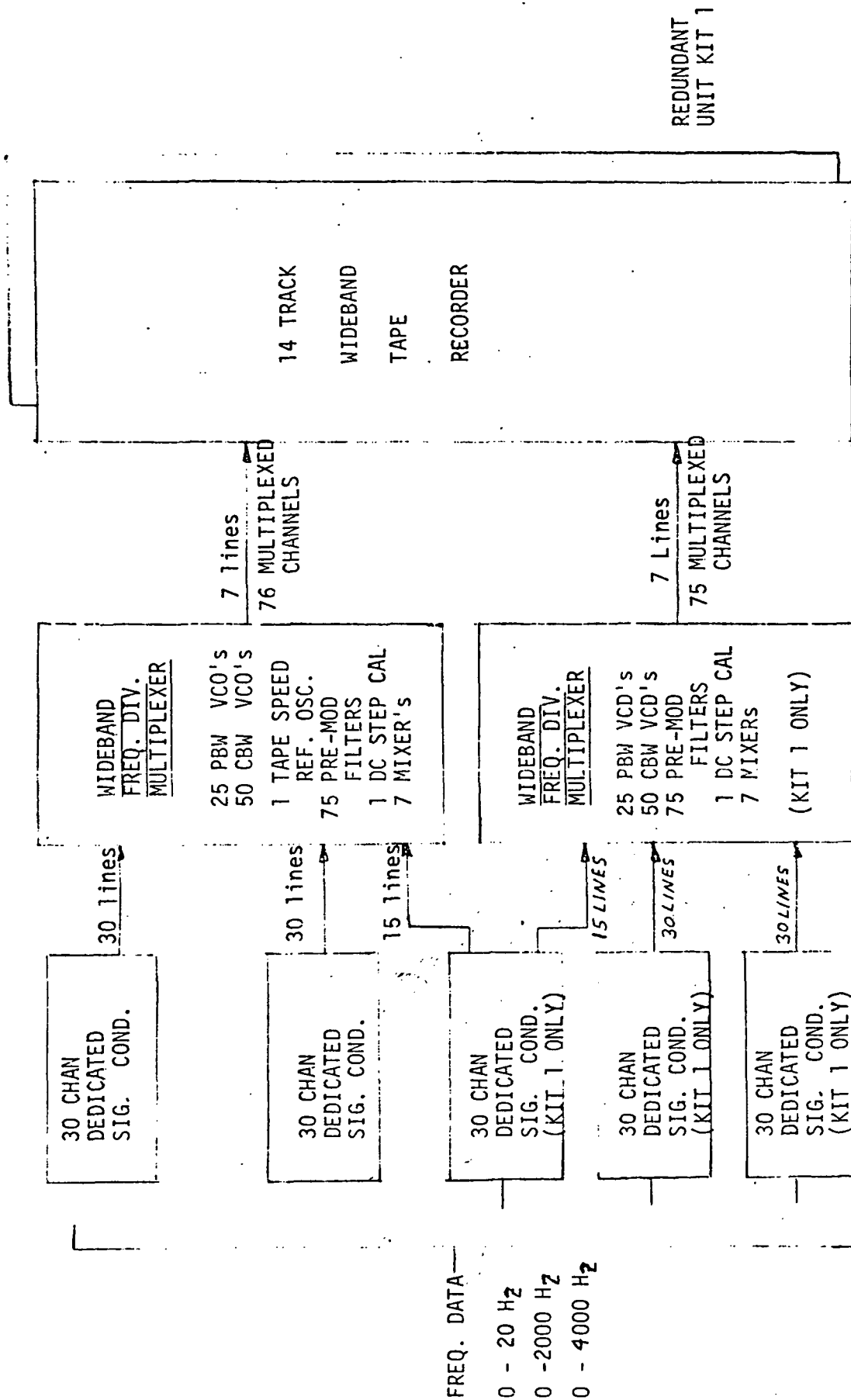


Figure 2.7.5-3. Wideband Data Acquisition and Recording



dedicated voltage controlled oscillators. Standard IRIG channels are used from proportionate as well as constant bandwidth channels. Approximately 11 VCO's will be mixed into each of 14 composite signals. Each composite signal will be recorded on a separate track of a 14 channel wideband recorder.

A typical VCO frequency mix per channel is shown below.

<u>IRIG Channel</u>	<u>VCO Frequency KHz</u>	<u>Data Response HL</u>
7 (PBW)	2.3	0 - 35
8 (PBW)	3.0	0 - 45
9 (PBW)	3.9	0 - 59
10 (PBW)	5.4	0 - 81
11 (PBW)	7.35	0 - 110
1A (CBW)	16.0	0 - 1000
2A (CBW)	24.0	0 - 1000
3A (CBW)	32.0	0 - 1000
5B (CBW)	48.0	0 - 2000
7B (CBW)	64.0	0 - 2000
11C (CBW)	96.0	0 - 4000

In addition, a reference frequency of 120 KHz will be mixed in one channel for tape speed and flutter compensation. Pre-modulation filters for each VCO channel is provided in the frequency division multiplexers. Also, each multiplexer will have built in DC step calibration for the VCO's, actuated by external command.

The wide band recorder is a 14-track wide band analog recorder. Tape capacity is 4500 feet, allowing one hour of recording at a tape speed of 15 IPS.

Timing Unit - The timing unit provides time and synchronization to the communications and data systems. The equipment contains a time accumulator and provides mission elapsed time on parallel lines plus modified IRIG "B" serial output. Parallel time is furnished to the PCM equipment and serial time is recorded by the tape recorders. In addition to the time outputs, 1 Hz and 512 KHz frequencies are provided to the PCM equipments (OFI and DFI) for data frame synchronization, allowing time correlation between OFI and DFI data. 512 KHz is also furnished to the communications subsystem, from which two subcarrier frequencies are derived.



2.7.6 Summary

A baseline system using independent OFI and DFI has been described. Maximum use of developed hardware; i. e., sensors, signal conditioners, and PCM equipment, recorders, and timing equipment, has been specified. Minor modifications and repackaging is required for some hardware. Common hardware is used as much as possible in both systems. All measurement data except certain dedicated displays are recorded. Black box redundancy is employed in the PCM master units and in wideband recording to prevent catastrophic loss of data. Critical measurements will have dual sensors and dual paths.

Further investigations are required in refining the measurement lists. Detailed information such as types of measurements, and their location, are needed to finalize the number of remote multiplexers, signal conditioners, and their respective capacity. Measurement sampling rates and their mission time phase effectivity must be further defined in order to establish a firm data rate and effective utilization of the stored programmable PCM format.



2.8 ELECTRICAL POWER DISTRIBUTION & CONTROL (EPDC)

2.8.1 Requirements

Figure 2.8.1-1 is an Avionics load profile for significant Orbiter flight phases, comparing power requirements of the Alternate Avionics System, ("new baseline") with the Phase B Avionics power levels, ("old baseline"). Table 2.8.1-1 compares the old and new baseline load levels for each Avionic subsystem during the peak load flight phase, with an explanation for the significant load changes. The column: "Inverter AC Load Increase" lists new ac loads which must operate in orbit and therefore utilize the inverter system power output. The 1939 W. added inverter load is due primarily to the change to off-the-shelf aircraft type Avionic equipment designed to use ac power. As shown, an increase in inverter system operational capacity of 2250 VA is necessary to accommodate this added load. The last column in the table lists that part of the new baseline Avionics load that must be cooled by the Environmental Control System (ECS).

The Electrical Power Generation System capacity, operation, and redundancy remain unchanged from the Phase B baseline: three 7/10 KW fuel cells operate from pre-launch through postlanding. Three APU driven 20/30 KVA generators operate from pre-launch through ascent and again from entry through postlanding to supply ac loads during these periods. Two 10-AH nickel-cadmium batteries supply power to primary power controls for system reset and restart in event of inadvertent loss of all main dc bus power, and are an isolated power source for pyrotechnic loads. For horizontal flight test of Mark I vehicles, and for ferry flights, the APU's and fuel cells will not be used. A kit will add three constant speed drives and 20/30 KVA alternators to drive pads on the turbojet engines.

All Alternate Avionics system power requirements can be supplied in the two standard power forms: 28 vdc and 115/200 V, 400 Hz ac.

2.8.2 Alternate EPDC Configurations Studied

1. Inverter Subsystem. The Phase B baseline inverter system (six 500 VA single phase inverters connected for three phase output with an operational capacity of 1500 VA) had insufficient capacity for the alternate system total orbit ac load (3041 watts maximum). Also, the 500 VA inverter did not meet the alternate avionics system criteria of maximum use of off-the-shelf equipment.

2. Power Control Multiplexing vs. Hardwire. Since in the alternate Avionics system selected, there is no central DCM data bus with a capability of multiplexing power controls, the only alternatives remaining were a dedicated EPDC multiplexing system or hardwiring of power controls. The first flight qualified dedicated EPDC multiplexing system presently planned is the B-1 EMUX (for Electrical Multiplex System). With delivery of the first flyable system scheduled for June, 1973, it does not meet the "off-the-shelf" criteria.

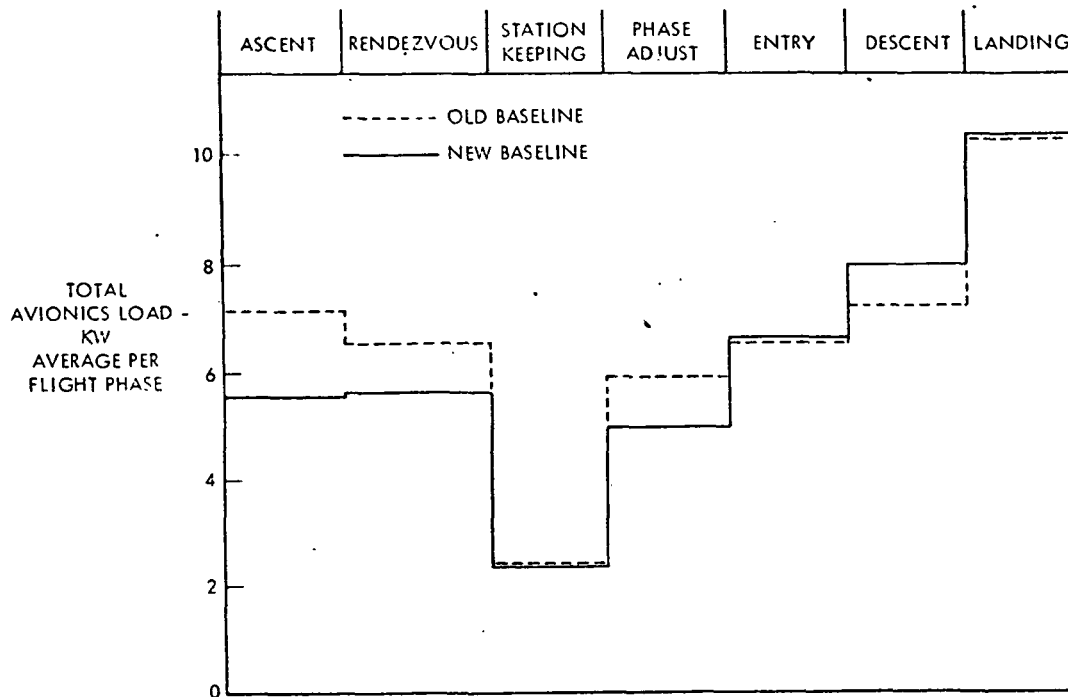


Figure 2.8.1-1. Avionics Electrical Load - Orbiter

Table 2.8.1-1. Orbiter Avionics Electrical Load Changes - Watts

SUBSYSTEM	LOADS-LANDING PHASE		LOAD CHANGES	INVERTER AC LOAD INCREASE	ECS PRESS. COMPT COOLING LOAD
	OLD BASELINE	NEW BASELINE			
D&CM	1,594	0	NO DATA BUS, COMPUTER IN GN&C	0	0
GN&C	732	2,525	ADDED SAS	1020 COMPUTERS	2389
COMM	795	748	REMOVED RADAR ALT., PRS, VHF-FM, ADD TACAN	360 TACAN	748
D&C	4,601	4,294	DEDICATED D&C	266 DISPLAYS	1294
OFI	1,010	760	REDUCED SIGN. COND., ADDED PCM	0	445
EPD&C	1,551	1,724	PWR & CONTROL RELAYS, SEQUENCERS	0	1258
ECS DELTA	0	283	INCREASED AVIONICS AIR COOLING	283 FANS	0
OPERATIONAL TOTAL	10,283	10,334		1939	6134
DFI	1,316	1,475		0	725
DEVELOPMENT EQ TOTAL	11,599	11,809		1939	6859
INVERTER SYSTEM OPERATIONAL CAPACITY, VA	1,500	3,750		2250	1008



3. Remote Power Controllers. The solid state and hybrid remote power controllers used in the Phase B baseline have only been produced in bread-board form and tested in aircraft ground electrical simulators. They also do not meet the "off-the-shelf" criteria.

4. Electrical Connectors and Wiring. Flat conductor cable (FCC) and connectors were considered and would provide greater durability, simpler fabrication and weight savings. However, additional development and qualification of FCC connectors would be necessary, and off-the-shelf equipment would require extensive modification to use FCC receptacles.

Aluminum cable in electro-conductive grade alloy is presently used in aircraft, which have incurred flexibility and corrosion problems. New developments in high strength aluminum alloy cables with corrosion resisting finishes are promising, but do not meet the off-the-shelf criteria.

The Apollo teflon insulated round wire is not considered durable enough for the 10 year operating life of the Shuttle.

2.8.3 Selected Configuration.

Figure 2.8.3-1 is a simplified schematic of the alternate avionics EPDC configuration. Changes from the Phase B baseline are:

1. Inverter Subsystem. Four 1250 VA, 3 phase Apollo type inverters are used, three operating connected to three isolated inverter buses to provide 3750 VA operational capacity. The fourth inverter is an idle standby to provide fail operational capability. Fail safe capability after two inverter failures is provided by switching off redundant ac loads to stay within the 2500 VA capacity of the remaining two inverters. Sufficient operational inverters will be available as surplus from the Apollo program to meet Shuttle Orbiter requirements without manufacturing any new inverters.

2. Hardwired Power Control. Multiplexing is replaced by hardwired power control from the D&C panel, together with both conventional pushbutton circuit breakers on the D&C panels and remote control circuit breakers (RCCB).

3. Remote Control Circuit Breakers (RCCB). An estimated 50 RCCB's would be used for large dc and ac loads where wire weight and space savings are possible and where D&C panel switch and circuit breaker maximum ratings are inadequate. Qualified aircraft RCCB's (such as used on DC-10 and L-1011) can be used.

4. Subsystem Relay and Sequencer Logic and Timing Functions. Increased use of relays and timing devices in EPDC relay boxes and sequencers will be employed to replace logic, sequencing and timing functions performed by the central DCM computer in the Phase B baseline.

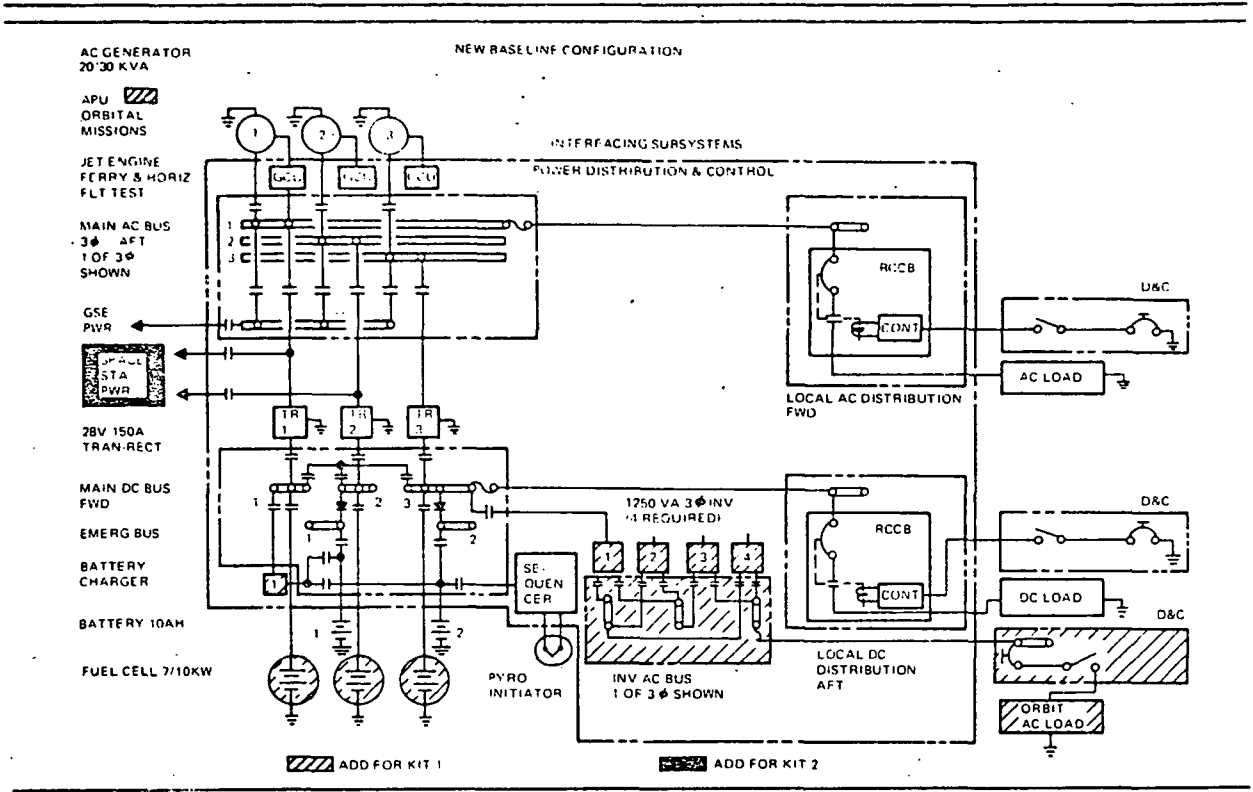


Figure 2.8.3-1. Space Shuttle Orbiter Electrical Power Distribution & Control Subsystem

5. Electrical Connectors and Wiring. Table 2.8.3-1 lists typical black box receptacle types which would be used, on off-the-shelf equipment and new designs. Harness plugs designed to MIL-C-83723 will mate with all these types (except rack and panel connectors) while still permitting use of standardized pins, sockets, and crimping, insertion and removal tools.

General purpose interconnecting wire will be MIL-W-81381/12 and 14 (8 mil insulation thickness) for power and MIL-W-81381/8 and 10 (5.5 mil insulation) for signal. This is a teflon impregnated Kapton insulated, nickel plated copper wire.

Figure 2.8.3-2 shows Orbiter launch facility and separation plane umbilical requirements. Figures 2.8.3-3 and 2.8.3-4 show electrical feed-through connectors. Apollo types may be used here.



Table 2.8.3-1. Shuttle Elect. Connectors

<u>BLACK BOX RECEPTACLES</u>	<u>WIRE HARNESS PLUGS</u>
<u>OFF-THE-SHELF</u>	
GENERAL PURPOSE (POWER) (MIL-C-5015)	UNIVERSAL TERMINATION SYSTEM (MIL-C-83723)
MINIATURE BAYONET (NAS 1599)	
STANDARD BAYONET (MIL-C-26500)	
RACK & PANEL RECTANGULAR	RACK & PANEL RECTANGULAR
<u>NEW DESIGNS</u>	
MINIATURE BAYONET (NAS 1599)	UNIVERSAL TERMINATION SYSTEM (MIL-C-83723)

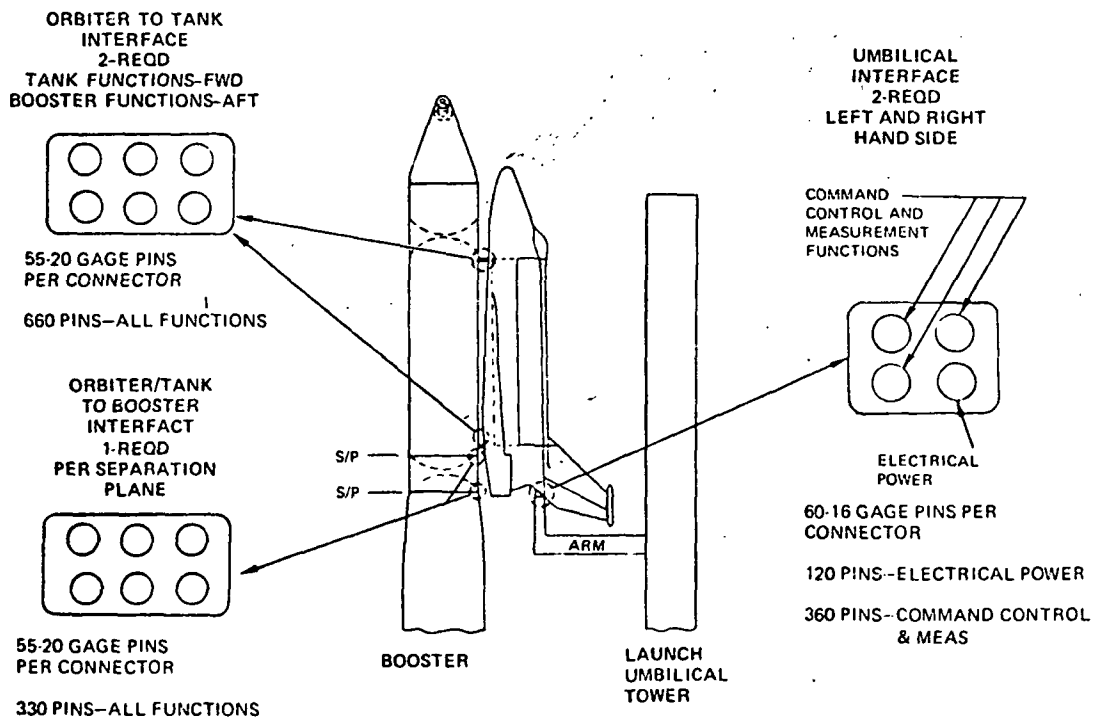
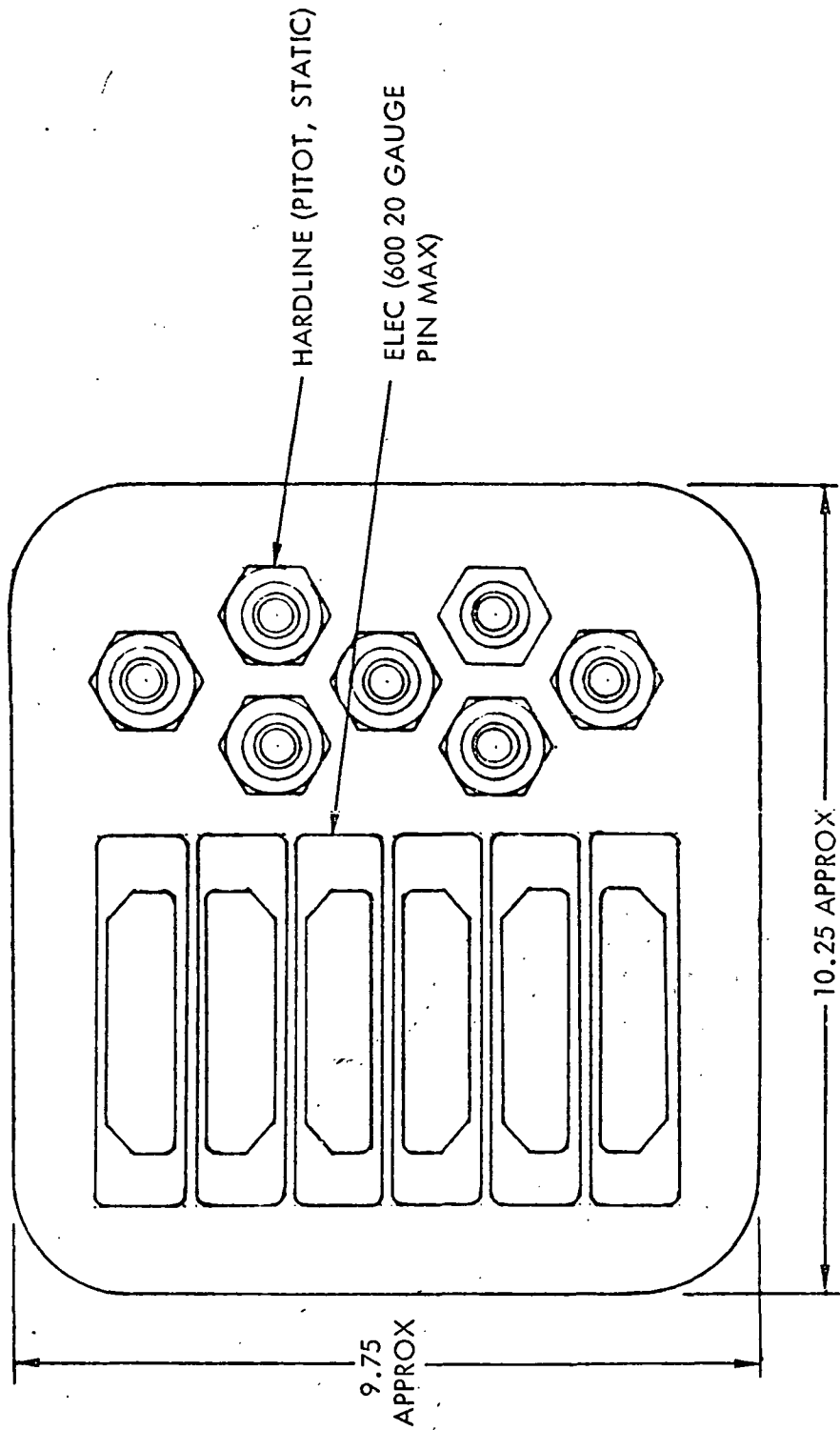


Figure 2.8.3-2. Space Shuttle Orbiter Umbilical & Separation Plane Interfaces

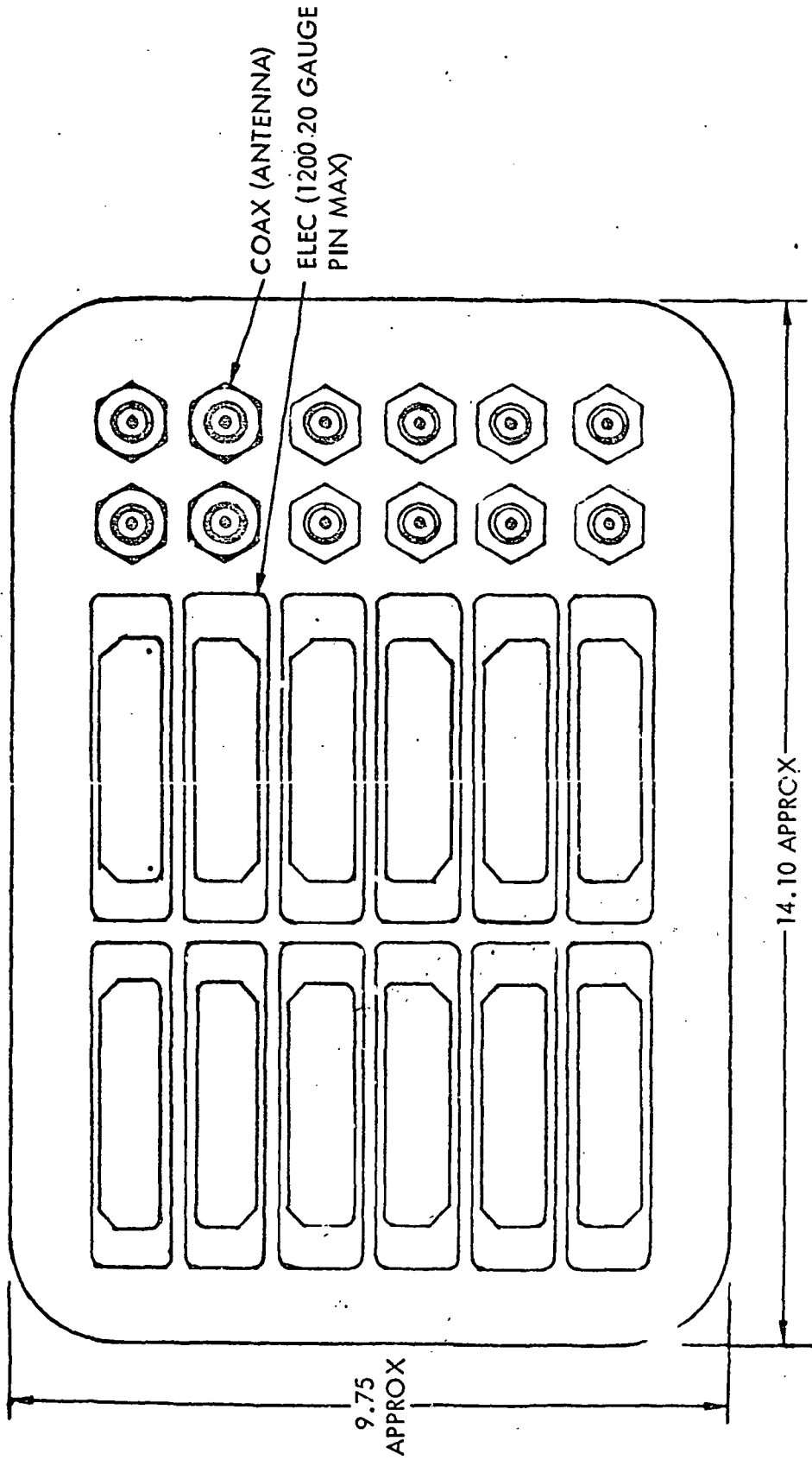


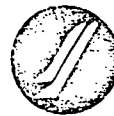
INSTALLATION
TYPICAL FWD PRESS. BKHD FEED THRU





INSTALLATION
TYPICAL AFT PRESS. BKHD FEED THRU





2.8.4 Emergency Power Study.

Since the Alternate Avionics System configuration has no central DCM system which must be powered through the mission, (as in the Phase B Baseline), and Avionics loads may now be drastically reduced in a power emergency, the requirement for and configuration of the emergency power source was reviewed. A summary of the baseline and alternates studied is listed in Table 2.8.4-1. The electrical power generation (EPG) redundancy and isolation concept is shown in Figure 2.8.4-1. An FMEA of the EPG and EPDC subsystems revealed no single failure points in the design. Also, during critical flight phases, (boost, entry and aerodynamic flight and landing), there are six primary electrical power sources. For these reasons and the extra weight and cost penalty of providing emergency power for extended periods, the minimum emergency battery system of the Phase B Baseline was retained.



TABLE 2.8.4-1. ELECTRICAL POWER GENERATION & DISTRIBUTION
EMERGENCY POWER

<u>EMERGENCY POWER SOURCE</u>	<u>EMERGENCY LOADS POWERED</u>	<u>OPERATION</u>
<ul style="list-style-type: none"> • BASELINE: BATT: 2 - 10AH NI-CAD SECONDARY 70 LB 	PRIMARY POWER CONTROLS ONLY (+ SEQUENCERS & PYRO SQUIBS)	RESTORATION OF MAIN BUS POWER, AFTER INADVERTENT LOSS
<ul style="list-style-type: none"> • ALTERNATE STUDIES: BATT: <u>AG-ZN PRIMARY</u> 891 LB (AC: GEN & INV ARE MUTUAL EMERG SOURCES DURING CRITICAL FLIGHT PHASES) BATT: <u>AG-ZN SECONDARY</u> 130 LB (AC: SAME AS ABOVE) 	ESSENTIAL DC & AC LOADS ONLY MAIN DC BUSES ESSENTIAL AC LOADS ONLY	TRANSFER ESSENTIAL LOADS FROM PRIMARY TO EMERGENCY SOURCE, IF PRIMARY FAILS PARALLEL EMERG BATT & FUEL CELLS FOR CRITICAL FLIGHT PH PHASES. TRANSFER ESSENTIAL AC LOADS TO EMERG AC SOURCE IF PRIMARY FAILS

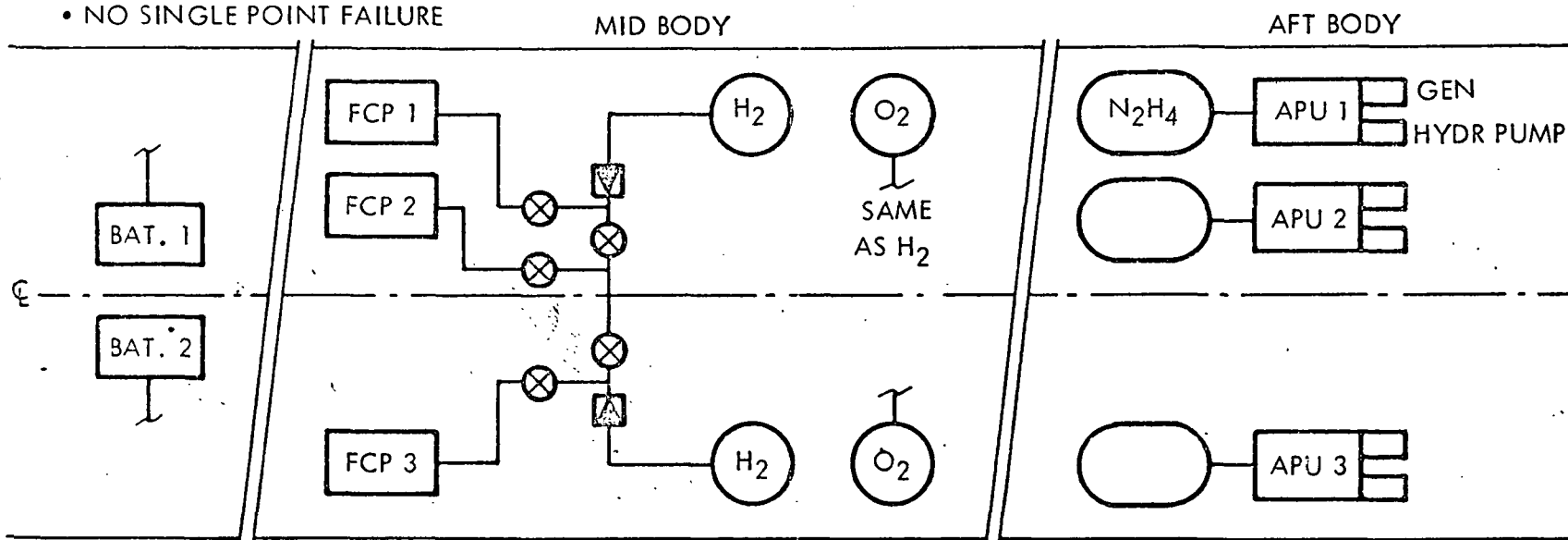
Figure 2.8.4-1.

POWER GENERATION SYSTEM ISOLATION & REDUNDANCY CONCEPT



ISOLATION

- PHYSICAL SEPARATION
- CIRCUIT ISOLATION
- NO SINGLE POINT FAILURE



REDUNDANCY

- FO/FS - FUEL CELLS
- APU'S & GENERATORS
- APU TANK & FEED SYSTEM
- CRYO TANK & FEED ACTIVE COMPONENTS
- FS - CRYO TANKS & DISTRIBUTION LINES

SD 71-346



2.9 AVIONICS SYSTEM INSTALLATION & SYBSYSTEM INTERFACES

2.9.1 Avionics Equipment Installation.

The majority of Avionics equipment is installed in two main areas: the Forward Avionics Bay in the pressurized compartment and the Aft Avionics Bay. These locations are shown in the figures "Orbiter Forward Avionics Bays" and "Aft Avionics Bays", Section 1.0.

Avionics equipment requiring locations outside the above two areas include: Antennas, GN&C body rate sensors (located for suppression of body bending modes) and the star trackers and horizon sensors. The latter must be located to provide a field of view of stars and horizon without light transmission through a window. This dictates locations outside the pressurized compartment near doors in the heat shield. Star trackers, horizon sensors and IMU's must all be mutually aligned within $+ 0.1$ arc minute. The IMU's are, however, aircooled and must be located inside the pressurized compartment. A tentative location meeting all these requirements will mount the star trackers on the outside of the pressure compartment, looking through the space station docking door, with the IMU's mounted inside the pressure compartment just opposite the star trackers.

The Aft Avionic Bay location is preferred only for that avionics equipment which must be located near propulsion and power system electrical loads and measurement transducers to reduce wire harness weight to a practical minimum. This includes AC central power distribution boxes, dc local distribution boxes, DFI and OFI PCM remote units and signal conditioning units. Location in this proximity will, for example, save an estimated 900 lbs in OFI wire harness weight. Equipment located aft must be capable of operation in vacuum, engine acoustic and vibration levels and the temperature extremes obtainable with passive temperature control.

The remaining avionic equipment, including all that requiring active ECS cooling, is located inside the pressurized compartment. Only D&C equipment requiring crew access is in the crew compartment. All other Avionics equipment is located in three redundant compartments in the pressurized area: two on the left and right sides below the flight deck floor, and one aft of the galley. These Avionics compartments are sealed from the crew compartment with pressure tight access doors and an ECS ΔP pressure regulator maintaining only 0.36 psid, with the crew compartment always at higher pressure to eliminate the toxicity hazard from the avionic equipment.

Figure 2.9.1-1 illustrates a typical shelf with the various equipment types used. Shelf wire harnesses will be located inboard to facilitate inspection and repair. The environment provided will limit the pressure, temperature, acoustic, vibration, etc., extremes to within that for which the equipment was qualified on past programs. No modification or further ground environmental testing will therefore be necessary on this equipment.

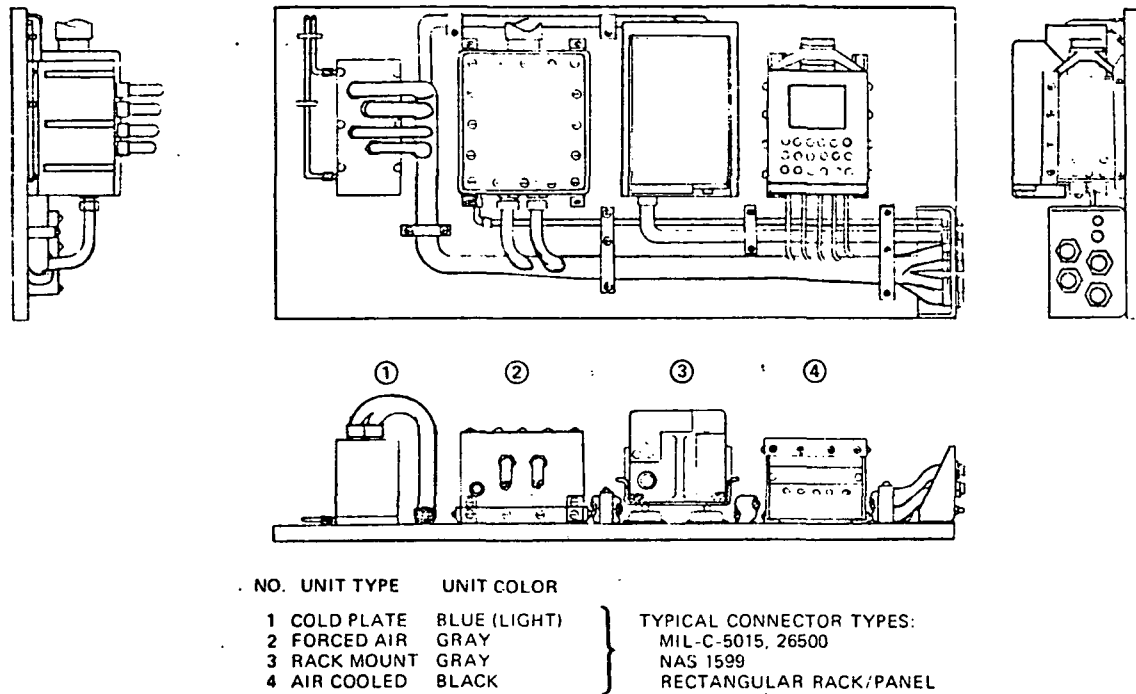


Figure 2.9.1-1. Avionics Equipment Shelf

Access in both forward and aft Avionics bays is provided in both horizontal and vertical Orbiter attitudes. Equipment mounting shelves are horizontal in the Orbiter horizontal attitude since most maintenance activity is expected to occur in the hangar. To replace equipment on the launch pad, ladder rungs are built into the compartment access doors for the ground crew. Also, the access doors open to form access platforms as shown in the Section 1.0 figure: "Orbiter Forward Avionics Bay".

Figure 2.9.1-2 shows a GSE removal shelf which may be installed to facilitate heavy equipment replacement on the launch pad. The equipment may be hoisted to the airlock for removal through the space station docking port, using the GSE electric winch shown in Figure 2.9.1-3.

2.9.2 Impact on Other Subsystems

Electric Power Generation (EPG). The power and energy required for the Alternate Avionic subsystem on the Mark II seven day space station rendezvous mission is about 800 W. less (during the fuel cell peak load period) and 47 KWH less than that for the Phase B baseline IAS. These figures include the ECLSS power increase for Avionics and distribution and inversion losses.

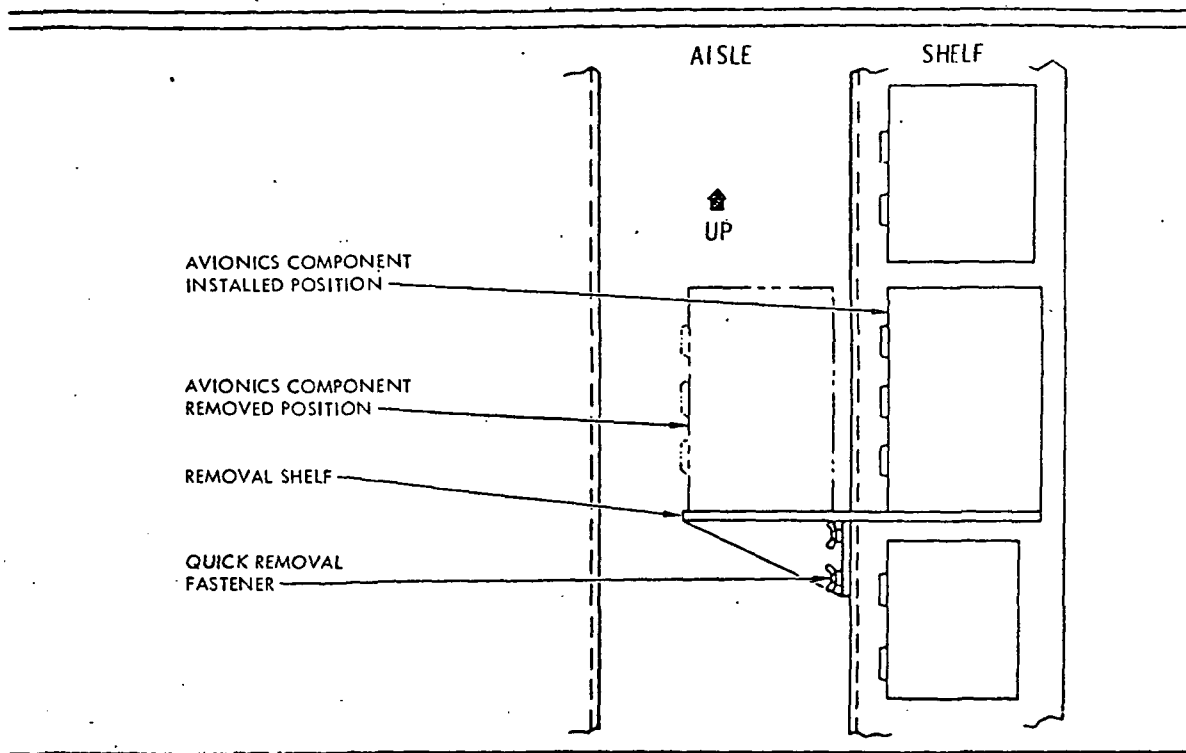


Figure 2.9.1-2. Equipment Removal Vertical Position



AVIONICS BAYS

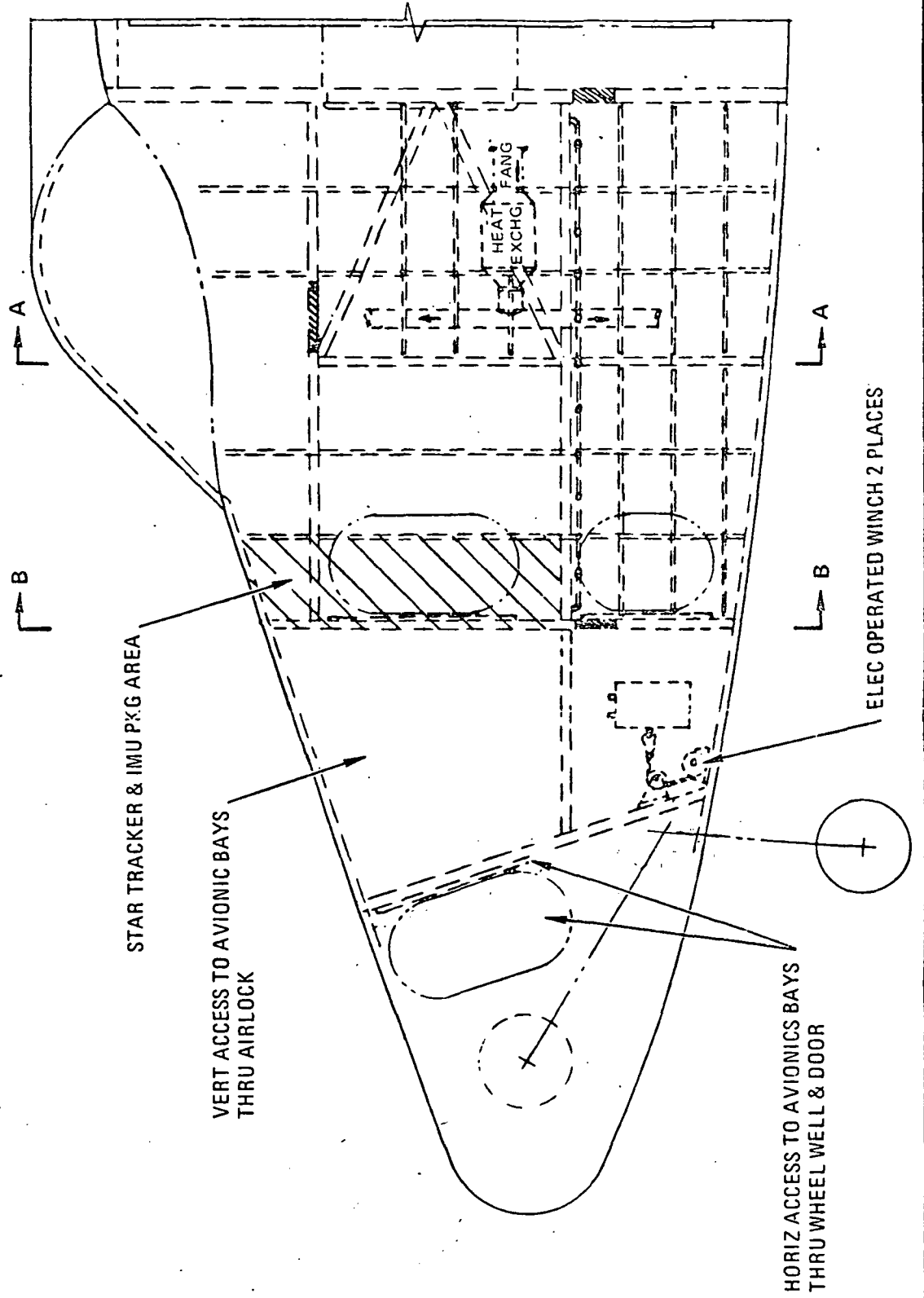


Figure 2.9.1-3



The resultant impact on EPG weight is:

Fuel cell system:	H ₂ system	- 15.4 lbs
	O ₂ system	- 54.6
	Fuel Cells	- 42.0
		<hr/>
	Total	- 112.0 lbs

The ac generator peak load is increased 422 watts but has negligible impact on EPG, because of the short operating time on ac generator power, and since the generator rating of 20/30 KVA has more than 100 percent reserve (being selected for commonality with the booster).

Environmental Control and Life Support Subsystem (ECLSS). The impact of Avionics cooling requirement changes on ECLSS is:

Coldplate load:	-2257 watts
Non-coldplate load:	-1346
Forced air cooling load:	+2616
	<hr/>
Net cooling load change	- 987 watts

The requirement for active ECLSS cooling outside the pressurized compartment was removed. The resultant impact on ECLSS weight is a 198 lb decrease, and on power is a 252 watt increase (for fan load).

The new ECLSS coolant circuit is shown in Figure 2.9.2-1. Avionics cooling has been removed from the water loop to the freon loop, reducing the required interchanger capacity. Separate heat exchangers are provided for the Avionics bays, resulting in crew cabin cooling load reduction to the extent that the humidity heat exchanger is adequate and the cabin air heat exchanger is removed. The separate Avionic Bay pressure is maintained at 0.36 psi lower than the crew cabin by a ΔP pressure regulator and controlled 1 lb/day bleed overboard. Smoke detection and remote control fire extinguishing are provided.

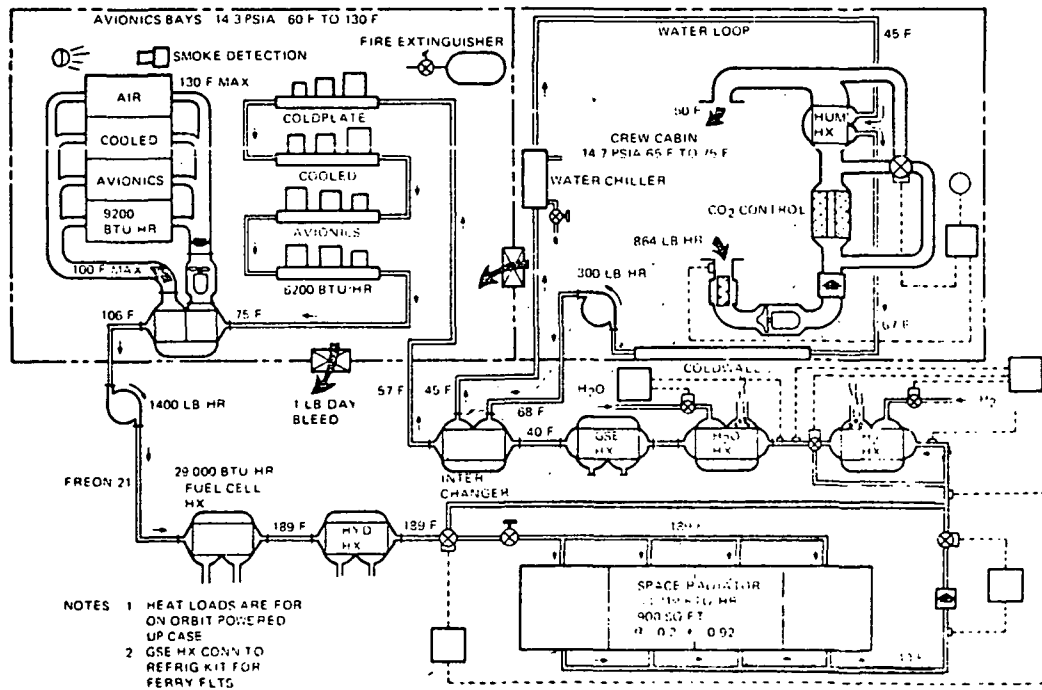
Redundancy is not indicated in Figure 2.9.2-1. There are two avionics heat exchangers, each having three redundant fans and two redundant freon loops. One heat exchanger will cool one redundant avionics equipment bay, while the other heat exchanger will cool two redundant bays. From the latter, the two outlet ducts will be provided with shut-off valves to shut off one bay in event of smoke detection while maintaining air cooling to the other bay.

Weight and Structure Impact. The alternate avionics system weight is listed in Table 2.9.2-1. Weight figures include secondary structure for the avionics subsystems. The principal causes for the weight changes are listed in Table 2.9.2-2. As stated in Section 1.0, the net Orbiter weight impact of the Avionics change on all subsystems is:



Avionics and wiring	-123 lbs
Flight Control Mechanical Equipment	+600
Electrical Power Generation	-112
ECLSS	-198
Net	+167 lbs

This is only 2.4 percent of the Avionics system weight. The resultant impact on primary structure is considered negligible.



Reproduced from best available copy.

Figure 2.9.2-1. ECLSS Coolant Circuit



Table 2.9.2-1. Orbiter Avionics Weight

<u>SUBSYSTEM</u>	<u>NEW BASELINE</u>
D&CM	0
GN&C	1,252
COMM	378
D&C	1,272
OFI	555
OFI WIRING	1,490
EPD&C	1,968
TOTAL AVIONICS - MARK II	6,915
DFI	1,184
DFI WIRING	3,900
ORBITER TOTAL	11,999
EXPENDABLE TANK	276

Table 2.9.2-2. Orbiter Avionics Subsystem Weight Changes
Old Baseline to New Baseline

DCM:	COMPUTER COMPLEX NOW DEDICATED TO GN&C. 3 COMPUTERS WERE 4. NO MASS MEMORY. NO DATA BUS
GN&C:	ADDED SAS & HORIZON SENSOR
COMM:	REMOVED RADAR ALTIMETER, PRS, VHF-FM. ADDED TACAN
D&C:	CHANGED INTEGRATED D&C TO DEDICATED D&C
OFI:	ADDED PCM
OFI WIRING:	LESS REMOTE MULTIPLEXING.
EPD&C:	CENTRAL BUSES WERE DISTRIBUTED. NO MULTIPLEXING OF POWER CONTROL. NO SOLID STATE REMOTE POWER CONTROLLERS. MORE RELAYS & SEQUENCERS. LARGER INVERTERS
DFI WIRING:	LESS REMOTE MULTIPLEXING



2.10 DIGITAL SYSTEMS AND SOFTWARE

The following paragraphs summarize the results of preliminary analysis of a low cost approach to those elements of digital hardware and software applicable to a simplified Shuttle Avionics System. Three areas are covered in this section: (1) Digital Hardware, (2) GN&C Software, and (3) Display and Control Software. A fourth area, Ground Software, is discussed in Section 4.4. Section 2.10.1 includes a brief summary of the rationale for selection of the new baseline computational concept, as well as a summary of pertinent characteristics. Sections 2.10.2 and 2.10.3 simply present the particular results of the preliminary design analysis. Selection rationale may be found in the sections pertaining to GN&C and Display and Control, respectively.

2.10.1 Digital Hardware

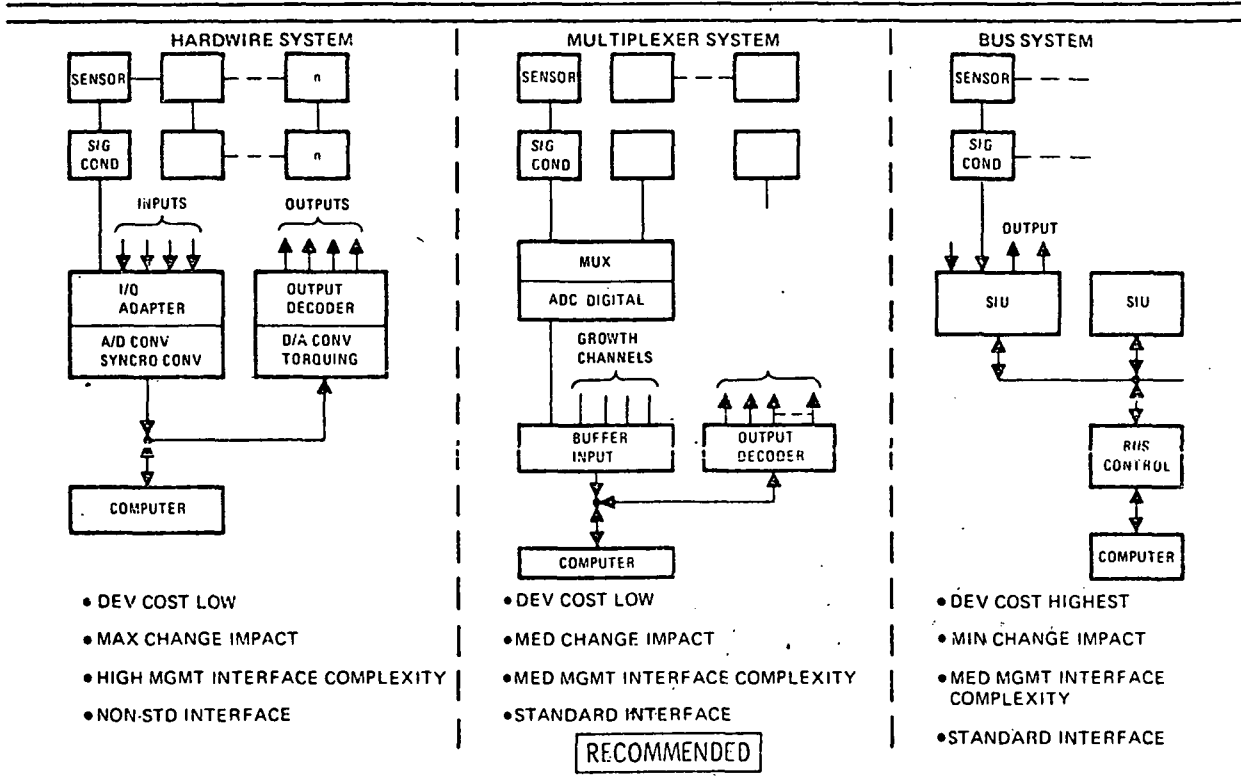
Introduction

A study was conducted during the initial four-month extension to the Space Shuttle Phase B Program contract to identify alternate approaches to providing a 'low-cost' avionics system. Part of that study included an evaluation of the extent to which existing off-the-shelf computers could be utilized for tasks identified for the operational orbiter. Also considered during the study were the specific requirements identified requiring on-board computation. Conclusions drawn at the completion of the study pertaining to the system definition consisted of separating all GN&C functions into an independent system with digital and analog components and reduction of other computer operations to those associated with the processing of performance monitoring data for display on an independent performance monitoring display.

The major area reviewed during the four-month extension pertained to the specific requirements and means of acquiring GN&C data and making these data available for use in the GN&C computing facility. Figure 2.10.1-1 illustrates in simplified block format the three concepts considered. The bus system is essentially that defined as the Phase B Data Bus concept.

Figure 2.10.1-2 shows the derived subsystem cost estimate as a function of the number of data points to be monitored. A major cost reduction factor (between the Data Bus concept and the others) is the elimination of power/cooling cost factors and the elimination of the remote standard interface units distributed throughout the vehicle. Other factors causing the wide separation in initial costs are the extent of mandatory hardwire implementation of control functions, vehicle size, equipment type and location, etc.

Figure 2.10.1-1. GN&C COMPUTER
INTERFACE TECHNIQUE COMPARISON



101SV14909A

Figure 2.10.1-2. I/O TECHNIQUES COMPARISON
(DESIGN, EQUIPMENT, & WIRE ONLY)

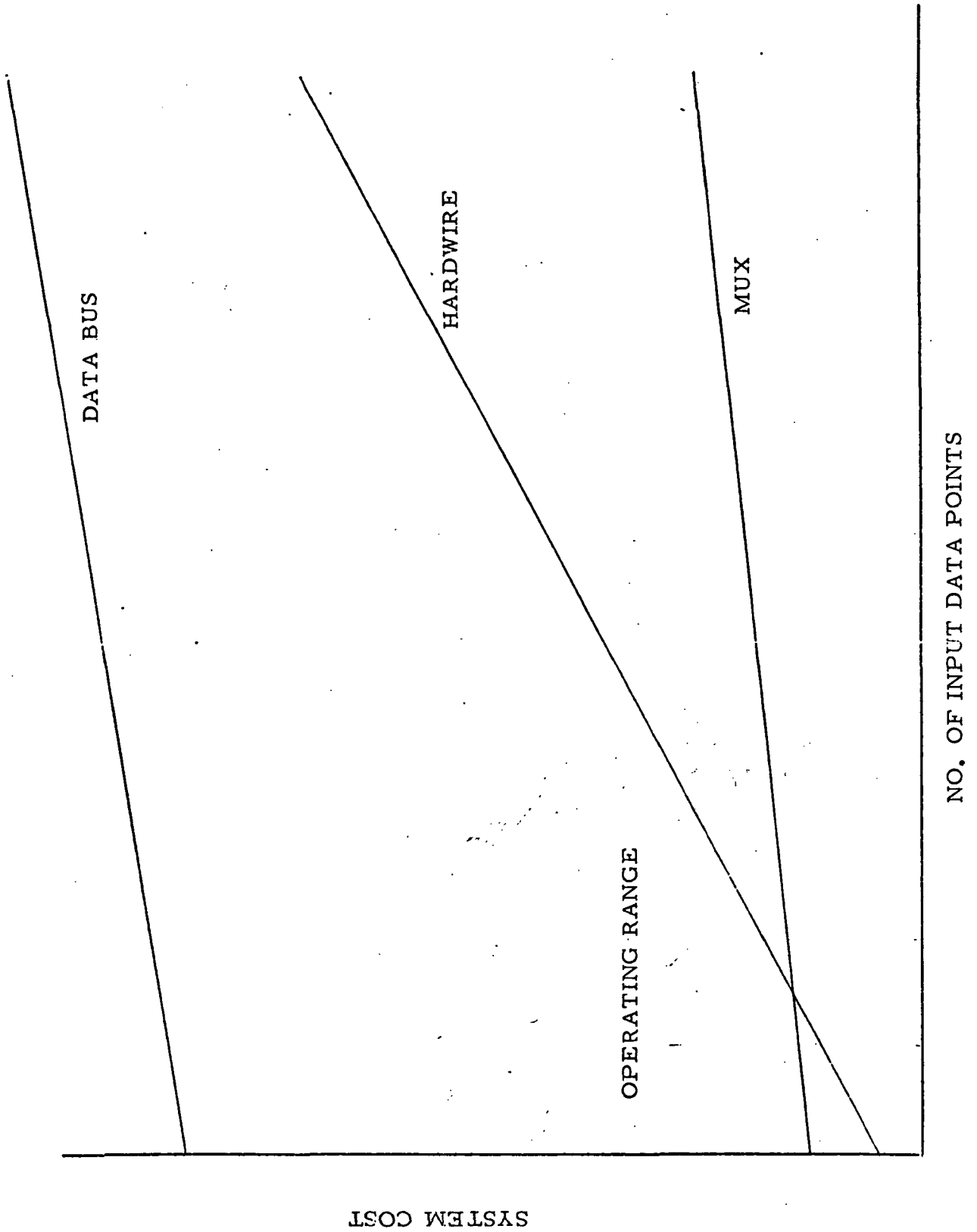




Figure 2.10.1-3 provides a listing of special features which are desirable in the selected GN&C computer so as to provide for self-test, checkout and in-flight and/or ground maintenance. Figure 2.10.1-4 lists potential candidate equipments.

A simplified block diagram of the selected GN&C computation subsystem is given in Figure 2.10.1-5. The block diagram shown does not completely identify the entire avionics system but is limited to those elements that will be discussed in this section.

The new GN&C computer baseline is comprised of four basic elements: (1) a remote multiplexing unit; (2) an input buffer; (3) an output decoder; (4) a 48K by 32 bit high speed computer. An additional component, a Display/Keyboard (DSKY), is also discussed briefly.

Design Requirements

The following design requirements were established for the selected configuration:

- Display Processor is simplex
- Display Processor software costs to include format generation and limit checking only. Language/compiler for format generation required.
- No interconnection between subsystem display CRT and G&N computer interface.
- GNC computer input is single string - no cross strapping.
- 600 signals may be limit checked in the Display Processor in any flight mode-- different limits for each measurement.
- 50 formats in Display Processor
- Computer/Processor input buffers designed and populated for input from OFI PCM master controllers.
- Special purpose inputs may be included in computer/processor input buffers if cost effective on an exception basis.

Configuration

Data Acquisition. Data is distributed to the G&N computers via the GN&C data acquisition subsystem. After signal conditioning has been performed, data from the various subsystems is distributed to one of two multiplexers. The input data is multiplexed and routed to the buffer units of the computers. Each multiplexer accepts: (1) 96 discrettes, (2) 1 serial

RECOMMENDED COMPUTER ERROR DETECTION/
DIAGNOSTIC CAPABILITY

- HALF-WORD MEMORY PROTECTION
- MAIN STORE TO CPU TRANSFER PARITY ERROR
- MAIN STORE TO I/O
 - PARITY
 - TIMED RESPONSE
- I/O CHANNEL ERROR
 - INCORRECT DATA TRANSMISSION
 - INCORRECT I/O CHANNEL OPERATION
 - INCORRECT PARITY ON RECEIVED DATA
 - INCORRECT TIMING ON RECEIVED DATA
- MULTIPLE I/O COMMANDS
 - IMPROPER I/O CHANNEL SELECT SEQUENCE
- GO/NO-GO COUNTER RUNOUT _ PROGRAM CONTROL ERROR
- SOFTWARE DIAGNOSTIC ROUTINES - GROUND & FLIGHT
 - CPU CHECK
 - MEMORY CHECK
 - I/O CHECK
- UNDER/OVER VOLTAGE DETECTION & STORAGE PROTECTION
- REGISTER CONTENT PRESERVATION

Figure 2.10.1-3.

ALTERNATE EQUIPMENT OR SOURCES

COMPUTER: IBM 47/AP-1

CDC ALPHA MINOR MODS

SINGER SKC-2000

INPUT/OUTPUT: IBM

TELEDYNE NEW

SCI

NEW USING EXISTING
HARDWARE COMPONENTS
& TECHNOLOGY

MUX: TELEDYNE DS-704

SCI - SPIS

MINOR MODS

Figure 2.10.1-4

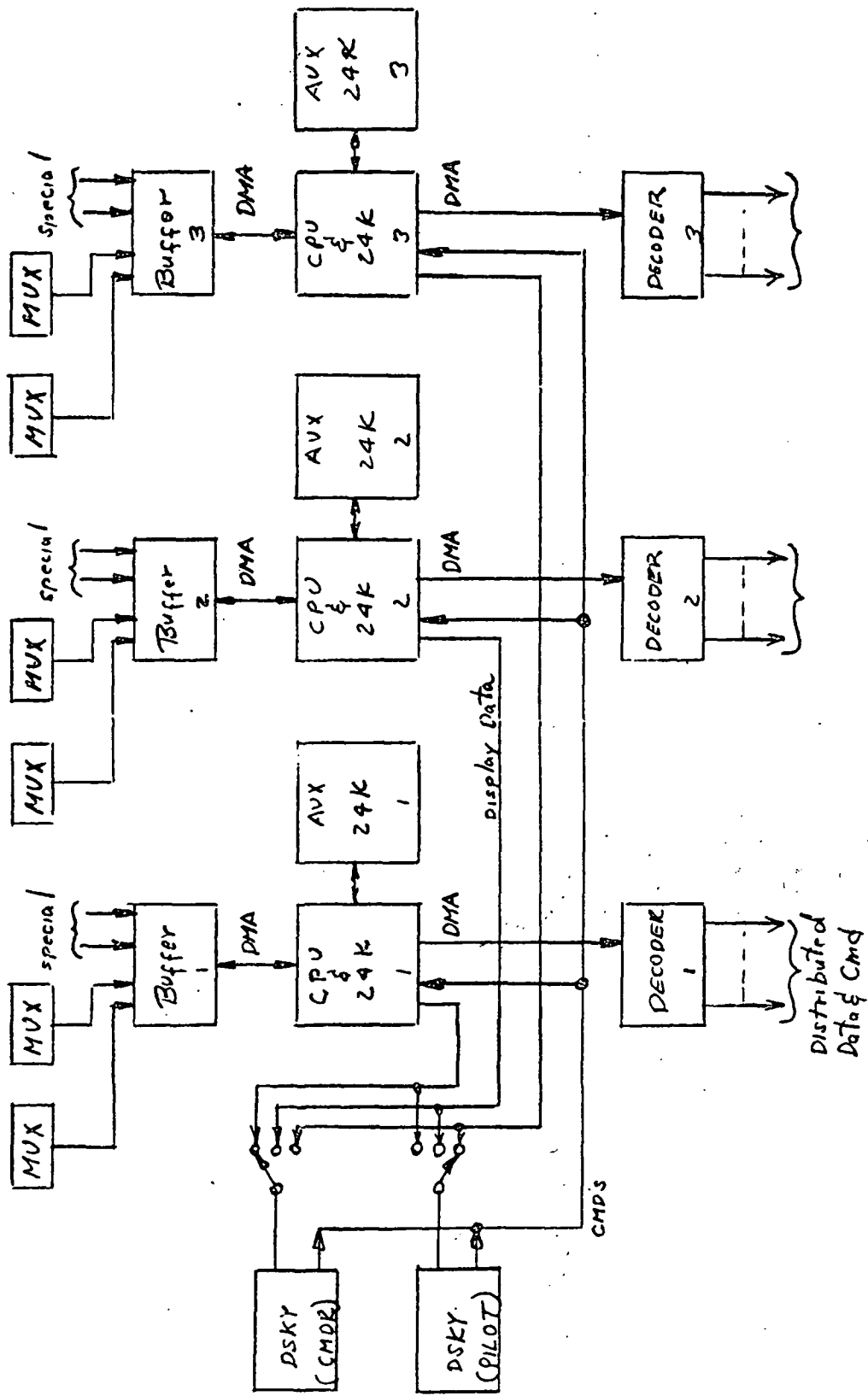


Figure 2.10.1-5. GN&C COMPUTER CONCEPT



data channel, (3) 80 high level analogs, and (4) 50 differential analog signals. The multiplexer's sample rate is under internal control; its output is 8-bit plus parity serial data.

The buffer units accept input data from the multiplexer at the bit rate programmed by the multiplexer programmable read only memory. The buffer stores the input data in an internal memory. Each multiplexer input has a separate starting address. Two multiplexer channels are stored simultaneously (16 bits).

G&N Computer System. The buffers used in conjunction with the G&N computers receive data from the various subsystems via the multiplexers described above. Using input data and the stored program, the computer system performs the required G&N functions. Computer outputs are obtained either under direct software control or under decoder control via direct memory address through the parallel channel. Nominal decoder output capacity is as follows: (1) 15 serial signals, (2) 100 analog signals, and (3) 200 discrettes. Figure 2.10.1-6 describes a possible implementation approach based upon an IBM 477/AP-1 computer system and describes what modification may be required.

The alphanumeric display and keyboard units provide the crew with access to the G&N computers. Computer commands and data requests are first presented on the display so that a crew member can visually verify the data prior to computer entry. The keyboard represents the crew's basic computer interrogative device and can be used typically for the call-up of a particular display and the entry of data into the computer. Information from each of the two keyboards is routed to all G&N computers simultaneously; the source of the data presented on the displays is selected by the crew.

Display Processor System. The Display Processor System is described in detail in Section 2.5.4.

Performance and Equipment Characteristics

Remote Multiplexing Unit (RMU). The selected multiplexer is typified by an off-the-shelf Teledyne Remote Multiplexer Unit (RMU) Model DS-704 with minor modifications. The RMU is the device which interfaces the user subsystems to the GN&C computers, via the appropriate Buffer Unit. It consists of a time division multiplexer, A/D converter, programmable read only memory, and the necessary control/timing logic and modulator circuitry required to putput multiplexed subsystem data into a PCM format for transmission to the Buffer Units. The modification required is the addition of a serial digital input channel(s).

GN&C COMPUTER SYSTEM

EQUIPMENT

COMPUTER (AP-1)
(24K x 32 BIT)

AUXILIARY STORAGE
(24K x 32 BIT)

INPUT BUFFER/
OUTPUT DECODER

MUX (TELEDYNE DS-704)

NOTE: COMPUTER & AUXILIARY STORAGE

INPUT BUFFER/OUTPUT DECODER

MUX

MODIFICATION

ADD: PARALLEL I/O CHANNEL (DMA)
OVERVOLTAGE INTERRUPT
REGISTER PROTECTION
HALF-WORD STORAGE PROTECTION

MOD: MEMORY ADDRESS EXPANSION

MOD: REMOVE CPU FROM BASIC AP-1

NEW: BACK PANEL HARNESS

NEW: SP-1 COMPUTER CASE, BACK PANEL,
PAGE HARDWARE, LOGIC COMPONENTS
SKY-LAB POWER SUPPLY

MOD: ADDED SERIAL INPUT CHANNEL

115V 400 HZ

28 VDC

28 VDC

ALL MIL-E-5400 CLASS II, AIRCOOLED

Figure 2.10.1-6.



The following is a description of the functional capabilities of the RMU.

Inputs:

- . High level (00 to +5VDC) Analog: 80 single ended channels
- . Low/High Level Analog: 50 differential channels
- . Discretes: 96 points
- . Serial Digital: Total input capability is limited to 142 equivalent 8 bit words, i. e. , 1 Analog Input = 1-8 bit word, 8 discrete inputs = 1-8 bit word. The number of serial digital 8 bit words inputted to the RMU will subtract from the existing analog/discrete capability, i. e. , - 1 serial digital word block consisting of 142 - 8 bit words would delete any analog/discrete input capability for that RMU.
- . Format Select: Two lines specifying one of four formats to be read from the programmable read only memory.
- . Power: +28 +10% VDC @ 400. ma.

Outputs:

- . Data Output: 9 bit (8 bit + parity) words being transmitted at approximately 30 Kbps with Bi-phase - L Modulation.

Functional Operation. The RMU will operate according to the mode stored in the programmable read only memory and selected by the Format Select lines. Figure 2.10.1-7 presents a simplified functional block diagram of the RMU. One of four modes can be externally selected which specify: output bit rate, input channel selection, output modulation, output word length, and gain selection for the differential analog input channels. The RMU will then scan the appropriate input channels, perform the required A/D conversion for formatting, add the necessary parity, modulate the resulting serial data stream, and output this data stream for transmission to the appropriate Buffer Unit.

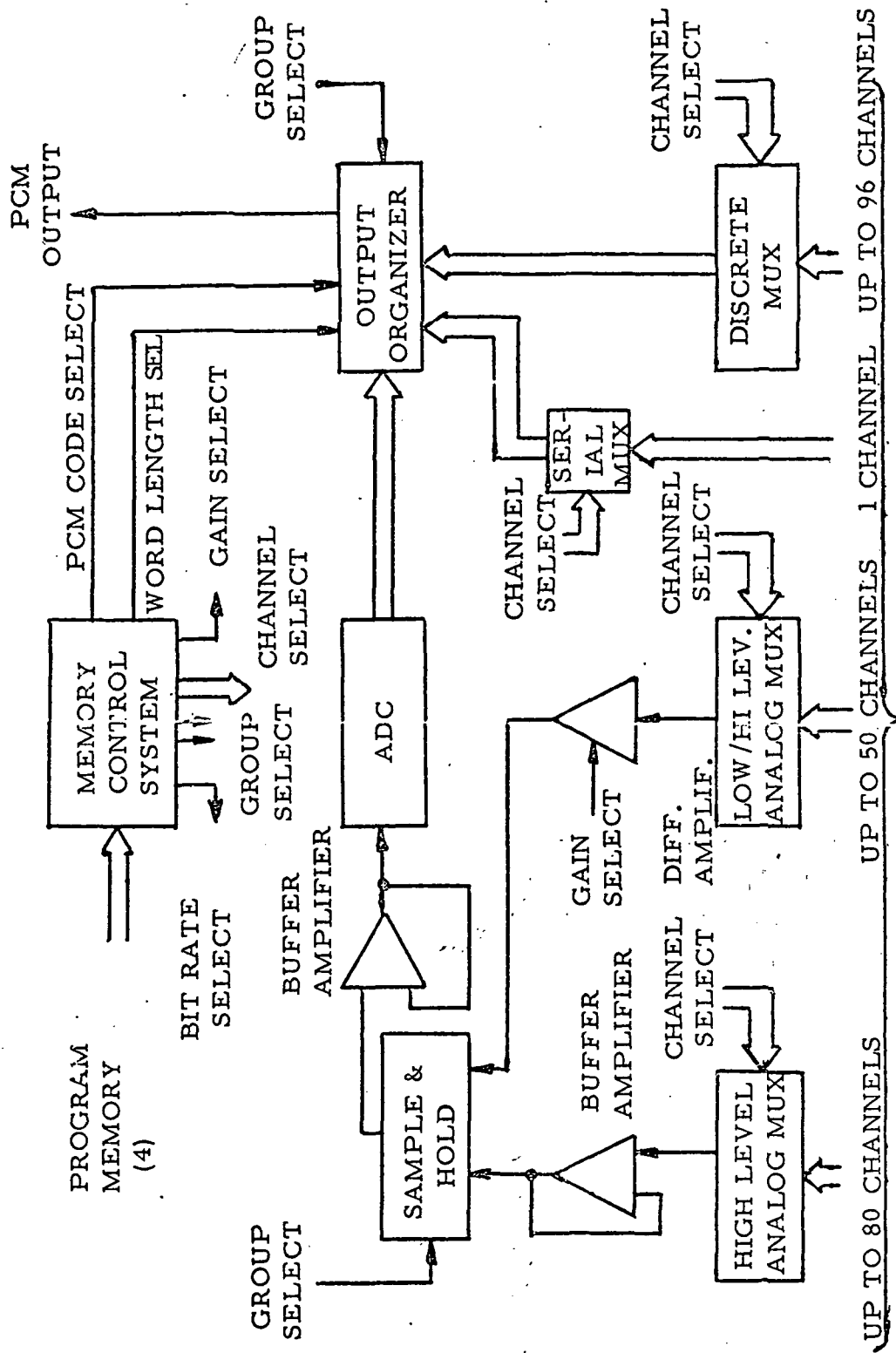
Functional Configuration. Two functional RMU's are provided within one case. These RMU's may be used redundantly by external cross-strapping or to expand the total number of input channels.

Physical Characteristics. (for Dual RMU's)

Power:	20 Watts (both units ON)
Weight:	10 lbs.
Dimensions:	8.5 H x 4.5 W x 3.5 L (Preliminary estimate)
Volume:	0.08 cu. ft.

Figure 2.10.1-7 - SIMPLIFIED BLOCK DIAGRAM, REMOTE MULTIPLEXER

(TELEDYNE DS-704)





Input Buffer. The input buffer contains all of the circuitry and storage necessary to accept, decommutate, and store the inputs from four multiplexers which are outputting serial data streams. In the baseline system only two will be used. The multiplexers are assumed to be free running and operating asynchronously. As each pair of words of data are received from a multiplexer channel, the data will be transferred to a pre-determined location of core storage in the buffer unit, or the computer.

In addition to the four PCM input channels, provisions will be made to custom interface, on an exception basis, those specific inputs where significant cost savings may accrue. Typical candidates for consideration are data inputs from TACAN and Up-Data Link equipments.

The input buffer for the G&N computers will have a buffer-computer interface compatible with the AP-1 computer parallel Direct Memory Access (DMA) channel. The computers will initiate data fetches from the buffer under control of software. By use of an I/O call, a starting buffer address and work count will be transferred and then sequential fetches of up to 24 locations will begin. Transfers from the buffer core to main memory will take place via DMA.

The input buffer also contains circuitry to share the DMA channel with the Output Decoder. An alternate mode of data acquisition still under consideration would operate the data acquisition cycle on a demand basis rather than the continuous PCM telemetry style described previously. In this mode the computer executive would initiate an acquisition cycle timed to be completed just prior to the transfer of data from the buffer memory to the computer main store. This approach would guarantee that all data acquired or transferred from the buffer would be time correlated. In the baseline mode a request occurring in the middle of the repetitive input cycle could possibly result in sets of data from two different time cycles.

A simplified block diagram of the selected input buffer is shown in Figure 2.10.1.8.

Physical Characteristics

Weight	24 lbs.
Dimensions:	4.1 H x 10.1 W x 15.5 D
Volume:	0.25
Power:	100 watts

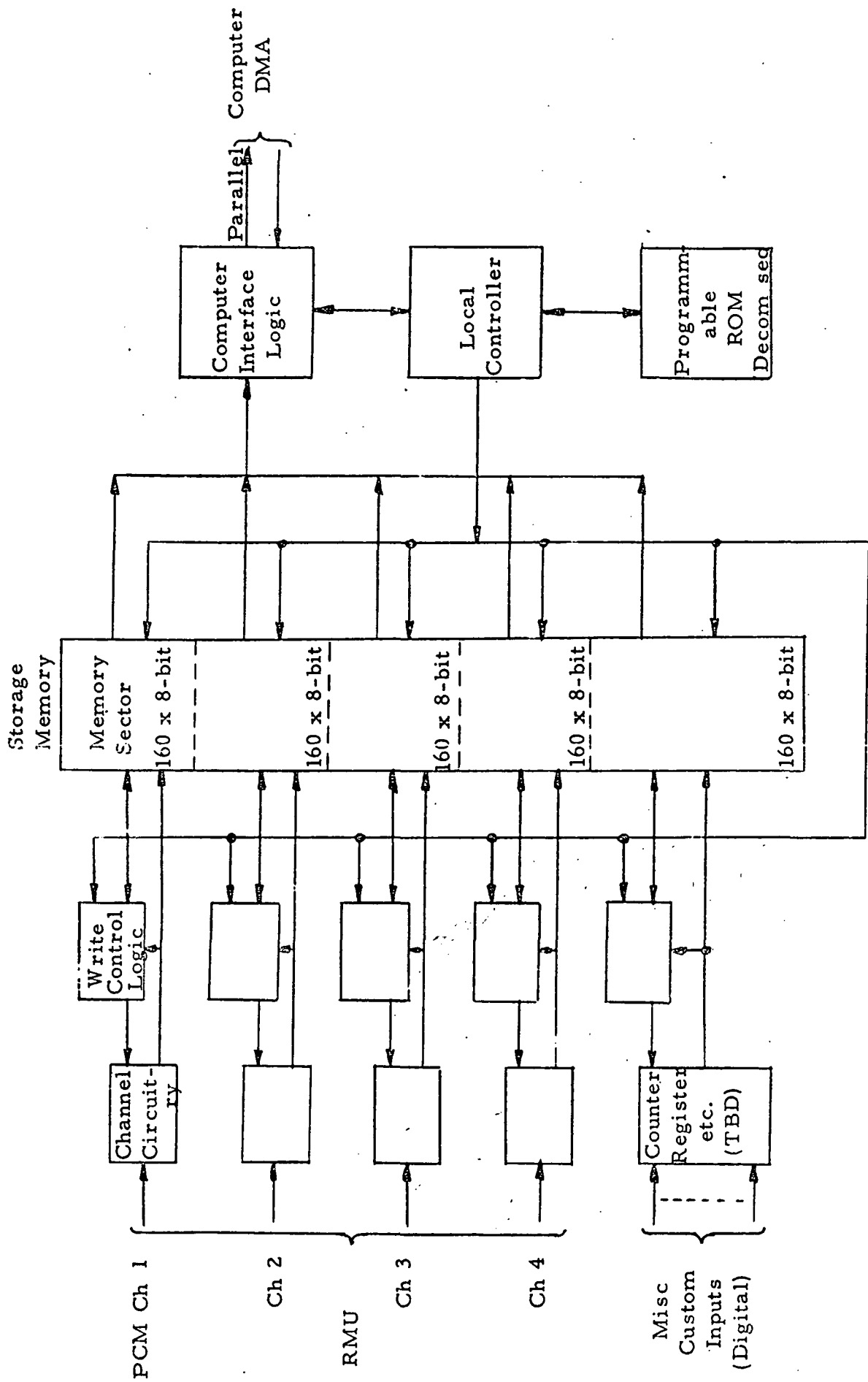
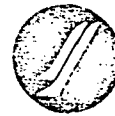


Figure 2.10.1-8 New Baseline Input Buffer



Output Decoder. The output decoder provides a means of distributing and/or formatting the G&N computer output. The output decoder has the following capabilities.

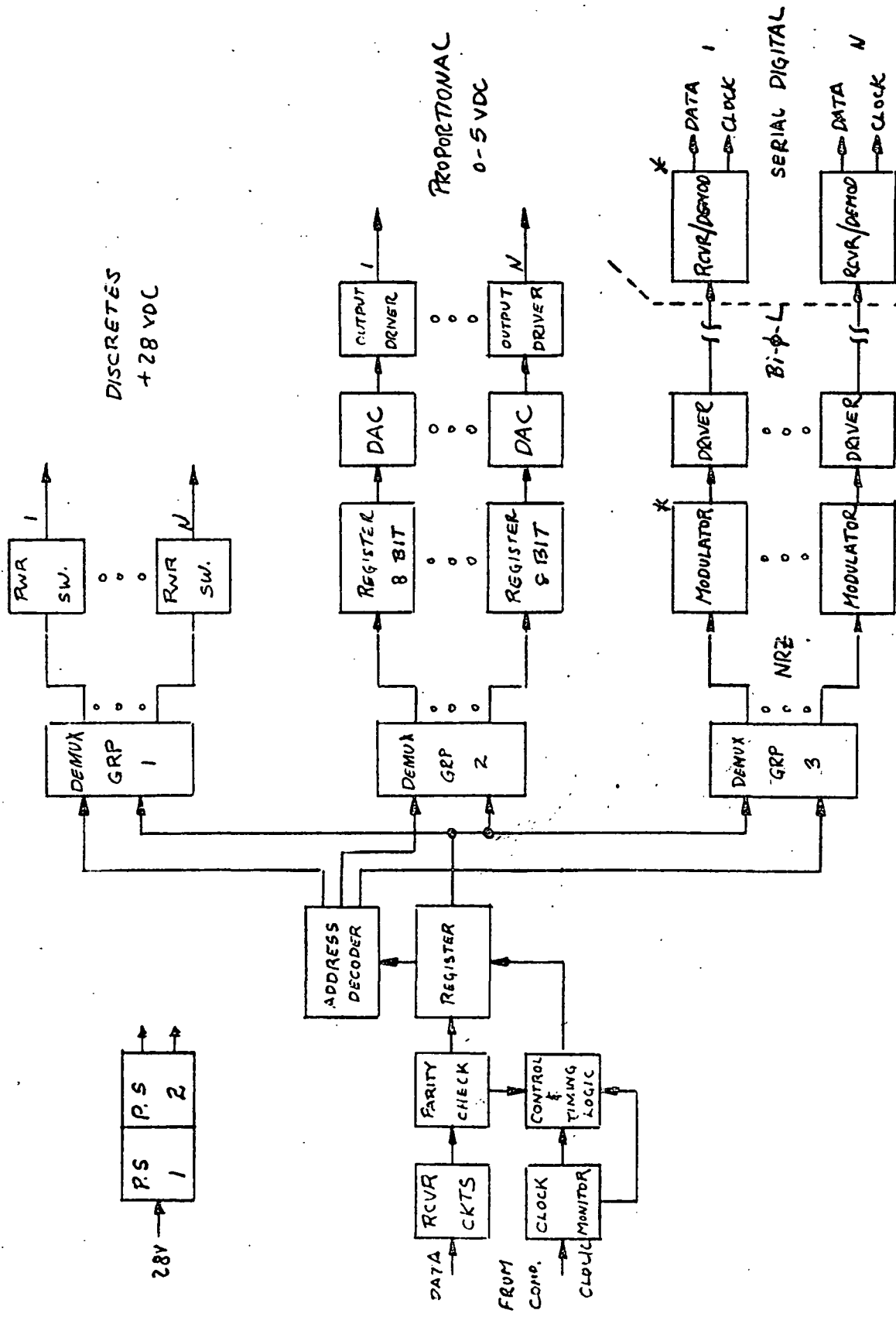
- 16 independent serial channels -- each serial channel is loaded via a direct I/O instruction.
- 128 discrettes ($0 - 5^V$) -- four fixed main memory locations are fetched via the DMA at selected intervals and their contents are used to update the discrete outputs.
- 64 DC analogs ($0 - 5^V$ @ 8 bit accuracy) -- 16 fixed main memory locations (4 outputs per location) are fetched via the DMA at selected intervals and their contents are used to update the digital to analog converters.
- Gyro Pulse Torque Commands to gyro -- a three-channel gyro pulse torquer converts command signals from the computer to precise binary torque pulses which may be used to drive the gyro packages. Each pulse delivered to the gyro represents an increment of angle at the gyro input axis.
- 48 bit register to telemetry -- one and one half fixed memory locations are fetched via the DMA at software controlled intervals and their contents are used to reload the 48 bit register. The register is made available for periodic sampling by the telemetry data acquisition system.

The Output Decoder interfaces with the parallel DMA channel of the AP-1. The DMA facilities of the computer are shared with the Input Buffer.

Figure 2.10.1-9 depicts a simplified block diagram of typical functional paths which may be provided by the Output Decoder.

Physical Characteristics

Weight	27 lbs
Dimensions:	4.1 H x 10.1 W x 15.5 D
Volume:	0.25
Power:	80 watts



* NOT REQUIRED IF NRZ SMR IS ADEQUATE

FIGURE 2.10.1-9 DECODER UNIT (COMMAND GROUPS)



GN&C Computer. The selected configuration cost and configuration analysis utilized an IBM AP-1 computer with selected options for the GN&C function in order to have a specific realistic example, not necessarily for a final source selection.

Although the designation AP-1 has been used in other previously available documents, the AP-1 will be designated AP-101 for the Space Shuttle GN&C application. This is because the basic AP-1 does not include options such as auxiliary storage and half-word storage protect.

AP-1 Characteristics

Type	General purpose, stored program, parallel
Organization	Binary, fixed point, fractional
Data Flow	32-bit parallel
Storage	Random access, nonvolatile, destructive readout, core
Storage Size	49,152 18-bit halfwords (including parity and Store Protect bit) (24K AP-101) 49,152 18-bit halfwords 24K Aux Storage 98,304 18-bit halfwords TOTAL
Storage Cycle Time	1.0 us
Addressable Unit	16-bit halfword
Instruction Word Length	16-bit and 32-bit
Data Word Length	16-bit and 32-bit
General Registers	Two sets of eight 32-bit hardware registers, selectable
Interrupt Facilities	4 externally controlled 12 internally controlled 9 levels of priority



Program Loadable Counters	Two 6-second program loadable countdown counters One 3-second program loadable/readable countdown counter
I/O Channels	1 parallel channel, up to 4 devices per channel One DSKY Adapter Channel

Shuttle Configuration Features

1. Expanded Memory Storage size increased to 98K halfwords
CPU addressing increased to 128K halfwords
2. Store Protect A store protect bit set on the selected halfwords during memory load through the AGE interface prevents accidental overwrite of protected halfwords.
3. Parallel Channel A high-speed parallel channel with support for 4 devices is provided. This to interface with external I/O Buffer and Decoder.
4. DSKY Adapter To interface directly with a customer specified DSKY a customized data channel replaces the standard AP-1 serial data channels.

Expanded Memory (48K)

Main storage expansion to 98,304 18-bit (16 data/instr + 1 parity + 1 STG protect) halfwords (48K fullwords) is accomplished in two modules.

Module 1 consists of the basic AP-1 computer CPU containing 48K halfword (24K fullword) storage, and connector to interface to Module 2. Module 2 contains 48K halfword (24K full word) storage, an additional storage interface logic page (SIL), and necessary power modules. It is referred to as 24K AUX STORAGE.

The combination (module 1 and module 2) appears as 1 computer system with interfacing through Module 1. Two PSW bits (Bit 21 and 22) are used for address expansion. This provides address capability to 64K fullwords.



Memory (Storage) Protect. A store protect bit is provided which prevents accidental overwrite of protected halfwords. The bit is set on selected halfwords during memory load through the AGE interface. It is not program resettable, assuring maximum integrity of the protected data. If an attempt is made to write in a protected area, an interrupt will be issued and serviced by the interrupt routine.

Parallel Channel. A high-speed parallel channel with support for four devices (four pairs of unique request and acknowledge lines) is provided for shuttle application. This channel provides both general register and direct memory access data transfers and allows channel to channel interface and compatibility with other advanced Systems 4 Pi computers.

DSKY Adapter. To match the selected DSKY interface customized input and output data channels will be provided within the AP-101. This will allow DSKY direct access to the AP-101 without going through the external buffer or decoder.

Physical Characteristics

Basic Module	(24K 36-bit memory)
Weight	54
Dimensions	7.6 H x 127 W x 17.2 D
Volume	0.87
Power	245
Auxiliary Memory Module	(24K x 36-bit memory)
Weight	44
Dimensions	7.6 H x 12.7 W x 17.2 D
Volume	0.87
Power	100

Display Processing Computer. The computer candidate considered for display processing evaluation purposes is the IBM SP-1. The following summarizes the SP-1 computer characteristics:



SP-1 Characteristics

Type	General-purpose, stored program, parallel
Organization	Binary, fixed point, fractional, two's complement
Data Flow	16-bit parallel
Storage	Random access, nonvolatile, destructive readout, core
Storage Size	16,384 17-bit words
Storage Cycle Time	1.3 sec.
Addressable Unit	16-bit word
Instruction Word Length	16-bits
Data Word Length	16-bit, including sign, 32-bit double-word, including sign
Registers	4
Instructions	41
Interrupt	One externally controlled
I/O	Externally controlled Direct Memory Access (DMA) 16-bit buffered Program controlled Direct I/O Data Transfer Rate - 150,000 to 600,000 words per second

DSKY Adapter. An additional channel will be provided in the SP-1 I/O to permit data transfer between the SP-1 and its dedicated DSKY. The proposed modification will be similar in nature to that proposed for the AP-101/DSKY interface.

Physical Characteristics

Weight	11.0 lbs
Dimensions	7.32 H x 6.70 W x 8.81 D
Volume	0.25
Power	87 watts



Display and Keyboard (DSKY). The display and keyboard is comprised of a Display Electronics Unit (DEU), Display Unit, and Keyboard.

Display Electronics Unit (DEU)

Functionally, the DEU interfaces with the selected computer; and (1) stores alphanumeric display formats in its 16K, 16-bit memory; (2) provides CRT display refresh; (3) performs the operations necessary to position characters and symbols; (4) provides the appropriate deflection and blanking signals to the Display Unit; (5) decodes keyboard data entry and generates the data on the CRT for verification; and (6) encodes and transmits keyboard/CRT line select control data to the computer.

The interface consists of digital input and output channels compatible with the selected DSKY DEU parts. Data from any of the two GN&C keyboards are entered or made accessible to all three GN&C computers simultaneously. The source of the DSKY display is selectable at the option of each crew member. Each GN&C computer is capable of simultaneously deriving appropriate information for two different display formats although the crew may optionally display identical data from the same or different computer sources.

Further details regarding the DSKY are given in Section 2.5.

2.10.2 Orbiter GN&C Operational Flight Computer Program

The operational flight program for the Shuttle on-board GN&C computer consists of an executive and a set of "mission" modules (see Figure 2.10.2-1 which can be assembled to build up an operational program to accommodate all Shuttle baseline missions. The computer program for the GN&C computer will be devoted to the solution, display and performance monitoring of the Guidance, Navigation and Control problems of the Shuttle Orbiter. Peripheral support such as displays and performance monitoring of non-GN&C orbiter subsystems will be provided by a dedicated processor which is divorced from the GN&C processor.

The GN&C Operational Flight computer program description covers the operational software functions for the orbiter from prelaunch through landing. The GN&C software functions include Guidance, Navigation, Control, Attitude reference, Executive, Display Processing, and status monitoring. The defined functions within the above functions for all mission phases are:

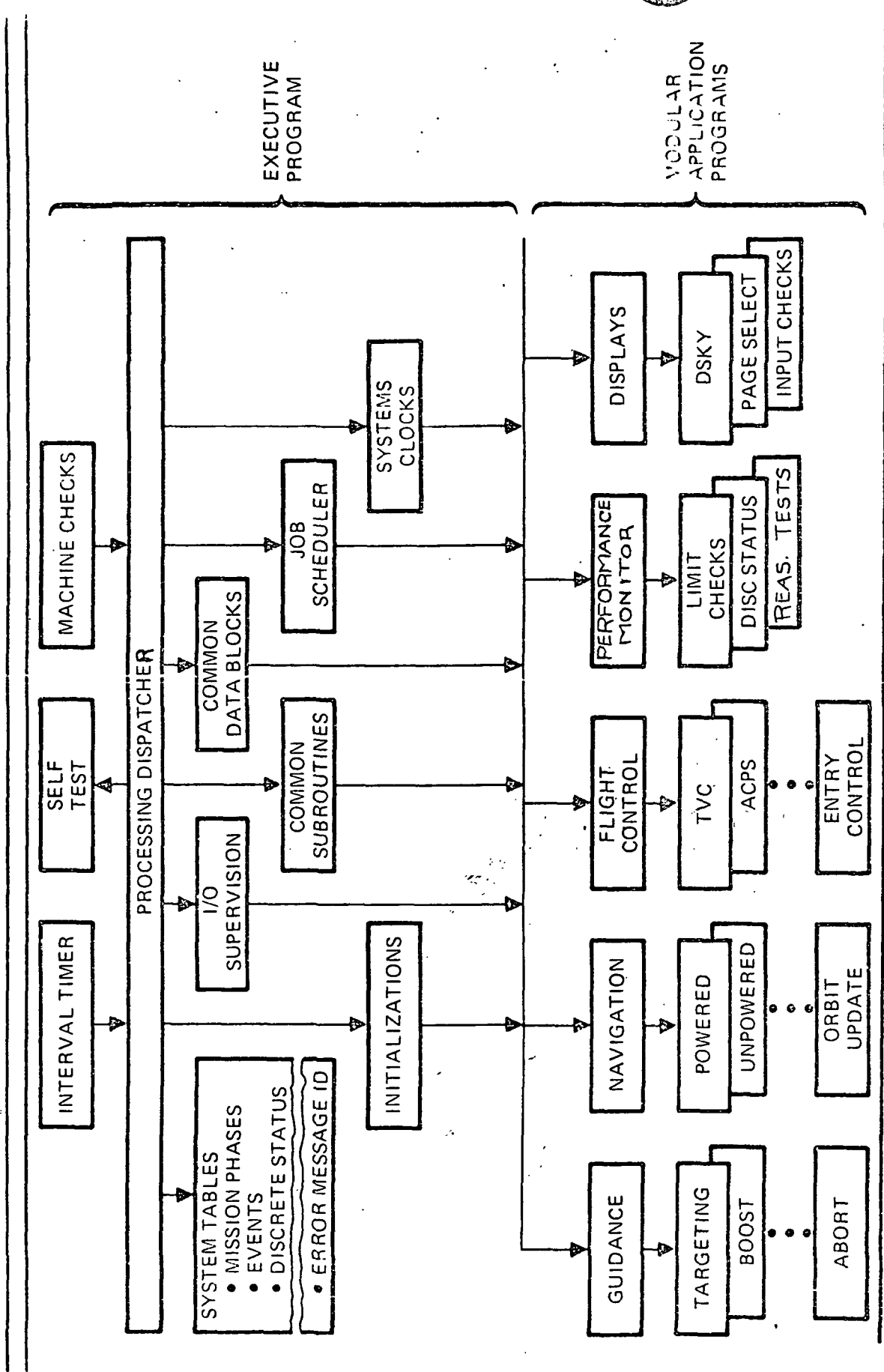


Figure 2.10. 2-1. GN&C Flight Software Program Structure



- . Guidance
 - Prelaunch Targeting
 - Atmospheric guidance during boost
 - Orbit injection guidance
 - Prethrust targeting for rendezvous and deorbit
 - Cross-product steering
 - Entry guidance
 - Approach guidance

- . Navigation
 - Prelaunch alignment
 - Powered and unpowered navigation
 - Earth relative navigation
 - On-orbit updating (horizon sensor)
 - Target navigation
 - Entry navigation
 - Atmospheric navigation (TACAN Updating)

- . Attitude Reference
 - Direction cosine and Euler angle computation
 - Platform updates (Star Tracker)
 - Gyro compassing during prelaunch

- . Flight Control
 - Main engine thrust vector control
 - ACPS Control
 - Thrust on-off control
 - Entry program control

- . Executive
 - Task control
 - I/O control
 - Common routines
 - Tables

- . Display Processing
 - Keyboard interface
 - Display I/O
 - Display format processing
 - Tables



- . Status Monitoring
 - Maintain system status
 - Computer self test
 - Command/response tests
 - Limit checks
 - Reasonableness tests

In addition, the software functions include sequencing and interfacing with systems other than GN&C.

The overall control of the operational program is provided by the executive sub-program. The Executive sub-program provides for the flow of information between programs and the external environment and provides for standardization of interfaces in the presence of a variety of modules and for management of the program resources to assure the response required by the system. In addition, the executive program will provide an internal environment which will permit application programs to be constructed and executed independently of one another and allow a modular program structure which emphasizes operational flexibility, ease of modification, and capability for growth.

All computer program instructions will reside in protected memory locations while all computer variable data will reside in unprotected memory locations. Protected memory is defined as read-only memory while unprotected memory is defined as read-write memory. The protected and unprotected memory locations are predetermined during the program development process and will not change during a mission time span. All unused memory locations will be assigned as protected memory. If any program attempts to write into protected memory, the computer will enter a fail state.

Individual program will be responsible for all internal program calculation scaling and arithmetic manipulations in order to attain sufficient accuracy as required for the Shuttle Orbiter configuration for all mission phases. Extended precision arithmetic functions will be used as required. The fixed point scaling must be sufficient to accommodate the maximum expected values and maintain the required parameter resolution.

Software Elements

The following paragraphs describe design concepts for the on-board Shuttle GN&C Flight Program.



a. On-board Executive System. This section describes the on-board executive control system.

1. Basic Structure. The executive structure will be based on a timer interrupt, fixed schedule, time slice mode of operation. In effect, this type of executive does not require, nor allow, instantaneous response to external interrupts.

The timer interrupt interval will be under program control and will provide the initialization and reference for the basic executive time structure. When the executive detects a timer interrupt, it will process its schedule tables to determine which set of program jobs are to be executed during the next program interval.

All program intervals will be approximately equal in execution time. This will be accomplished by predetermining the time required for the longest path in each job and shuffling jobs to balance the time intervals. Each time interval will contain portions of various rate jobs as required to satisfy the time and rate constraints.

The program intervals occur 25 times per second, based on the maximum system sample rate. Each interval contains all 25 per second jobs, every other interval contains 25 and 12.5 per second jobs, every fourth interval contains 25 and 6.25 per second data, etc.

2. Job Schedule. The Job Schedule function manages the sequence in which programs are executed. System considerations determine the priority of the routines driven by the schedule function. The job schedule function is designed to respond quickly to execution requests on a pre-determined priority basis.

All external interrupts are treated by the scheduler as discrete inputs (ON-OFF). The scheduler will mask the interrupt and sample the interrupt status on a periodic basis. An interrupt detected as ON will result in the scheduler modifying its scheduled jobs or setting a flag to be used by one of the sub-programs.

Internal interrupts such as the Internal Timer or Machine Error interrupts will be handled as true interrupts, as follows:

(a) Internal Timer Interrupt - This interrupt will occur when a pre-set timer counts down. The program will set the time to count down in 40 milliseconds (based on accepted maximum sample rate of 25 per second). This interrupt will cause the program to exit its present scheduled



job and return to the basic scheduler entry point. The program structure will be designed so that this interrupt will always occur during a low priority (background) job or during an idle state.

(b) Machine Error Interrupt - This interrupt indicates that the computer circuitry has detected a problem in the computer hardware. This interrupt will force the program into an error detection self test mode. Verification of the machine error will cause the computer to be shut down. One of the redundant computer systems will take over control of the orbiter vehicle by manual or automatic means.

3. Input/Output Supervision. The input/output supervision function controls all the operations associated with the input/output buffer device. Effectively, the input/output supervisor will input data at interval n to be used in calculations in interval $n + 1$. The output data at interval n is for data calculated during interval $n - 1$. For time critical functions, the input-calculation-output process will be optimized to satisfy the required time constraints.

The input and output data is normally a function of mission phase, sequencing, control panel selection, job scheduling, etc.; as such, the data set is variable and is a function of the jobs schedule by the executive.

The input/output data will be transmitted to or received from the input/output buffer utilizing predetermined unprotected main memory locations.

4. Systems Services. Systems services provides the function of system and applications program interface with the operator of the system. This function will sample all input control media such as control panel switch positions at a predetermined and constant rate, nominally 6 samples per second. If the panel switch inputs are in a changed state for two or more successive samples, the services facility will either change the job schedule to satisfy the inputs or set a flag for usage by the subsystems program.

5. Systems Clocks. This facility will provide the capability of maintaining unique clock facilities for system usage. The clock facility will be utilized for display functions, sequencing, or subsystem usage. System clock accuracy and update rate will be consistent with accuracy constraints on the onboard system. The clock facilities will include Time From Launch, GMT, Launch Local Time, Time to Go to Event, and Time in Mission Phase. Maximum clock update rate for a minimum number of



time bases will be 25 per second. Nominal clock update rate will be 1 per second. The basic time information for calibrating the Software clocks shall be provided by the Master Timing Unit or the Display Keyboard.

6. Common Subroutines. This service facility will contain all common subroutines to be used by the various subprograms. Included in the common subroutines are the standard mathematical functions to determine sine, cosine, tangent, square root, root mean square, matrix inversion, etc. In addition to the standard set of subroutines, this facility will contain other routines that are used by more than one subprogram. The subprograms will branch to the appropriate subroutine as necessary; exit from the subroutine will be to the using subprogram.

7. Common Data Blocks. The Executive program will maintain common data blocks of unprotected memory. The Common Data Blocks will be available for usage by all programs, subprograms or routines. Inclusion of data in the Common Data Blocks implies that data is used across program interfaces. Data that is unique to individual programs such as scratch pad memory, constants, etc., will be maintained in individual local program data blocks which are not accessible to other programs.

8. Self Test. The Self Test function will be executed on a minor cycle basis to determine and verify the operational state of the GN&C computer. This function will be designed to execute a maximum number of computer program instructions and to compare results of mathematical manipulations against a predetermined result. Failure of the Self Test function in a particular computer will result in a shutdown of that computer.

9. System Tables. The executive program structure will be designed to maximize the usage of System Tables in conjunction with simple subroutines for execution of the System Table data. The System Table concept simplifies the programming, checkout and validation of the GN&C Flight Program in addition to providing a relatively simple means of incorporating program changes and updates.

b. Guidance, Navigation, and Control Programs. The GN&C software consists of the operational software functions from prelaunch through landing. The GN&C software functions include Guidance, Navigation, Control, attitude reference, Executive, Display processing, and status monitoring. These functions are required for both the orbiter and the booster of the flyback (LOX/RP) configuration.



However, a pressure fed booster which is controlled by the orbiter and which is towed to the launch site after a free fall return would not require any software functions; the orbiter GN&C computer would be augmented to provide the pressure fed booster control during the mated boost phase.

The LOX/RP booster configuration contains its own GN&C computer with the same functions as the orbiter during prelaunch, mated boost, and separation. A period of booster coast and return to atmospheric flight requires computer Guidance, Navigation and ACPS control. These functions are performed by the computer until the pilot initiates manual control.

Table 2.10.2-1 summarizes the orbiter GN&C program functions by mission phase.

c. GN&C Display Systems Software. The Display subprogram in the GN&C computer provides the capability to selectively support specific crew requested displays. This support includes arithmetic conversion, BCD conversion, formatting and transmission of GN&C data to the crew station displays. For output data, the display support software has the capability to process one parameter per line for twelve lines on each of fifty pages. In addition, the display software also processes input data passed from the crew station to the GN&C computer via an alphanumeric keyboard. The processing of the input data includes interpretation, conversion, limit testing and reasonableness testing. The input processing function is capable of processing 100 parameters. The display system subprogram is capable of generating alphanumeric display messages only, i. e., a vector symbol generation capability does not exist in the display hardware.

1. Display System Subprogram. The Display System Software performs the function of presenting the Orbiter GN&C system status information to the crew via a combination CRT and keyboard (DSKY) display. The DSKY consists of numeric and special character keys which the crew will use to communicate with the Display subprogram. The CRT display will accommodate 14 lines of alphanumeric data with 24 characters per data line. The 14 lines of data are dedicated to 1 title line, 12 measurement data lines, and 1 scratch pad line. Fifty CRT page formats, each containing 14 lines of data, are required to accommodate the GN&C display requirements.



Table 2.10.2-1. Orbiter Mission Phase Requirements Summary

<u>PHASE</u>	<u>GN&C FUNCTIONS</u>
Prelaunch	Targeting Update Powered Navigation Prelaunch Nav Update Direction Cosines Computations Prelaunch Gyro Compassing
Mated Boost	Powered Navigation Earth Relative Navigation Direction Cosines Computations Open Loop Guidance Closed Loop Guidance Abort Guidance
Separation/Orbit Insertion	Powered Navigation Earth Relative Navigation Separation Guidance Abort Guidance Injection Guidance Separation Control Thrust Vector Control Thrust On-Off Throttle Control Direction Cosine Computations Euler to Body Transforms
Rendezvous/Unpowered	Orbital Navigation Direction Cosine Computations Update Platform Attitude Euler to Body Transform ACPS Control Thrust On-Off Control Rendezvous Targeting Prethrust Maneuver Targeting Targeting Conic Routines
Rendezvous/Powered	Powered Navigation Direction Cosine Computations Euler to Body Transform Thrust On-Off Control Cross Product Steering
Docking	Unpowered/powered Navigation Manipulator Arms Target Relative Navigation Direction Cosine Computations Euler to Body Transform ACPS Control



Table 2.10. 2-1. Orbiter Mission Phase Requirements Summary (Cont.)

<u>PHASE</u>	<u>GN&C FUNCTIONS</u>
Deorbit Maneuver Sequence (Unpowered)	Unpowered Navigation Earth Relative Navigation Direction Cosine Computations Euler to Body Transform Attitude Update Deorbit Targeting Target Conic Routines Prethrust Maneuver ACPS Control
Deorbit Maneuver Sequence (Powered)	Powered Navigation Earth Relative Navigation Direction Cosine Computations Euler to Body Transform Thrust Vector Control Thrust On-Off Control Cross Product Steering
Entry	Powered Navigation Earth Relative Navigation Direction Cosine Computations Body Axis Error Computations Attitude Control
Aerodynamic	Approach Guidance Earth Relative Navigation Atmospheric Nav/Aids Updates Aero Flight Control Attitude Reference



2. Page Formats . The 24 characters for each of the 12 measurement lines is subdivided into a group of 12 characters for measurement identification, 5 characters (including decimal point) for data, 4 characters for units, and 3 blank separator characters. The page skeleton information containing the page title, measurement ID's and units is contained in the DEU memory. The DEU memory contains 16K, 16 bit words. The memory required to perform data display calculations is included in the GN&C Display program.

3. Display Selection The two keyboards are each dual redundant. Data entered on each DSKY is directed toward each of the 3 GN&C computers simultaneously - so that all programs receive the same input data. Either display can be switched to read the outputs of any one of the 3 GN&C computers. The program structure is designed such that a different page of display data can be directed toward each of the CRT displays. Either DSKY can be coded to direct either a common or a unique display to each of the CRT displays.

As data is entered on the DSKY, the DEU will interrogate the DSKY and display the keyed data on the scratch pad line of the CRT display. The program will not react to keyed data requests until the Enter button is depressed.

4. Input Data Checks . The program will perform input data tests on a selected set of DSKY input data. The tests will attempt to insure that erroneous or unrealistic data is not accepted by the program. In the event of a failure of the input data tests, the program will notify the crew of the failure condition and will not act on the entered data. The crew has the option of overriding the test results and forcing the program to accept the data. The input data checks include 50 limit tests and 50 reasonableness tests.

5. Display Types . The program will provide for 50 unique data page displays where each data page contains 12 system measurements, 1 title line and 1 scratch pad line. The 50 display pages are further defined as follows:

(a) 15 Systems Status Display Pages - To provide for 125 GN&C system measurements including 40 analog, 60 discrete and 25 system status measurements.

(b) 35 Parametric Data Displays - To provide a capability for displaying approximately 400 measurements, internal computer calculations, or program constants. The 400 lines of parametric display data are not presently defined.



6. Failure Indications. Any system failures detected by the GN&C Flight program will be presented to the crew via a caution and warning indication in addition to a failure message on the scratch pad line of the CRT display. The scratch pad line will inform the crew of the failure type and present a page number to be entered via the keyboard to obtain a full set of CRT display data for the failed subsystem. Multiple failure indications will be provided for by the program.

d. Performing Monitoring Software

1. Analog Processing - A Performance Monitor module will be provided to verify that all GN&C system analog parameters are within a predefined set of high/low limits. An out-of-tolerance condition will cause notification to be made to the crew via subsystem annunciators and/or CRT displays. The current estimate is that 40 analog parameters will be processed by the routine. Each parameter will have four sets of high/low limits as an average. One set of these limits will be active during the various flight phases or modes.

The GN&C computer control program will perform all input checks at either a 25/second or 2/second rate. The input data sources are the input buffers:

Upon completion of the input via the executive program, the control program will dispatch the Performance Monitoring tasks. After routine initialization, active interrogation of the current parameter value to the active limits is initiated. If no parameter is found out of tolerance, all parameters will quickly be processed and control passed back to the control program.

If a parameter is found out of tolerance, a test is made to determine if the parameter has been out of tolerance "n" times. The currently assumed value of "n" is two. This test prevents noise from causing superfluous no-go indications. If "n" successive conditions have been encountered, the type of crew notification required is interrogated. If an annunciator activation is specified, an 8-bit annunciator address code is placed on the control program output data list for transmission to the annunciator panel at the end of the processing cycle.

Measurements that are out of tolerance can be designated as disregard for error notification by the crew. This capability is useful for stopping continuous error displays caused by invalid measurement readings. The disregard designation is made by requesting display of the format containing the measurement in error and utilizing the line designator switch on the DEU to identify the measurement. A cancel disregard designation is



made in a similar manner.

If a CRT display is specified several actions occur. First, an indicator is set in the format skeleton associated with the out of tolerance parameter. When this format (page) is displayed, the appropriate parameter will have appended the out of tolerance flag which allows the crew to quickly see those out of tolerance conditions associated with the subsystem. For analog measurements, the flag will indicate the direction, high or low, of the out of tolerance condition.

The routine will then request a flag message display which will appear on the scratch pad (line 14) of the display the next time the Display Support routine updates the current display. The message will notify the crew that Performance Monitoring has detected an out of tolerance condition and will specify the page number on which the condition can be viewed. If the scratch pad is currently being used by the crew to enter a command, the display will be delayed until the area is free. A related annunciator will notify the crew of the condition. The scratch pad line may then be cleared via the keyboard to free it for the error message.

In addition, a one line message is added to a queue of messages that reflect a summary of all out of tolerance conditions existing at the current time. These one line messages are displayed when the crew requests the Performance Monitor Status page and will reflect the 12 most recent out of tolerance conditions. Earlier malfunctions may be viewed by "paging" through the Performance Monitor Status displays via a special keyboard command. Provision will be made to accommodate up to 48 of the one line malfunction messages (4 x 12).

If both panel and CRT crew notification action is specified for a particular parameter, both of the above described panel and CRT sequences occur.

When a parameter that has previously been out of tolerance comes within tolerance, the converse of the above steps is performed. The annunciator is reset and/or the one line message in the Performance Monitoring Status format is purged and the remaining entries compressed. The crew notification scratch pad line message is removed if applicable.

The above discussion described the comparison of the current parameter value to the active limits. The active limits are established by a separate module identified as the Performance Monitor Phase Initializer. This module is called by the control program upon flight phase changes, or as commanded by the crew. Additionally, measurement limit changes required for vehicle status change (e.g., engine firing) will be effected by



monitoring discrete signals for the change in status. The Performance Monitor Phase Initializer will have tables that associate the phase to the appropriate limits to use for each of the parameters for that phase. The tables have been sized to accommodate an average of four sets of high/low limits.

2. Discrete Processing - The Performance Monitor module will be provided to verify the correct status of certain subsystem input discretets. Similar to the analog processing module, if a discrete is found to be in an incorrect state, the crew will be notified via panels and/or CRT displays.

The current assumption is that 60 discrete parameters will be monitored at a 25/second or 2/second rate.

The Executive control program will perform all input operation and pass control to the discrete processing module. After module initialization, groups of 16 discretets will be tested to determine if any one of the 16 discretets has changed. If none have changed, other groups are tested until the scan is complete.

If a discrete has changed a table entry is interrogated to determine the type notification specified. Again, this can be via annunciators, CRT, or both. If annunciator notification is specified, an eight bit annunciator address code and an associated on/off bit are queued for output.

If CRT notification was specified, an "allow/disallow" table is interrogated to determine if the change was allowable. If so, no additional action is taken and the scan of other discretets continues. Presumably, the discrete for the current phase is not considered a part of the performance monitoring. If the discrete change is not allowed and is currently in the incorrect state, the same sequence as was done in analog processing is performed. The "out of tolerance" indicator is set in the format skeleton on which the discrete appears, a notification is placed on the scratch pad line, and a one line summary message is inserted in the Performance Monitoring Status format buffer. The use of this presentation is the same as described in the analog description. If the discrete change is to the correct state the converse of the above sequence is performed.

The initial discrete status and the control routines access to the proper "allow/disallow" and "should be" tables is established by the Performance Monitor Phase Initializer. This is the same routine mentioned earlier in the analog processing section. A set of allow/disallow and should-be tables are provided for each flight phase.



e. Memory Size and Speed Estimates

The memory size and speed estimate methodology is presented in Figure 2.10.2-2. The memory and speed estimates generated during the Phase B study effort were utilized where practical. Where system requirements were different for the Phase B¹ study, the following methodology was pursued.

1. Program requirements were generated to define the GN&C system operation during various mission phases.
2. Program modules, flowcharts, mechanizations and logic were defined to satisfy the system program requirements.
3. The program sizing methodology was reverified by comparison against an existing program. Using flow charts from a Saturn program, estimates were made to determine memory and speed requirements to accommodate the Saturn phase requirements. Comparison of the estimated values against the actual programs showed a maximum variation of 15% for individual subprograms and 5% for the cumulative programs.
4. Using the previously defined program requirements, the GN&C subcontractor and the software subcontractor made independent estimates of the memory and speed requirements. The independent estimates for the total program were within 10% of each other. The selected Orbiter GN&C memory size and speed requirement as depicted in Figure 2.10.2-3 was a comparison between the subcontractor estimates. The memory and speed driver functions are as illustrated in Figure 2.10.2-4. Figure 2.10.2-5 depicts the Booster (LOX/RP) memory and speed estimates.
5. Memory size and speed estimates for the Docking and Cargo Handling System (see section 2.12) are not included in the following tables. These estimates will be derived in subsequent studies. It is anticipated that this function will require approximately 1,000 32-bit words.

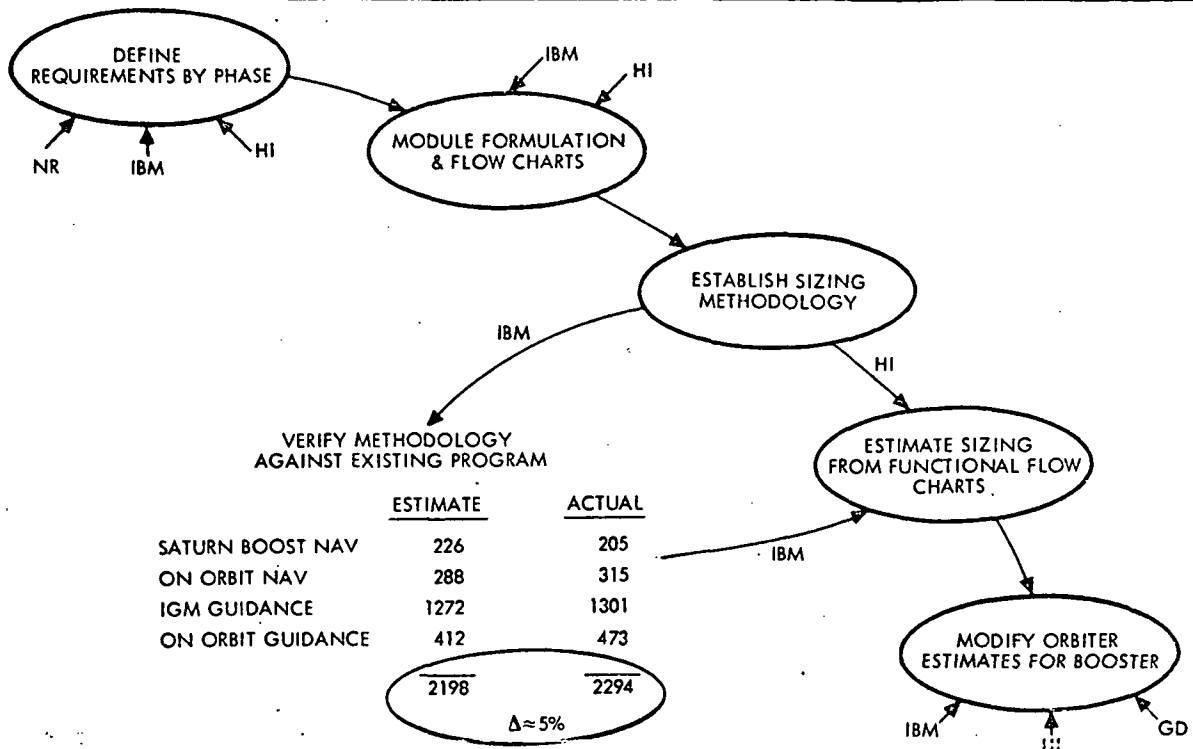
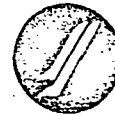
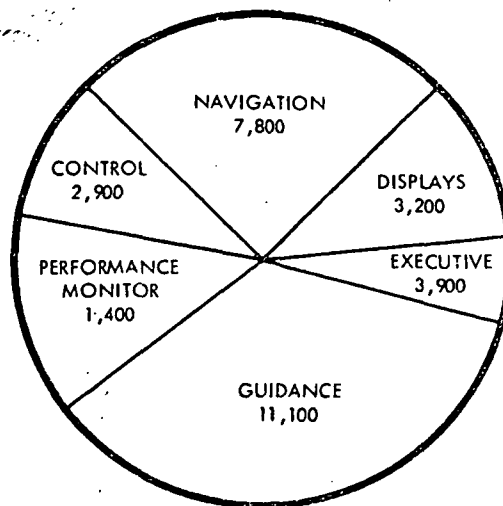


Figure 2.10.2-2. GN&C Memory Sizing

SPEED (KADS)	
BOOST	60
ORBIT INSERTION	85
RENDEZVOUS	55
REENTRY	80
AERODYNAMIC	55



SPEED REQUIREMENTS = 85 KADS

MEMORY REQUIREMENTS = 30,300 32 BIT WORDS

Figure 2.10.2-3. GN&C Orbiter Flight Software

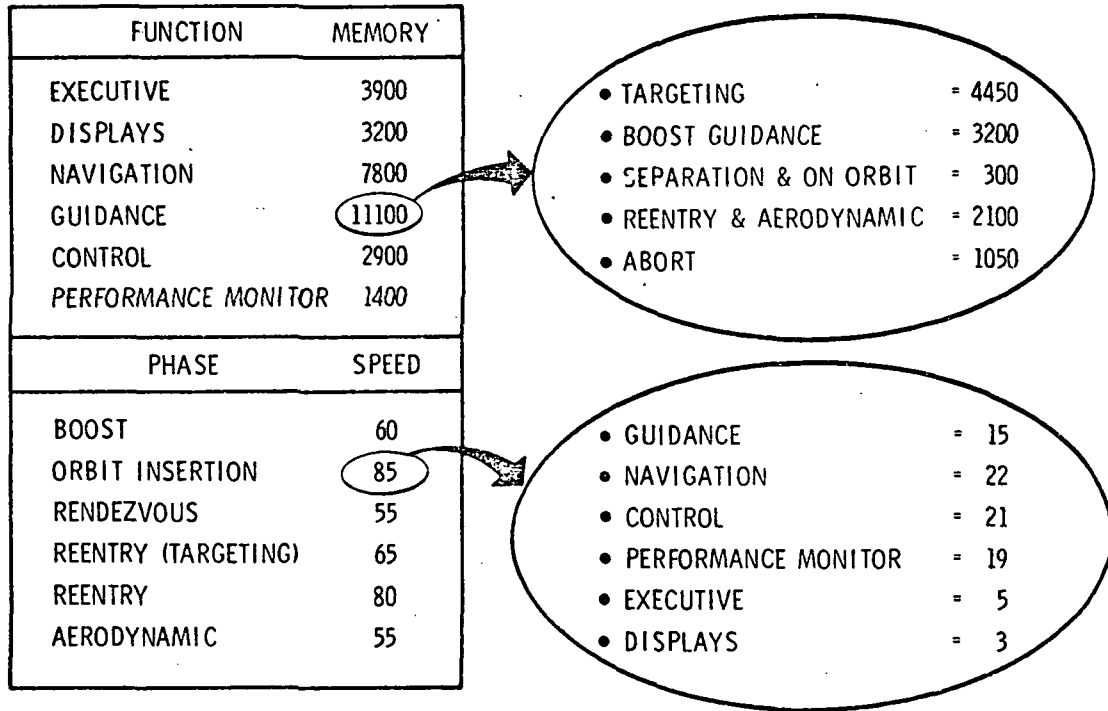
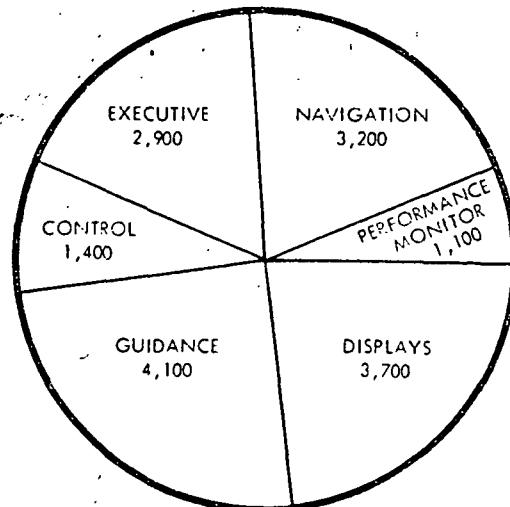


Figure 2.10.2-4. GN&C Memory & Speed Driver Functions

SPEED (KADS)	
PRELAUNCH	40
MATED BOOST	65
REENTRY	60



SPEED REQUIREMENTS 65 KADS

MEMORY REQUIREMENTS = 16,400 32 BIT WORDS

Figure 2.10.2-5. GN&C Booster Flight Software Memory & Speed Estimates



2.10.3 Display Processor Computer Software

Software shall be provided to execute in the Display Processor Computer and shall perform the following functions:

- . Limit check analog parameters
- . Verify correct discrete status
- . Communicate results of the above two functions to CRT and panel annunciators
- . Display crew selected subsystem status formats on CRT

The combination of the above software functions along with the assumed hardware configuration will allow the crew to verify and make judgments as to subsystem status. Figure 2.10.3-1 shows the equipment configuration that is assumed.

The software shall consist of a minimal control program, limit checking routines, display support routines and tables. Additionally, pre-defined format skeletons shall be supplied which are stored in the Display Electronics Unit (DEU).

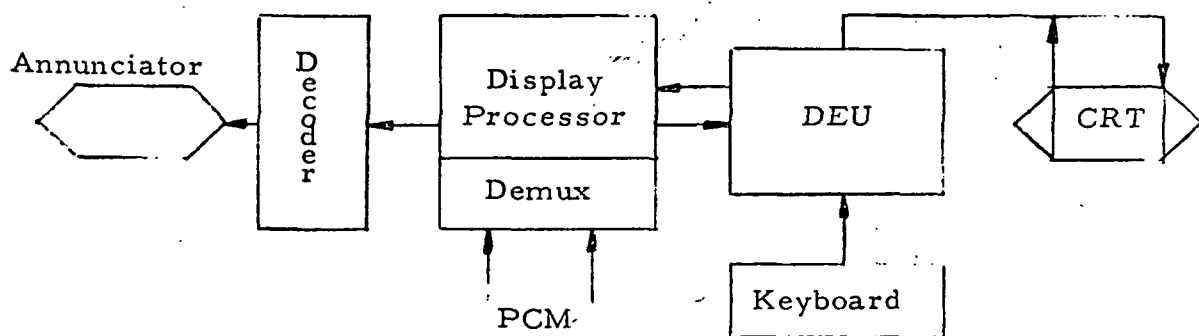


Figure 2.10.3-1 Stored-Program Processor Approach



On-Board Display Processor Software Elements

The following paragraphs functionally describe the on-board display processor software programs and present size data for each program. Supporting assumptions and rationale are included as required.

Display Processor Control Program. The control program for the display processor schedules the display processor software tasks for execution during a predefined fixed time slice processing cycle occurring every TBD ms, or at a preselected integral PCM frame count. The display processor cycle is shown in Figure 2.10.3-2. Two control program subprograms cycle control and data input, are executed prior to dispatching the performance monitor and display processing applications programs. Execution of data output and self-check subprograms complete this cycle. The control program subprograms are discussed in the following paragraphs.

1. Cycle Control - The cycle control function determines major cycle timing. It is presently planned that the occurrence of PCM master frames will be counted and that the processing cycle will be entered after that count reaches a value of two (a measurement sample rate and program iteration rate of 2/second was assumed for sizing purposes). An alternate approach is to update a software time clock in the cycle control routine, and to initiate execution of the major cycle when that clock count reaches a preselected (presently 500 millisecond) value. In either case, the counter (or clock) is reset immediately before the major cycle is entered.

2. Data Input - Data input processing consists of initiating an I/O sequence to read the required PCM measurements into the display processor main storage. Discrete measurements are assumed to be 1 bit each, and analog measurements are assumed to be 8 bits each. For sizing purposes, it was assumed that 300 analog and 300 discrete measurements are obtained for performance monitor and display processing.

3. Data Output - The data output subprogram transfers the display update data to the DEU and transfers on/off signals to the subsystem annunciators as determined by the performance monitor software. DEU and annunciator output lists are generated by the display processing and performance monitor application programs, respectively. These lists are processed by the control program to effect data output.

4. Self-Check - The self-check subprogram is executed as the last task in each processing cycle. Main storage, internal CPU, and I/O functions are tested. Anomalies detected by self-check are communicated by an external discrete indication and/or an error code to the DEU.

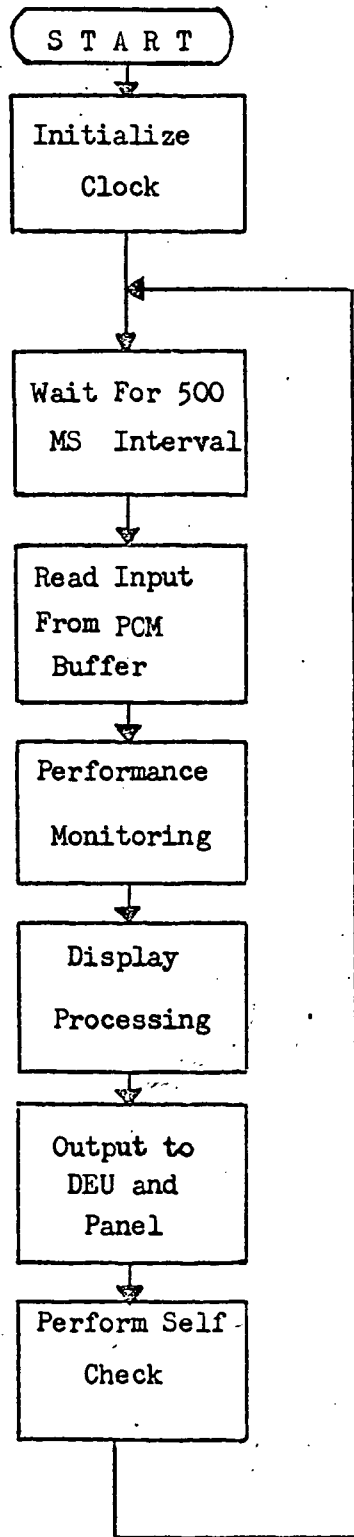


FIGURE 2.10.3-2 - DISPLAY PROCESSOR OVERALL FLOW

ENTER FROM
CONTROL PROGRAM

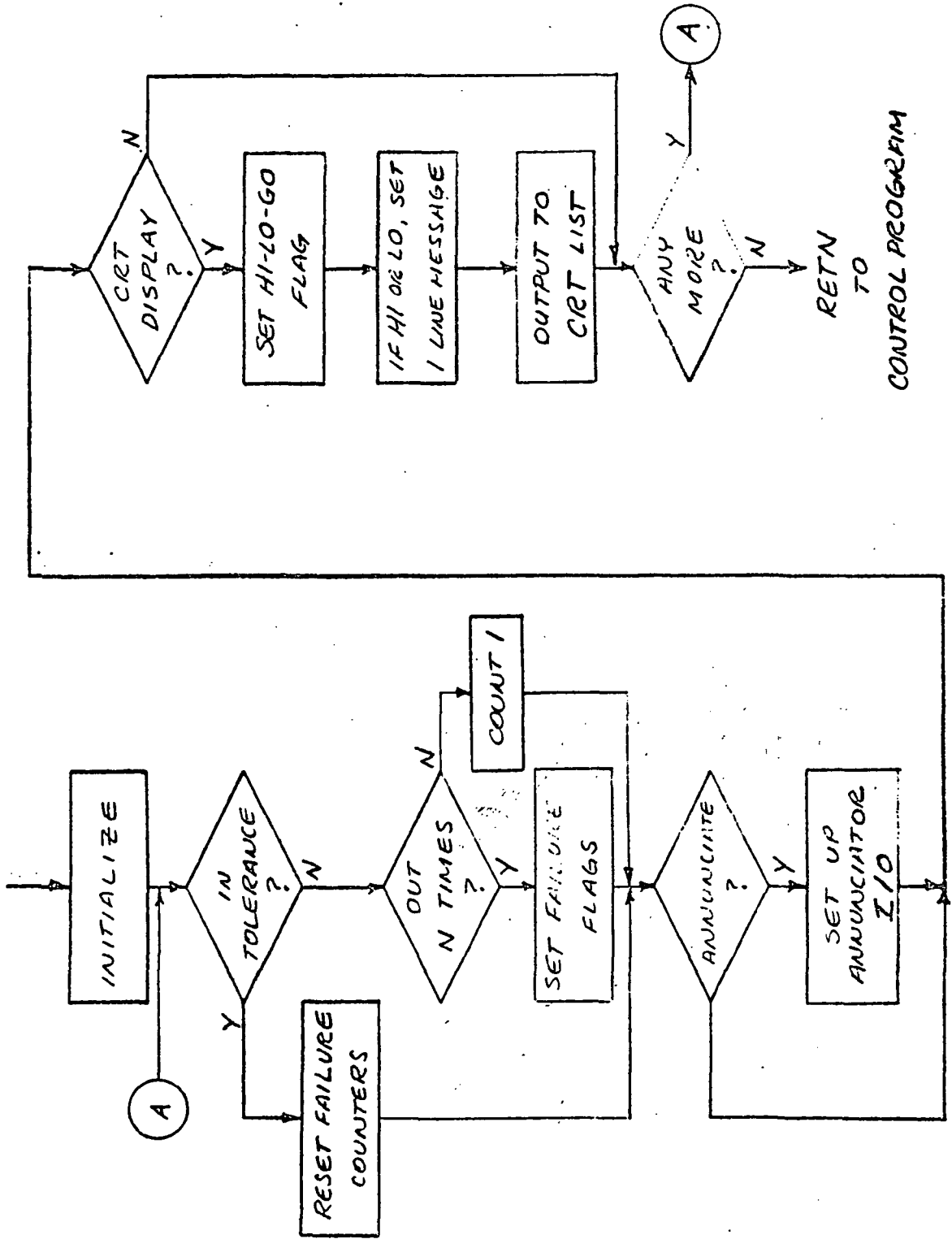


FIGURE 2.10.3-3 ANALOG PROCESS MONITOR



Performance Monitoring Software.

1. Analog Processing - A performance monitor module shall be provided to verify that all non-GN&C subsystem analog parameters are within a predefined set of high/low limits. An out-of-tolerance condition will cause notification to be made to the crew via subsystem annunciators, CRT display, or both annunciator and CRT. The current assumption is that 300 analog parameters will be processed by this routing. Each parameter will have three sets of high/low limits, on the average. One set of those limits will be active at a time.

The control program will input an entire set of the selected PCM measurements at a fixed predetermined rate, as described earlier. The input data source is the OFI Master Controller. Upon completion of the input routine, the control program will dispatch the performance monitor tasks.

After routine initialization, the analog measurements will be converted to engineering units, and in that form will be tested with respect to the active limits. Refer to the general program flow shown in Figure 2.10.3-3. If no parameter is found out of tolerance, all parameters will quickly be processed and control passed back to the control program. If a parameter is found out of tolerance a test is made to determine if it has been out of tolerance "n" consecutive times. This test prevents noise from causing superfluous no-go indications. If the error count has not yet reached "n" for that parameter, its error counter is incremented and processing of other parameters continues. If the error count had reached "n", the type of crew notification required is interrogated. If an annunciator activation is specified, an 8-bit annunciator address code is placed on the control program output data list for transmission to the annunciator panel, at the end of the major processing cycle.

Measurements that are out of tolerance can be designated as "disregard for error notification" by the crew. This capability is useful for stopping continuous error displays caused by invalid measurement readings. The disregard designation is made by requesting display of the format containing the erroneous measurement, and utilizing the line designer switch on the DEU-CRT assembly to identify the measurement. A "cancel disregard" designation is made in the same manner.

If a CRT display is specified several actions occur. First, an indicator is set to identify the no-go condition as "high" or "low". The routine then sets up a message for line 14 of the current display, which specifies the format number associated with the faulty parameter. The message notifies the crew that an out of tolerance condition has been detected and will specify the page number on which the condition can be viewed. If the scratch pad (line 14) is currently in use, the display will be delayed



until the area is free and a discrete notification of the conflict will be issued to the DEU. Next, a one-line message is added to a queue of messages at the current time. These one-line messages are displayed when the crew requests the Performance Monitor Status page, and will reflect the 12 most recent out of tolerance conditions. Earlier malfunctions may be viewed by "paging" through the four-page Performance Monitor Status displays via a special keyboard command. Provision will be made to accommodate up to 48 of the one-line malfunction messages (4 pages x 12 measurements per page). If both panel and CRT crew notification action is specified for a particular parameter, both of the above described panel and CRT sequences occur.

When a parameter that has previously been out of tolerance comes within tolerance, the converse of the above steps is performed. The annunciator is reset and/or the one line message in the Performance Monitoring Status format is purged and the remaining entries compressed. The crew notification scratch pad line message is removed if appropriate.

The above discussion described the comparison of the current parameter value to the active limits. The active limits are established by a separate module identified as the Performance Monitor Phase Initializer. This module is called by the control program upon flight phase changes, or as commanded by the crew. Additionally, measurement limit changes required for vehicle status change (e. g., engine firing) will be effected by monitoring discrete signals for the change in status. The Performance Monitor Phase Initializer will have tables that associate the phase to the appropriate limits to use for each of the parameters for that phase. The tables have been sized to accommodate an average of three sets of high/low limits.

2. Discrete Processing - A Performance Monitor module shall be provided to verify the correct status of certain subsystem input discretets. Similar to the analog processing module, if a discrete is found to be in an incorrect state, the crew will be notified via panels, CRT or both panel and CRT.

The current assumption is that 300 discrete parameters will be monitored at twice per second rate. The Discrete Performance Monitor module is briefly described. Refer to Figure 2.10.3-4 for the general program flow.

The control program will perform all input and pass control to the discrete processing module. After module initialization, groups of 16 discretets will be tested to determine if any one of the 16 discretets has changed. If none have changed, other groups are tested until the scan is complete.

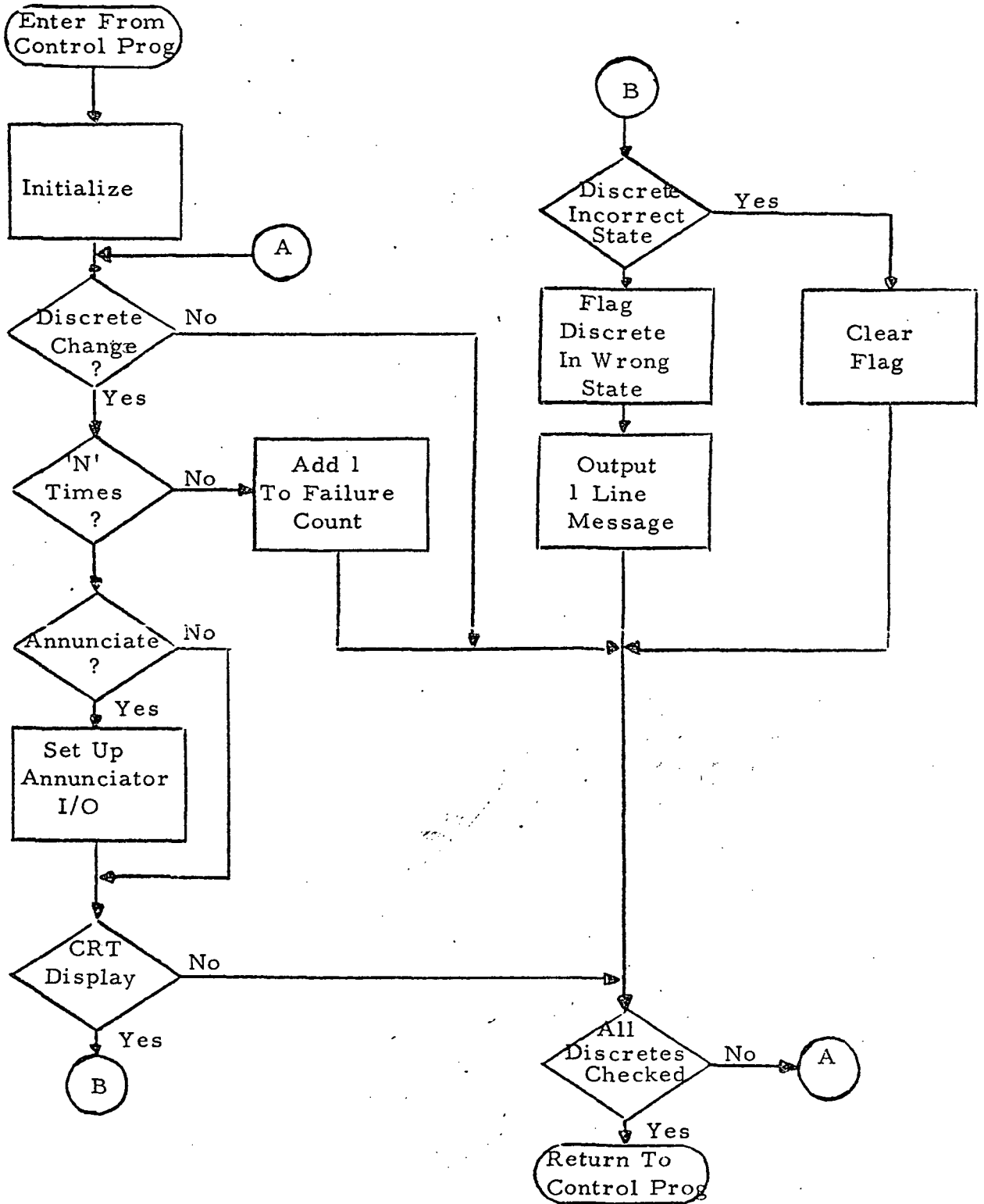


Figure 2.10.3-4. Discrete Processing Flow



If a discrete has changed, a table entry is interrogated to determine the type notification specified. Again, this can be via annunciators, CRT, or both. If annunciator notification is specified, an eight bit annunciator address code and an associated on/off bit are queued for output.

If CRT notification was specified, an "allow/disallow" table is interrogated to determine if the change was allowable. If so, no additional action is taken and the scan of other discretets continues. Presumably, the discrete for the current phase is not considered a part of the performance monitoring. If the discrete change is not allowed and is currently in the incorrect state, the same sequence as was done in analog processing is performed. The "out of tolerance" indicator is set in the format skeleton on which the discrete appears, a notification is placed on the scratch pad line, and a one line summary message is inserted in the Performance Monitoring Status format buffer. The use of this presentation is the same as described in the analog description. If the discrete change is to the correct state, the converse of the above sequence is performed.

The initial discrete status and the control routines access to the proper "allow/disallow" and "should be" tables is established by the Performance Monitor Phase Initializer. This is the same routine mentioned earlier in the analog processing section. A set of allow/disallow and should-be tables are provided for each flight phase.

Display Software. The display software performs the function of organizing and presenting engineering unit subsystem data for CRT display. The following paragraphs describe the display software subprograms, which are stored in the display processor, and the display formats, skeletons of which are resident in the DEU memory.

1. Format Description - Subsystem status information is presented on the CRT in predefined logical groups. A single group or format is presented at a time. Each format consists of a title line, 1 to 12 measurement lines, and a scratch pad line. Each line may contain up to 24 characters. The measurement lines each contain three fields; for sizing purposes a 12-character measurement descriptor, a 5-character measurement value, and a 4-character engineering unit label are assumed. The scratch pad line is used for the display of keyboard input data and messages from performance monitoring software. The format skeletons (exclusive of the actual parameter values) are stored in the DEU memory, and contain the alphabetic measurement name, engineering units (for example, "PSIA"), decimal points, display edit information, and Format Control Words (FCW's). The FCW's specify character generation control information, x and y CRT position data, and mode control data.

An "interface dictionary" (table of data), which is resident in the



display processor, is associated with each format. It contains data which describes and controls the processing unique to each measurement on a given format. The dictionary contains the measurement number, conversion scale factors and biases, output codes, test limits, and certain flags and pointers.

The maximum DEU storage requirement for a 12-measurement format is:

13 lines of 24 characters + carriage return	165 words
x, y position and control words	4 words
Format Control and edit information	<u>18 words</u>
Total 16-bit words for one format:	187 words

The display processor storage requirement for the interface dictionary data associated with a 12 (analog) measurement format is:

Measurement designators, 12 x 12 characters	72 words
Scale factors (1 # byte) + Bias (2 bytes)	18 words
Output codes (9 CRT, 3 annunciator)	2 words
Test Limits, 3 sets at 1 word/set	<u>36 words</u>
Total 16-bit words for one format:	128 words

Total interface dictionary size for 35 formats: 4480 words

2. DEU Keyboard Processing - The DEU keyboard and line-select pushbuttons are used by the crew to select formats for display, and to specify flight measurements which are to be excluded from performance monitor error notification. Activation of the data entry-format select keys result in the storage of the key information in an input buffer in the DEU and in its display on line 14 of the CRT. The line-select keys are also stored, but not displayed. Activation of a specific key will result in a signal to the display processor, which responds by reading the input buffer data and routing it to the proper processing program. Figure 2.10.3-5 describes the display processing flow.

3. Display Update Processing - Display update processing consists of determining which format has been selected for display, assembling and organizing the appropriate parameter values and status flags, and scheduling data for output to the CRT and/or annunciator. When the display format has been identified, the interface dictionary is searched to determine which of the parameters are to be displayed. The engineering unit values

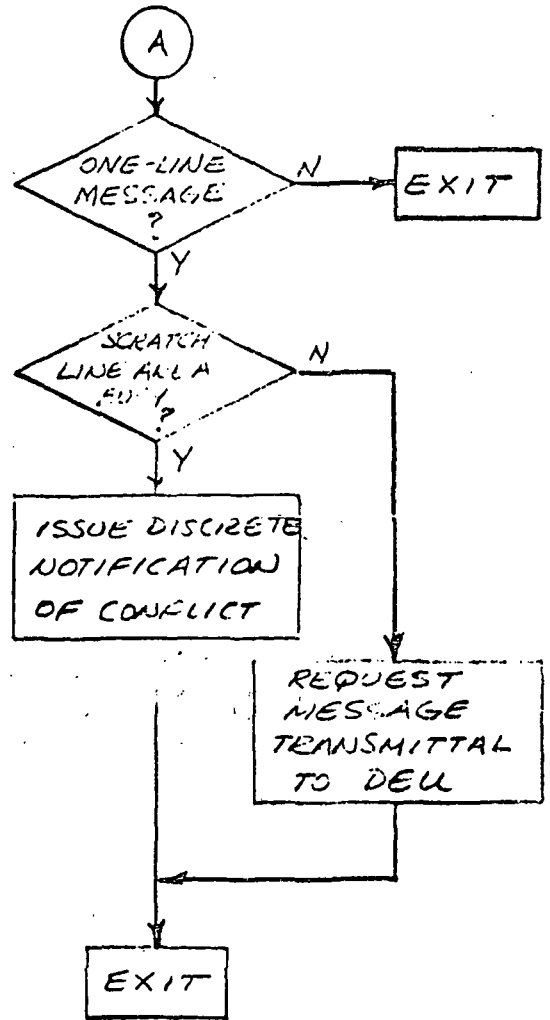
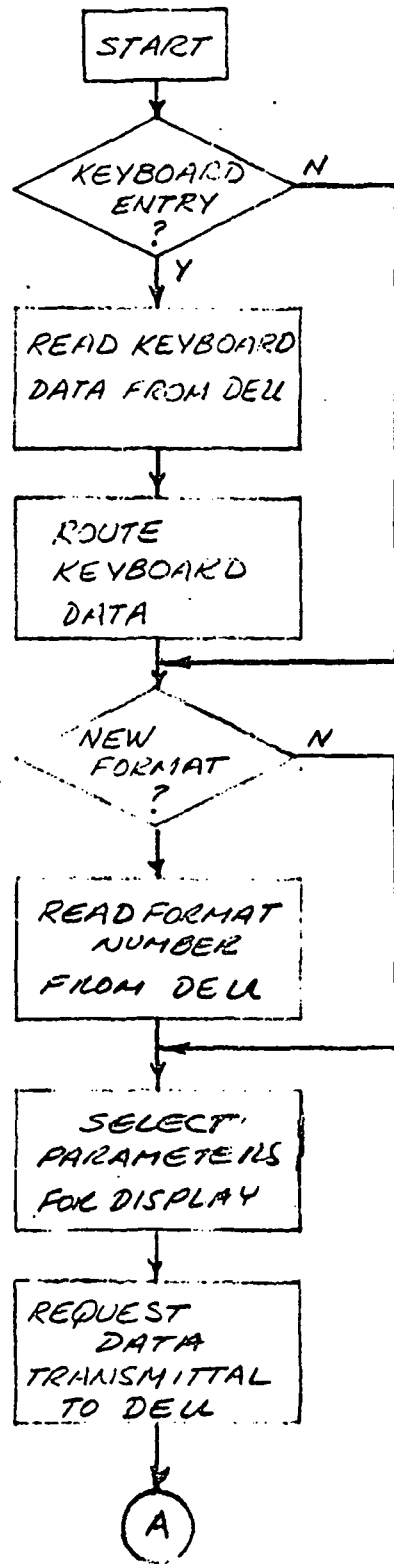


Figure 2.10. 3-6. Display Processing Flow



(previously stored by the performance monitor routines) are queued for output to the DEU by the output program. The status (go or no-go) of each parameter is inspected and if appropriate, a one line message is queued for transmittal and display on one of the four performance monitor status pages.

Memory Size and Speed Estimates

Certain assumptions were made in estimating the amount of memory capacity and computing speed required by the above-described tasks. These are summarized below:

- . 300 8-bit analog and 300 1-bit discrete measurements will be processed.
- . 35 A/N status type formats are stored in DEU.
- . Conversion to engineering units will be linear, of the form $Ax + B$.
- . 900 analog test limits will be used.
- . The DEU refresh is controlled by Format Control Words in the DEU storage.
- . The display processor can read and write DEU storage. Write operations are restricted to the DEU storage area reserved for the active formats.
- . Each display format can contain up to 14 lines of 24 characters.
- . The DEU has no vector graphic capabilities at present, although it is expandable to include that capability.
- . Keyboard entry data is buffered in the DEU until read by the display processor upon receipt of an ENTER or other command signal from the DEU.

Using the SP-1A architecture and instruction repertoire as a basis, an estimate of the number of instructions and data words required to perform the previously described functions was made. A summary of the sizing estimates is shown in Figure 2.10.3-6. The discrete and analog processing tasks were flow-charted to estimate size and speed required. It was found that (worse case) the discrete processing function will require not more than 10,000 instructions per second, the analog processing tasks will require not more than 20,000 instructions per second, and it is estimated that system control and overhead functions will require fewer than 10,000 per sec.

DISPLAY PROCESSOR				DISPLAY ELECTRONICS UNIT			
FUNCTION	INST.	TABLES DATA	16-BIT WORDS	FUNCTION	INST.	TABLES DATA	16-BIT WORDS
Control Program (includes interface tables)	600	5000	5600	Display Formats (35)		6500	6500
Performance Monitoring				Symbol Stroke Table		500	500
Analog processing	300	500	800	Keyboard Input Buffer		20	20
discrete processing	100	100	200	Active Format Area		200	200
Displays				Message Line Area		20	20
Display control	100		100				
Display update	200	500	600				
I/O	50	50	100				
TOTALS	1350	6150	7500			7140	7140

SD 71-346



2.11 PAYLOAD INTERFACE

The new baseline configuration possesses the capability to accommodate certain specific payload interface functions as delineated below.

1. Power distribution; orbiter to payload via hardwire.
2. Two-way communications; orbiter to payload via hardwire.
3. Payload access to orbiter S-band antenna subsystem.
4. RF and internal hardwire two-way data transfer.

In addition, a volume of approximately 29 cubic feet is presently allocated in the avionics bay for payload management equipment. Dependent upon the specific payload requirements, this volume can be utilized for data conditioning, for display and use by crew members, or retransmission as required. This equipment is presently considered chargeable and unique to the specified payload, until better definition of requirements is available.

Provisions at the orbiter/payload interface (physical and functional) for cooperative satellite rendezvous will be defined as part of the continuing study effort.

It is recognized that some minimum number of critical payload signals will require display to the orbiter flight crew during the launch and orbital mission phases with payload aboard. A special payload caution and warning display will be developed for this purpose as payload requirements become defined.



2.12 DOCKING AND CARGO HANDLING SYSTEM (DCHS)

The Docking and Cargo Handling System is described here because of its interfaces with the D&C and GN&C subsystems.

Docking, payload deployment, and payload retrieval are accomplished through the use of a pair of manipulator arms, as shown in Figure 2.12-1. A manipulator operator station is integrated with the flight deck of the crew module, as discussed in Section 2.5. A combination of direct viewing for gross arm control and closed-circuit television for fine arm control provide visual cues for manipulator operation.

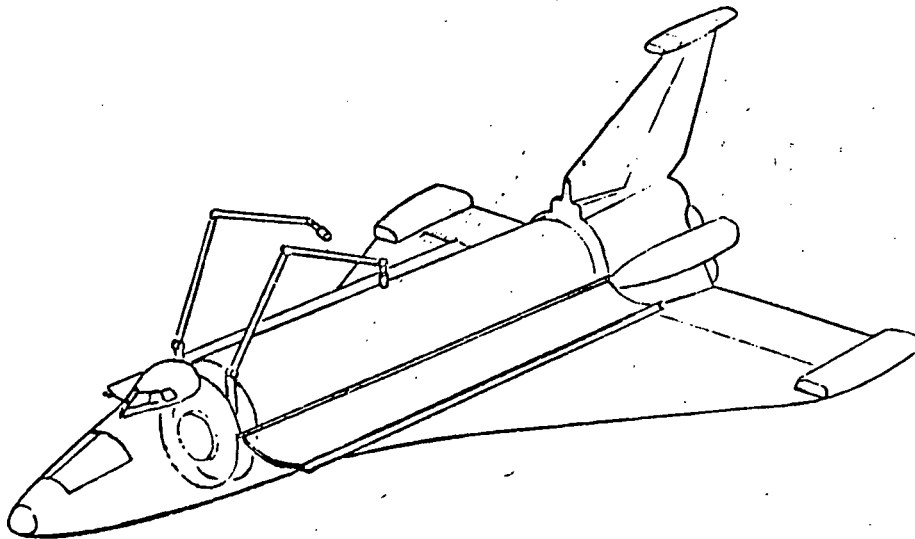


Figure 2.12-1. Docking and Cargo Handling System Manipulator Arms



The total DCMS consists of the manipulator system, a payload retention assembly, a docking port integrated with the airlock, a personnel transfer port between the crew module and the cargo bay, illumination, and closed-circuit TV. The DCMS displays and controls are all located at the manipulator operator station.

NR's concept for the manipulator system has the following central features:

- (1) Each arm is 50 feet long, articulated at shoulder, elbow, and wrist, with maximum joint rates of 2, 4, and 8°/sec, respectively. Maximum torque capacity is 500 ft-lbs at the shoulder. Typical operation times are on the order of 5 to 15 minutes.
- (2) The wrist of each arm includes a television camera (zoom lens), floodlight (controllable intensity), axial wrist extension feature, and interchangeable end effector.
- (3) As a result of piloted simulation studies, the manipulator operator station combined orbiter controls (translation and attitude) with manipulator controls. The operator flies a combination of the orbiter and the arms to achieve the desired results. The principal direction of manipulator operation is the orbiter's -Z axis, with most operations constrained to one plane (generally the orbiter pitch plane).
- (4) The manipulator operator has manual control of one arm at a time for all operations, but generally relies on computer control for (a) moving an attached payload through prescribed motions, (b) directing the camera on the second arm to track the activities of the first arm, thereby providing a view normal to the plane of operations, and (c) transforming input translation commands for tip motion into the proper angular commands for joint motion. The manipulator operator has exclusively manual control of all attachment and release functions. Computer processing is accomplished by the GN&C computer.

Design of the manipulator's control system is in an early conceptual stage, with the principal design issues having been identified: (1) to what extent are master-slave and/or feedback features required, (2) what influence does the orbiter's attitude control system have, (3) are simple direct viewing and television adequate for the position, angle, and rate information, closing the control loop for satellite retrieval through the operator, (4) what are the back-up modes of system operation (can the operator perform all tasks manually), and (5) can tip translation control be restricted to planar motion, or must 3 dimensions be incorporated?

/ 3.0 BOOSTER AVIONICS STUDY

The studies discussed in this report were undertaken in response to a NASA Statement of Work, which required identification of an alternate booster avionics configuration. This alternate configuration was primarily envisioned as a means to reduce both cost and technical risk. The resultant system configuration is planned to be that minimal amount which will still provide a safe, flyable vehicle.

The objectives of the study, as discussed in Section 1, were to establish a viable avionics system which minimizes development, reduces management/integration complexity, and has a redundancy philosophy based on criticality, experience, and cost. As a part of the study the reduction of automated functions, single string redundancy management, and phased development were studied.

The final configuration of the booster has not been determined at this time. For the purpose of this study two basic configurations, the LOX/RP/F-1 and the Pressure Fed Booster, were considered. The latter has two possible configurations, recoverable and nonrecoverable expendable. The alternate configurations are shown in Figure 3.0-1. The LOX/RP/F-1 will be called B18 throughout the remainder of this report. It is characterized as a delta wing, manned booster with deployable air breathing engines. The length is 155.5 feet from the gimbal block to the nose. Figure 3.0-2 is an inboard profile of B18.

The Pressure Fed Booster, or B19 as it will subsequently be referred to, is a less complex vehicle in which the propellants are forced into the combustion chamber by internal tank pressure without the aid of boost pumps. There are aerodynamic fins but no wings or crew compartment. The vehicle is guided by the orbiter during mated ascent. After separation the vehicle is

oriented during entry by the use of drag flaps from the aft skirt. After entry it is slowed by parachutes for a 150 feet per second splashdown. It is subsequently recovered by a recovery ship, removed from the water and cleaned enroute to the launch site. Figure 3.0-3 is an inboard profile of the B19 showing the large helium spheres in the forward cone. The helium is used to pressurize the tanks.

The B18 will enter and deploy its air breathing engines approximately 100 to 150 miles down range as shown in Figure 3.0-4. The B19 impact will be approximately 250 miles down range. The mission profile for the B18 indicates that it will remain within line of sight communication with the launch site throughout the one hour mission. The B19 will be over the horizon during the final phase of atmospheric flight but could be in communication with the recovery ship.

Summary

The avionic systems described in Sections 3.1 and 3.2 have been adopted as the baseline for B18 and B19, respectively.

The B18 system represents a major revision to the original 360 day baseline. The requirements for a central computer diminished to a level such that it was possible to eliminate the Data Management subsystem and transfer the remaining functions to a dedicated computer in the GN&C subsystem. The data bus was replaced by a combination of hardwired controls and data gathering and a multiplexed operational instrumentation system. The overall avionics weight is now 9422 pounds and the avionic electric power requirements increased to 10.6 kw.

The B19 Booster is a totally new vehicle and therefore the avionics is not just a subset of either the orbiter or the flyback booster. The vehicle is

TASK

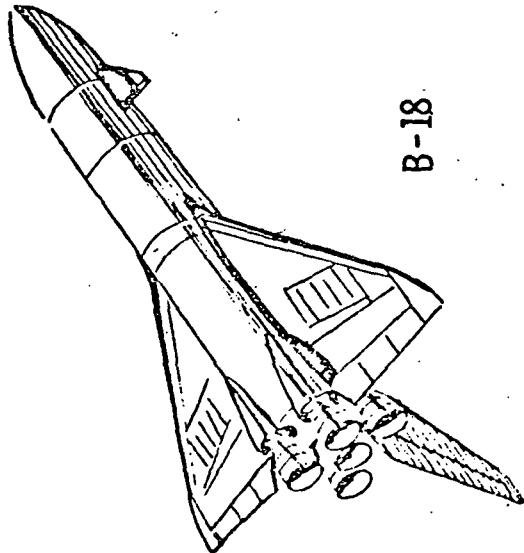
- AVIONICS ALTERNATE APPROACH STUDY

RATIONALE

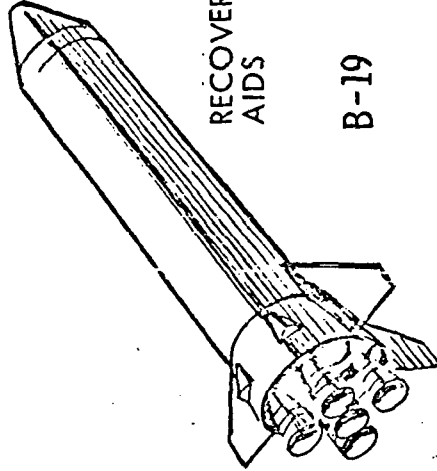
- CHANGES IN NON-AVIONICS SUBSYSTEMS SINCE 360 DAY BASELINE HAVE IMPACTED

- D&C
- DCM
- SOFTWARE
- EPG&D

- ACPS
- APU
- ABES

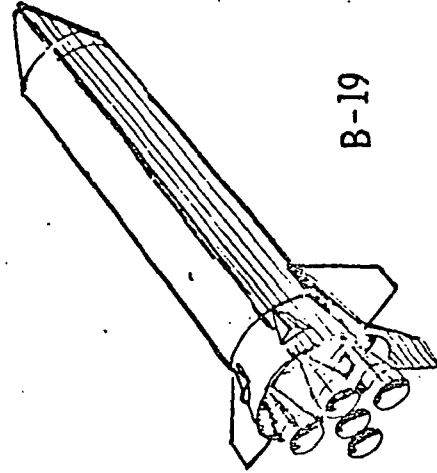


B-18



RECOVERY AIDS

B-19



B-19

<p>(A) F-1, FLYBACK</p>	<p>(B) PRESSURE FED</p>	<p>(C) PRESSURE FED</p>
<p>REUSABLE</p>		<p>EXPENDABLE</p>

Figure 3.0-1. Avionics Alternate Approach Study

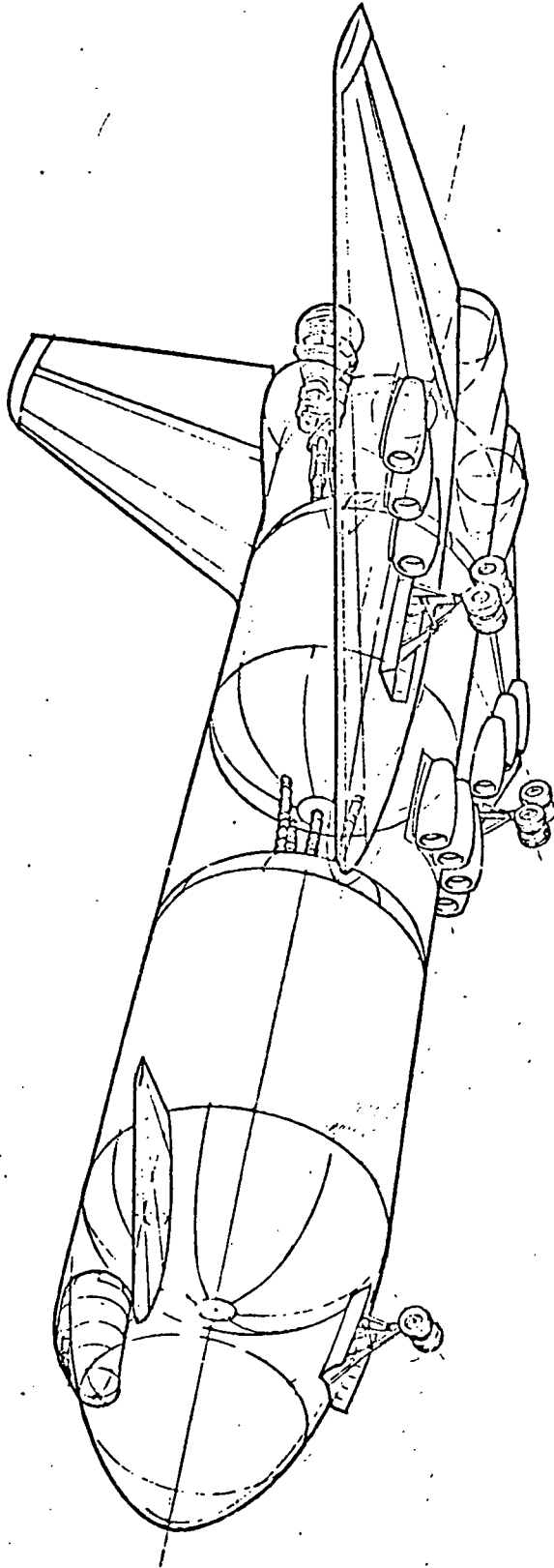


Figure 3.0-2. B-18E3 Booster (LOX/RP -F1)

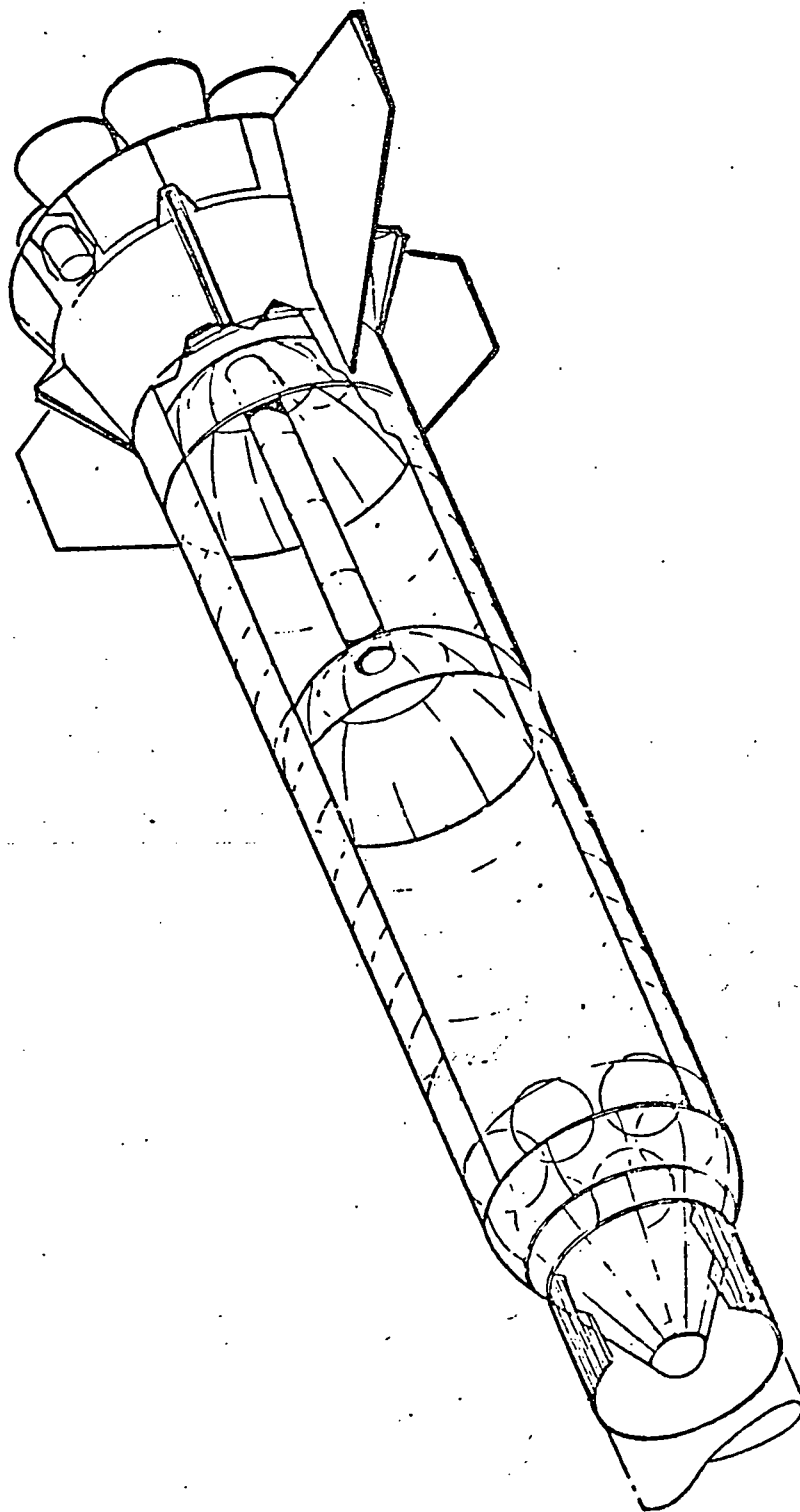


Figure 3.0-3. B-19 Booster

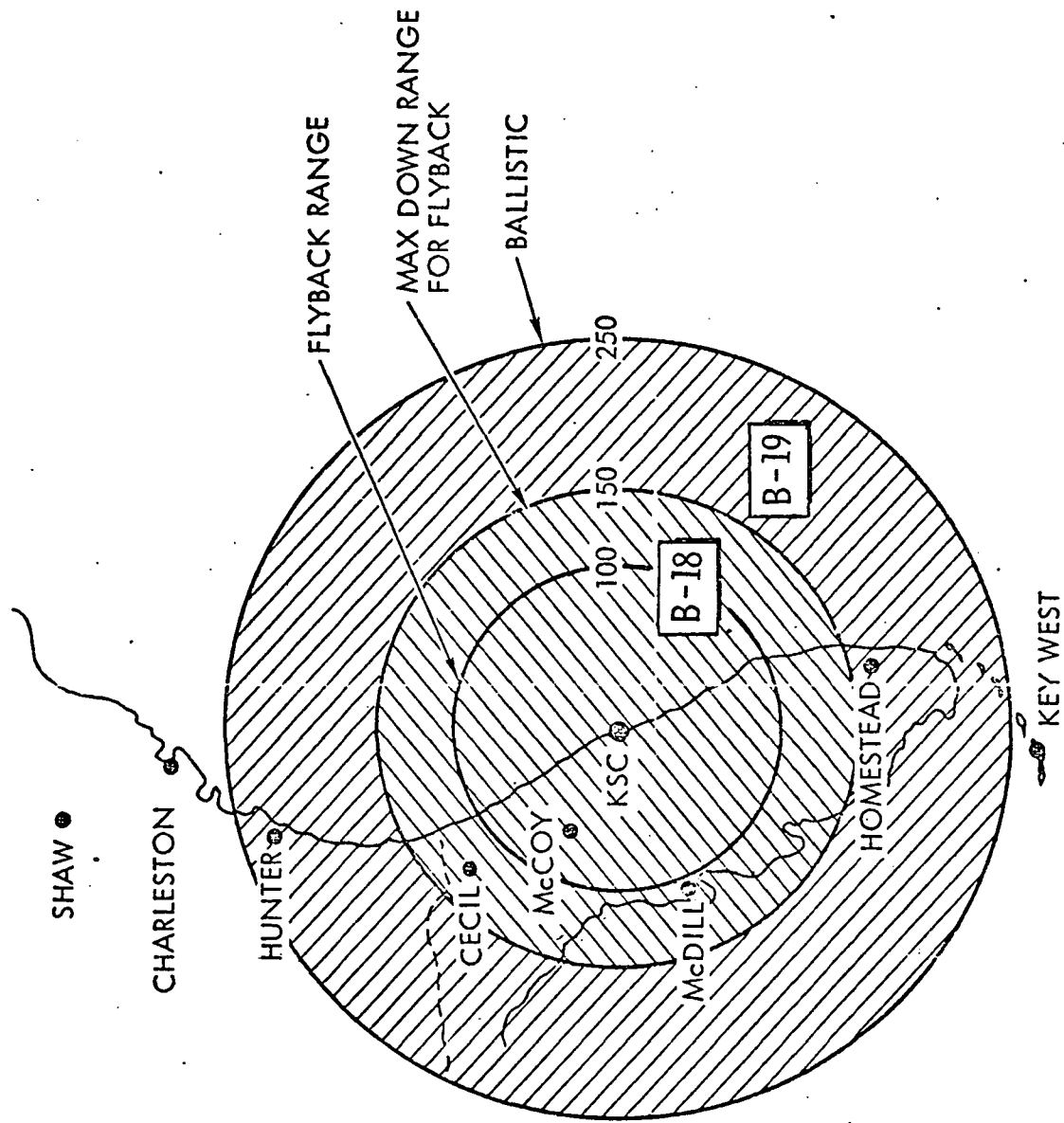


Figure 3.0-4. Booster Range Considerations

far less complex and therefore the avionic support to the non-avionic subsystems is reduced. Another major factor in reducing the B19 booster avionics is the transfer of guidance and flight control functions to the orbiter. The orbiter GN&C subsystem provides thrust vector control during mated ascent. The B19 avionics weighs 2287 pounds and requires 115 watts average and 2700 watts peak electric power.

3.1 B-18 BOOSTER - INTRODUCTION

In these studies a number of factors were considered relating to the complexity of the system, utilization of the crew, low cost proven designs, etc., however, the primary source of requirements to which these factors are subsidiary is the booster configuration and the total mission profile. In addition to the requirements for a low cost minimum risk system, changes in the non-avionics systems in response to changing mission and other requirements have impacted the avionics. The described configuration, therefore, represents the influence of all of these factors.

In conjunction with crew activities this system provides functional control, display, and management of the vehicle operational subsystems. The baseline configuration minimizes development cost and risk, simplifies integration complexity, provides manual backup for all flight control functions, and sufficient dedicated controls and displays for the single string redundancy management. The results of the alternate avionics approach trade study has shown this system to be less costly than the original Phase B avionics system.

In general, the avionics elements are dedicated to either the aerodynamic flight phase or the spacecraft phase. Figure 3.1-1 depicts an avionics block diagram with the equipment added for the first manned orbital flight Kit 1 accented with shading. Most of the avionic equipment will be installed in the forward avionics equipment bay.

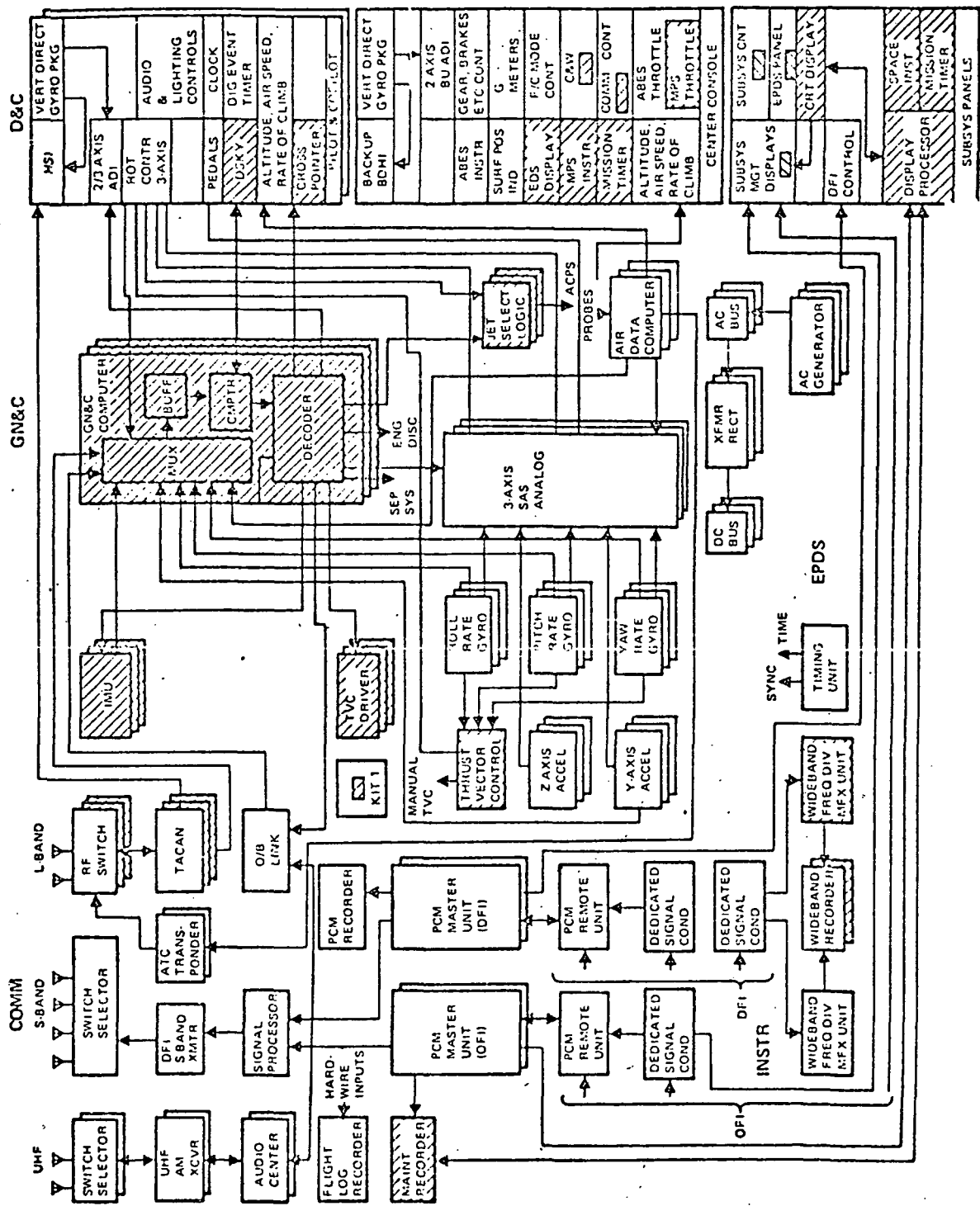


Figure 3.1-1 Space Shuttle Avionics Interim Configuration - Booster

The equipment will be installed in physically separated compartments to minimize the single point failure modes. In general, the equipment will be connected in three independent strings with a minimum amount of both cross-connections between strings and switching of elements within a string.

3.1.1 CONFIGURATION

Alternates

The avionics subsystems described in this report are direct results of the Alternate Avionics Trade Study. This alternate configuration was primarily envisioned as a means to reduce both cost and technical risk. The study originally analyzed 11 alternate configurations which were summarized in Section 1; of these, the team selected the concept of Number 5.

In the ensuing weeks this system was developed in detail and was ultimately adopted as the baseline system. In addition to developing an off-the-shelf avionics system it was necessary to make major revisions to the Development Plan. It was necessary to adopt a phased approach in which only those avionic elements necessary for horizontal flight be developed first. This was to be followed by Kit No. 1 which contained the avionics necessary for the vertical flights. In the case of the orbiter another modification, Kit No. 2, would be added for the more sophisticated Mark II Orbiter. There were no schedule problems identified within the major avionics subsystems.

Kits

Figure 3.1-2 shows the avionics required for the aerodynamic phase of flight. The boxes that have been crossed out are those items found on the orbiter and not on the booster. The rotational control and S-band DFI

are peculiar to the booster. The equipment is conventional aircraft equipment. Unlike the orbiter, the new booster baseline does not include a mechanical flight control path because the total impact for the booster has not been established. This requires further study.

Figure 3.1-3 is an overlay to Figure 3.1-2 and depicts the added avionics necessary for vertical flight. Again the crossed out boxes show the equipment not required for the booster. The numbers in parenthesis indicate the level of redundancy. The "X" in the upper right hand corner indicates that the rotational controller was added to the orbiter in Kit 1 and therefore is not peculiar to the booster as indicated on Figure 3.1-2.

When the Mark II orbiter is phased in the booster will be considered fully operational and the DFI equipment will be removed. The vertical displacement gyro package, which was used as a sensor for the ADI and HSI instruments before the IMU was added in Kit 1, is also removed. The "Xs" on Figure 3.1-4 which is also an overlay to Figure 3.1-2, are intended to show the removal of these elements. The horizon scanner added to the orbiter in Kit 2 is not required for the booster.

Equipment Lists

The equipment elements required for implementation are tabulated in Tables 3.1-2 through 3.1-8. They are segregated by subsystem and show the number of LRU's, size, weight, power and utilization by flight phase.

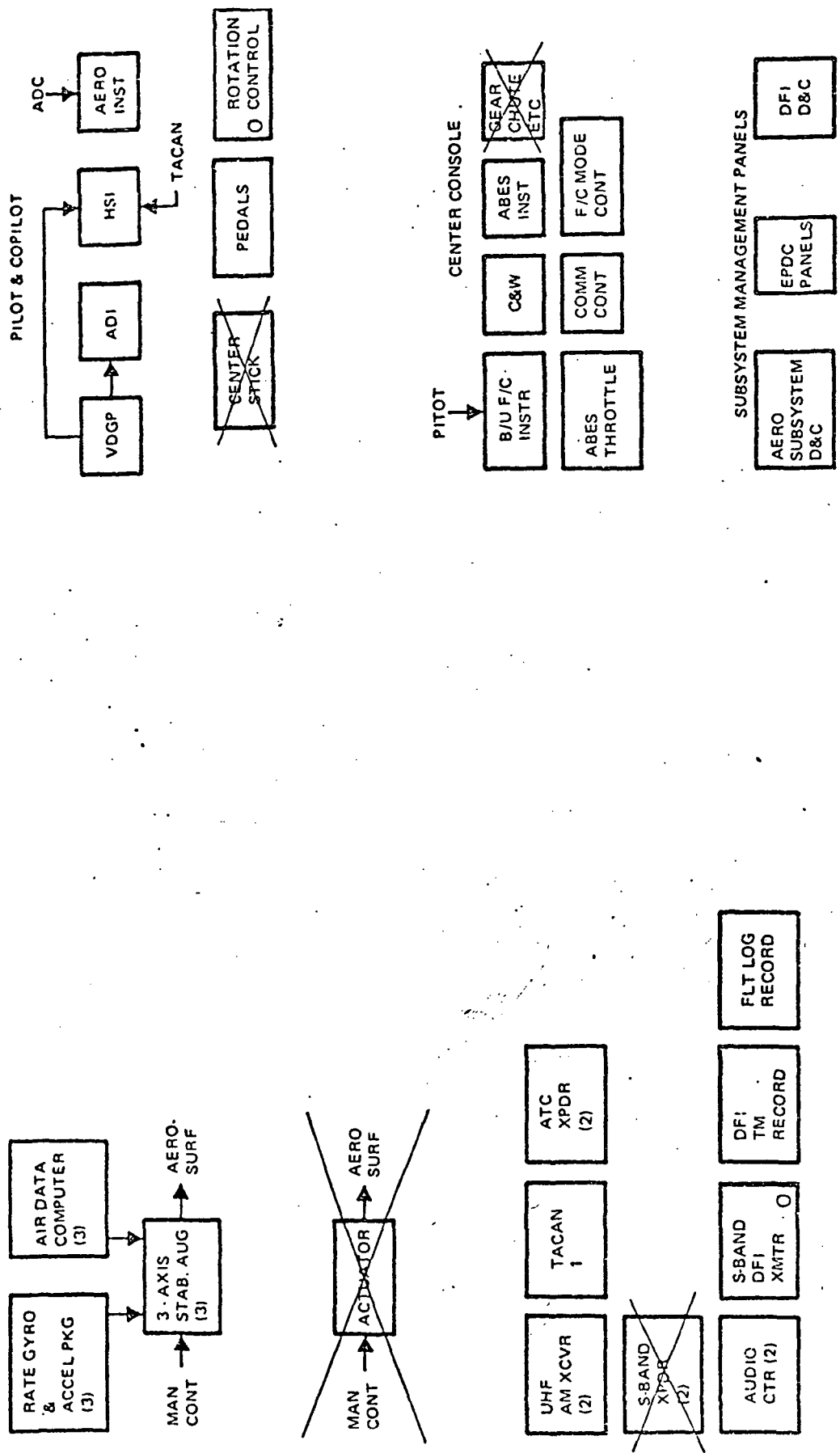


FIGURE 3.1-2. INTERIM CONFIGURATION A/C AVIONICS BOOSTER

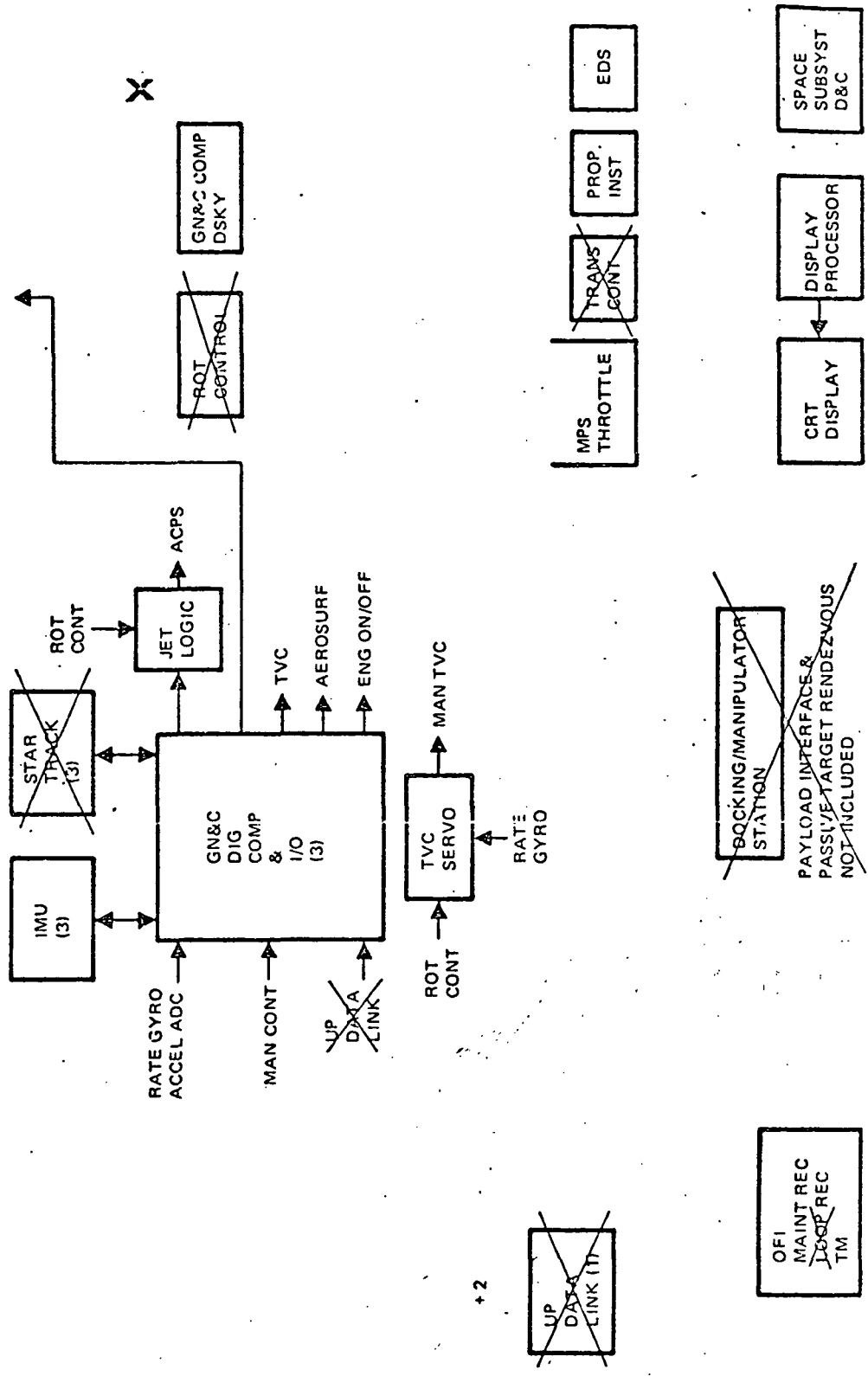


FIGURE 3.1-3 OVERLAY #1

HORI: JIN
SCAPE: EF
(3)

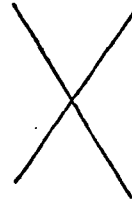
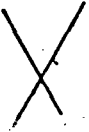
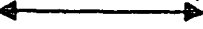


FIGURE 3.1-4. OVERLAY #2

Table 3.1-2 EPG D&C

B18 BOOSTER IAS EQUIPMENT

SUBSYSTEM	EQUIPMENT CHARACTERISTICS & USAGE										FLIGHT PHASE, DURATION												
	HORIZON	KIT 1	QUANTITY	WT(LBS)	VOL(V)	SIZE (IN.)	SHOCK MOUNT			ENVIRON. Temp/F	QTY Location			TOTAL NO.	NO. OPERATE.	LOAD TYPE (1)	TOTAL LOAD - W	COOLING (2)	ASCENT	SEPARATION	ENTRY	CRUISE DESCENT	LANDING
							Connector	Mounting	SURFACE		Min. Operate. Press. (PSI)	LOW	HIGH										
ELECTRIC POWER GENERATION DISTRIBUTION & CONTROL EQUIPMENT	3		.3	9	.05	.15	3.5	5.5	4.5	x	.4	-65	210	3	3	A1	150	A	1.0	1.0	1.0	1.0	1.0
	9		.3	2.7	.005	.045	2	2	2	-	.4	-65	175	9	9	A1	45	A	0	0	0	0	0
	3		1.1	3.3	.02	.06	2	3	6	-	.4	-65	175	3	3	A1	15	A	1.0	1.0	1.0	1.0	1.0
	3		24	84	.16	.48	7	7	7	7	.003	-65	110	3	3	-	LD	OIL	0	0	0	0	0
	2		35	70	.19	.36	7	7	6.5	-	.003	50	90	1	2	-	LD	A	0	0	0	0	0
	6		20	120	.22	1.32	6.1	6.3	10	-	.4	-65	160	3	6	A1	LD	AF	0	0	0	0	0
	3		20	60	.87	2.61	10	10	15	x	.4	30	140	3	3	D1	60	A	1.0	1.0	1.0	1.0	1.0
	3		20	90	1.42	4.26	12	12	17	x	.4	-65	175	3	3	D1	90	A	1.0	1.0	1.0	1.0	1.0
	3		10	30	.44	1.34	8	8	12	x	.4	-65	80	3	3	D1	30	A	1.0	1.0	1.0	1.0	1.0
	3/3		20	120	1.73	5.19	10	10	15	x	.4	30	110	3	6	D1	120	A	1.0	1.0	1.0	1.0	1.0
3		30	90	.16	.48	7	7	7	7	.003	-65	210	3	3	-	LD	OIL	0	0	0	0	0	
3		4	12	.07	.21	4	0	5	x	.4	-65	200	3	3	A1	120	A	1.0	1.0	1.0	1.0	1.0	
1		5	5	.31	.31	5	5	18	x	.4	-65	175	1	1	D1	60	A	0	0	0	.5	.5	
1		2	2	.01	.01	4	4	4	-	.003	-65	300	1	1	-	-	PS	0	0	0	0	0	
2		15	30	.21	.42	6	6	10	x	.4	30	140	2	2	D1	30	A	1.0	1.0	1.0	1.0	1.0	
10		2.5	25	0.01	.40	3	4.7	5.3	x	.4	-65	175	10	10	D1	100	A	0	0	0	1.0	1.0	
3		3	9	0.01	.12	3	4.7	5.3	x	.4	30	175	3	3	D1	75	A	1.0	1.0	0	0	0	
TOTAL				2143		23.01											A1		285	285	285	285	285
																	D1		405	405	405	420	420

(1) A1: APUA C; A2: INVERTER AC; DI: DC; (2) CW: COLDPLATE; PS: PASSIVE, UNPRESS.; A: AIR, CONVECTION; AF: AIR FORCED; LD: LOAD DEPENDENT

Table 3.1-3 OFI
B18 BOOSTER IAS EQUIPMENT

SUBSYSTEM OPERATIONAL INSTRUMENTATION EQUIPMENT	EQUIPMENT CHARACTERISTICS & USAGE										ENVIRON.					QTY Location					FLIGHT PHASE, DURATION							
	HORIZON	KIT 1	QUANTITY	WT(LBS)	VOL(T)	SIZE (IN.)	SHOCK MOUNT			Min. Operate. Press. (PSI)	Temp(F)		CREW	Avionics Bay	APU BAYS	AIRLOCK	OTHER	TOTAL NO.	NO. OPERATE.	LOAD TYPE (1)	TOTAL LOAD - W	COOLING (2)	ASCENT	SEPARATION	ENTRY	DESCENT	LANDING	
							Connector	H	W		L	LOW																HIGH
UNIT	UNIT	TOTAL	UNIT	TOTAL	UNIT	TOTAL	UNIT	TOTAL	UNIT	TOTAL	UNIT	TOTAL	UNIT	TOTAL	UNIT	TOTAL	UNIT	TOTAL	UNIT	TOTAL	UNIT	TOTAL	UNIT	TOTAL	UNIT	TOTAL	UNIT	TOTAL
PCM REMOTE UNIT	3	2	6	36	.14	0.81	4.7	3.1	5.8	-	0	-67	175				6	6	DI	300	CW/PS	1	1	1	1	1		
PCM MASTER UNIT	2	2	20	40	.52	1.95	9	10	10	-	0	-65	175				2	1	DI	60	CW	1	1	1	1	1		
SIGNAL COND. UNIT	3	3	15	20	.28	1.68	6	8	10	-	0	-65	175				6	6	DI	210	CW/PS	1	1	1	1	1		
MAINT. RECORDER	1	1	50	50	1.85	1.85	5	20	32	X	0	0	130				1	1	DI	60	CW	1	1	1	1	1		
FLIGHT LOG REC'DR	1	1	50	50	1.85	1.85	10	16	20	X	0	0	130				1	1	DI	60	CW	1	1	1	1	1		
TIMING UNIT	1	1	20	20	.41	.41	7	10	10	-	0	-65	175				1	1	DI	30	CW	1	1	1	1	1		
TRANSDUCERS	500	500	15	180	-	-	-	-	-	-	-	-	-	X	X	X	1200	1200	-	0	-	0	0	0	0	0		
SECONDARY STRUCTURE																												
SHOCK MOUNT & TRAY																												
BOLTS, SCREWS, ETC.																												
TOTAL																												
TOTAL																												

(1) A1: APJA C; A2: INVERTER AC; DI: DC; (2) CW: COLDPLATE; PS: PASSIVE, UNPRESS.; A: AIR, CONVECTION; AF: AIR FORCED;
LD: LOAD DEPENDENT
APU: APU COOLED
APPD

DATE 11-5-71

Table 3.1-4 DFI

B18 BOOSTER IAS EQUIPMENT

SUBSYSTEM	EQUIPMENT CHARACTERISTICS & USAGE															FLIGHT PHASE / DURATION											
	QUANTITY (KIT 1)		HORIZON	TOTAL		UNIT	TOTAL	CONNECTOR		SHOCK MOUNT	ENVIRON.		QTY LOCATION			TOTAL NO.	NO. OPERATE.	LOAD TYPE (1)	TOTAL LOAD - W	COOLING (2)	ASCENT	SEPARATION	ENTRY	DECENT	LANDING		
	2000	1000		H	W			L	Min. Operate. Press. (PSI)		Temp. F	Avionics Bay	APU BAYS	AFT BAY	OTHER												
DFI																											
	INDICERS	2000	1000																								
	PCM REMOTE UNIT	7	5	6	72	1.18	1.68	4.7	9.4	5.8	-	0	-67	175	-	7	-	5									
	PCM MASTER UNIT	2		20	40	.53	1.95	9.0	10.0	10.0	-	0	-65	175	-	2	-										
	PCM RECORDER	1		50	50	1.85	1.85	5	20	32	X	0	0	130	-	-	-										
	SIGNAL COND UNIT	10	7	15	255	.24	4.74	6	8	10	-	0	-65	175	-	10	-	7									
	WIDE BAND FDM UNIT	1	1	15	30	.27	.56	6	8	10	-	0	-65	175	-	1	-	1									
	WIDE BAND RECORDER	1	1	50	100	1.85	3.70	5	20	32	X	0	0	130	-	2	-										
	S-BAND TRANSMITTER	1		10	10	.29	.59	6.5	6.5	12	X	0.6	-17	100	-	1	-										
	INSTALLATION				200																						
	SECONDARY STRUCTURE				10																						
	SCREWS, ETC.				10																						
	ANTENNA SELECTOR	2		2-1/2	5	.02	.01	2	4	5	-	0.6	-67	160	-	2	-										
ANTENNA	4		2-1/2	10	.01	.01	3.5	12.1	3.1	-	0	160	-	1	-												
SIGNAL PROCESSOR	1		15	45	.20	.20	7.0	4.7	10.6	-	0	35	135	-	1	-											
SUBTOTAL					1247		14.15																				
CABLE/WIRE																											
					3415																						
DFI TOTAL					1662																						

(1) A1: APUA C; A2: INVERTER AC; DI: DC; (2) CW: COLDPLATE; PS: PASSIVE, UNPRESS.; A: AIR, CONVECTION; AF: AIR FORCED; LD: LOAD DEPENDENT APU: APU COOLED

APPD *21/1/55* DATE *11-5-71*

Table 3.1-5 Communications
BOOSTER IAS EQUIPMENT

SUBSYSTEM	EQUIPMENT CHARACTERISTICS & USAGE										FLIGHT PHASE, DURATION																	
	HORIZON	KIT 1	QUANTITY	WT(LBS)	VOL(FT)	SIZE (IN.)	SHOCK MOUNT		ENVIRON.		QTY Location			TOTAL NO.	NO. OPERATE.	LOAD TYPE (1)	TOTAL LOAD - W	COOLING (2)	ASCENT	SEPARATION	ENTRY	DESCENT	LANDING					
							Connector	Mounting	Surface	Surface	TempF	Min. Operate Press. (PSI)	LOW											HIGH	CREW	Avionics Bay	APU BAYS	OTHER
EQUIPMENT	UNIT	UNIT	TOTAL	UNIT	TOTAL	H	W	T	H	L	TempF	LOW	HIGH	CREW	Avionics Bay	APU BAYS	OTHER	TOTAL NO.	NO. OPERATE.	LOAD TYPE (1)	TOTAL LOAD - W	COOLING (2)	ASCENT	SEPARATION	ENTRY	DESCENT	LANDING	
UHF/AN	2	-	.28	55	.45	.90	7.7	7.7	13.0	x	0.6	-07	160	-	2	-	-	-	2	1	A1	100	AF	1.0	1.0	1.0	1.0	
TACAN	2	-	25	50	.45	.90	8.0	7.5	13.0	x	0.6	-07	160	-	2	-	-	-	2	1	A1	120	AF	0.0	0.0	1.0	1.0	1.0
ATC TRANSPONDER	2	-	27	54	.28	.56	8.0	4.3	13.0	x	0.6	-07	160	-	2	-	-	-	2	1	A1	54	AF	0.0	0.0	1.0	1.0	1.0
AUDIO CENTER	2	-	8	16	.14	.28	6.0	4.7	8.7	-	0.0	35	135	-	2	-	-	-	2	1	D1	20	CW	1.0	1.0	1.0	1.0	1.0
ANTENNAS																												
L-BAND TACAN	2	-	1.5	7	.024	.05	5.0	1.74	1.75	-	0.0	A/10/A/R	-	-	-	-	-	-	2	2	-	0.0	PS	0.0	0.0	1.0	1.0	1.0
L-BAND ATC	2	-	1.5	7	.024	.05	5.0	1.74	1.75	-	0.0	A/10/A/R	-	-	-	-	-	-	2	2	-	0.0	PS	0.0	0.0	1.0	1.0	1.0
HIF	2	-	5	10	.30	.78	4.0	13.0	13.0	-	0.0	A/10/A/R	-	-	-	-	-	-	2	2	-	0.0	PS	1.0	1.0	1.0	1.0	1.0
ANTENNA SELECTORS																												
L-BAND TACAN	2	-	2.5	5	.023	.05	2.0	4.0	5.0	-	0.6	-07	160	-	2	-	-	-	2	1	D1	5.0	CW	0.0	0.0	1.0	1.0	1.0
L-BAND ATC	1	-	2.5	2.5	.023	.02	2.0	4.0	5.0	-	0.6	-07	160	-	1	-	-	-	1	1	D1	5.0	CW	0.0	0.0	1.0	1.0	1.0
UHF	2	-	2.5	5	.023	.05	2.0	4.0	5.0	-	0.6	-07	160	-	2	-	-	-	2	1	D1	5.0	CW	1.0	1.0	1.0	1.0	1.0
COAXIAL CABLE																												
INSTALLATION																												
SECONDARY STRUCTURE																												
MOUNTING																												
BOLTS, SCREWS, ETC.																												
SUBTOTAL				171.5		3.64																399						

(1) A1: APUA C; A2: INVERTER AC; DI: DC; (2) CW: COLDPLATE; PS: PASSIVE, UNPRESS; A: AIR, CONVECTION; AF: AIR FORCED; LD: LOAD DEPENDENT

APU: APU COOLED

APPD

DATE

4/10/71

Table 3.1-6 GN & C

B18E3 BOOSTER IAS EQUIPMENT

SUBSYSTEM	EQUIPMENT CHARACTERISTICS & USAGE										FLIGHT PHASE, DURATION													
	HORIZON	KIT 1	QUANTITY	WT(LBS)	VOLTA	SIZE (IN.)		SHOCK MOUNT	ENVIRON.		Crew	Avionics Bay	Other	TOTAL NO.	NO. OPERATE.	LOAD TYPE (1)	TOTAL LOAD - W	COOLING (2)	ASCENT	SEPARATION	ENTRY	DESCENT	LANDING	
						CONNECTOR	H		W	T														Mounting
EQUIPMENT																								
INU																								
GUIDANCE & NAVIGATION & FLIGHT CONTROL																								
PLATFORM	3		25.0	75.0	.45	1.35	12	8	11A.					3	3	A1	210	A	1.0	1.0	1.0	1.0	1.0	
POWER SUPPLY	3		22.0	66.0	.35	1.05	12	8	8					3	3	D1	150	A	1.0	1.0	1.0	1.0	1.0	
TVC DRIVERS (Pib, Yaw, Roll)	3		5.0	15.0	.10	.20	T	D						3	3	A1	29	CW	1.0	0	0	0	1.0	
TVC CONTROL-MANUAL	1		5.0	5.0	.10	.10	T	D						1	1	D1	10	CW	1.0	0	0	0	1.0	
ACES LOGIC/DRIVER	7		8.0	56.0	.15	1.20	T	D						7	7	A1	10.3	CW	0	1.0	1.0	0	0	
AERO SURFACE DRIVER	5		12.0	60.0	.27	1.35	T	D						5	5	A1	100.4	CW	0	0	1.0	1.0	1.0	
STAB. AUG. SYSTEM	3		13.0	39.0	.29	.87	T	D						3	3	A1	57.24	A	0	1.0	1.0	1.0	1.0	
AIR DATA PKG Incl Computer	3		25.0	75.0	.52	1.36	5	12	7.5					3	3	A1	266	A	1.0	1.0	1.0	1.0	1.0	
BODY RATE SENSOR PKG	3		3.0	10.5	.04	.12	T	D						3	3	A1	12	PS	1.0	1.0	1.0	1.0	1.0	
ANGLE OF ATTACK XDCR	3		10.0	30.0	.15	.45	T	D						3	3	A1	30	A	1.0	1.0	1.0	1.0	1.0	
ACCELEROMETER PKG	3		1.0	3.0	.02	.06	T	D						3	3	A1	2	PS	1.0	1.0	1.0	1.0	1.0	
APES THROTTLE DRIVER	10		2.0	20.0	.01	.30	T	D						10	10	D1	10	A	0	0	0	0	1.0	
COMPUTER, 4SK12 units/sec	3		50.0	300.0	.96	5.76	7.6	12	17.3	114.7	55	155		3	3	D1	1020	AF	1.0	1.0	1.0	1.0	1.0	
BUFFER	3		24.0	72.0	.30	.90	4.1	10	15.5	114.7	55	155		3	3	D1	300	AF	1.0	1.0	1.0	1.0	1.0	
DECODER	3		27.0	81.0	.30	.90	4.1	10	15.5	114.7	55	155		3	3	D1	240	AF	1.0	1.0	1.0	1.0	1.0	
MULTIPLEXER, DUAL	3		10.0	30.0	.08	.24	8.5	4.5	3.3	114.7	55	155		3	3	D1	60	A	1.0	1.0	1.0	1.0	1.0	
SUBTOTAL																								
INSTALLATION																								
SECONDARY STRUCTURE																								
SHOCK MOUNT & TRAY																								
BOLTS, SCREWS, ETC.																								
TOTAL																								

(1) A1: APUA C; A2: INVERTER AC; D1: DC; (2) CW: COLDPLATE; PS: PASSIVE, UNPRESS.; A: AIR, CONVECTION; AF: AIR FORCED; LD: LOAD DEPENDENT

Table 3.1-7 D&C

B18 BOOSTER IAS EQUIPMENT

SUBSYSTEM	EQUIPMENT CHARACTERISTICS & USAGE										FLIGHT PHASE, DURATION										
	QUANTITY	WT(LBS)	VOL(FT ³)	SIZE (IN.)			TOTAL	UNIT	TOTAL	UNIT	TOTAL NO.	NO. OPERATE.	LOAD TYPE (1)	TOTAL LOAD - W	COOLING (2)	ASCENT	SEPARATION	ENTRY	DESCENT	LANDING	
				H	W	T															SHOCK MOUNT
EQUIPMENT	HORIZON	KIT 1	CONNECTOR	SURFACE	MOUNTING	SURFACE	SHOCK MOUNT	Min. Operating Press. (PSI)	LOW	HIGH	TempF	CREW	Avionics Bay	APU BAYS	OTHER	ASCENT	SEPARATION	ENTRY	DESCENT	LANDING	
PRIMARY FLT DISPLAY SET																					
3-AXIS ADI	2	-	10	20	.37	.74	8	8	10			2	-	-	-	2	2	2	2	2	2
HSI	2		8	16	.13	.26	5	5	9			2	-	-	-	2	2	2	2	2	2
IAS - MACH IND	2		3	6	.06	.12	3.25	3.25	9			2	-	-	-	2	2	2	2	2	2
ALT	2		4.5	9			3.25	3.25	8			2	-	-	-	2	2	2	2	2	2
RATE OF CLIMB	2		1.5	3			3.25	3.25	4			2	-	-	-	2	2	2	2	2	2
2-AXIS G METER	2		4				3.25	3.25	6			2	-	-	-	2	2	2	2	2	2
SLIP INDICATOR	2		1	2			1.0	1.25	4			2	-	-	-	2	2	2	2	2	2
FLAIR INDICATOR	2		2	4			1.75	3.75	6			2	-	-	-	2	2	2	2	2	2
BACKUP FLT DISPLAY SET																					
3-AXIS ADI	1		9				5.25	5.0	10.7			1	-	-	-	1	1	1	1	1	1
VERT/DIRECT CYRO PKG	3		10.5	31.5	.21	.63	6.5	6.5	8.5			3	-	-	-	3	3	3	3	3	3
ABESS DISPLAY SET	22		2.5	53.0			7.3	1.0	6			22	-	-	-	22	22	22	22	22	22
ROTATIONAL HAND CONTR.	2		6	12	.31	.62	3	6	10			2	-	-	-	2	2	2	2	2	2
THROTTLE QUAD	1											1	-	-	-	1	1	1	1	1	1
CONTR/ENCODER/COUPLER	2		12	24	.37	.74	8	8	10			2	-	-	-	2	2	2	2	2	2
CRT DISPLAY UNIT	2		20	40	.71	1.48	8	9	17			2	-	-	-	2	2	2	2	2	2
COMPUTER ENTRY KEYBRD	2		7	14	.15	.30	6	7	6			2	-	-	-	2	2	2	2	2	2
CAUTION/A WARNING UNIT	1		28	28	.22	.44	4	4	24			1	-	-	-	1	1	1	1	1	1
EVENT TIMER	2		1	2	.01	.02	1	5	5			2	-	-	-	2	2	2	2	2	2
MISSION TIMER	2		1.5	3	.02	.04	1	7	5			2	-	-	-	2	2	2	2	2	2
SUBSYSDISP PROCESSOR	1		11	11	.25	.25	6	9	8			1	-	-	-	1	1	1	1	1	1
PEDAL CONTROL ASSY	2		50	100								2	-	-	-	2	2	2	2	2	2
PANELS																					
MAIN FWD G-DTR-R	1																				
CTR CONSOLE	1																				
L-SIDE CONSOLE																					

(1) A1: APUA C; A2: INVERTER AC; D1: DC; (2) CW: COLDPLATE; PS: PASSIVE, UNPRESS; A: AIR, CONVECTION; AF: AIR FORCED; LD: LOAD DEPENDENT

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11-3-71

Table 3.1-8 D&C Continued
BOCSTER IAS EQUIPMENT

SUBSYSTEM DISPLAY & CONTROL	EQUIPMENT CHARACTERISTICS & USAGE												FLIGHT PHASE, DURATION									
	HORIZON	KIT 1	QUANTITY	WT(LBS)	VOL(FT)	TOTAL	UNIT	TOTAL	UNIT	TOTAL	NO. OPERATE.	LOAD TYPE (1)	TOTAL LOAD - W	COOLING (2)	ASCENT	SEPARATION	ENTRY	DESCENT	LANDING			
																				SIZE (IN.)		ENVIRON.
EQUIPMENT																						
R SIDE CONSOLE	1																					
OVERHEAD	1																					
CIRCUIT BRKR PANEL	2																					
WIRING, CLAMPS, BKTS, ETC					225																	
STRUCT & SHOCK MOUNT					200																	
INTERIOR LIGHTING (SET)					65																	
PRIMARY (PANEL) SET	Set																					
SECONDARY (FLOOD)	10												700		1.0	1.0	1.0					
HI INTENSITY	6												450		0	0	0					
EQUIP. BAY	6														0	0	0					
AVIONICS (FLOOD)	8														0	0	0					
ESTERIOR LIGHTING (SET)					100	100									0	0	0					
ANTI COLLISION	3														0	0	0					
FUSELAGE CLEARANCE	4														0	0	0					
WING & TAIL POSITION	4														0	0	0					
LANDING/TAXI LITES	3														0	0	0					
															0	0	0					
															0	0	0					
															0	0	0					
															0	0	0					
															0	0	0					
															0	0	0					
															0	0	0					
															0	0	0					
															0	0	0					
TOTALS																						
						1185																

(1) AI: APUA C; A2: INVERTER AC; DI: DC; (2) CW: COLDPLATE; PS: PASSIVE, UNPRESS.; A: AIR, CONVECTION; AF: AIR FORCED; LD: LOAD DEPENDENT

APPD

DATE 11-3-71

Interior Layout

The avionic equipment will be installed in either the forward crew station/avionics compartment or in one of the two APU compartments near the wing roots as shown in Figure 3.1-5. The number of areas in which avionics equipment is located was held to an absolute minimum to minimize the impact on the environmental control system.

A large portion of the avionics will be located in the crew station/avionics compartment shown in Figure 3.1-6. The equipment will be shelf mounted with the connectors easily accessible. The equipment will be installed such that maintenance can be performed in either the vertical or horizontal position.

Interface With Orbiter

The avionics interface between the B-18 booster and orbiter is accomplished through three redundant hardwired umbilical type connections. During the boost phase the booster controls the mated vehicle. System status, abort commands, and event sequence timing are transmitted both from booster to orbiter and vice versa. Body rates are transmitted one way, from orbiter to booster, while booster propellant depletion is supplied to the orbiter.

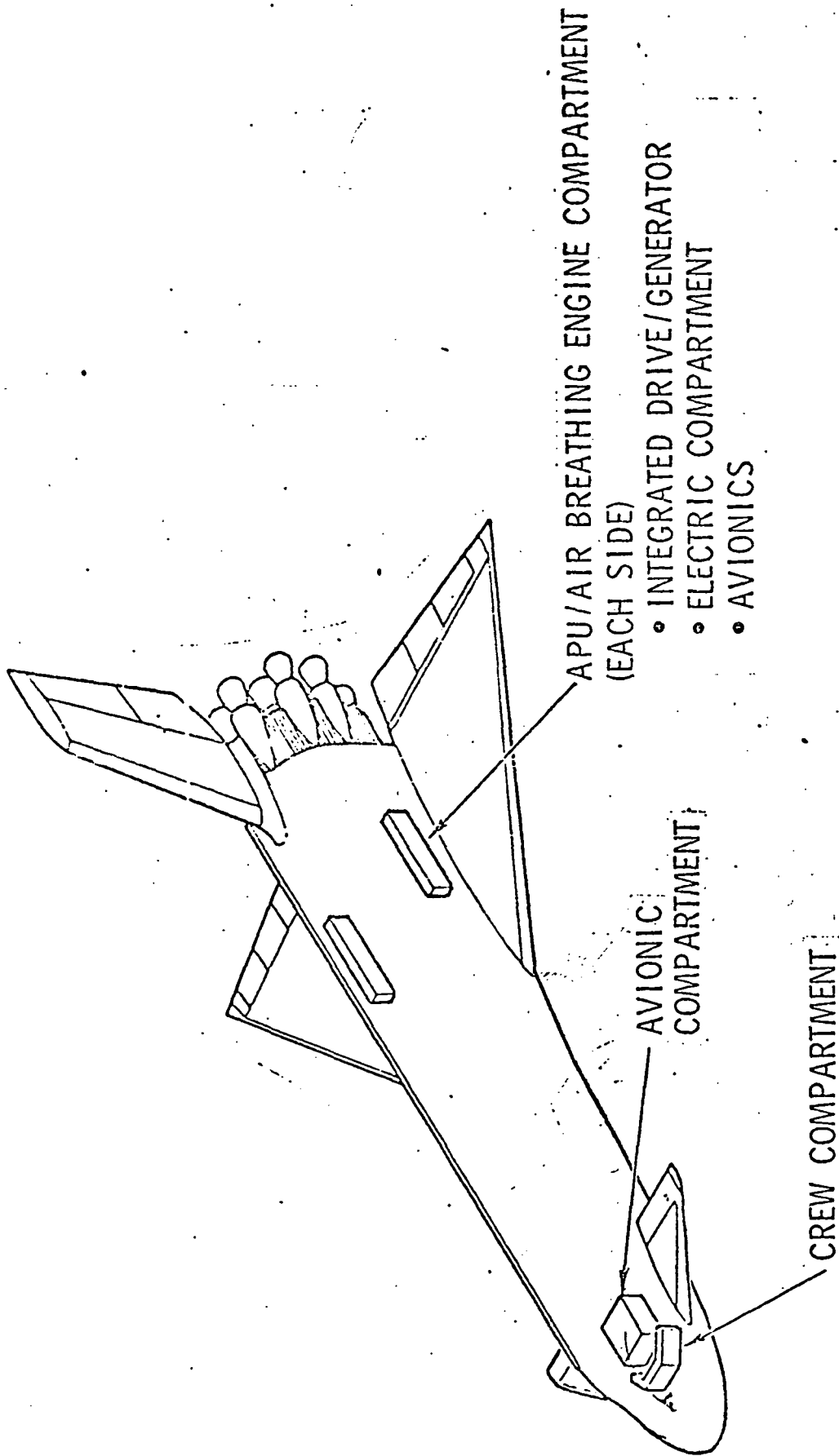
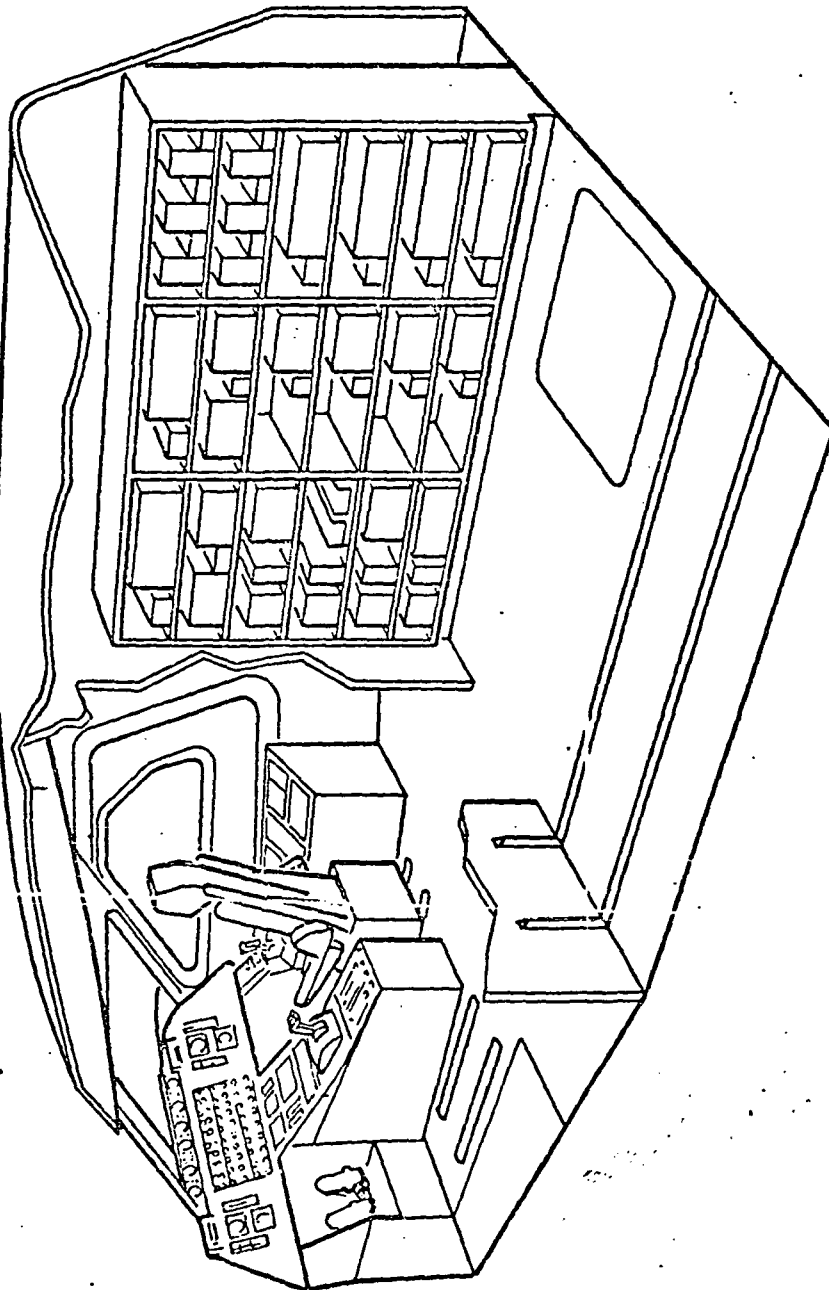


FIGURE 3.1-5. B-18 BOOSTER AVIONIC EQUIPMENT LOCATION



PRIMARY FEATURES:

- 2 MAN CREW
- CONVENTIONAL/DEDICATED INSTRUMENTS
- ACCESS TO AVIONIC EQUIPMENT VEHICLE VERTICAL & HORIZONTAL

FIGURE 3.1-6. E-18 INTERIOR LAYOUT

3.1.2 GUIDANCE NAVIGATION, AND CONTROL

Requirements

The primary requirements of the B 18 GN&C system hardware relative to the various mission phases are shown in Table 3.1-7. Each hardware group of the system is discussed in the following paragraphs in relation to its prime and secondary functional requirements throughout the various mission phases:

IMU & Computer - The IMU and the computer are used throughout the total booster flight for navigation. The prime requirements which dictate the use of this equipment aboard the booster come in the separation and entry phases. For a normal stage separation, engine gimbal schedule during separation and the booster coast-to-apogee pre-entry schedule (including the post-burnout maneuver) need to be contained in the onboard guidance storage. For abort separation, additional storage requirements exist such as a post-separation acceleration schedule, post maneuver guidance to recovery altitude with constraint monitoring, and a cruise burn maneuver to burnout.

During the total mission, the IMU and computer are used for navigation, guidance and flight control from lift off through reentry (approximately 20K feet altitude). For the cruise back and landing phases, this equipment will be used only for navigation and display of vehicle flight parameters.

Rate Gyros - The body mounted rate gyros are also used throughout the total flight for flight control/stability augmentation purposes. During the ascent phase through the high dynamic pressure regions of flight, true body rate at specific vehicle locations is required for stability augmentation purposes and cannot be obtained from derived data from the IMU platform.

Jet Select Logic - The jet select logic for the ACPS is required only in the separation and entry phases where dynamic pressure is below 50 psf and the aerodynamic surfaces are essentially ineffective for vehicle control. The pitch and roll logic is required in the entry phase down to approximately 140K feet

FUNCTION	MISSION PHASE UTILIZATION						RATIONALE
	ASCENT	SEP	ENTRY	CRUISE	LANDING		
IMU & COMPUTER	X	⊗	⊗	X	X		POST-SEPARATION MANEUVER GUIDANCE; CRITICAL TO INTACT ABORT
RATE GYROS	⊗	X	X	X	X		DERIVED RATE NOT USEABLE THROUGHOUT ASCENT DUE TO IMU LOCATION
JET SELECT LOGIC		X	X				
STABILITY AUGMENTATION SYSTEM			X	X	⊗		AUGMENTATION MANDATORY FOR ACCEPTABLE STABILITY MARGINS
AERO CONTROL FLY BY WIRE			X	X	X		APPROACH STABILITY PLUS WIDE TEMPERATURE RANGE MULTI-MODE OPERATION & LOW COST DICTATE USE OF FBW SAS

○ DRIVERS

Figure 3.1-7 Booster B-18 GNC Requirements

altitude however, the yaw logic is required down to approximately 60K feet altitude due to the vehicle high angle of attack which blanks the aerodynamic tail.

Stability Augmentation System (SAS) - The analog SAS, which drive only the aerodynamic surface controls, is required in all phases of flight after separation. The most critical requirement comes in the landing phase where vehicle stability augmentation is mandatory in order to achieve acceptable stability margins.

Fly-by-Wire (FBW) - The FBW control augmentation system is used as the prime control mode after separation. The use of a FBW approach for the manual control mode as opposed to a conventional aircraft mechanical control system is primarily dictated from design consideration associated with the vehicle. Expansion and contraction characteristics of the vehicle due to temperature excursions and structural deformation due to loads render a mechanical system of push-pull rods and/or cables impractical if not impossible. In addition, the inertia effects of the system when subjected to the "g" environment would be difficult if not impossible to tailor to meet acceptable flight control characteristics. Other factors including a large reduction in weight and overall program cost savings support the FBW approach for the booster. However, as discussed on p. 3-39, further evaluation of a mechanical system is planned.

GN & C Rationale - The GN & C system for the B 18 booster evolved from mission requirements, redundancy and complexity tradeoffs, cost considerations, and overall program design, development, and test schedules. The system concept was based upon providing a minimum amount of totally automatic functions and a maximum amount of crew utilization. In this manner a minimum amount of computer software development is required reducing cost and complexity. Adequate redundancy, utilizing independent equipment strings down to a using source, has been maintained to provide a fail operational capability. In addition, emergency manual override functions have been provided for all vehicle flight control systems allowing the crew to have a direct manual path to the control mechanisms.

The flight control hardware associated with all flight phases (except the computer controlled ascent phase) is separated from the guidance and navigation equipment. This includes all hardware associated with stability/control augmentation, air data computation, and the necessary electronics for interfacing with the aerodynamic surface servos, TVC actuators, and ACPS controls. This will allow the development and test of the necessary hardware in a time frame which is consistent with the vehicle horizontal flight test program without the dependence of the IMU/computer complex.

G&N System Configuration - The B 18 guidance and navigation, system, as shown in Figure 3.1-7a, consists primarily of an inertial measurement unit (IMU) and a GN&C computer. The system is triple redundant providing three independent IMU/computer functional paths and independent sources of heading, attitude, inertial velocity, position, angle of attack (α), and angle of sideslip (ρ).

Navigation position update is provided through the use of TACAN. Triple TACAN receiver/transmitters are installed, each of which interface with a computer and with the horizontal situation indicators (HSI) in the crew compartment. Multiple station range and bearing data can be input to the computer and/or displayed for homing or relative positioning.

Flight Control System Configuration - The B 18 flight control system is totally a fly-by-wire system. All control signals for the normal modes of operation (other than emergency manual takeover and aerodynamic surface stability augmentation) are generated in the GN&C computer. The control signals are used for automatic guidance utilizing computer control of the vehicle or for displaying axis deviations of the vehicle, relative to the nominal trajectory, which require pilot corrective action. Flight control functions relative to the various mission phases are described as follows:

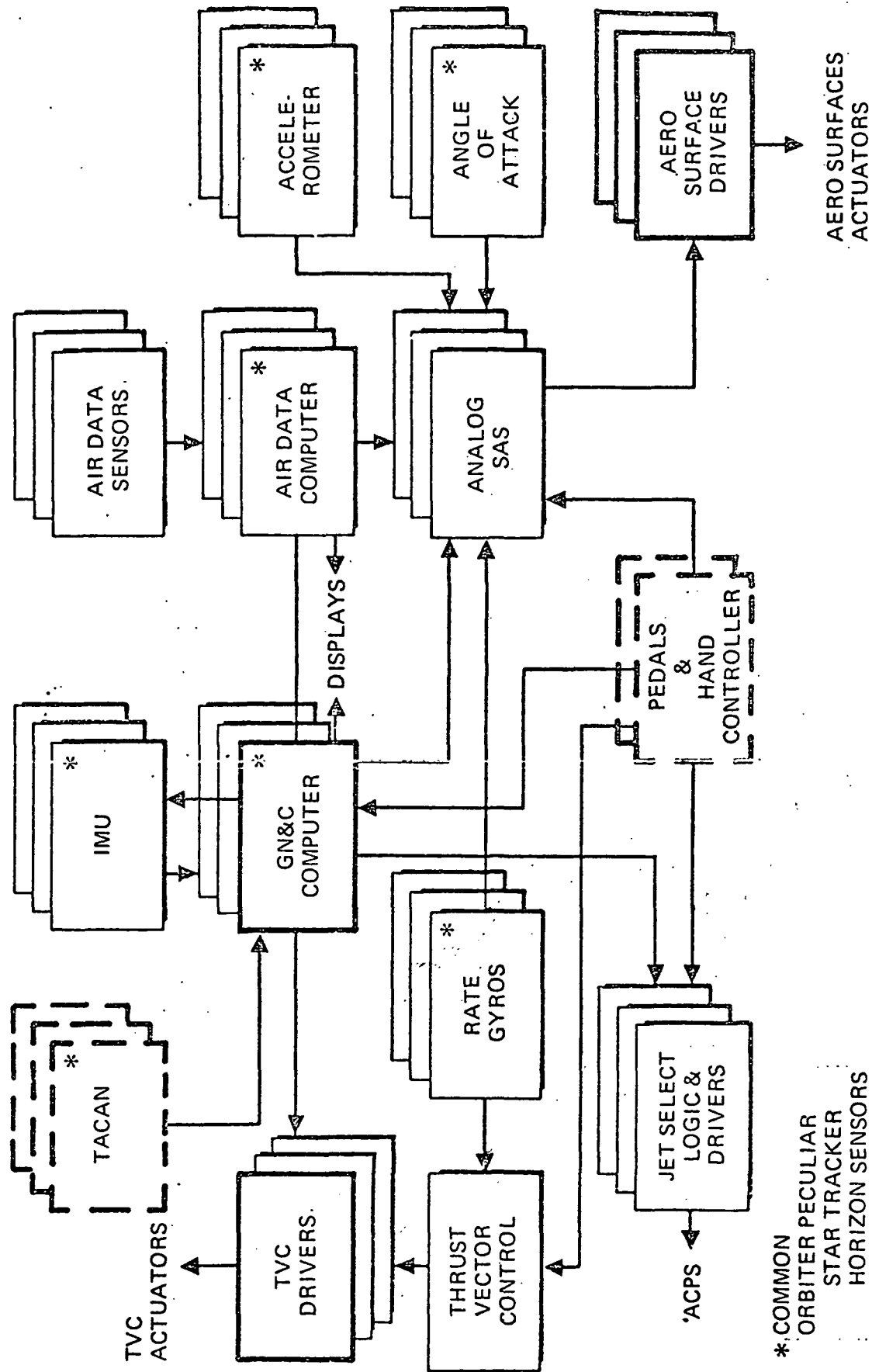


Figure 3.1-7a Guidance, Navigation & Flight Control

Ascent - In the ascent flight phase the normal mode of control will be through the GN&C computer. The computer generates flight control commands utilizing computed navigation data and stored guidance parameters. These control commands are output to the main engine thrust vector control (TVC) electronic drivers in the form of actuator position commands. The drivers contain the required electronics for median value signal select and for power amplification and isolation to interface with the actuator electro-hydraulic servos. In addition, the drivers contain the comparator networks for monitoring actuator position feedback versus position command and determining an out of tolerance condition. Fault tolerance commands will be output from the drivers for actuator centering purposes.

The computer also outputs ascent stability augmentation signals which are superimposed on the flight guidance control commands. Stability augmentation signals are generated using three axis vehicle rate data from triple redundant body mounted rate gyro packages, vehicle acceleration data derived from the IMU platform accelerometers, and air data parameters supplied from triple redundant air data computers (ADC). The ADC's will output the following data to the GN&C computer for flight guidance and stability agumentation signal generation and to the crew station for display:

<u>Function</u>	<u>Accuracy</u>
Mach No.	$\pm 10\%$ (Max of 5)
Dynamic Pressure (Entry Only)	$\pm 10\%$ (Max of 10 PSF error)
Indicated Airspeed (Subsonic)	$\pm 2\%$
Barometric Altitude	$\pm 3\%$
Angle of Attack, α	$\left\{ \begin{array}{l} \pm 1.0 \text{ deg, } \alpha > 20 \text{ deg.} \\ \pm 0.5 \text{ deg, } \alpha < 20 \text{ deg.} \end{array} \right.$
Angle of Sideslip, β	$\pm 0.5 \text{ deg.}$

An emergency manual override capability is also mechanized utilizing the pilot/copilot three axis rotation controllers. This mode of control would be used only in an emergency situation where an abort condition was eminent. The rotation controllers provide pitch, yaw, and roll commands to the non-redundant manual thrust vector control unit. This unit contains the necessary electronics to generate proportional vehicle rate commands which are output to the TVC drivers in the form of actuator position commands.

Separation to Entry - In the exo-atmospheric phase of operation the normal mode of control will again be through the computer. In this phase, which is initiated at separation, the computer will furnish flight control commands which are derived from the stored post-burnout and coast-to-apogee pre-entry guidance schedules. This auto guidance mode of control will be terminated in the entry phase when dynamic pressure (q) has increased to approximately 50 PSF. During this low q flight phase, the IMU supplies α and β data to the computer to an accuracy of ± 0.5 degrees. The computer outputs flight control commands in the form of vehicle body axis corrections which are transferred to the attitude control propulsion system (ACPS) jet select logic. This logic network contains the required electronics for median value signal select and for generation of the individual thruster commands to the 28 ACPS engines as a function of the input signal levels.

During this phase the aerodynamic surfaces are also active for the prime purpose of vehicle trim and stability augmentation. Trim commands are generated in the computer and output to the three axis analog stability augmentation system (SAS). The SAS receives vehicle rate and acceleration from the respective body mounted packages and dynamic pressure from the ADC. It outputs steady state trim commands with superimposed stability augmentation signals to the aero surface drivers. The drivers act as servo amplifiers for electrically closing the position servo loops. Auto trim commands are limited to the elevons and canards only. The SAS is triple

redundant providing a median value select position command on quadruple outputs to each aero-surface hydraulic servo-control. Each hydraulic servo-control has quadruple servo control. Each hydraulic servo control has quadruple servo channels and employs a force summing technique for output equalization. Each elevon servo control commands an individual elevon segment as a function of the pitch collective elevon and the roll differential elevon commands. In the flight regions above Mach 2, all segments on each wing are commanded for a full slab motion. Below Mach 2, the most outboard segments on each wing receive roll commands and the middle and inboard segments provide pitch control.

A manual override capability is provided throughout this flight phase for emergency takeover. Manual operation is the same as the entry phase and is described in the next section.

Entry - During the entry phase, atmospheric operation is initiated when the dynamic pressure has increased to about 10 PSF. Normal operation is pilot takeover at this point commanding pitch rate and bank rate. Three axis attitude deviation signals, derived in the computer from guidance and vehicle parameter data, are displayed on the three axis, Attitude Director Indicator (ADI). The pilot controls a specific angle of attack and bank angle to achieve the proper flight path.

Pilot pitch and roll control commands are input to the SAS operating in a control augmentation (control stick steering) mode. Elevon and canard position commands are generated in the SAS as a function of hand controller displacement and stability augmentation commands and input to the aero surface as described above. During the high angle of attack position of the entry phase (above 50K to 60K feet altitude) yaw control commands for coordinating turns, maintaining near zero sideslip, are generated in the computer and supplied to the ACPS jet select logic. ACPS yaw control is terminated at the pilot's option when the vertical stabilizer becomes aerodynamically effective. The rudder is then the sole source of yaw control.

Pilot input commands from the rudder pedals and turn coordination signals are generated in the SAS, in the same manner as are the pitch and roll commands, and input to the rudder aero surface drivers.

A manual bypass mode is also implemented during the entry phase in which the computer and the SAS are completely by-passed. This mode can be selected at the pilot's option but would not be used except under conditions of extreme emergency. Either total failure of the computer complex and/or the redundant SAS would have to occur. In this mode rotation controller and rudder pedal commands are input directly into the aero surface drivers and jet select logic for non-augmented vehicle control.

Cruise Back and Landing - During the termination of the entry phase (at approximately 20K feet altitude) the pilot initiates the deployment and start-up cycle of the air breathing engine systems (ABES). The pilot monitors deployment and start-up and can provide manual override in the event of abnormal operation. Thrust balance is also provided by the pilot.

The vehicle descends to a cruiseback altitude of approximately 13K feet. The cruise back phase utilizes aerodynamic surfaces only which are command from pilot inputs through the SAS. No computer steering signals are used in this phase. The pilot flies the vehicle using the standard flight instruments displaying attitude, heading, airspeed, barometric altitude, and angle of attack. TACAN homing and distance to the landing site is displayed on the horizontal situation indicator (HSI). In the final cruise portion of the phase, TACAN data, from single or multiple land stations, may be used to update the navigation problem if desired. During the let-down and landing phase, the pilot retains manual control following an approximate 3° glideslope, initiates flare and touchdown.

Booster Computer Memory Requirements - Tables 3.1-10 and 3.1-11 show a comparison of the computer memory requirements for the B 18 E booster and several other representative space vehicles. Major software functions are identified

TABLE 3.1-10 Booster Computer Memory Requirements

FUNCTION	APOLLO LGC	APOLLO CMC	SKYLAB	CENTAUR	SATURN SIMPLEX	LO2/RP BOOSTER
EXECUTIVE	1,598	1,721	1,712	1,167	4,867	2,500
DIGITAL AUTOPILOT	3,676	3,923	4,939	1,500	624	692
ATTITUDE MANEUVER	1,539	1,336	1,470			
ORBITAL INTEGRATION	2,602	2,599	2,115			
RENDEZVOUS NAVIGATION	1,515	2,812	2,643			
RENDEZVOUS TARGETING	1,769	1,974	2,255			
ALIGNMENT	2,658	2,241	1,931			1,663
IMU CONTROL	889	816	816			
POWERED FLIGHT	3,118	2,265	2,269	3,441	3,306	3,403*
PRELAUNCH		503	503	585	362 (4,608) (5,000)	1,546
LAUNCH MONITOR		518	512	870		
ENTRY		1,653	1,651			499

*ADD 631 WORDS FOR DOG LEG TO ACHIEVE POLAR ORBIT

TABLE 3.1-11 Booster Computer Memory Requirements (Continued)

FUNCTION	APOLLO LGC	APOLLO CMC	SKYLAB	CENTAUR	SATURN SIMPLEX	LO2/RP BOOSTER
LUNAR DESCENT	1,406					
LUNAR ASCENT	773					
ABORT CONTROL	289					700
ATM CONTROL			327			
INTERPRETER	2,211	2,203	2,205			
COMPUTER RESTART PROTECTION	968	1,123	1,117			EXECUTIVE
CREW-COMPUTER COMMUNICATION	4,237	4,015	4,033			3,200
RADAR SERVICES	2,039					
RETURN-TO-EARTH		1,337				
SPECIAL COMPUTATION FOR DISPLAY	1,360	1,642	1,498			
ORBITAL MATH ROUTINES	907	953	640			380
UPLINK & DOWNLINK	714	712	697	741	1,123	
SELF CHECK	314	312	312	15	794	1,051
ERASABLE MEMORY		2,048				
MEMORY WORD SIZE		16 BIT		24 BIT	26 BIT	32 BIT

TOTAL 15,634

separately to make the comparison more meaningful. Apollo LGC, Apollo CMC and Skylab are basically manned orbiting vehicles while Centaur, Saturn and the B 18 are boost vehicles. The Centaur and Saturn vehicles are unmanned while the B 18 is manned. These vehicle differences determine the software functions required to be implemented for a particular vehicle.

The B 18 executive software is sized at 2500 words. This is significantly larger than the executive program for all of the comparison vehicles except for the Saturn, especially when the memory word size is taken into account. The B 18 memory word is twice as long as that of the Apollo CMC. The variation in memory requirements can easily be accounted for by the difference in identifying executive functions such as computer restart protection which is included in the executive sizing estimate for the B 18 and is accounted for separately for Apollo and Skylab.

The digital autopilot function for the B 18 requires 692 memory words. This compares favorably with the Saturn vehicle but is significantly less than the other comparison vehicle numbers. The difference when compared with the Centaur requirements is because of the accuracy requirement for delivery a payload into orbit. Skylab and Apollo vehicles requires a complex autopilot because of the accuracy and flexibility of stabilization control.

IMU alignment for the B 18 booster requires 1663 words. This compares very favorably with the number of words required for the other vehicles for which this function is separately identified.

The powered flight memory allocation of 3403 words by the B 18 compares favorably with the other vehicles. It is somewhat larger because of flexibility to handle all azimuth launches.

Prelaunch software of 1546 words for the B 18 is significantly higher than the equivalent software for the other vehicles to minimize the expense of ground

support equipment and personnel.

The 499 words for the B 18 entry function is considerably lower than that for Apollo and Skylab. This number will probably increase when the software requirements for entry of the Shuttle booster is better defined.

Abort control is estimated at 700 memory words for the B 18. This estimate will very likely increase significantly when the abort control requirements are fully determined.

For the B 18 the computer restart protection function is included in the executive program estimate.

Crew - computer communication requires 3200 words for the B 18. This estimate is very close to the number of words required for the equivalent function on Apollo and Skylab.

Orbital math routines for the B 18 include the direction cosine computation and earth relative navigation and are estimated to require 380 words. The orbital math routines required for Apollo and Skylab are probably entirely different so no reasonable comparison can be made.

Self check software for the B 18 requires 1051 words and includes limit checks, reasonableness checks, command/response tests, maintenance system status and computer self test. Computer self test requires 356 words which compares favorably with the same function for Apollo and Skylab.

The interpreter used to reduce the memory required for the Apollo and Skylab computers is used for the orbital integration, rendezvous navigation, rendezvous targeting, lunar descent, lunar ascent, abort control, special computation for display, and orbital math routines functions. The only one of these functions that is common with software functions for the B 18 is the

orbital math routines and it is not valid for comparison.

The above comparison shows that most of the software functions identified for the B 18 booster are sized reasonably. One of greatest concern is the abort control which is not well defined at this time. The problem is much more complex than the abort control for the Apollo. The second area of greatest concern is the digital autopilot function which also needs better definition. The present software estimate is a total of 15,634 words. Sufficient growth is provided by sizing the GN&C computer memory available at 48,000 words to cover errors in estimates and the addition of new functions identified in the future.

The following functions are performed by the software identified in Tables 3.1-10 and 3.1-11:

Guidance

- Entry guidance
- Approach guidance
- Cross-product steering
- Prelaunch targeting
- Atmospheric guidance during boost

Navigation

- Prelaunch alignment
- Powered and unpowered navigation
- Earth and unpowered navigation
- Entry navigation
- Atmospheric navigation

Attitude Reference

- Direction cosine and Euler angle computation
- Gyro compassing during prelaunch

Flight Control

- Main engine thrust vector control
- ACPS control equations
- Thrust on-off logic
- Entry programs

Executive (GN&C)

Display Processing

Status Monitoring

- System Display
- Input Data Checks

In addition, the software includes aspects of mode sequencing by mission phase and computation time allocation through executive program control. Not included in the software functions and in the sizing estimates are the additional functions associated with abort, ferry and post flight operations.

Mechanical Control System Evaluation - During the second extension to the phase B study effort an in depth evaluation of a manual mechanical backup flight control approach will be performed. Vehicle stability and control aerodynamic data will be developed for all flight regimes and used to derive control system design parameters and constraints. In addition, the structural deformation of the vehicle in terms of deflection due to bending, elongation/contraction due to thermal effects and propellant loading will be evaluated throughout the mission profile.

Using the above data as design guidelines, a mechanical system layout will be made which minimizes control system travels and defines a suitable method for system compensation due to structural deformation. This design effort shall

include the study of cable tension regulator devices and push-pull rod position equalization mechanisms that will provide a practical solution to the problem. In addition, the hydraulic power controls operating in conjunction with the mechanical system shall be defined.

Parallel to the above effort, a control system simulation study will be performed to determine the vehicle design requirements with and without the incorporation of a stability augmentation system (SAS). From these simulation results the necessary vehicle modifications will be defined so as to be compatible with the mechanical system without SAS incorporation.

At the conclusion of the evaluation, a flight control system consisting of a minimum amount of SAS avionics combined with a mechanical system will be defined. This system will be traded-off against the present B 18 configuration.

Canard On/Off Study - The phase B second extension study efforts will also consist of an evaluation relative to the use of canards on the booster vehicle. This study will define the wing area and location and other vehicle parameters for a vehicle configuration not using a canard system. The control requirements of the two configurations (canard - no canard) will be identified. These requirements will consist of the necessary hinge moments and surface rates, stability augmentation functions, and the hydraulic drive system complexity. In addition, the vehicle handling qualities will be evaluated for each configuration. The results of the study will define the overall vehicle weight and cost differences and establish a recommended approach.

ACPS Aerosurface Study - The use of aerodynamic surfaces and ACPS engines for flight control will also be evaluated during phase B second extension. This study will be initiated with an examination of the booster flight trajectory in an effort to raise the q profile during the low q regions of flight. As a result of

this effort, the requirements for the ACPS will be minimized (or possibly eliminated) and various combinations of aerosurface/ACPS modes of control can be identified. The handling qualities, design complexity, cost, and weight of each of the combinations will then be evaluated which will result in the recommendation of the most appropriate configuration.

Altitude vs. Time B 18 - Figure 3.1-7b represents an altitude versus time plot of the B-18 configuration. The plot highlights specific flight parameters and events occurring during the ascent, separation, and entry phases. The maximum q condition is reached at 54 seconds at an altitude of approximately 23K feet. This compares to the B 9U configuration at 80 seconds and 42K feet. Separation takes place at 145.7 seconds (154 K feet altitude) and ascends to an apogee of approximately 205K feet. This compares to the B 9U configuration at 216.3 seconds and 244K feet with an apogee of approximately 266K feet.

During reentry, the B 18 reaches its maximum down range distance (159 NM) at 325 seconds when the vehicle is at 77K feet altitude. This compares to the B 9U configuration at 456 seconds and 84K feet. The end of the entry phase (20K feet altitude) where the airbreathing engines are deployed and started occurs at 654 seconds with the vehicle approximately 124 NM down range. This compares to the B 9U configuration at 657 seconds with the vehicle approximately 399 NM down range.

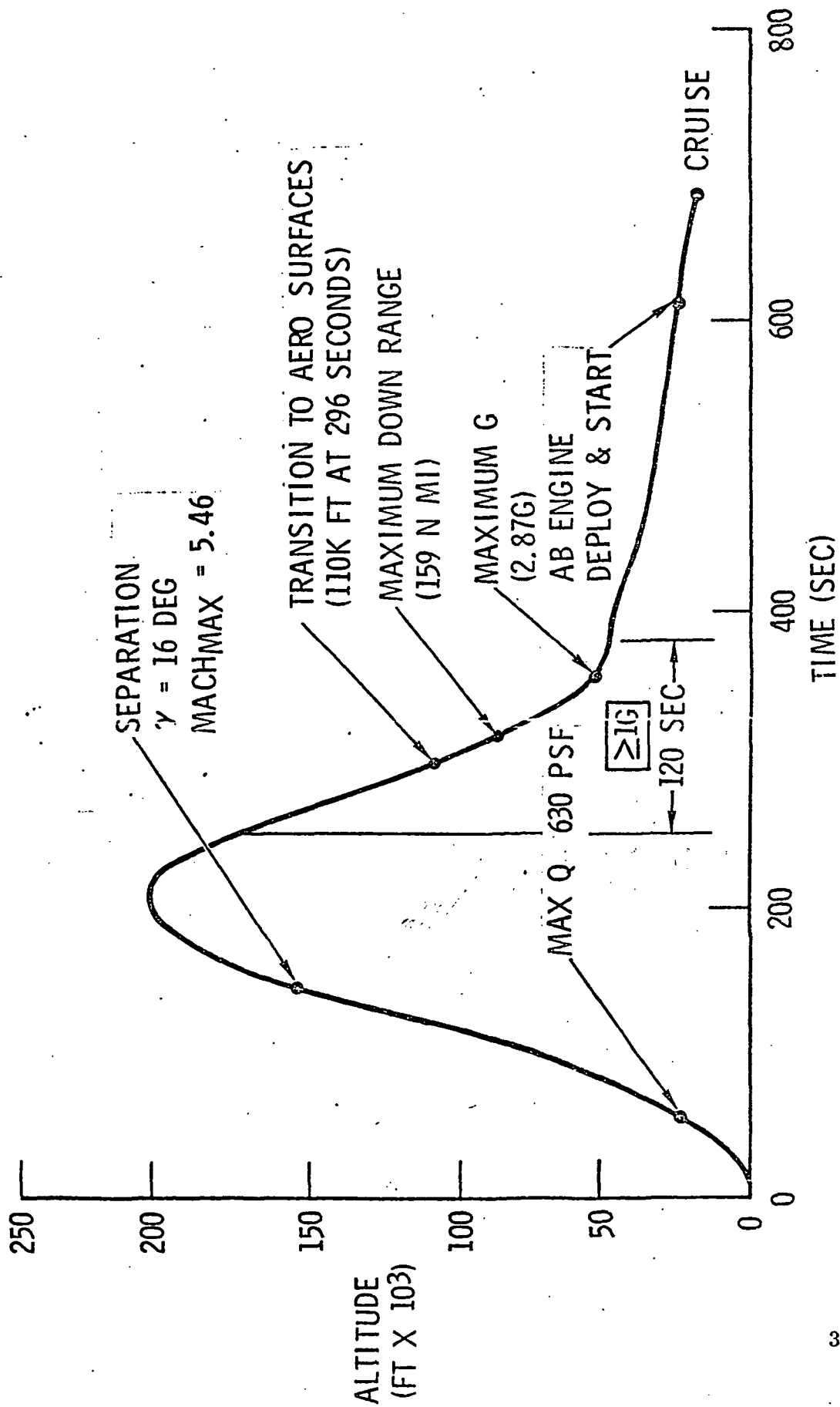


Figure 3.1-7b Altitude vs. Time

3.1.3 REDUNDANCY MANAGEMENT AND ONBOARD CHECKOUT

Redundancy Levels

Redundancy levels for booster avionics were arrived at through individual subsystem examinations. They vary from triply-redundant, as for example in GN&C and electric power elements, to two levels, as for example ATC transponder, to single non-redundant elements such as the S-band transmitter. Selections were made on the basis of criticality, reliability, and an understanding of the failure modes. In general, the redundancy level selector philosophy applied to the booster is the same as for the orbiter, at the same time recognizing the differences in booster and orbiter tester as well as flight duration.

Redundancy level selection decisions taken into account:

Criticality, that is the effect of the functional failure as determined through FMEA.

Credentials, that is the extent to which the hardware has proven itself.

Historical precedent, that is the practice or standard that exists for the nearest similar vehicles, i.e., large aircraft.

Interfaces, that is the compatibility with the redundancy management method.

Cost effectiveness, that is the practicability of the redundancy level in terms of costs, considering the amount of hardware, complexity, operations, weight, power, and tooling.

Individual subsystem redundancies are shown in the subsystem block diagrams.

An economic analysis was made of the effects on mission costs by varying the avionics fault tolerance levels. The analysis addressed the probabilities, at a given fault tolerance level, of mission abort/failure and of vehicle loss contributed directly by the avionics, and the added costs to accomplish those missions by additional launches and vehicle replacements. The numerical results shown in Table 3.1-12 represent probable program dollars lost by lowering fault tolerance levels from a baseline of FO/FO/FS, when not considering for the moment the difference in the avionics equipment costs.

Table 3.1-12. Redundancy Level Analysis — Avionics

CONCEPT	FAULT TOLERANCE		
	FS	FO/FS	FO/FO/FS
LOX/RP BOOSTER			
• NO ORBITER ABORT CAPABILITY	1,233.5M	\$76.5M	0
• ORBITER ABORT CAPABILITY	446.6M	28.7M	0
PRESSURE FED BOOSTER			
• NO ORBITER ABORT CAPABILITY	15.1M	0	0
• ORBITER ABORT CAPABILITY	0.4M	0	0

NOTE: DOLLARS SHOWN REPRESENT THOSE NASA SHOULD BE WILLING TO SPEND ON AVIONICS TO ACHIEVE THE NEXT HIGHER FAULT TOLERANCE LEVEL - BASED UPON THE PROBABILITY OF MISSION FAILURE & MISSION COSTS

Restating this in the other direction and staying strictly in the domain of economics, the results show that going successively from FS to FO/FS to FO/FO/FS that as long as the additional avionics costs are lower than the dollar shown in a particular column, the next higher fault tolerant avionics(next column to the right) is economical. Based on expected avionics costs the best economic choices are shown by the boxes. For example, for the B-18 Booster with orbiter abort capability, FS avionics will result in 446M more mission costs than FO/FS avionics. Further, that if FO/FS avionics costs less than 446M more than FS avionics, FO/FS avionics is a better economic choice, and that based on mission costs and avionics costs FO/FS avionics is the best choice.

Reliability Consideration

Reliability considerations above dictate that there shall be no critical single failure points; that flight safety systems have at least FO/FS capability; that nonflight systems has at least FS capability; and that once a functional path is lost, this loss will be detected.

Redundancy Management

Redundancy management follows the concept that the crew manages the subsystems with assistance from a performance monitor system. The performance monitor utilizes the output of the PCM system for its raw data regarding the vehicle subsystems. This data is processed, scaled and limit checked in the cockpit display processor. Pertinent failure messages are displayed on the cockpit CRT display. This display of out of tolerance data is supplemented with the standard Caution & Warning display. The crew accepts the failure message, from either source, evaluates the data and then makes decisions regarding the operational status of the subsystem elements.

Checkout and Fault Isolation

To maintain as much autonomy as practical with conventional off-the-shelf avionics equipment and at the same time minimize ground base flight support, onboard performance monitoring to the functional path level is used during flight. To achieve

the desired turnaround with minimal ground crew and ground equipment, it is necessary to be able to identify most problems to individual replaceable units. In contrast to the methods suggested in the main Phase B effort, fault detection in the alternate avionics scheme relies substantially on hardware in addition to software; diagnosis and correction are essentially crew oriented. For example, if fluid system exceeded hi-low limits are displayed and the crew isolates the failed functional paths by controlling valves; LRU's which employ BITE flag their failed status, and the crew manually switches to their replacements; hydraulic pressures and tank quantities are metered and the crew tracks their values knowing their adequacy.

3.1.4 DISPLAYS AND CONTROLS

Alternate Design Impacts - Within the framework of this low cost avionics study, a large number of alternate designs were studied which had potential impact on the crew station. For example a third crewman was considered; was then shown unnecessary by workload analysis and finally rejected as uneconomical. In another variation a shortened vehicle and placement of canards had suggested a possible need to place the crew compartment and the avionics equipment bay in a stacked configuration. This was also rejected. The final B 18 configuration allowed a return to the original single pressure vessel design with only a slight shortening of the avionics equipment bay required due to a revised canard location. Another vehicle change was to a tandem configuration which allows overhead crew ejection. This change simplified considerably previous requirements to rotate the seats outboard during the ejection sequence.

A change of the mission profile, including a lower staging velocity, produced a shorter mission and reduced g load during the entry maneuver. Impact on the crew includes reduced stress from acceleration and a shorter flyback.

These changes were considered along with changed subsystem requirements and specific low cost avionic system alternate approaches. The phase B displays and controls design aids were used as a baseline to study the new overall layout.

Displays and Controls Layout - Starting with the same panel working area (3400 sq. inches), signal and measurement information gathered from the subsystems was analyzed to assess the new requirements. The rationale used for space allocation was much the same as in previous layouts. A key factor is provision for an emergency backup capability. This requirement has always existed even in the previous integrated display baseline. One difference however in the present baseline is that controls and displays are hardwired instead of tied to the data bus and central computer. Prime flight displays, CRT's, keyboards and backup

flight instrumentation are generally common between orbiter and booster. Subsystem dedicated, hardwired controls and displays are arranged generally similar to previous layouts. The revised baseline displays and controls layout which has been produced requires approximately the same panel space except that two additional panels are required and are located out of reach from the seated crew positions. These panels are used for the location of 800 circuit breakers. Previously, remote power controllers were used but with the requirement to switch to low-cost aircraft type circuit breakers about 600 sq. inches of accessible panel space became necessary to hold these breakers. This created no special problem on the booster and was handled in a manner similar to that used in present aircraft design. Figure 3.1-8 illustrates the new booster displays and controls layout. The selection of the dedicated prime flight displays was made employing minimum development, low cost, proven design instruments. Ground ruling out the central computer and multipurpose displays caused other obvious changes to display and control selection, allocation and layout as indicated in the above referenced figure.

Crew Work Loading - A preliminary crew task/timeline was prepared in scenario form representing only nominal operating tasks. A preliminary allocation of tasks was accomplished to aid in display and control layout. Using this scenario times were established for major activity throughout the booster mission. Both pilot and commander times were tallied and these were maintained separately for operational (controlling or control associated tasks) monitoring tasks. Figure 3.1-9 illustrates this loading by mission phase. Considerable work load analysis is still required to define detailed operational procedures to handle contingencies and redundancy management. It was, however, possible to establish a reasonable allocation of crew tasks based on the selected displays and controls layout. The appearance of heavy work-load during airbreathing engines deployment and landing can be attributed to periods of close attention quite similar to normal large aircraft operations. It should be

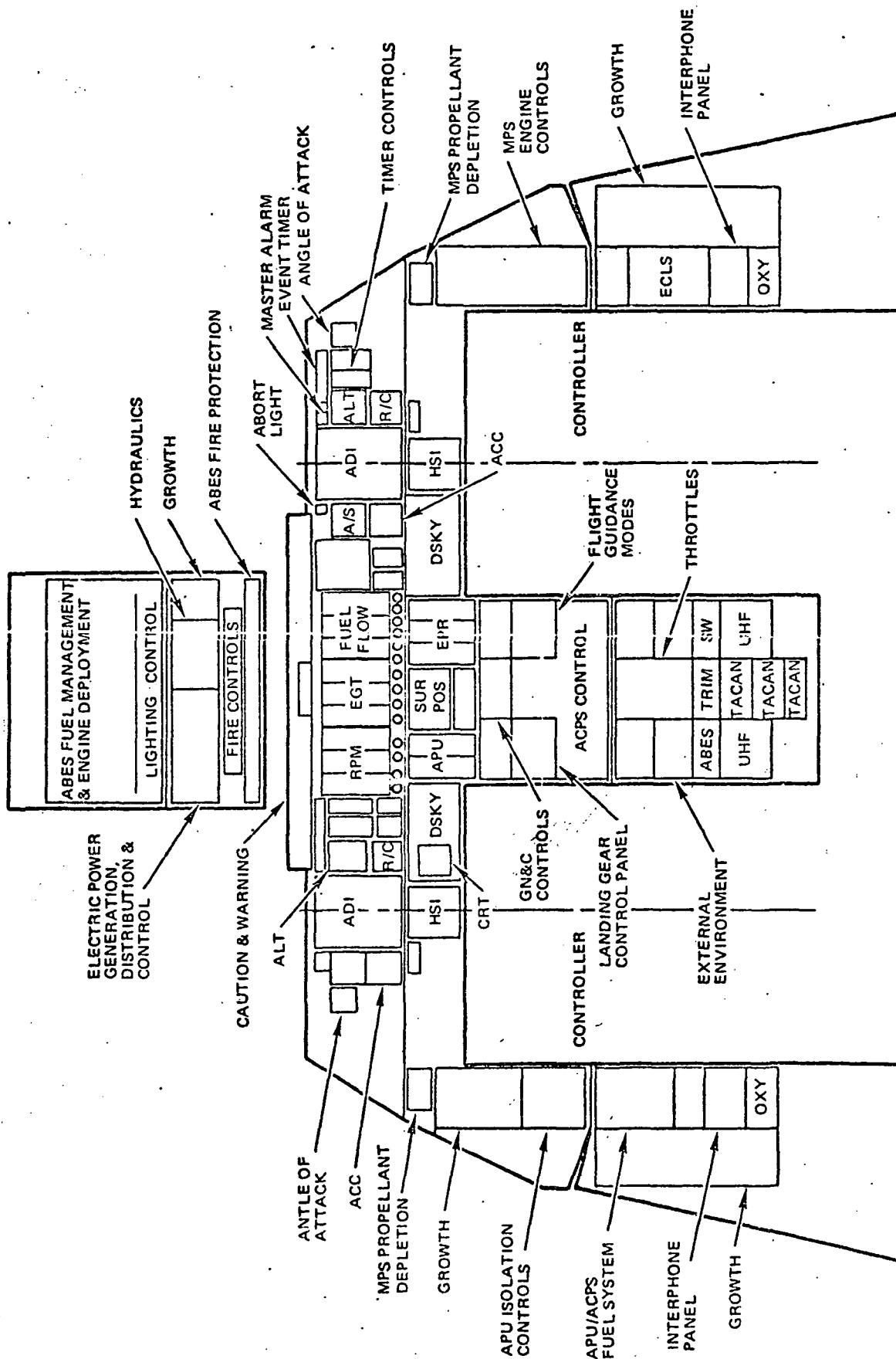


Figure 3.1-8 Space Shuttle Booster Crew Station

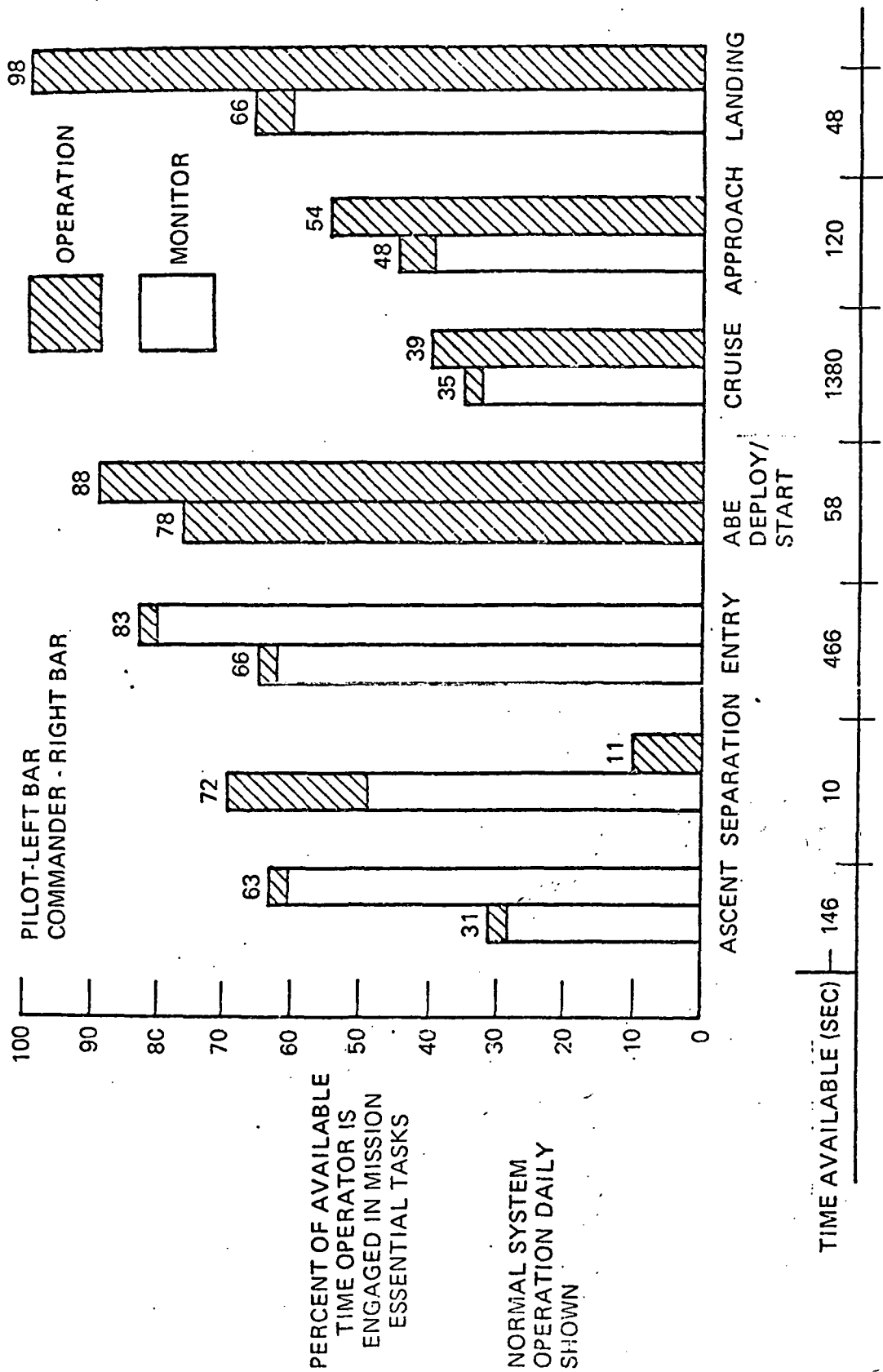


Figure 3.1-9 Booster Flight Crew Workload

noted that both of these mission phases are for very short durations of time where a heavier schedule for monitoring was considered as operation time. Several important points that require additional study came up during the analyses. For example it may be possible to shift task loading during these critical phases by deploying engines sooner thus allowing additional time for contingencies. In addition, an ILS system (mentioned as a design option) and a radar altimeter might be important landing aids which could reduce crew work load.

One area of crew workload which received a more detailed consideration was the airbreathing engine displays. The selectable time-shared display defined in the Phase B baseline was unacceptable as a new development item. Using a rough layout for the Booster main front panel it was determined that either round dial or tape-type instruments could be utilized in the space available. A workload study summarized in Table 3.1-13 indicated that a mix of vertical scale and round dial instruments was the optimum approach. RPM, EGT, FUEL FLOW, and OIL PRESSURE (10 round dials below fuel flow) are located in the best scan area. EPR which is not in the normal scan pattern was placed only slightly below the main parameters. While types of instruments were being studied, a comparison of all normal tasks associated with airbreathing engine deployment, startup and monitoring was also underway. Such time critical tasks were analyzed to determine the proper design approach. Although previously highly automated through the central computer (which is no longer a part of the baseline) these tasks were examined in relation to both the new mission profile and alternative ABES subsystem considerations. Phase B auto sequence had 12 engines deployed and up to full thrust in 87 seconds. It was determined that the crew could best handle contingencies if this sequence remained automatic. This baseline design presently provides manual initiation at 20,000 feet with all ten engines deployed and running in 57 seconds. The ABES Engine Deployment and Fuel Management Control Panel located overhead was only modified to reflect

Table 3.1-13 ABES ENGINE INSTRUMENT SELECTION

DESIGN CONSIDERATION (TYPES OF INSTRUMENT)	NUMBER REQUIRED	SPACE REQUIREMENT (CREW STATION)	ESTIMATED SCAN TIME
1. 2" ROUND DIAL	10 ENGINES X 5 PARAMETERS = <u>50</u>	280 sq. Inches	1 Sec/DIALS = <u>12</u> Sec.
2. VERTICAL SCALE (6.31" x 6.38" DISPLAY w/10 SCALES)	5 VERTICAL SCALE DISPLAYS	198.30 Sq. Inches	1 Sec/DISPLAY = <u>5</u> Sec.
3. MIX OF VERTICAL SCALE & 1" ROUND DIALS FOR OIL PRESSURE PARAMETER	4 VERTICAL SCALE DISPLAYS + 10 ROUND DIALS	173.08 sq. Inches	1 Sec/VERTICAL DISPLAY + 1 Sec/4 DIALS = 7.5 Sec.

Other Considerations: 1. Readability - Parallax becomes a problem with round dials when a large scan area is involved.
2. Differences in individual engine relationships can be detected faster on vertical scale displays as compared to round dials.

* Design choice - design consideration #3 has the lowest space requirement & a reasonable scan time requirement.

subsystem design changes. This panel continues to represent a logical sequential flow type of layout, easy to understand and allowing crew capability to manually override or reconfigure the system as necessary. Other subsystem panels are updated to carry the same design approach to aid in training proficiency and to reduce errors in time critical task areas.

3.1.5 COMMUNICATION AND TRACKING

Design Requirements

A requirement exists to provide for interchange of data by voice communication between the booster crewmen and between crewman and:

1. Ground personnel during prelaunch activities.
2. Ground personnel during flight.
3. Orbiter personnel during mated flight.

An additional requirement exists to provide for terminal phase area navigation and position update to the Inertial Measuring Units utilizing ground based radio aids.

A further requirement exists to provide for booster operation within FAA controlled airspace.

Equipment Configuration. The communications and tracking (C&T) subsystem comprises those facilities provided for communicating within the booster, with the orbiter, and ground facilities; and those for determination of vehicle position and identity in cooperation with ground installations. These include hardwired links to the orbiter, maintenance and crew stations, and launch complex as well as radio links to ground facilities. Figure 3.1-10 illustrates the C&T subsystem configuration.

Audio. A triply redundant two channel audio center is provided for mixing, amplification, control and distribution of audio signals originating at the crew and maintenance stations, and orbiter and launch complex umbilicals. Each crewman is provided with personal equipment which includes two independent microphones and earphones.

Radio Communications. A redundant air to ground voice communication link is provided for air traffic control by two simplex UHF-AM transceivers. Each of which can be connected to one of two complementary antennas to provide nearly spherical coverage.

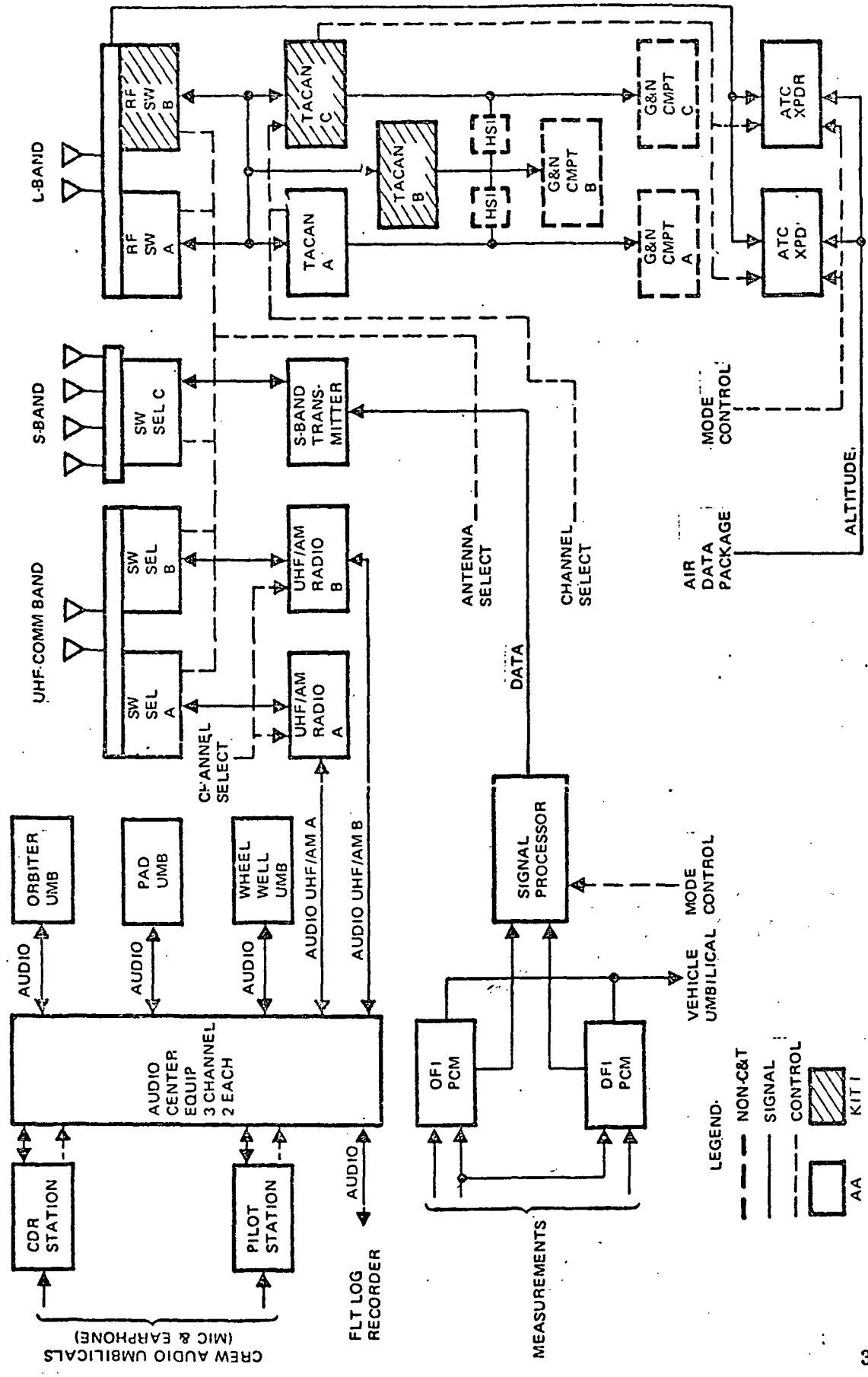


Figure 3.1-10. Booster Communications & Tracking Subsystem Configuration

Radio Navigation. Redundant terminal airway navigation capability is provided by two TACAN receiver-transmitters operating in conjunction with the vehicle's vertical/directional gyro package and horizontal situation indicators (HSI's) to display range, bearing and selected course deviation. TACAN range and bearing are also available to the G&N computers for position update if required. Each receiver-transmitter can be connected to one of two complementary antennas for nearly omnidirectional coverage.

Identification. Redundant identify and altitude reporting in response to ground interrogation is provided by two AIMS compatible ATC transponders, each of which may be alternately connected to complementary antennas for nearly spherical coverage. Encoded altitude is obtained from the central air data computer (CADC).

Antenna Installation. The antenna configuration, illustrated by Figure 3.1-11, provides the required spatial coverage and minimizes the effects of vehicle separation and control surface displacements on radiation patterns.

Two flush mounted antennas covering the upper and lower hemispheres are selectable to insure reliable UHF communications throughout the booster flight. Coverage in the upper hemisphere is required for the entry turn maneuver as well as for some launch orientations.

Two L-band antennas covering the upper and lower hemispheres are selectable and multiplexed for use with two ATC transponders and three TACAN receiver-transmitters. Blanking will be required between these equipments to guard against interference.

The S-band telemetry transmitter is provided with four antennas interconnected by hybrid junctions for protection against fading during the booster flight profile.

Rationale. Hardwired communication between booster and orbiter and between booster and launch complex have been selected as most cost-effective and to insure communications integrity during prelaunch activities when the vehicles are shrouded by the gantry.

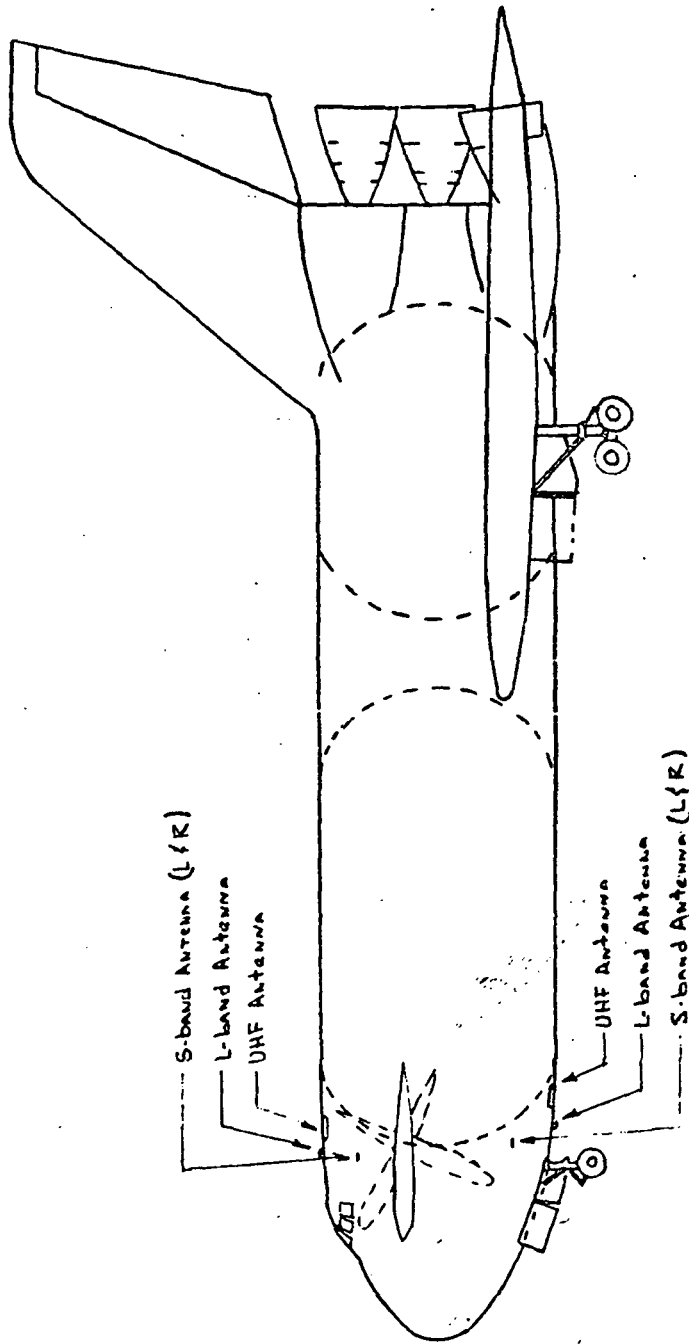


Figure 3.1-11. B-18 Antenna Locations

UHF radio communications are selected since the booster will never be beyond the radio horizon and reliance upon other than the standard line-of-sight air traffic control link is not justified.

The ATC transponders have been included in compliance with FAA regulations governing operation within controlled airspace.

TACAN has been selected to utilize existing ground installations for position update data as well as to provide terminal area navigation.

3.1.6 BOOSTER INSTRUMENTATION

Booster Signal Requirements

Booster signal requirements for the Operational Flight Instrumentation (OFI) is shown in Figure 3.1-12 and for the Development Flight Instrumentation (DFI) is shown in Figure 3.1-13. Booster instrumentation requirements consist of acquiring system and subsystem status, performance, maintenance, and trend data by means of sensors during checkout and operational modes; conditioning the data for analysis purposes; and timing signal generation and distribution. These requirements on sensors, signal conditioners, recording, and timing are discussed below.

Sensor Requirements. Of the booster measurements, about 40 percent are discrete on/off type that do not require sophisticated sensors. The remaining measurements may be acquired by about 25 different types of analog sensors over various ranges. The number of measurements requiring advanced state-of-the-art sensors is small and is typically predicated on environmental or accuracy requirements rather than the type of measurement itself.

Signal Conditioning Requirements. The analog measurements will need some form of signal conditioning. Approximately 80 percent of the measurements will originate in off-the-shelf hardware and some of those will contain the sensor and associated signal conditioning as a unit. Environmental conditions throughout the booster require a major portion of the signal conditioning hardware to be remotely located from the sensors and placed in one of the avionics equipment locations.

Recording Requirements. Maintenance data, including malfunction/failure data and parameters for trend analysis, and propulsion data will be recorded on the OFI flight data recorder. Incident investigation data, which is required to determine the cause of an accident, must be able to survive a catastrophic accident and will be recorded on the flight log recorder. Developmental flight data will be recorded in PCM form on a flight data recorder and in high frequency analog form on a FM wide band recorder.

SUBSYSTEM	BOOSTER TOTAL			COCKPIT DISPLAY		
	ANALOG	DISCRETE	TOTAL	ANALOG	DISCRETE	TOTAL
MAIN PROPULSION	182	83	265	7	36	43
ATTITUDE PROP. SYS	46	116	162	18	72	90
ABES	223	54	277	83	32	115
HYDRAULIC	24	122	146	24	26	50
APU	38	36	74	26	20	46
EC/LS	5	14	19	2	8	10
NON-AVIONICS SUBTOTAL	518	425	943	160	194	354
INSTR & DATA	0	15	15	0	15	15
ELEC PWR DIST	39	31	70	24	31	55
COMMUNICATION	2	12	14	0	8	8
GN&C	60	70	130	30	30	60
AVIONICS SUB TOTAL	101	128	229	54	84	138
GRAND TOTAL	619	553	1,172	214	278	492

Figure 3.1-12 B-18 Booster Signal Requirements

BOOSTER SUBSYSTEM	NO. OF MEASUREMENTS
BASIC TEST FUNCTIONS (FOR TESTING MOST SUBSYSTEMS)	102
STRUCTURES (VIBRATIONS, LOADS, TEMPERATURES, SEPARATION, ETC)	309
AIRBREATHING PROPULSION	373
AUXILIARY POWER UNIT	164
MAIN ROCKET ENGINES	155*
AUXILIARY PROPULSION SYSTEM	126*
EC/LS	30
PGD&C	258
HYDRAULIC POWER	45
GN&C	307
COMMUNICATION SYSTEM	67
LANDING GEAR	116
DFI TOTAL	2052

*NOT REQUIRED FOR HORIZONTAL FLIGHT TESTING

Figure 3.1-13 Development Flight Instrumentation No. of Measurements/
Booster Subsystem

Timing Requirements. Three time references are required by the booster: (1) Greenwich Mean Time (GMT) for data tagging, (2) Mission Elapsed Time (MET) for time-line sequencing, and (3) time independent of GMT/MET for sequence and operation timing.

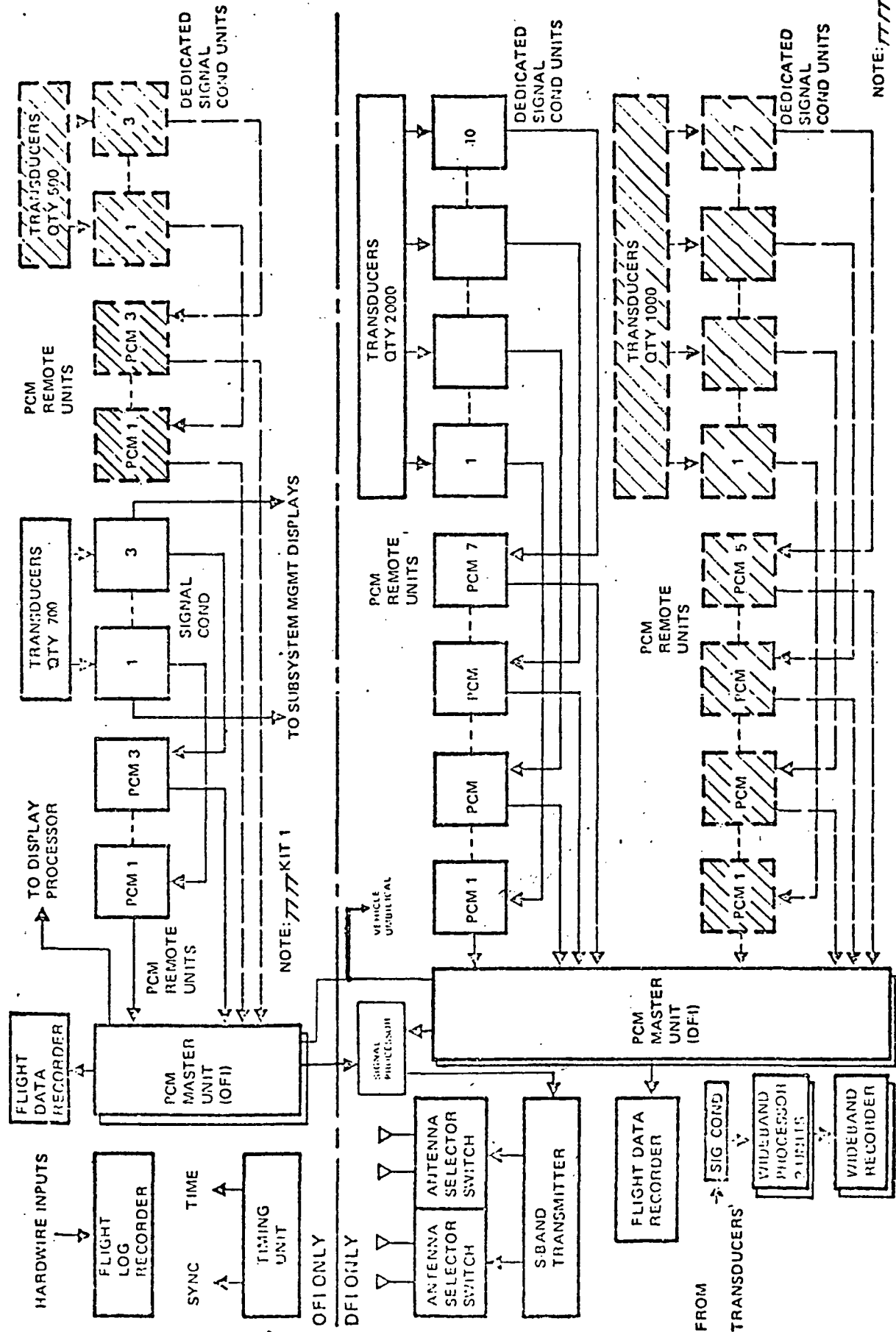
GMT and MET will be displayed on the time of year display and independent time (59 minutes, 59.9 seconds) will be displayed on the digital event timer. Timing synchronization is required by the following systems:

1. GN & C
2. Instrumentation
3. Cockpit D&C
4. Separation

DFI requires millisecond resolution, and clock tic phase shifting control to vernier data sampling.

Digital event time is required for crew use. It requires a range of 59 minutes, 59.9 seconds, with 0.1 second resolution, with pre-set capability and up-and-down count capability.

Instrumentation Configuration. A block diagram of the instrumentation subsystem is shown in Figure 3.1-14. The instrumentation subsystem configuration evolved during phase B extension places strong emphasis on minimum development and minimum risk concepts. To this end extensive use will be made of proven, off-the-shelf components. Use of this approach maintains the ability to achieve the goal of a high percentage of commonality between booster and orbiter instrumentation. The selected components have been configured into an instrumentation



NOTE: KIT 1

Figure 3.1-14 Instrumentation Subsystem Space Shuttle Booster Interim Avionics System

subsystem responsive to the booster subsystem operation and test philosophy and associated measurement requirements. A few of the components utilized in this system are different than those in the orbiter because of differing requirements (such as operation in an environment which is unique to the booster).

The system shown in Figure 3.1-14 uses PCM remote multiplexers to convert from normalized high level analog signals to digital signals. The PCM remote units are controlled by the PCM master unit. The formatted serial bit stream out of the PCM master unit is both telemetered by an S-band transmitter and recorded on a magnetic tape recorder. Similar systems have been configured for a number of recent space vehicles and high speed aircraft. The off-the-shelf designs and components for these systems are being applied to the space shuttle booster and orbiter requirements. A few channels of FM will be used on signals requiring high frequency response such as acoustic measurements. These will be recorded on a wideband recorder.

Pulse Code Modulation (PCM) was Chosen as the Primary Submodulation Method. In PCM discrete values of analog data samples are converted to binary coded words which are then assembled into a pulse stream for recording or transmission.

Reasons for choice of PCM over FM or other submodulation method.

1. Superior signal to noise performance
2. Greater accuracy
3. Lends itself well to microminaturization and integrated circuit technologies
4. Insensitivity to system nonlinearities after digitizing (thus can achieve high accuracy as required for guidance data)
5. Ready automation of ground equipment for digital data processing without further degradation
6. Versatility of test and data format, e.g., (1) single bit high-response "on-off" indications can be systematically

interlaced into the bit stream with minimum waste of information capacity; (2) pulse counting measurements can be readily implemented.

7. Flexibility of data channel frequency response through supercommutation and subcommutation channels and the ability to reprogram these if required.
8. Reduction in system volume (for large system 100 channels and up).
9. Reduction in number of RF links.
10. Highly flexible programming, lends itself to automatic checkout.
11. Less power and weight per vehicle.
12. Less cost per measurement.

3.1.7 ELECTRIC POWER GENERATION, DISTRIBUTION, AND CONTROL

Rationale

The major impact of the alternate avionics concept on the PGDC subsystem is that the data bus has been deleted. The computer/data bus/ACT concept was previously responsible for the control signals to the remote power controllers, both solid state and hybrid (EM + SS).

Conventional practice for control of electric power is to locate the load buses in the crew compartment and provide direct access to the thermal circuit breakers that provide wire protection. This requires that all the power be transmitted from the aft APU compartment forward to the crew compartment for redistribution to the aft loads and to other local loads. The weight and congestion of wire bundles makes this practice undesirable.

An alternate conventional approach used on high density aircraft, is to run feeders for local distribution near load centers with limiters for wire protection of the load circuits. Only the control wiring for the relay or contactor control runs from the crew compartment with an associated switch and low current circuit breaker. The only possible disadvantage is that the fuses are not accessible to the crew, and indication is not available.

An alternate approach that has been developed places the thermal circuit breaker at the remote load center. The circuit breaker is fitted with a solenoid to operate the contacts from a remote position. Several arrangements have been considered to provide indication of a tripped remotely controlled circuit breaker (RCCB) at the crew station. All of these depend on a small (1 ampere) thermal circuit breaker that will trip in response to the tripping of the RCCB.

The RCCB has been chosen for the base line to accommodate the majority of load switching requirements since it is developed and in use on commercial aircraft (DC-10), and provides indication to the crew of malfunction.

Power Generation

Isolated generator operation has been the baseline approach since it is inherently less complex and therefore less expensive, more reliable and easier to maintain and operate than a paralleled system. Isolated systems require one additional control to avoid the beat frequency, (i.e., the difference or low frequency) disturbance to the utilization equipment. These low frequency signals are easily coupled into avionic and sensitive control circuits.

Although synchronization of the APU's has been the baseline for the booster, the feasibility of providing this control is complicated by the fact that the APU's are not dedicated to generation of electric power, since nearly 90% of the APU horsepower is provided for hydraulic pumps. It is therefore prudent to avoid penalizing the APU by requiring it to control less than 10% of the power to the same very fine quality required of the electric power system.

Isolation of the generators from the transient frequency disturbances caused by hydraulic loads is readily accomplished by adding a constant speed drive between the APU and ac generator. A developed system for integrating the generator and drive (IDG) exists and has proven in-service reliability. These systems are also available with synchronizing controls (F-111).

Subsystem Design Requirements

General. The booster electrical power subsystem comprises power generation, conversion, distribution, and control elements required to supply electric power of the types, quantities, and quality necessary to reliably operate the utilization subsystems and their electrical, electronic, and electromechanical components. The subsystem configuration selected for the booster has been formulated on the basis of commonality with the orbiter, booster operational concepts, subsystem functional requirements, safety considerations, maintenance philosophy, trade studies, and risk assessments. Power source and conversion equipment sizing has been established from an analysis of power demands occurring during the various vehicle operational modes.

Primary Functions. The primary functions defined for the PGDC subsystem are:

1. Generation, regulation, and control of primary ac power.
2. Conversion of primary ac power to and control of dc secondary power.
3. Provision for and control of secondary batteries.
4. Protection of buses and distribution circuits against failures of power source and grounding/overload faults.
5. Control of ac and dc power to subsystems, from the interface with the DCM and/or the D&C subsystems.
6. Distribution and control of all generated and converted forms of electric power from source terminals to load terminals.
7. Generation of signals required by the D&C subsystem to provide status, configuration, fault, and caution and warning information.
8. Interconnection of subsystem LRU's.
9. Interconnection of GSE power source and controls to booster subsystems.

Requirements and Guidelines. The following general considerations are incorporated in the PGDC subsystem design:

1. Provide critical functions with fault tolerance service concepts consistent with integrated avionics system (IAS) and main propulsion engines.
2. Power transfer between orbiter and booster is not required.
3. Electrical power characteristics of booster PGDC to be essentially common with those of the orbiter.
4. Provide maximum commonality of PGDC hardware for orbiter and booster.
5. Provide for onboard checkout, including BIT as available and convenient use of conventional carry-on test equipment.

General Description

Salient Characteristics. The PGDC subsystem features the following:

1. Redundant functional components supporting a FO/FS hybrid fault tolerance concept serving critical subsystems.

2. Manual and automatic operational and test control.
3. Completely automatic fault (and failure) sensing, control, and distribution system management.
4. D&C monitoring of critical subsystem parameters for display, fault isolation, and maintenance action support.
5. Completely automatic, unmanned system startup provisions.
6. Backup dc power sources for critical management functions.
7. Component commonality and subsystem similarity with the orbiter.

The simplified block diagram (Figure 3.1-15) illustrates most of these features in a conceptual form.

The ac Subsystem. The primary electric power source consists of three ac generators, their associated control/regulator units, line contactors, and bus tie contactors. Each generator channel is connected to a main three-phase ac bus through a generator line contactor (GLC).

A three-phase transfer bus provides a means of interconnecting main buses and generators through bus tie contactors and also serves as the tie point for GSE power. The transfer bus status is monitored by the transfer bus monitor, which provides a discrete signal for switching logic. The ac distribution system consists primarily of feeders and local ac distribution buses located in the forward and APU and aft vehicle equipment compartments.

The dc Elements. The dc power required by the booster utilization subsystems is supplied from two sets of redundant transformer-rectifiers (TR's), three located at the forward load center and three at the aft load center. The output of each TR is connected to a main dc bus through a dc line contactor (DCLC). The main dc buses are normally isolated from each other but may be interconnected through the normally open dc bus tie contactors (DCBTC's), which form a ring-bus configuration. Two batteries, one near the forward equipment compartment and one in the APU compartment are also included in the dc elements of the PGDC.

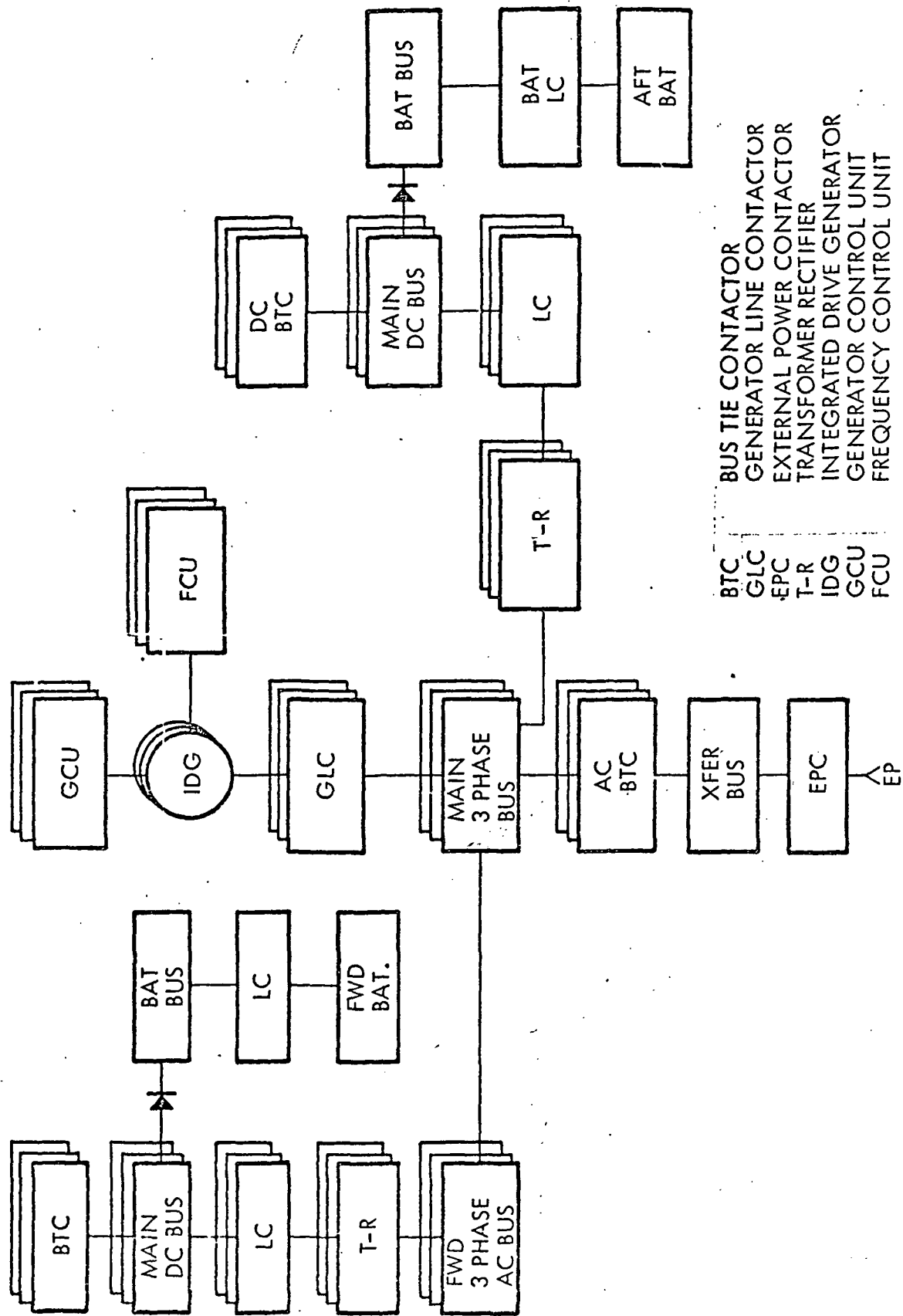


Figure 3.1-15. B-18 Electric Power Generation, Distribution & Control

Equipment Descriptions

The ac Generators. (See Figure 3.1-16). Each ac generator is rated 20/30 kva, 120/208 volts, 400 Hz and is capable of delivering 30 kva continuously and 40 kva for five seconds. The latter rating is achieved at the expense of some degradation in power quality.

A generator rated at 20/30 kva will accommodate the load profile of the booster. The "slash" rating means that short circuit current capacity and the 2.0 per unit, five-second load requirement are based on the 20 kva rating. Rated generator life is also based on 20 kva; however, the generator is capable of delivering 30 kva continuously with some degradation of life expectancy from resultant higher insulation temperatures.

The generator is driven by a hydro-mechanical constant speed drive (CSD). The CSD and generator are combined into an integrated unit for maximum reliability. A cooling oil system, completely independent of the APU gearbox is provided. External to the integrated drive — generator (IDG) the oil is cooled by an hydraulic oil heat exchanger. The IDG contains an oil pump to provide the required flow and pressure for the generator to lubricate the single antidrive end bearing. The cooling oil circulates through the stator and rotor passages to remove heat from the magnetics. Spray nozzles direct oil against the stator and rotor winding end turns and rectifiers. No rotating seals are required between the generator and the CSD.

The main generator field, ac exciter stator, rectifiers, and permanent magnet generator are mounted on a common shaft and form the rotor assembly. The rotor assembly is supported at the anti-drive end with one main bearing and an adjacent auxiliary bearing. A bearing failure detector senses loss of the main bearing. The smaller auxiliary bearing, normally inoperative, will operate a minimum of four hours at full generator load. The drive end of the rotor assembly is supported by the CSD bearing and spline assembly.

The generator housing contains the main three-phase stator, terminal block, electrical connectors, differential protection current transformers, exciter field, and

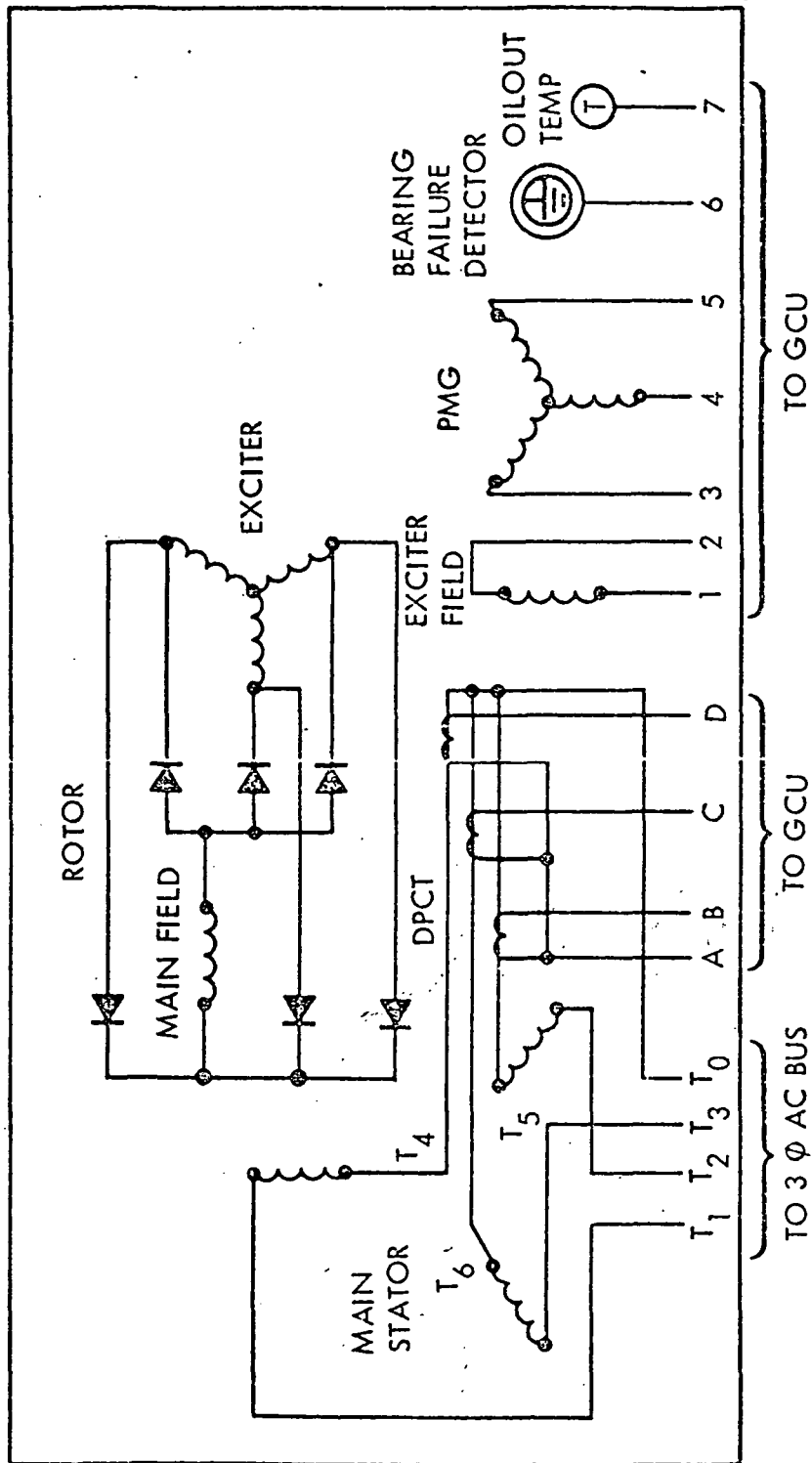


Figure 3.1-16. Simplified Block Diagram of ac Generator

permanent magnet generator (PMG) stator. The aluminum housing is bolted to the CSD, using locating dowel pins for alignment.

The CSD is a gear differential type, wherein the major portion (> 70%) of the transmitted power is carried through the drive mechanically, depending on input speed. The make up power is transformed hydraulically through a variable and fixed displacement unit, with the role of motor and pump changing with input speed. This hydraulic log unit operates in parallel with the mechanical transmission element and inputs to a gear differential to add or subtract speed as needed to maintain the generator at constant speed. This arrangement relieves the APU of precise speed control and provides a high degree of isolation for the generator from the sharp frequency transients generated by the three hydraulic pumps driven by the APU.

The CSD accessories include a basic governor for coarse speed control ($\pm 1\%$) which is trimmed by a frequency control unit (FCU) to maintain synchronization among the operating generators. A control cylinder senses governor demand and modulates the hydraulic log containing the reversible motor/pump. A scavenge pump collects oil from the transmission sump and returns the oil to the system through a oil filter. Sensing elements for external control and indication include output speed, underspeed, oil in and oil out temperature, and a low oil pressure switch. A manually activated disconnect solenoid retracts the CSD input shaft from the APU in the event of malfunction as sensed above.

Generator Control and Protection. (See Figure 3.1-17.) A static generator control unit (GCU) incorporates the regulator, protection, and switching functions required by the generator. The voltage regulator senses the three-phase voltages on the generator side of the line contactor and corrects the exciter field current to maintain 115/200 volts at the point of regulation. The necessary power required by the GCU is obtained from the PMG.

The voltage sensing leads also provide information for the protective functions: over/under voltage and over/under frequency. Two differential protection current transformers serve to protect the feeder zone between them against line-to-ground

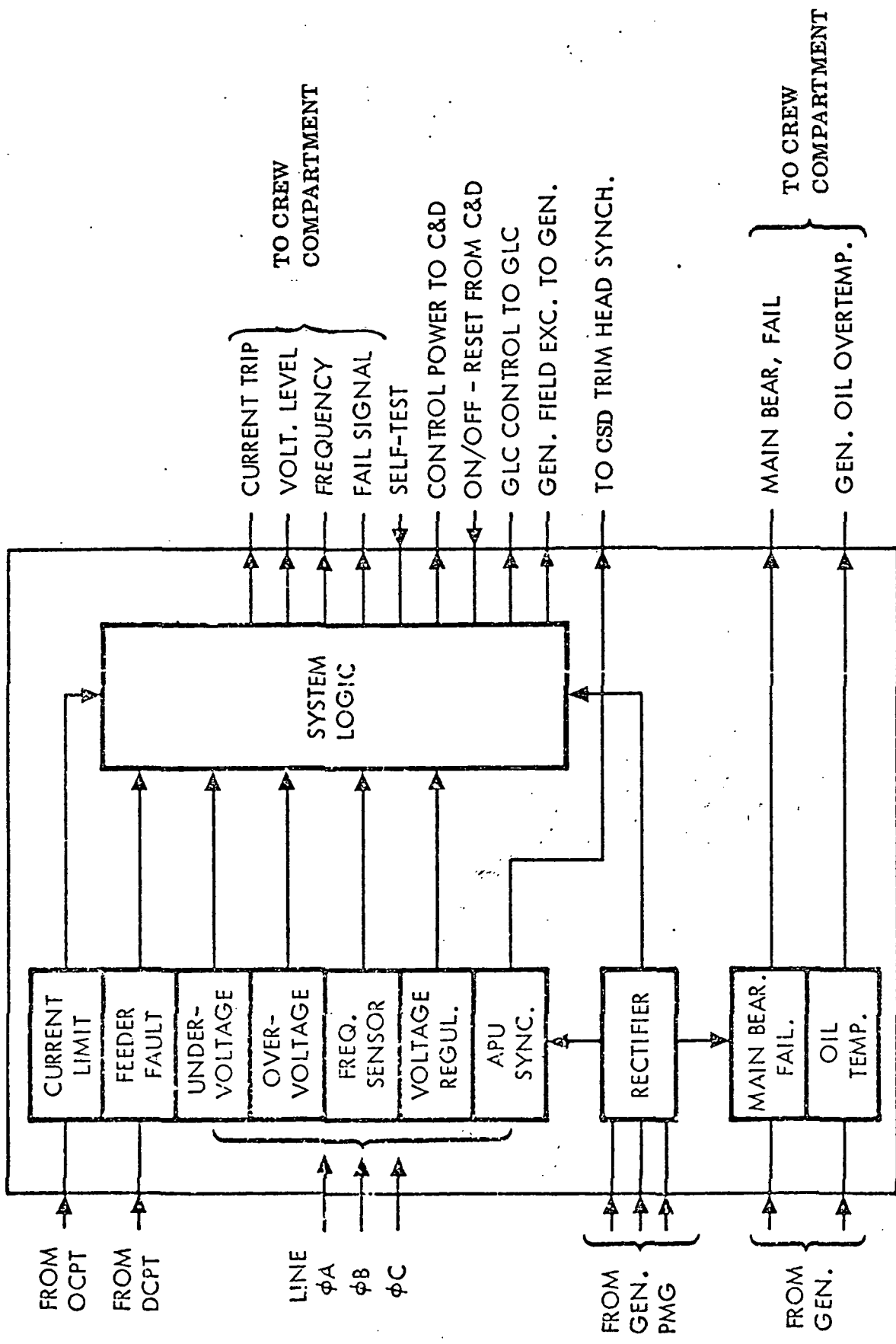


Figure 3.1-17. GCU — Simplified Block Diagram

and line-to-line faults. Over-current limit control is provided by a third current transformer to limit the maximum amount of current delivered by the generator. When the established limits are exceeded for any of the protective functions, the GCU operates to remove exciter field voltage and open the generator line contactor (GLC), thus isolating the generator. The GCU may be manually reset to re-energize the system (hard-wire) at the discretion of the crew. If the fault remains, the GCU trips and remains tripped until the reset signal is removed and re-applied. An anti-cycling circuit is provided for this function.

The GCU has provisions for self-test during ground checkout. A press-to-test button causes the logic circuits to carry out the necessary go/no go decisions, causing indicators to actuate. LRU levels indicated on the GCU are: CSD, AC generator, GCU, FCU and distribution system.

Conversion. Direct current is provided by transformer-rectifiers (TR). Each unit is rated 28 volts, 200 amperes, and can deliver 300 amperes for five minutes and 400 amperes for five seconds. The latter rating is achieved at the expense of lower output voltage and somewhat higher ripple content.

No active regulator is used to control the output voltage. The range of output voltage is proportional to the variation in ac input voltage and further by the inherent regulation due to the transformer and rectifier impedance drop variation with load current.

Six TR units are provided. Three are located aft near the boost engines to supply control, ignition, and instrumentation power during the boost phase. Three additional units are located forward in the compartment aft of the cockpit to supply the electronic and other loads in this area.

Batteries. Two nickel-cadmium batteries, each rated at 10 ampere hours, are provided: one is in the forward compartment and one is in the APU compartment. These are conventional excess-electrolyte batteries, fitted with nonventing cell caps. Special ventilating provisions are not required. The batteries are not charged on the vehicle.

Remotely Controlled Circuit Breakers (RCCB). RCCB shown in Figure 3.1-18 are used for switching and thermal protection of the wiring. These are thermal breakers with inverse current - time characteristics chosen to match the wire protection required. An electro-magnetic mechanism operates the mechanical contacts, operating from 28 volts switch control from the crew compartment. To indicate trip status, a separate thermal circuit breaker and indicator lamp is provided. A grounding switch in the RCCB operates if a thermal overload is experienced. Discretionary action by the crew may be taken to reset the breaker by opening the switch to the off/reset position, and reclosing to on. The breaker will trip free if the overload persists.

Distribution Boxes. Three main ac distribution panels are located in the proximity of the three APU-driven IDG's. Each has hinged panels to provide full access to the enclosure. The components of each panel are: generator control unit, line contactor, bus tie contactor, three over-current sensing transformers, differential protection current transformer (three phase) and forward bus feeder fuses.* These panels also provide power distribution buses consisting of the required three-phase and single-phase remote RCCB's, power controllers and the associated ACT units.

Remote dc distribution panels are provided in the crew compartment racks, and in the aft fuselage thrust section structure. These panels serve the local area and provide control power for the RCCB's.

Three dc conversion and distribution centers are located in the crew module electronics racks and three in the APU compartment. Each enclosure provides a transformer-rectifier, bus tie contactor, line contactor, a secondary distribution bus feeder fuse and RCCB's.

Operational Characteristics

Functional Description. (See Figure 3.1-15.) The ac Elements. The primary electrical power source consists of three IDG units each regulated, controlled, and

*The term "fuse" is used as a generic term to denote fusible link protective devices.

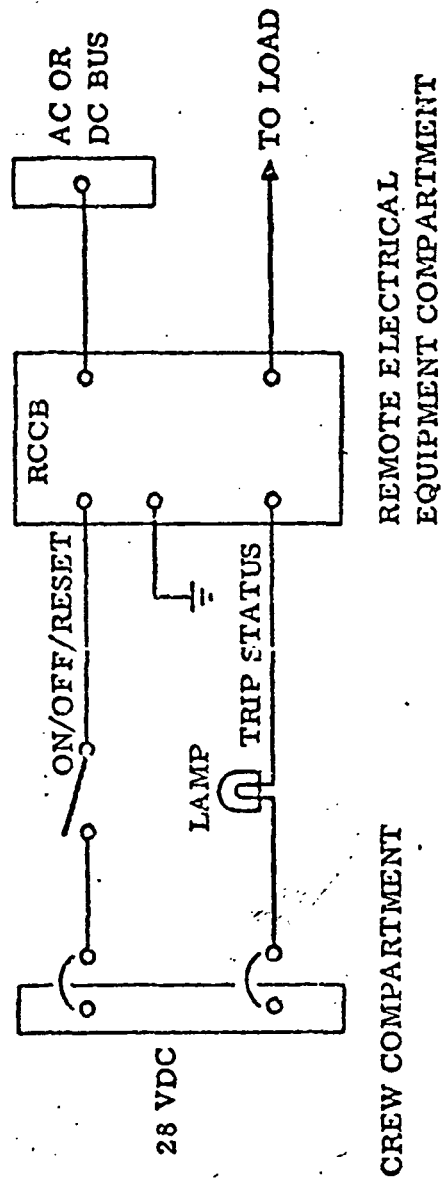


Figure 3.1-18. Remotely Controlled Circuit Breaker

fault protected by an associated generator control unit (GCU) and frequency control unit (FCU). The IDG's are driven by individual, N_2H_4 fueled APU's that are speed controlled as a function of generator output frequency. The output frequencies of the three machines are compared and an error signal is generated to regulate relative CSD speeds until synchronization of the generators is accomplished and maintained. Each generator channel is connected to a main three-phase ac bus through a normally open, GCU controlled, generator line contactor (GLC). The GLC's are located in close proximity to the main ac buses, and the length of feeder between the GLC and the generator terminals is protected from ground faults by differential protection current transformers (DPCT's). DPCT's for the ground side of the generator are an integral part of the machine. Over-current protection transformers (OCPT's) are also provided on the feeders to sense output currents in excess of TBD-percent rated output to protect the system in the event of a main ac bus line-to-ground or line-to-line fault. The fault protection features of the system are summarized as follows:

1. Over/under voltage.
2. Over/under frequency.
3. Feeder fault.
4. Overcurrent.
5. Anti-cycling.

The three main, three-phase ac buses are normally isolated from each other, each being energized by an associated ac generator. These buses may be interconnected only through a common transfer bus and then only under one of two conditions. The first condition is that a failed, faulted, or de-energized generator is automatically removed from the bus when its associated GLC opens; two ac bus-tie contactors (BTC's) are then closed to restore power to the de-energized bus, and a second generator assumes the load of the inoperative machine. The other condition is that with the vehicle generators inoperative, GSE (external) power is supplied to the main three-phase ac buses from the transfer bus through the ac BTC's. Over-current faults are automatically isolated and the bus cannot be restored by a second generator to prevent possible

cascading of fault conditions. The faulted generator may be reset and the bus restored by discretionary crew action.

The ac distribution system consists of distribution feeders and local ac distribution buses located in the forward and aft vehicle equipment compartments. These forward feeders are connected to the main ac buses through current limiters sized to clear/isolate distribution feeder faults.

As a part of the control logic required for bus system management, a discrete monitor is connected to the transfer bus. This unit senses the presence of voltage at the transfer bus and provides functional interlocks to prevent inadvertent paralleling of ac generators with each other or with GSE power. These controls cannot be overridden by the crew.

The dc Elements. The dc power required by the booster utilization subsystems is supplied from two sets of redundant transformer-rectifiers (TR) located near the forward and aft load centers. Each TR is energized from a separate local, three-phase ac bus through current limiters sized to clear/isolate TR internal and feeder faults. The output of each TR is connected to a separate main dc bus through a RCCB. The main dc buses are normally isolated from each other but may be interconnected through normally open dc BTC's which form a ring-bus configuration. The dc BTC's are closed under one of the following three conditions. First, a faulted, failed or de-energized TR is manually removed from the dc bus when the warning indicator operates. A dc BTC then can be closed to restore power to the de-energized dc bus, and a second TR assumes the load of the de-energized unit. Second, prior to discretionary switching of ac or dc buses, a dc BTC may be closed, paralleling two TR's until the switching sequence has been completed. This is done to prevent momentary power interruptions (MPI's) on the main dc buses during bus management operations. Third, loss of a local ac distribution bus by one or more blown current limiters results in the associated TR being de-energized. Its dc bus is then connected to an adjacent energized dc bus through a dc BTC. An MPI occurs during this switching operation.

The forward and APU compartment dc bus systems provide an additional dc bus identified as the BAT DC Bus. This bus is connected to one of the main dc buses through a power rectifier. The normally open contacts of the battery line contactor provide a means of connecting this DC bus to the battery. Neither battery is charged during mission operations and both are normally not loaded. Remote, automatic system startup uses both batteries for short periods during ground operations.

Remote Controlled Circuit Breakers Power Controllers. Generally, all electrical loads are supplied with power through a RCCB of the type described in the Equipment Descriptions section. This unit provides manually controlled switching. Overload protection functions are integral.

Displays and Controls. (See Figure 3.1-19.) A brief summary of the concepts as applied to the PGDC subsystem is included here to complete the subsystem configuration description. The complete dependence of the booster vehicle on the displays and controls (D&C) subsystems for command, control, and displays and in turn their dependence on the PGDC subsystem for electric power, establish a requirement for PGDC operating and control modes that are independent of the D&C during startup. GSE ground power and onboard batteries are the supply sources during this operating mode, and control functions are provided by manually operated switches.

An "on-off/reset" switch is provided for each ac generator/GCU. The manual "off" position is normally used only for ground maintenance and as a "reset" position after a "trip" condition has been experienced. The switch is normally "on" to delegate control of the generator and its associated GLC to the GCU. An "on-off" switch is provided for each ac BTC to provide for manual control during ground and flight operations including startup.

A manual-only external power switch is provided to control the application of GSE power to the vehicle transfer bus. Interlocking functions prevent paralleling GSE power with the vehicle ac generators. The "aft battery" switch provides manual control of the battery line contactor for ground operations and startup. Each TR is provided with an "on-off" switch to provide manual control of the TR's.

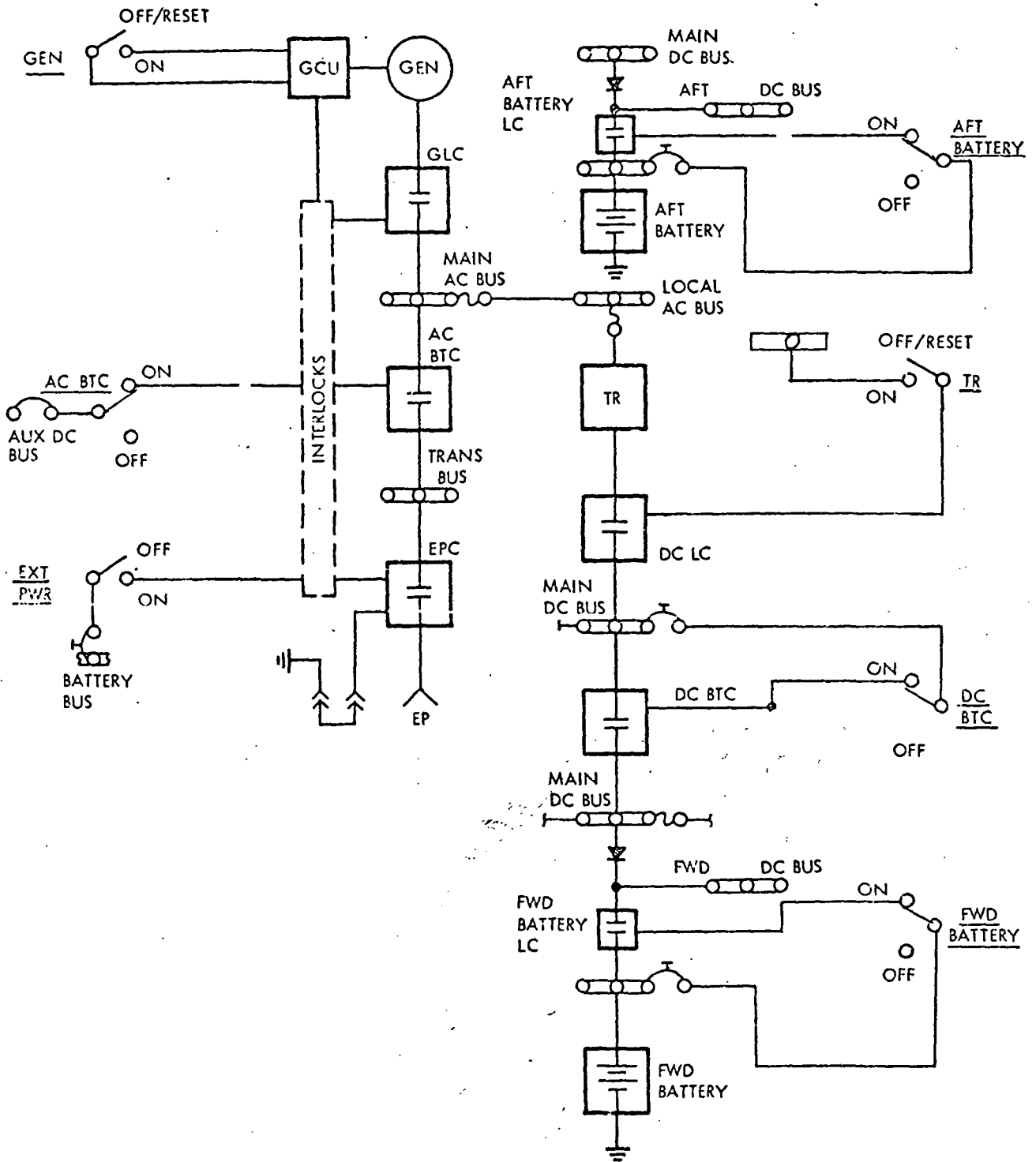


Figure 3.1-19. Booster PGDC Control Concept

A dc BTC switch permits individual, manual control of the dc BTC's during ground maintenance. This switch is an "on-off" type, normally in the "off" position during airborne operations.

The "fwd batt" switch provides control of the forward battery line contactor. Like the aft battery, this function is normally manually controlled during flight operations.

The following "caution" indications are provided to the crew:

- Generator Inop.
- Generator Bearing Failure
- Generator Oil Overtemp.
- Generator Overcurrent Trip
- TR Inop.
- Transfer Bus On
- Fwd Batt On
- Aft Batt On

Physical Characteristics. The estimated gross weight of the subsystem is:

Item	(Weight (pounds))
Electric power Distribution and control equipment	1123
Wiring and installations	1020
Total	<u>2143</u>

Note: This weight includes 244 lbs. of generation equipment.

Performance. Electric power having the following characteristic shall be provided at the load interface.

Power (dc)	Main Bus
Nominal voltage	28 v
Steady-state limits	24 to 30 v
Transient limits	(See Figure 3.1-20)
Ripple voltage	4 v peak-to-peak (MIL-STD-704A)

Power (ac)	Main Bus
Phases	3, 120-degree displacement
Nominal voltage	115 v L-N, 200 v L-L
Steady-state limits	109 to 117 v L-N
Wave shape	Sinusoidal
Nominal frequency	400 ±20 Hz
Transient limits	(See Figure 3.1-21)

Avionic Load Profile. An avionic load profile summary covering the various flight phases is presented in Figure 3.1-22. Also shown in Figure 3.1-23 is a comparison of avionics power for B9U versus B18E-3.

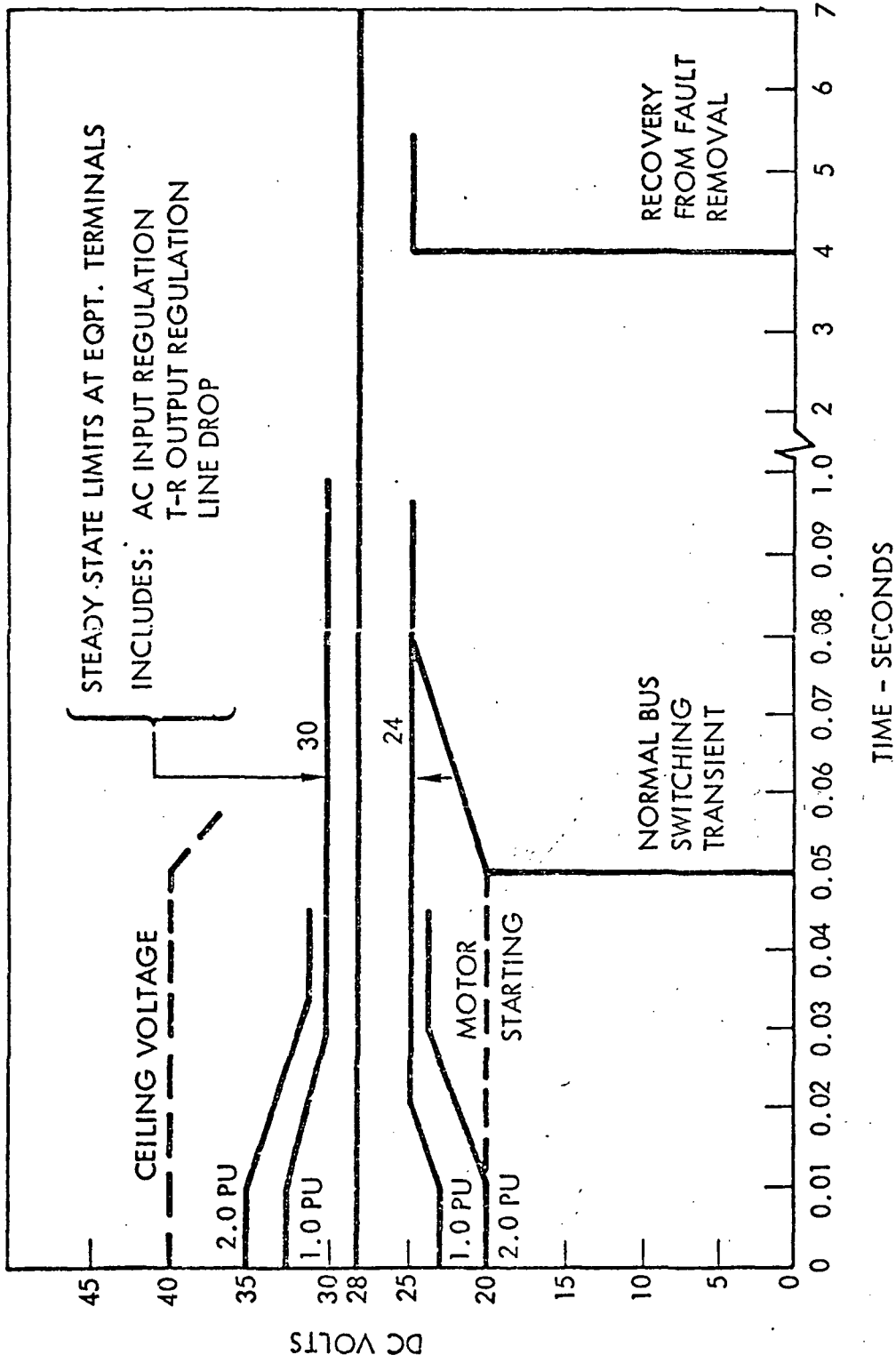


Figure 3.1-20. Transient and Steady-State Voltage Limits (dc)

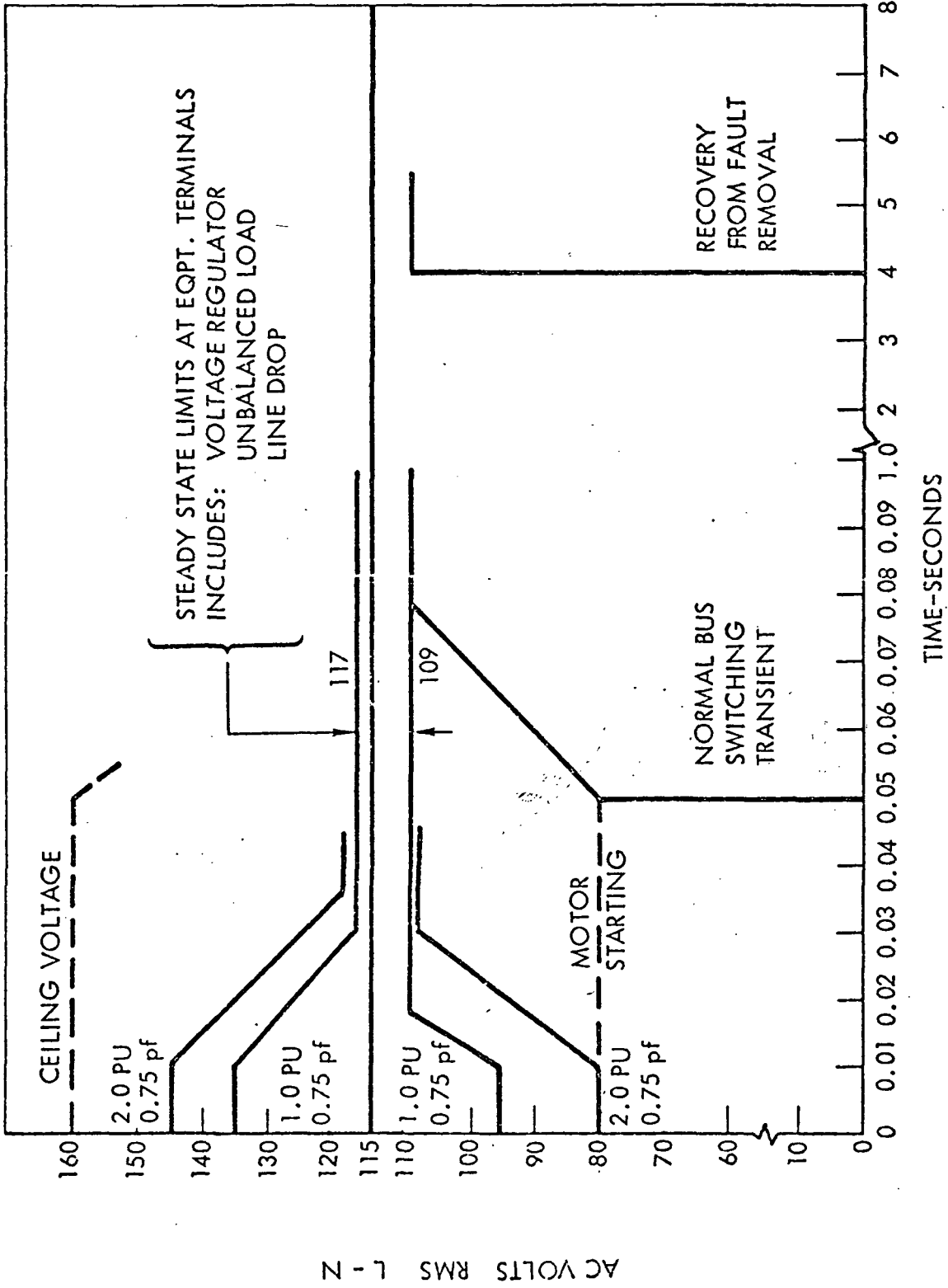
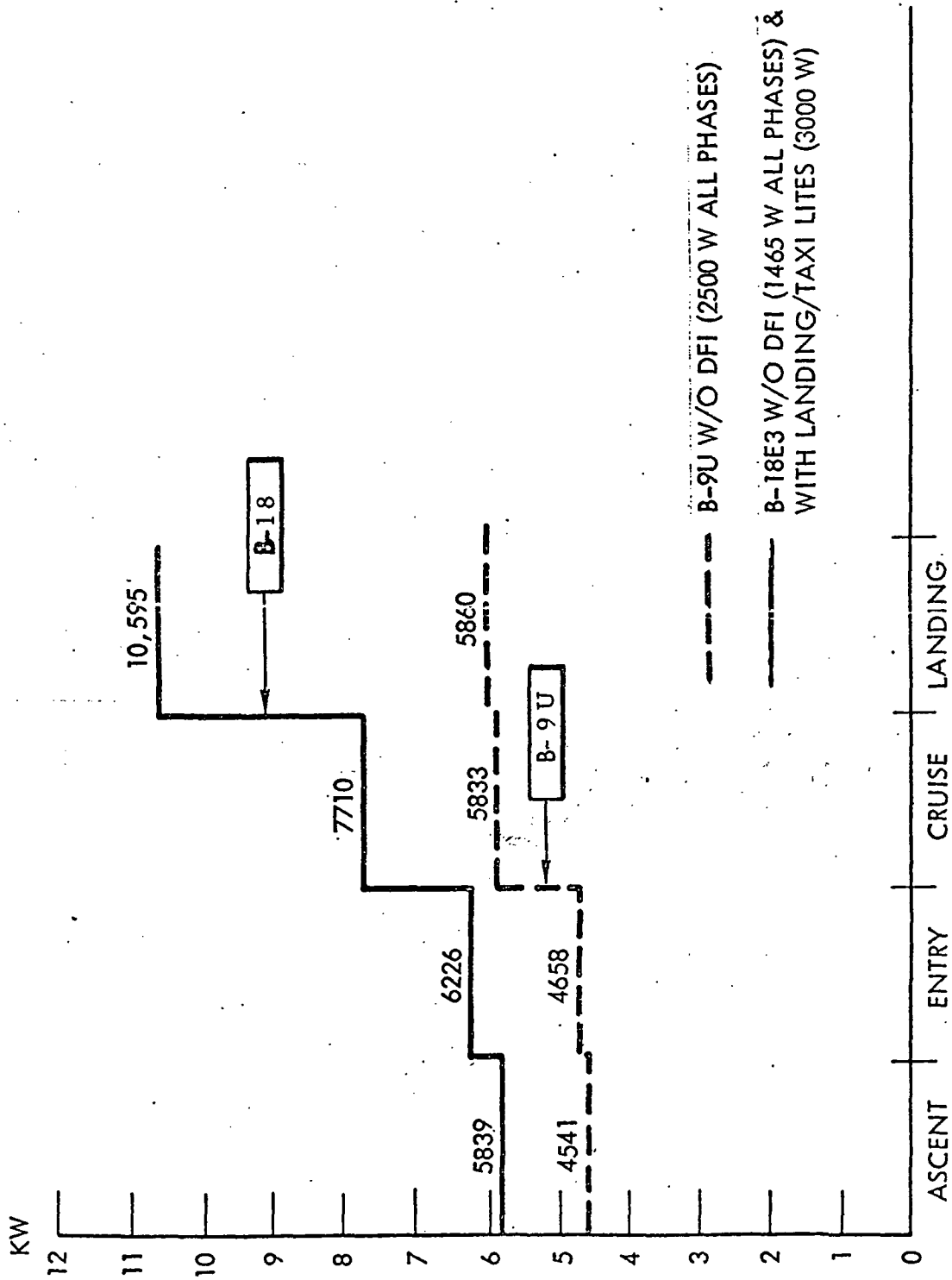


Figure 3.1-21. Transient and Steady-State Voltage Limits (ac)



- - - B-9U W/O DFI (2500 W ALL PHASES)
 — B-18E3 W/O DFI (1465 W ALL PHASES) & WITH LANDING/TAXI LITES (3000 W)

Figure 3.1-22. B-18E3 Booster Avionic Power Profile

SUBSYSTEM	B-9U B PHASE BASELINE	B-18E B' AVIONICS STUDY
DCM	1,824	
PD&C	1,105	575
GN&C	685	2,678
COMM	676	602
D&C	1,570	5,980*
OPERATIONAL INSTRUMENTATION		760
TOTAL OPERATIONAL	5,860	10,595
DEVELOPMENT FLIGHT INSTRUMENTATION	2,500	1,465
TOTAL	8,360	12,060

* INCLUDES 3000 W LANDING LIGHTS

Figure 3.1-23. Booster Avionics Power

3.1.8 BOOSTER AVIONICS INSTALLATION

The avionics equipment is installed in three locations on the vehicle (see Figure 3.1-24). The primary location is a compartment which is aft of and a portion of the pressurized crew station module. The area contains most of the avionics and electrical distribution equipment with the modules mounted in either one of two racks. Redundant equipment is placed in opposing racks on opposite sides of the vehicle to provide space diversity in the event of a localized disaster. The racks are sectioned into three areas. Each section is separated from its neighbor by a vertical firewall and horizontal cold plates. A bulkhead is provided between the aft compartment and the flight deck so smoke or toxic contamination from the avionics cannot enter the flight deck.

The crew compartment is the second avionic location and contains primarily the displays and controls for control of the vehicle and operation of the avionics.

The third area is in the aft portion of the vehicle adjacent to the APU's near the wing root. There are two sealed compartments, one on each side of the vehicle. These compartments contain generator control units, frequency control units, electrical distribution panels, and the electronic drivers required to operate the TVC, aerodynamic surfaces and altitude control thrusters. This area is electrically connected to the forward avionic compartment by two wire ways running over the top of the vehicle on either side of the centerline.

The avionics are cooled primarily with air cooled cold plates with some air flow in the crew compartment and the forward avionics bay. The cold plates are configured to provide sufficient thermal capacity to prevent equipment overheat during boost and entry when no active cooling will be supplied. The crew compartment and the forward avionics bay have internal recirculating fans in each compartment to transfer the heat into the air and then to the structure by forced convection. This cooling mode is used during the boost and entry with air being recirculated independent of each compartment. A check valve in the dividing bulkhead is used to prevent air flow to the crew compartment. During prelaunch activities the refrigerated air is supplied by the ground support equipment. Flylock conditioned air is supplied from the engine bleed air refrigeration system.

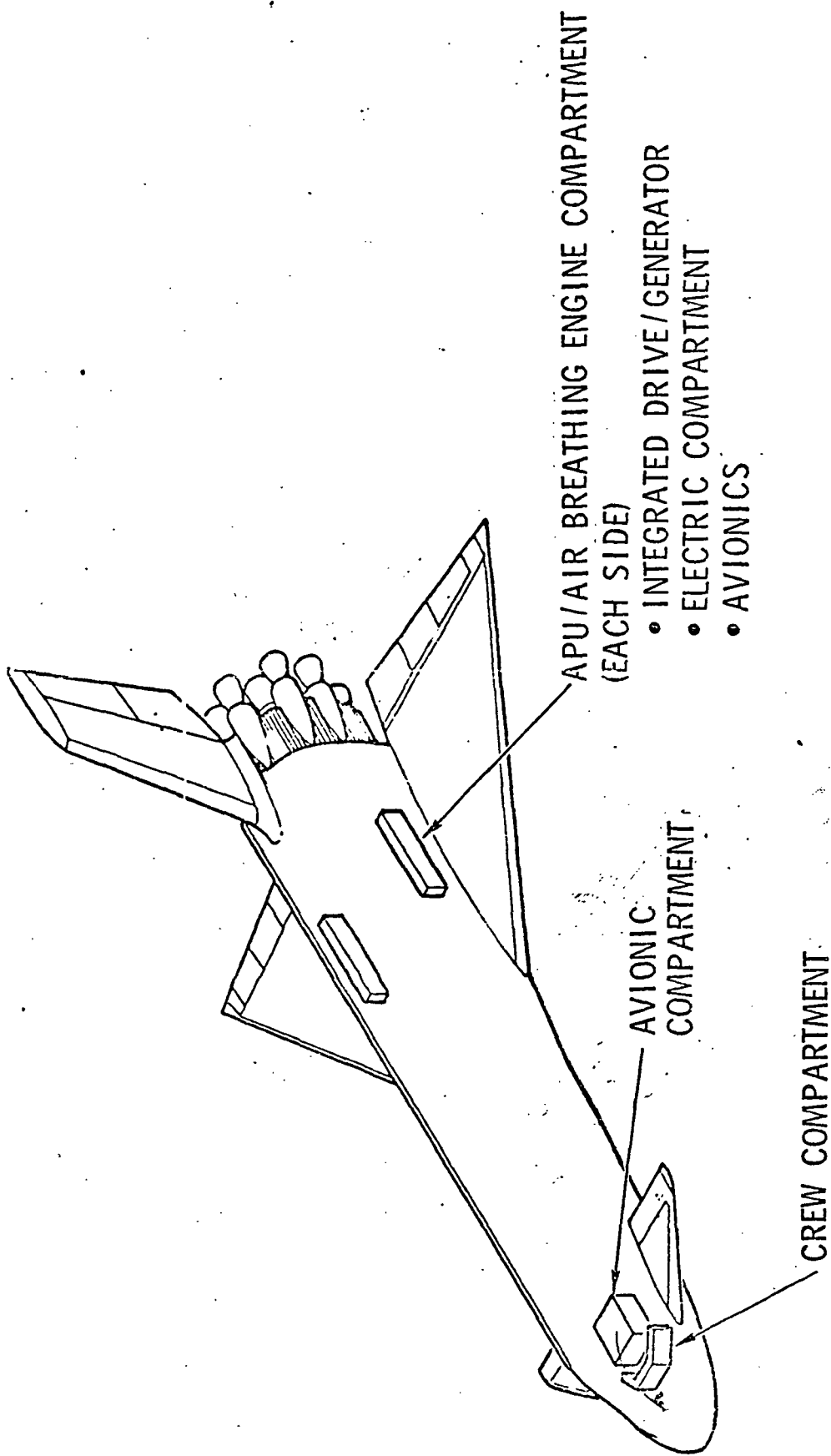


Figure 3.1-24 B-18 Booster Avionic Equipment Location

Two independent fire control systems are provided for the avionics racks. The extinguishing agent is Freon 1301. The agent is deployed by the pilot after the initiation of the fire warning light.

Access of the forward avionics, on the ground, is from the nose wheel well. Initial access is from a GSE stand then a ladder to the lower section of the sealed crew compartment.

A hatch is removed from the pressurized compartment for access. Equipment can be raised or lowered through the access latch through the wheel well to the ground. Access to the avionics when the booster is in a vertical position will be through the crew compartment.

A comparison of the avionics weights with the B9U baseline is provided by Figure 3.1-25. Generator weights have been deleted from previously published baseline data to provide a uniform basis of comparison.

SUBSYSTEM	B-9U B PHASE BASELINE	B-18E B' AVIONICS STUDY
DCM	3,301	1,899
PD&C (WIRE FOR POWER ONLY)	1,448*	1,244
GN&C	723	371
COMMUNICATIONS	553	1,185
DISPLAYS & CONTROLS	755	586
OPERATIONAL INSTRUMENTATION	250	4,137
OFI WIRING		
SUB TOTAL	7,030	9,422
DFI	5,332	1,147
DFI WIRING		3,515
TOTAL WEIGHT	12,362	14,084

* REMOVED POWER GENERATION EQUIPMENT

Figure 3.1-25 Booster Avionics Weights

3.1.9 IMPACT ON GROUND OPERATIONS

The booster configuration is a prime driver and will have significant impact on the ground operations. The LOX/RP and the pressure fed vehicles have major differences and which vehicle is ultimately selected is not a function of the avionics. The Alternate Avionics Study reflects some dollar differences that are driven by vehicle configuration rather than the avionics configuration. For example, the unmanned pressure fed booster will not require training for a flight crew and this was an approximately \$50 million item. Regardless of the avionics configuration the dollar required for training, which is a part of the ground operations, will be much less for the pressure fed vehicle than for the LOX/RP vehicle. In trying to access the impact of the avionics configuration on the ground operation be careful to be certain the impact is avionics driven and not driven by the vehicle configuration.

The configuration of the ground systems required to support Launch, Maintenance, (Level I & Level II), Safing Manufacturing, Development & Training are generally common with the orbiter. The detailed description of the checkout support equipment is contained in the orbiter section of the Alternate Avionics Study but outlined in general terms in the table below.

Operation	Checkout System Configuration
Launch, KSC	Modified ACE with 9 new control consoles, and using existing PLS equipment. Considerable new software. Launch readiness checkout
Maintenance M&R Level I	Modified ACE with 4 new control consoles, manual servicing and test equipment as required. Two phased booster M & R bays. All new software. Premate checkout and mated checkout.
Maintenance M&R Level II	Commercial automatic Test Equipment. Bench set up or STE. Phased with manufacturing. Considerable new software.

Operation	Checkout System Configuration (continued)
Safing	LCC modified ACE (shared). Minor new software.
Manufacturing	Modified ACE with 4 new control consoles, manual servicing and test equipment as required. Considerable new software.
Development	Special test equipment and existing facilities. Shared SIL with the orbiter. Minor software for data processing.
Training	Same concept as B phase report. Less equipment for B 18 and much less for B 19.

Checkout Support Equipment Impact

A summary of the checkout support equipment impact is shown in Table 3.1-14. The comparison of the checkout support equipment configuration between the B 9U, B phase baseline, and the B 18 & B 19 as well as the comparison with the orbiter checkout concept is presented. The dollar impact is categorized as nil, less than \$5 million; minor, \$5 to \$10 million; and major, over \$10 million. The major dollar impact in the training equipment is due to the unmanned configuration and isn't due to the impact of the avionics. With the exception of this training impact there is no significant dollar impact on the ground between the B 9U & the B 18 - B 19 configurations.

The capability of the avionics in the LOX/RP booster and the pressure fed booster to isolate faults is limited by the capability of the off-the-shelf avionics. Failures to 56% of the functional paths on the booster and orbiter can be isolated onboard with the type of equipment being considered. Failures to 24% of the functional paths can be isolated after the vehicles return to the M&R facility, through the analysis of the onboard recording and a visual inspection of the vehicle. This leaves failures in 20% of the functional paths to be determined by tests in the M&R bay through the manual or automatic ground support equipment. The B phase baseline avionics system was able to isolate 95% of the LRU failures onboard.

The numbers of LRU's involved in the LOX/RP booster is considerably more than the number involved in pressure fed booster but less than the B phase

TABLE 3.1-14

CHECKOUT SUPPORT EQUIPMENT IMPACT

	BOOSTER		BOOSTER/ ORBITER %CONCEPT	\$ IMPACT
	B 9U	B 18/B 19		
LAUNCH	NEW & EXISTING	ACE MODIFIED & EXISTING	SAME	NIL
M&R LEVEL I	NEW	ACE MODIFIED	SAME	NIL
M&R LEVEL II	NEW	COMMERCIAL ATE	SAME	MINOR -
SAFING	NEW	ACE MODIFIED	SAME	NIL
MANUFACTURING	NEW	ACE MODIFIED	SAME	MINOR +
DEVELOPMENT	NEW & EXISTING	STE & EXISTING	SAME	NIL
TRAINING	NEW OPERATIONAL & DEVELOPMENT	B 18 - LESS B 19 - MUCH LESS		B 18 MINOR- B 19 MAJOR -

avionics system. It is believed that the ground test time for the LOX/RP and the pressure fed booster will be about the same and in both vehicles is driven by the non-avionics subsystems such as ABES, structures, hydraulics, etc., rather than the configuration of the avionics. In comparison of the LOX/RP and the pressure fed boosters time lines with the B phase vehicle indicates more tests will be required and consequently more test time will be required for Level I maintenance with the LOX/RP and pressure fed booster than the B phase vehicle, but again this appears to be driven by the non-avionics subsystems.

An analysis of the bench maintenance checkout indicates a majority of the LRU types are testable on automatic test equipment that is commercially available. It appears that due to the fewer LRU's involved, the RDT & E hardware, and software will be less significant for the conventional avionics than the B phase baseline. Using commercial automatic test equipment has a low technical risk but some compromise of NASA QA/QC requirements may be required in order to use off-the-shelf equipment. A small percentage of the LRU types will require bench set-up or special test equipment but can be phased with manufacturing.

3.2 B-19 BOOSTER

INTRODUCTION

The overall block diagram, Figure 3.2-1 shows the B-19 booster avionics and its relationship to the orbiter avionics. It should be noted that the elements, which provided the-man-machine interface, have been replaced with subsystems providing booster to orbiter and booster to ground interfaces.

3.2.1 AVIONICS REQUIREMENTS AND LOCATION

As shown in Figure 3.2-2, the avionics are housed in a sealed compartment between the forward and aft thrust bulkhead. This compartment isolates the avionic equipment from the hostile environment. The compartment is suspended to minimize the vibration and acoustic noise. Temperature control is achieved by ground supplied cold air during launch operations. Excess thermal capacity allows for passive cooling during flight until shortly after separation when most of the avionics equipment is shut down. The compartment is mounted so as to absorb the shock of a sea water landing and to protect the equipment installed from immersion and sea water corrosion.

Access to the compartment for checkout and maintenance will be via hatches which allow for the removal and servicing of the avionics and system batteries. Communication with the orbiter will be through hardwires located in a trough extending to the orbiter tank interface adapter.

3.2.2 PRESSURE FED BOOSTER AVIONICS

Introduction - Where possible the B-19 avionics is common to the orbiter, but since it is an unmanned vehicle there are few subsystems other than instrumentation that have many common items. All non-common units are either off-the-shelf or are adaptations of existing designs and technologies. Tables 3.2-1 through 3.2-5 are breakdown of the equipment required for each subsystem.

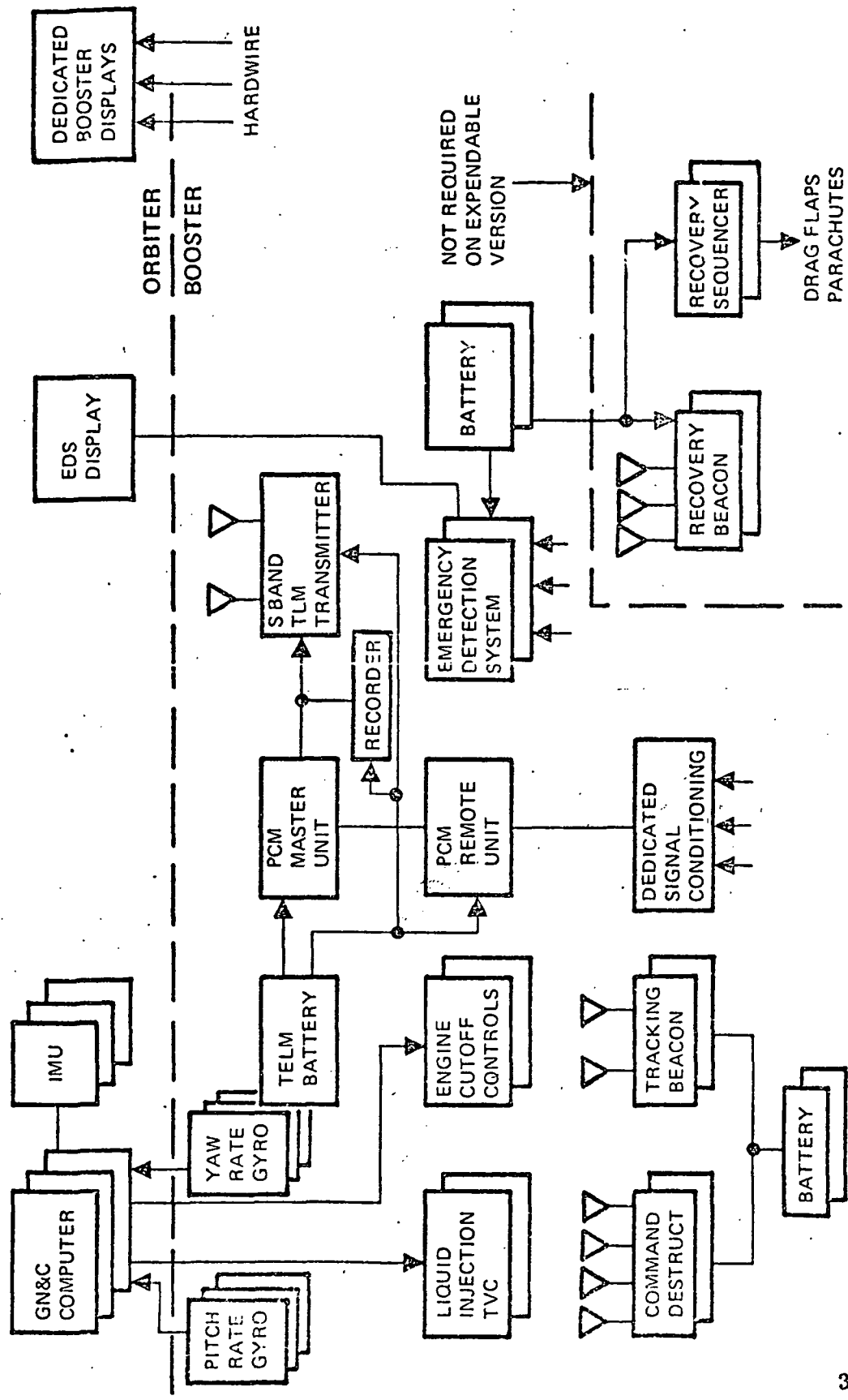


Figure 3.2-1 Pressure Fed Booster B-19 Avionics

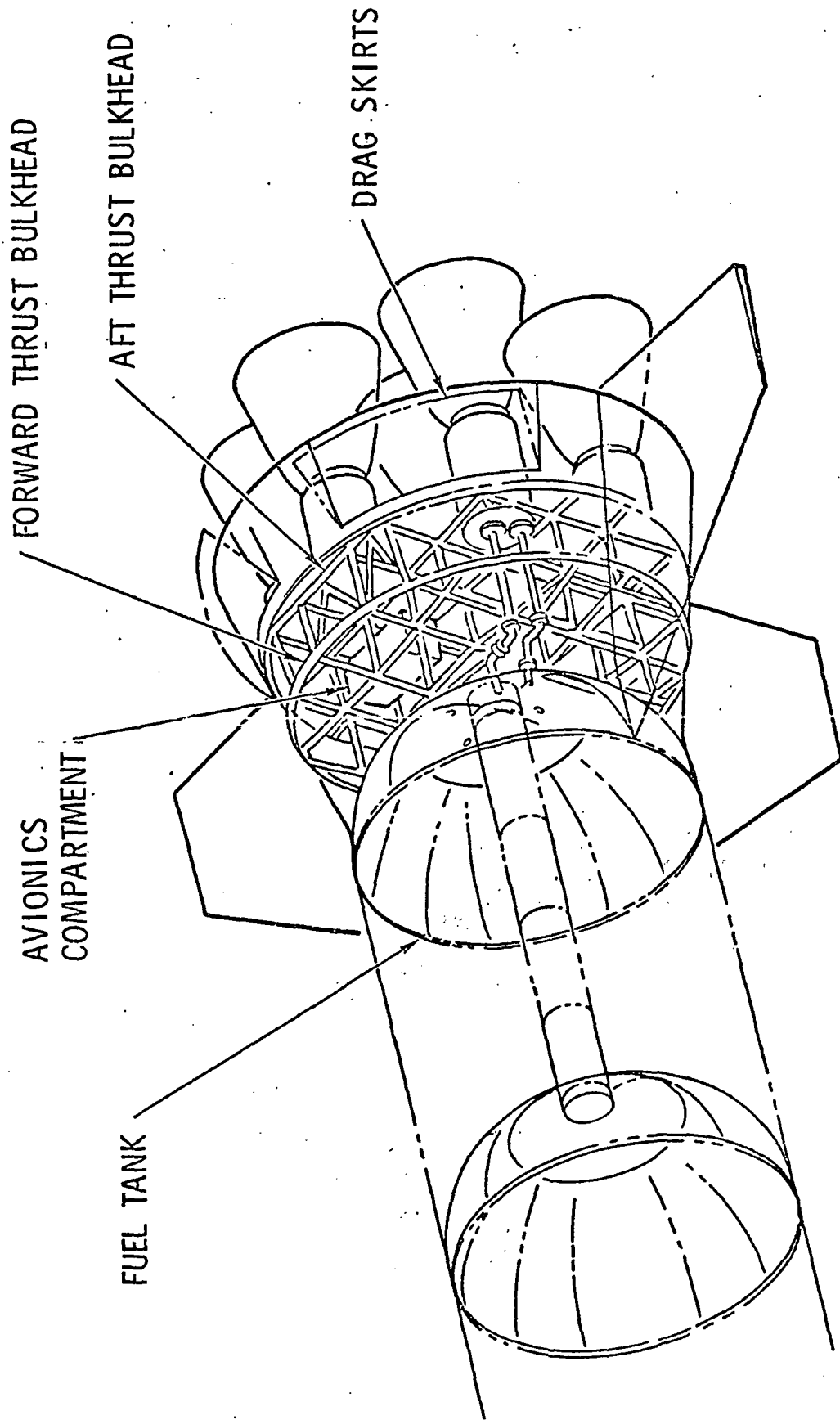


Figure 3.2-2 Avionics Bay Location - B-19 Pressure Fed Booster

Table 3.2-2 Data Management & OFI

PFB BOOSTER IAS EQUIPMENT

SUBSYSTEM DATA MANAGEMENT & OFI EQUIPMENT	EQUIPMENT CHARACTERISTICS & USAGE							ENVIRON.					FLIGHT PHASE, DURATION								
	QUANTITY	WT(LBS)	VOL(FT ³)	SIZE (IN.)		SHOCK MOUNT	Temp F		QTY Location			TOTAL NO.	NO. OPERATE.	LOAD TYPE (1)	TOTAL LOAD - W	COOLING (2)	ASCENT	SEPARATION	ENTRY	DESCENT	LANDING
	UNIT	UNIT	UNIT	Connector	H	W	T	Surface Mounting	Surface	Min. Operate Press. (PSI)	Low										
PCM REMOTE UNITS	2	30	40													100					
PCM MASTER UNIT	1	15	15													30					
SIGNAL CONDITIONING	2	15	30													70					
MAINT. RECORDER	1	50	50													60					
TIM TRANSMITTER	1	10	10													50					
O/B DATA LINK	2	13	26													40					
TRANSDUCERS	500	15	75													-					
TIMING UNIT	1	20	20													30					
EMERGENCY DETECTOR	2	20	40													30					
BATTERY	1	10	10													-					
RECOVERY SEQUENCER	2	20	40													30					
SUBTOTAL																					
SECONDARY STRUCTURE																					
SHOCK MOUNT & TRAY																					
BOLTS, SCREWS, ETC.																					
WIRING																					
TOTAL																					
			321																		

(1) AI: APU/C; A2: INVERTER AC; DI: DC; (2) CW: COLDPLATE; PS: PASSIVE, UNPRESS.; A: AIR, CONVECTION; AF: AIR FORCED; LD: LOAD DEPENDENT

Table 3.2-3 DFI

PFB BOOSTER IAS EQUIPMENT

SUBSYSTEM DEVELOPMENT FLIGHT INSTRUMENTATION EQUIPMENT	EQUIPMENT CHARACTERISTICS & USAGE		ENVIRON. Temp Min. Operate Press. (PSI)	QTY Location				SHOCK MOUNT	NO. OPERATE.	LOAD TYPE (1)	TOTAL LOAD - W	FLIGHT PHASE, DURATION						
	QUANTITY	WT(LBS)		VOL(EFT)	SIZE(IN)	H Connector	W SURFACE					L SURFACE	TOTAL	ASCENT	SEPARATION	ENTRY	DESCENT	LANDING
	UNT	UNT		UNT	UNT													
	UNIT	UNIT		UNIT	UNIT													
	UNIT	UNIT		UNIT	UNIT													
PCM REMOTE UNIT	3	20	60						DI	150								
PCM MASTER UNIT	1	15	15						T	30								
SIGNAL CONDITIONER	3	15	45						T	105								
WIDE BAND PROCESSOR	1	15	15						T	50								
WIDE BAND RECORDER	1	50	50						T	60								
PCM RECORDER	1	50	50						T	60								
TRANSDUCERS	1000	15	150						-	-								
BATTERY	1	10	10						-	-								
SUBTOTAL			395															
SECONDARY STRUCTURE			80															
SHOCK MOUNT & TRAY			80															
BOLTS, SCREWS			37															
WIRING			80															
TOTAL			672							355								

(1) A1: APUA C; A2: INVERTER AC; DI: DC; (2) CW: COLDPLATE; PS: PASSIVE, UNPRESS; A: AIR, CONVECTION; AF: AIR FORCED; LD: LOAD DEPENDENT

APPD _____ RW _____ DATE 10/19/71

Guidance, Navigation and Control - The B-19 booster relies completely upon guidance and flight control data transferred from the orbiter vehicle. Figure 3.2-3 is the GN&C block diagram. Booster flight control commands, derived from navigation data and stored guidance schedules, are output from the orbiters triple redundant GN&C computer system. These commands, in the form of low power level on-off discrettes, are input to the liquid injection thrust vector control (LITVC) unit which in turn transfers high current (1 amp) commands to the engine mounted controls. A total of 84 separate commands (12 to each of the 7 engines) are transferred, however, not more than 21 (3 to each engine) will exist at any point in time.

Booster flight path is controlled to the point of separation and then the vehicle follows a non-powered, non-controlled ballistic trajectory. During the ascent phase, the orbiter computer also outputs booster engine low power cutoff signals which are transferred to the engine cutoff control unit. This unit, operationally equivalent to the LITVC, provides high current commands to cut off two engines at 50 seconds (approximately 19K feet altitude), two additional engines at 130 seconds (approximately 130 K feet altitude), and the three remaining engines at 150 seconds (approximately 162K feet altitude). Separation occurs approximately 2.5 seconds later and the booster then assumes a ballistic trajectory.

Stability augmentation signals are also generated in the orbiter computer and are superimposed on the guidance signals. Booster body mounted pitch and yaw rate gyros provide triple redundant body rate data to the computer for the generation of these signals. The gyros have no electrical interface with the booster.

Communication and Tracking - The communication and tracking (C&T) subsystem includes the equipment required to implement range safety commands, facilitate in-flight tracking and recovery of the vehicle. Figure 3.2-4 illustrates

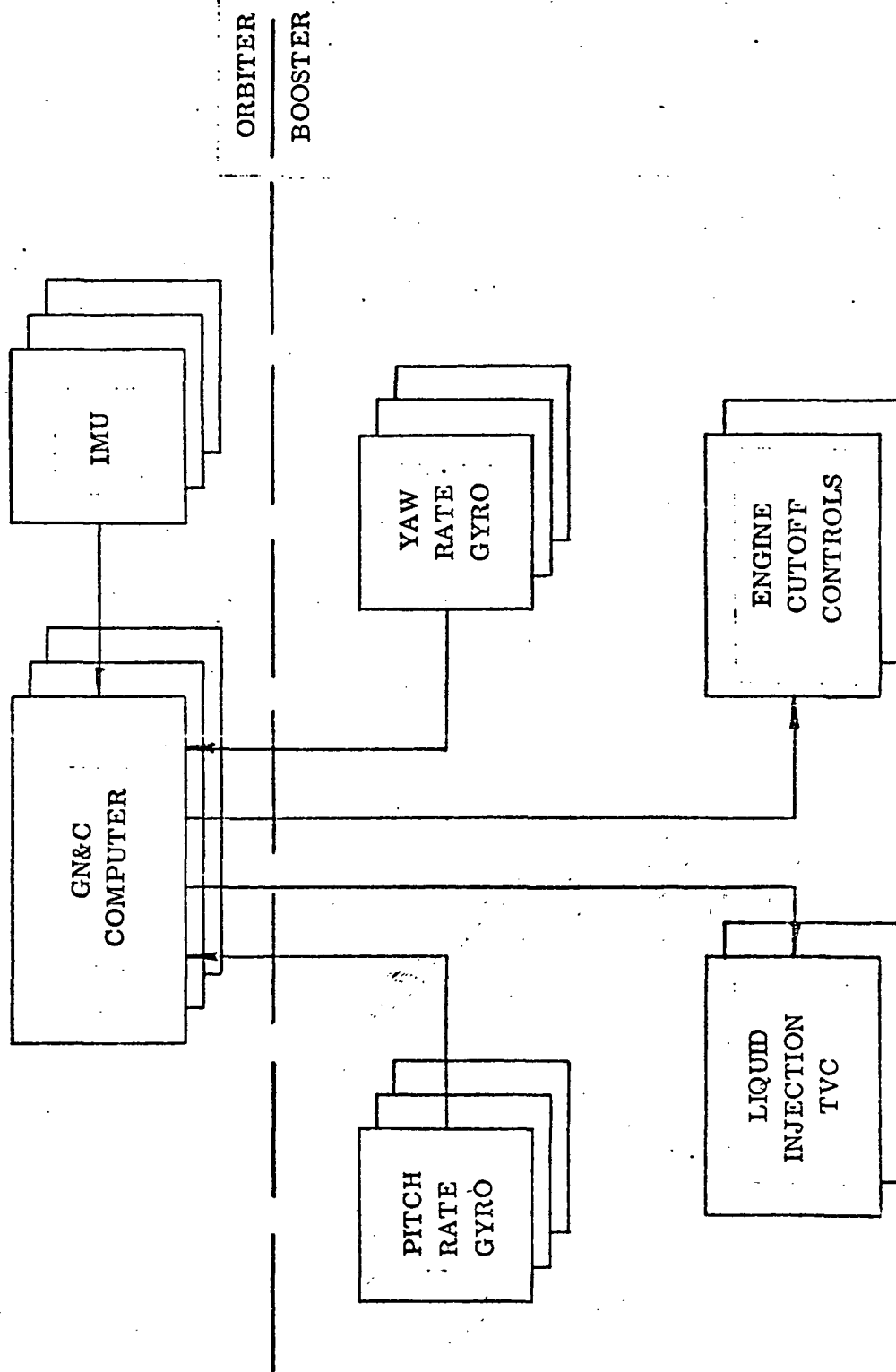


Figure 3.2-3 Pressure Fed Booster B-19 GN&C

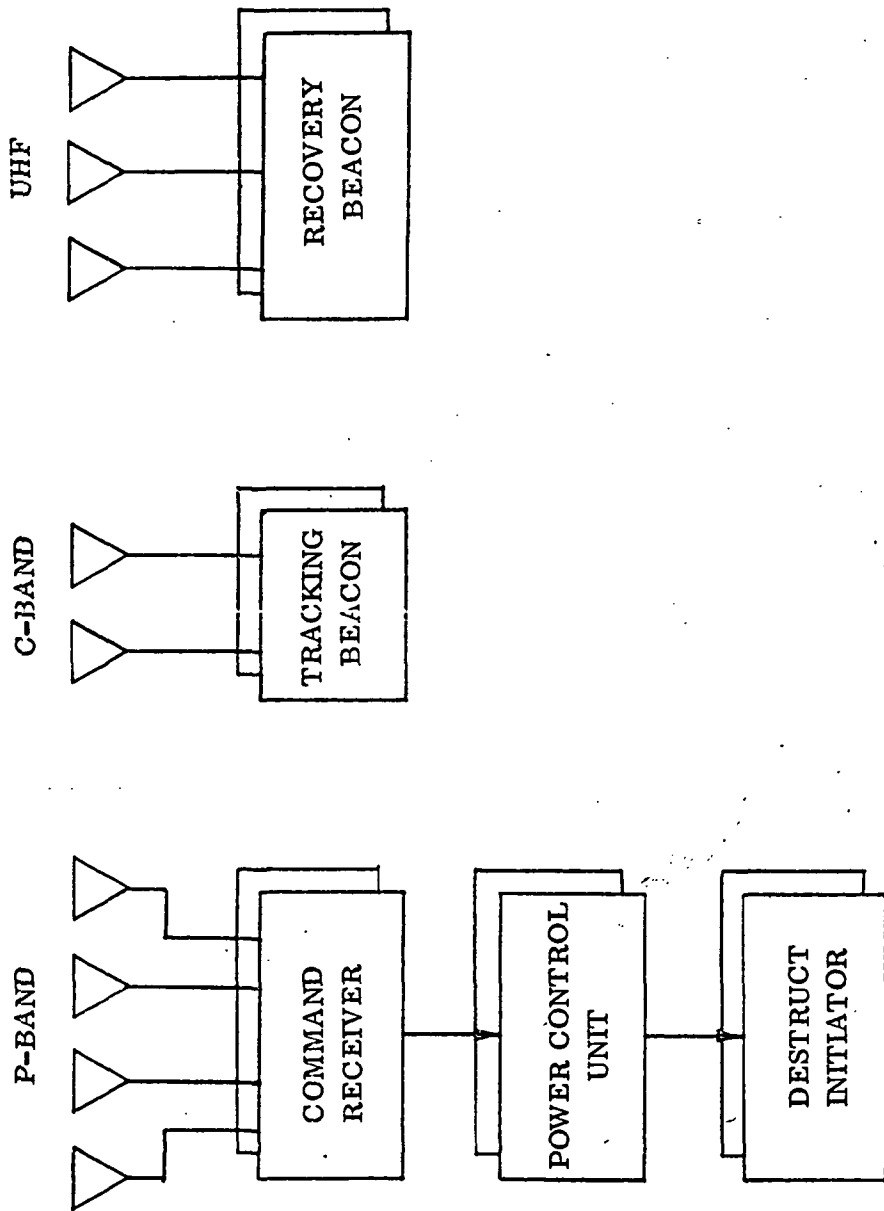


Figure 3.2-4 Communication and Tracking Subsystem Configuration

the C&T subsystem configuration.

Command Destruct System - The command destruct system is dually redundant, each consisting of a command receiver, power control unit, and destruction. The receiver, power control unit, and destruction. The receiver is a P-band single channel receiver for reception of tone sequential coded range safety commands, including safing, engine shutoff, destruct etc. The power control unit provides for switching vehicle electrical power in response to received commands. The destructors are on integral assembly of pyrotechnic initiators and safe-arm devices.

Tracking Beacon - The tracking beacon is a C-band transponder which responds to pulse space coded interrogations by one to several ground based radars. Redundancy is provided by a dual installation.

Recovery Beacon - The recovery beacon is a tone modulated radio transmitter operating on the UHF-AM guard frequency (243 MHz) at a duty ratio of 0.4. Redundancy is provided by a dual installation.

Antennas - Hybrid junctions are used to inter-connect the several antennas and equipments to achieve nearly spherical spatial coverage.

Instrumentation - The instrumentation subsystem of the pressure fed booster avionics is configured as shown in simplified form within the avionics system block diagram of Figure 3.2-1. The expendable and recoverable versions of the pressure fed booster would use similar instrumentation except that only the recoverable version has on-board data recorders.

The instrumentation subsystem performs the functions of conditioning data for orbiter display and of gathering data for subsequent analysis. Certain selected data are to be displayed and the balance is recorded and/or telemetered. Maintaining the same basic instrumentation concepts developed for the B-18 booster and

the orbiter, it uses the same building blocks where similar functions are performed. Separate batteries are provided for the instrumentation system to provide isolation from operational hardware which is being monitored.

The recovery sequencer will provide the necessary sensors, logic, and drivers to control the fins, drag flaps, and parachute release. The Emergency Detection System (EDS) gathers certain critical parameters and transfers them to the orbiter for display and action. The EDS adheres to the concepts and guidelines of the EDS of Saturn.

An orbiter/booster link provides an interface for signals (e.g., display signals) between the booster and orbiter. It contains low impedance driver circuits to protect outgoing circuits against shorts in the link, keep the source circuits from being loaded, and protect the orbiter side against possible shorts in the booster.

Electric Power Generation, Distribution & Control

Rationale - The identified loads and short mission time of the pressure fed booster make batteries the logical choice for a power source, particularly since an APU will not be provided. Several space qualified battery systems are available from past NASA programs that can provide the required performance without development costs. Since some quantity of pressure fed boosters will not be recovered, the re-useability aspects of the battery need not be considered, therefore, a primary type, silver zinc battery was selected from the existing inventory for availability and lowest cost. Qualification and operating experience records are available that may be examined to assure that the batteries will be applied to the booster within the performance and environmental limits. The modularity of the batteries may be readily used to provide additional power as required.

If re-useability of the batteries is desirable for later recoverable boosters, either secondary silver zinc or nickel cadmium batteries having a developed status can be substituted for the primary batteries. Although a greater weight penalty would occur, the economics of reuseability may offset the minor weight penalty.

Subsystem Design Requirements - The booster electrical power system consists of energy storage, distribution and control elements necessary to supply electric power of the type quantities and quality necessary to operate the utilization subsystems and their electrical, electronic and electro-mechanical components. The subsystem configuration selected has been formulated on the basis of simplicity, operational concepts subsystem functional requirements, safety considerations, maintenance philosophy, trade studies and risk assessment. Power source sizing has been established from an analysis of power demands occurring during the various vehicle operational modes.

Primary Functions. The primary functions of the electric power subsystem are:

1. Provision for and control of primary batteries.
2. Protection against battery and feeder faults from endangering the main buses and the redundant source.
3. Control of dc power to subsystems from the interface with the engine propellant computer and/or the orbiter control signals.
4. Distribution of power from the source terminals to the load terminals.
5. Generation of signals required by the Orbiter D&C subsystem to provide status, and caution/warning information.
6. Interconnection of subsystem LRU's
7. Interconnection of GSE power source and controls to the Orbiter subsystems.

General Description - The electric power sources for the vehicle and engine subsystems are provided using two electrically independent redundant battery groups. This separation of functions eliminates transient voltage influence of the engines from affecting the avionic loads. In each group, two paralleled batteries are provided to provide fail safe operation.

Vehicle Batteries (Refer to Figure 3.2-5). Manually activated primary silver zinc batteries are provided. These batteries are qualified for use on previous space programs. Either battery has sufficient capacity to supply the total vehicle load during ascent and entry. Under normal conditions, each battery supplies essentially half the system connected load. Each battery is connected to the main bus through a reverse current sensing contactor. Contactor control is achieved through switch control in the Orbiter. In the event of battery failure caused by cell shorting or a fault of the feeder between the battery and the bus, the reverse current coil senses the condition at a very low level of current and automatically removes the fault. The redundant battery and main bus are thereby isolated from the fault. Reset may be initiated from the orbiter, however the contactor will trip free if the fault remains.

During system initialization, either contactor may be closed first. The second contactor will not close until the voltage differential from battery to bus exceeds 0.5 volts. Normally this will occur immediately since the voltage of the first battery will decrease with application of bus load. Contactor position is sensed and displayed in the Orbiter crew station.

Engine Batteries. Engine batteries will be the same type as the vehicle batteries. These units normally provide a 30 day wet life and can be recharged 3-5 times with minimum degradation of reliability of the separation material. The

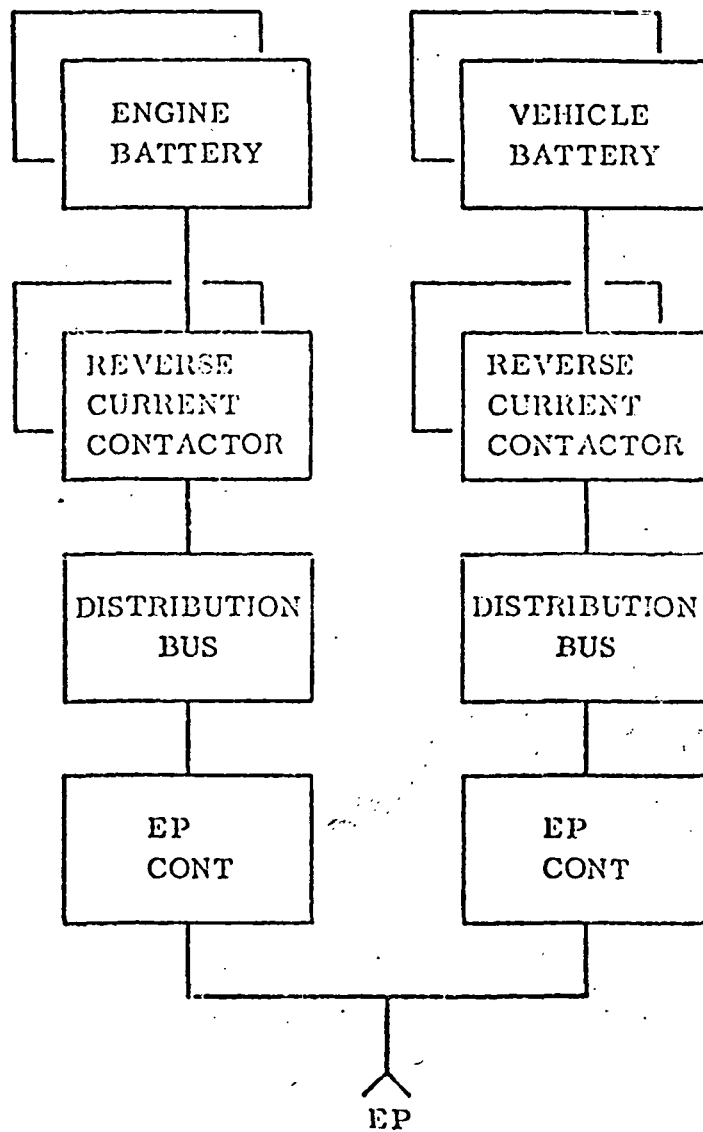


Figure 3.2-5 B-19 Electric Power

proposed battery is qualified for service on a previous space program. The connection arrangement between the battery and main load bus is the same as described for the vehicle battery subsystem.

Distribution Circuits. The distribution circuits for both the vehicle and engine loads are located primarily in the aft area of the vehicle. Circuit protection for wiring is not required for the majority of the loads. Since redundant loads are operated at all times, switching control is not required for load management or re-configuration. All utilization equipment identified operates from direct current, making converters unnecessary and thereby increasing the utilization efficiency of the battery energy as well as increasing system reliability.

External Power. An external power receptacle is provided for servicing and checkout. Separate external power contactors permit independent control of either subsystem from the Orbiter D&C subsystem. Auxiliary contacts provide confirmation of control commands.

Interface with Orbiter - The avionics interface between the orbiter and the pressure fed booster is accomplished through three separate and redundant hardwired umbilicals. In this configuration the orbiter controls the mated vehicle through the boost phase. Approximately 100 commands are transmitted from orbiter to booster. In return the booster sends engine thrust, body rate, propellant depletion, and dc power status signals.

3.2.3 POWER PROFILE, WEIGHT

Electric Power Profiles - Vehicle Battery Loads. (Refer to Figure 3.2-6)

Loads identified are: UHF recovery transmitter, emergency detector, O/B data link and recovery sequencer. Pyrotechnic loads are: separation bolts, drogue chutes and parachutes. Momentary valve loads for the control of drag flaps are required after separation and prior to water landing. Steady state voltage will not drop below 26 volts, however the pyrotechnic pulse loads may cause transient voltage dips to approximately 24 volts during entry.

Engine Battery Loads. (Refer to Figure 3.2-7). Loads identified for the seven engines and vehicle propellant system are: lox and fuel inlet valves, LITVC valves, as well as a propellant utilization computer and a small load for engine instrumentation. Steady state and transient voltage excursions due to valve loads will not drop below 26 volts.

Weights- Avionics weights are listed on Table 3.2-6. These weights include the weight of canisterising for salt water protection. Wire weight is included as part of the electrical power weights. Weights necessary for shock and vibration mountings are included as part of the vehicle basic structure and not listed on this figure.

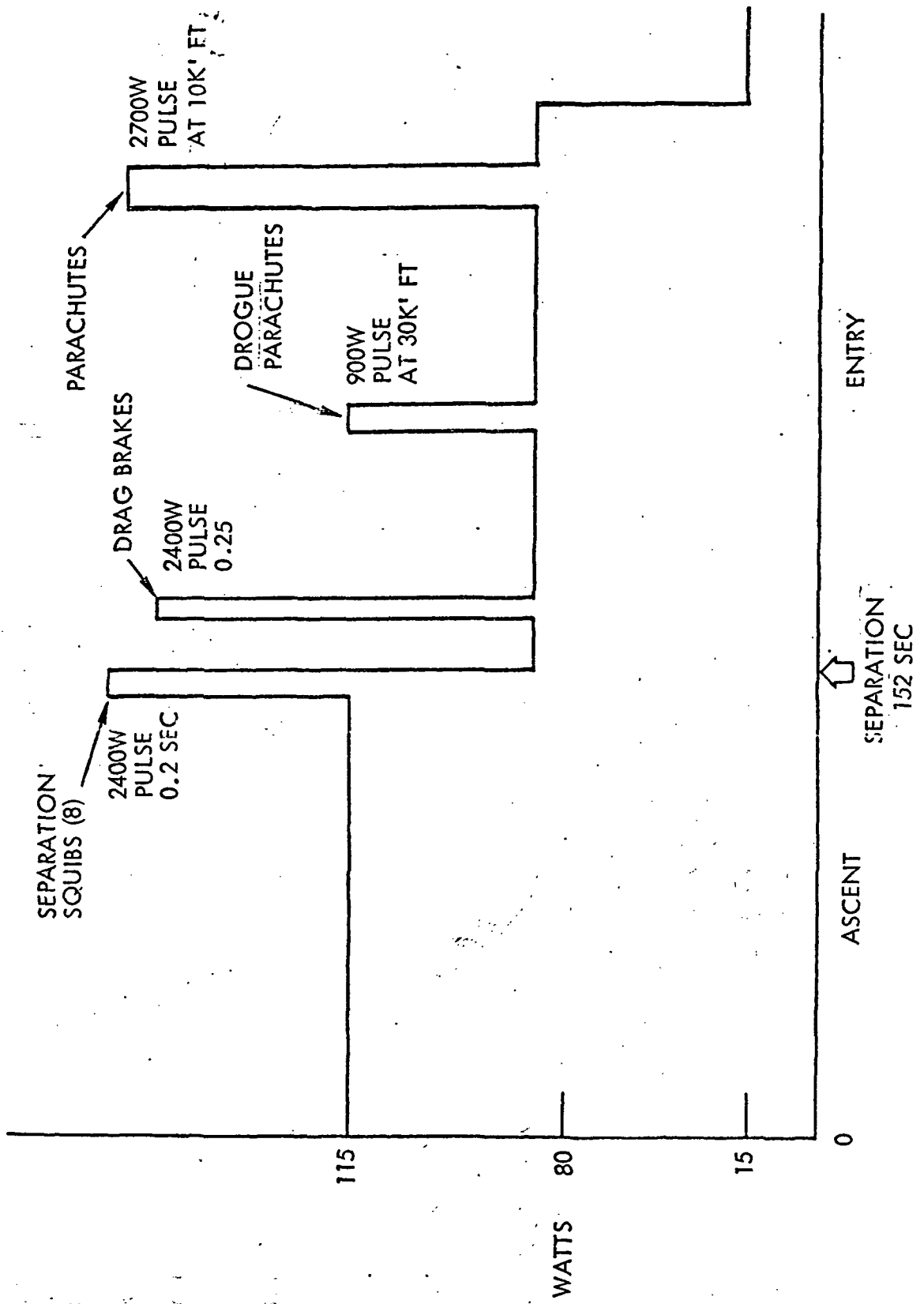


Figure 3.2-6 B-19 Booster Avionic Power Profile

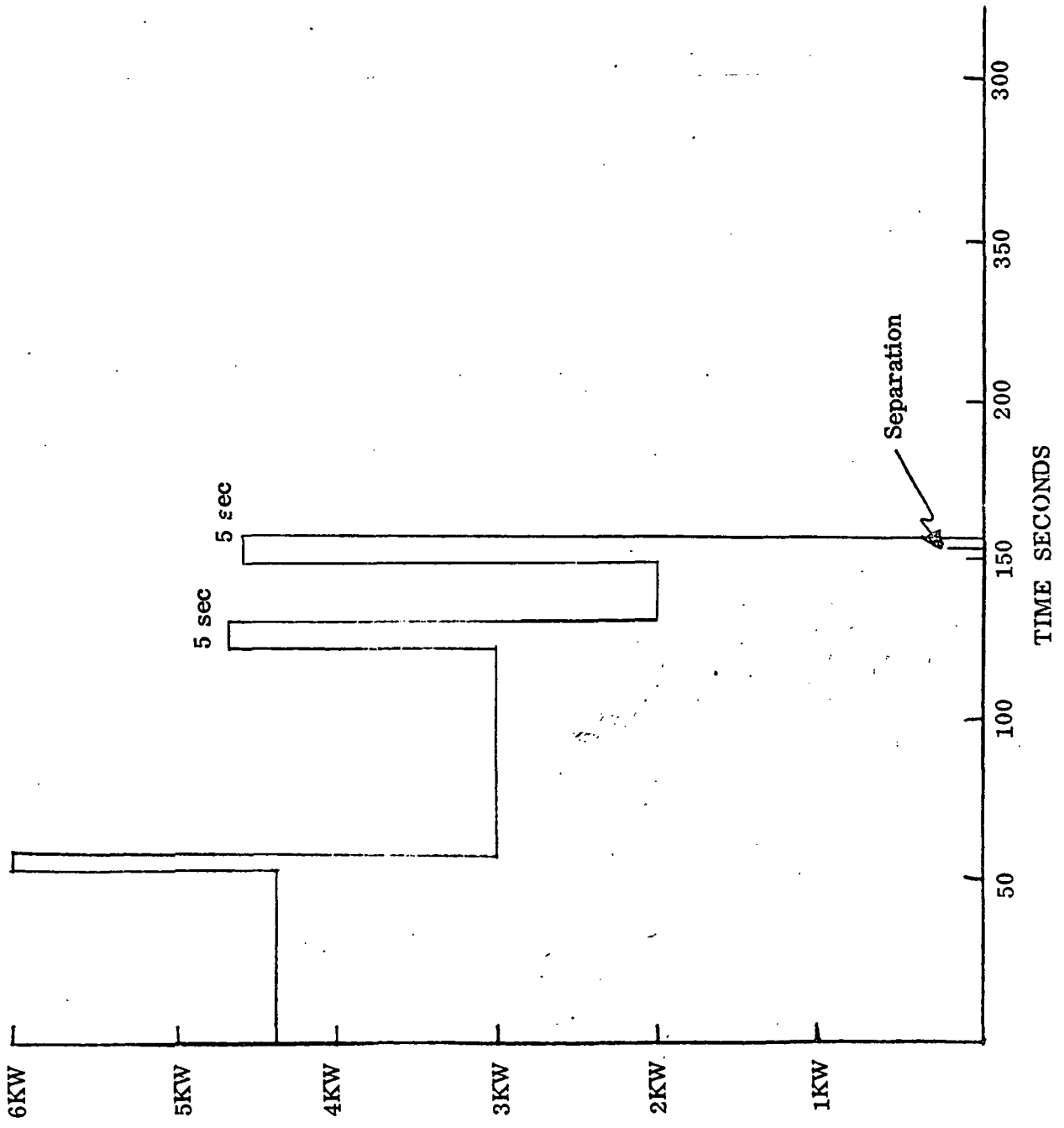


Figure 3.2-7 PFB Engine Power Loads Engine Battery

Table 3.2-6 B-19 Subsystem Weight

	<u>POUNDS</u>
COMMUNICATIONS	174
GUIDANCE, NAVIG & FC	31
OFI & DATA MGMT	821
ELECTRICAL POWER	589
SUBTOTAL	<u>1,615</u>
DFI	<u>672</u>
TOTAL	2,287

3.2.4 IMPACT ON GROUND OPERATIONS

The ground support equipment configuration to sustain launch, maintenance, manufacturing and development tests of the B-19, pressure fed booster, are similar to the equipment required to sustain the B-18, LOX/RP flyback booster. The write up on the ground support equipment impact on the B-19 (Section 3.1.9) dealt with the relationship between the ground configuration, vehicle configuration and avionics configuration and included some discussion on the B-19.

There are several noticeable differences between the configuration of the B-18 and B-19 ground support equipment. All of the differences are vehicle driven and generally not a function of the avionics configuration. The B-19 is an unmanned vehicle and this has a major impact on the training equipment and the training program. This is probably the largest dollar impact and has been estimated at a \$50 million dollar savings. In fact, an analysis of all the other ground equipment in the B-19 configuration ends up a push on dollar impact. There are some more expensive items, such as the ocean going tug used to tow the vehicle home. However, the expenses are offset by savings in other areas; for example, the saving realized in the air-breathing engine development and support program.



4.0 IMPACT ON GROUND OPERATIONS

The Space Shuttle checkout and ground operations requirements, and the previous concepts developed under the Phase B study, were re-evaluated as a part of the alternate avionic system study. The study was conducted in conjunction with the orbiter and booster vehicle avionic studies described in other report sections. The report sections describe the study approach and results in the following order:

- . Checkout and Launch Operations Requirements
- . Avionic Subsystem Requirements to Facilitate Checkout
- . Candidate Checkout and Launch Operation Systems
- . Selected Checkout and Operations Systems
- . Ground Software Analysis and Description
- . Booster Impact Summary
- . Cost - Schedule Impact Summary

4.1 CHECKOUT AND LAUNCH OPERATIONS REQUIREMENTS

The operational requirements that must be incorporated into the design of subsystems to facilitate efficient checkout of the Space Shuttle are presented here to establish design objectives. In general, a rationale is provided when these objectives cannot be met. Operational requirements are divided into two general categories, servicing and checkout, and considered as separate functions.

The terms used in this document are defined for clarity.

Checkout - Observation of performance of subsystems, assemblies, sub-assemblies or components, to determine operability; this function may take place during flight, or on the ground; ground checkout can take place in the quiescent state, or during application of artificial stimuli.

Servicing - Cleaning, replacement of consumables, replacement or replenishing fluids such as drinking water, coolant or hydraulic fluids; replacement or cleaning of filters or strainers; calibration of gages or transducers.

Measurement - Parametric information used to evaluate the operating performance or physical status of a system or subsystem, e.g., events or quantities displayed or telemetered.

Test Point - An access point for instrumentation to be connected, usually temporarily, to determine operating status of a system or subsystem and usually used during troubleshooting or isolating malfunctions.

The following ground rules have been utilized in defining the shuttle systems operations.

1. Full utilization will be made of flight data to minimize checkout requirements.



2. Onboard checkout capabilities will not be duplicated on the ground, but will be supplemented with ground equipment when more detailed checkout is required for fault isolation.

3. Subsystem checkout will be autonomous to the degree practical.

4. Checkout will be performed end-to-end and will consist of functional checks of longest practical functional loops. Primary and alternate loops will be selected by cockpit control.

5. Built-in test will be incorporated when economically feasible.

6. Checkout will be performed with the vehicle subsystems in functional flight configuration.

7. All vehicle measurements which are routed to the ground will be conditioned and digitized onboard. Test points will be protected by isolation circuitry as required for safety of equipment but not necessarily conditioned and digitized.

8. Multiple fixed electrical circuit redundancy implemented only to increase the MTBF of the circuit will be checked out only during bench maintenance when failures are suspected.

9. In cases where redundancy is implemented by parallel functional loops that are not switchable, test points must be provided to checkout each functional loop.

10. The booster and orbiter interface will be similar to the Saturn/Apollo.

11. The vehicles will become essentially autonomous after a number of R&D flights.

4.1.1 General Requirements and Ground Rules

The following General Requirements pertain to shuttle checkout and Launch Operations:

Checkout -

For the operational vehicle subsystem checkout will be "on-condition" in nature, meaning that only those functional paths suspect from flight data or violated by LRU replacement will undergo detailed checkout.

Initially, however, the lack of historical data will require a significant amount of subsystem verification. This verification will consist of exercising all primary and alternate subsystem functional paths while monitoring and recording parameters to verify correct response. Since this capability will be limited to those functions operational in flight and summary in nature, there will be a need to access the subsystem at certain intermediate points to introduce stimuli and/or monitor parameters. The function of the ground system during subsystem checkout will be to supply test discrettes, to stimulate unavailable subsystem operational interfaces, to provide a real-time display capability of selected measurements and to acquire subsystem data for future analysis and comparison. Since the checkout operation will involve a combination of onboard and ground actions, voice communications will be required to coordinate test operations.



Launch Operations -

During prelaunch and countdown operations the launch crew will be located in the LCC and will be occupied with the monitoring of operational vehicle measurements. These measurements will consist of only those parameters needed to launch a fully operational vehicle and are as follows:

1. Measurements needed for propellant loading.
2. Measurements needed to safe the vehicle with no crew onboard.
3. Subsystem go/no go status indications as presented by ground exception monitor routines.
4. Servicing equipment status.

In the early program phases the above measurements will not satisfy the necessary requirements to commit a vehicle to launch. In order to provide a capability to monitor more measurements and to call up individual parameters for fine grain investigation, the CIF will be utilized, manned by subsystem experts in support of the launch crew.

As the program matures, the role of the CIF support crew will be phased out along with the onboard DFI subsystem. The ultimate goal for the operational vehicle is to evolve to a mode requiring only the minimal support provided by the LCC system.

In addition to a monitor role the launch crew must have the capability to issue certain commands to the onboard subsystems. These commands are as follows:

1. Control of vehicle valves for propellant loading.
2. Final Nav. updates to the GN&C computer
3. Control of onboard electrical power distribution for safety purposes.
4. Control of onboard purges for safety purposes.
5. Provide a cut-off command to the engines for safety purposes.

Command capability must be available to control ground servicing equipment such as the following elements:

- . Vehicle ground power supplies
- . Ground cooling unit
- . Pneumatic servicing unit
- . Environmental control support
- . Service arms
- . Hold-down mechanism
- . Purge system
- . Hypergol loading servicing equipment

4.1.2 Factory Operations, Ground Rules and Requirements

The operations, ground rules, and candidate ground control system/equipment requirements considered in this study for the vehicle contractor factory checkout are identified below:

Operations

1. Subsystem Checkout
2. Combined Subsystem Test



3. Troubleshooting and repair
4. Horizontal Preflight Checklist
5. Static Checkout of ABES
6. Checkout of Inflight Monitoring Capability

Ground Rules

1. For acceptance purposes, end-to-end checkout of installed subsystems will be performed.
2. Subsystem performance data will be recorded for post-test analysis and historical purposes.
3. Stimuli will be introduced as required to completely exercise the onboard subsystem.
4. A telemetry ground station, capable of open or closed loop operation will be used for RF system checkout as well as for receiving and logging test data.

Candidate Ground Control System/Equipment Requirements

1. The ground system will provide recording and real-time monitoring capabilities for subsystem performance data.
2. The ground system will provide an operational and monitor interface for the test engineer.
3. The ground system will utilize the operational umbilical and test connector interfaces.
4. The ground system will have the capability to introduce onboard subsystem stimuli.
5. Voice communications will be provided to correlate cockpit/ground functions.

4.1.3 Horizontal Flight Test (HFT) Site Operations, Ground Rules and Requirements

The operations, ground rules and candidate system/equipment requirements considered in this study for the horizontal flight test site are identified below:

Operations

1. Subsystems Checkout
2. Troubleshooting and Repair
3. Load JP fuel, Pneumatics and other HFT consumables
4. Static Operation of ABES
5. Preflight Checklist
6. Inflight Monitoring of Data

Ground Rules

1. All vehicle functions required for checkout and servicing operations will be controlled from the cockpit.
2. Capability will be provided for data monitoring of flight vehicle subsystems and support equipment as required for hazardous operations.
3. All flight data will be monitored and/or recorded on the ground in real time via telemetry.



4. Capability will be provided for complete checkout and troubleshooting of all subsystems installed in the vehicle during HFT.
5. Support equipment utilized for HFT will be designed to allow maximum re-use at the alternate landing and ferry sites.
6. Checkout (subsystem and/or integrated) will verify all redundant paths applicable or contributing to safety of the HFT program.

Candidate Ground Control System/Equipment Requirements

1. Support equipment units will incorporate provisions for local control as practical and consistent with ground rules 1 and 2.
2. Correlation of vehicle and support equipment functions will be accomplished procedurally, using the Operational Intercom System (OIS).
3. A telemetry ground station, capable of open loop or closed loop operation, will be used for RF system checkout as well as for receiving flight data and support of other subsystem checkout.

4.1.4 Launch Site Checkout Bays Operations, Ground Rules and Requirements

The operations, ground rules and candidate system/equipment requirements considered in this study for launch site checkout bays are described below:

Operations

1. Subsystem Checkout
2. Combined Subsystem Test (Pre-Mate Checkout)
3. Troubleshooting and Repair
4. Modification and Reverification
5. Flight Program Loading
6. Maintenance

Ground Rules

1. For the fully operational shuttle, subsystem checkout will result from in-flight anomalies, mods or LRU replacement.
2. Subsystem checkout will consist of verification of LRU's and functional paths within the subsystem.
3. Pre-mate checkout will consist of end-to-end checks and will be functional in nature.
4. Performance data will be recorded for post-test analysis and for historical purposes.
5. Stimuli will be introduced as required to exercise the onboard subsystems.
6. A telemetry ground station will be used for RF system checkout as well as for receiving and logging test data.

Candidate Ground Control System/Equipment Requirements

1. The ground system will provide recording and real time monitoring capabilities for subsystem performance data.
2. The ground system will provide an operational and monitor interface for the test engineer.
3. The ground system will utilize the operational umbilical and test connector interfaces.



4. The ground system will have the capability to introduce onboard subsystem stimuli.

5. Communications will be provided to correlate cockpit/ground functions.

4.1.5 Launch Complex Operations, Ground Rules and Requirements

The operations, ground rules and candidate system/equipment requirements considered in this study for the launch complex are identified below:

Operations

1. Connect and verify ML/Pad interfaces.
2. Verify proper control of vehicle subsystems and support equipment from the Firing Room. (End-to-end control, compatibility, etc.)
3. Umbilical leak checks.
4. End-to-end checkout of ML/Pad facility type systems.
5. Purge cryogenic systems (Drop tank, MPS and PRDS)
6. Condition and load hypergolic propellant systems
7. Load other mission consumables
8. Load PRDS and drop tank cryogenics
9. Crew ingress
10. MPS propellant system and engine conditioning
11. Subsystem readiness verification
12. Lift-Off
13. ML/Pad securing

Ground Rules

1. Checkout will be limited to:
 - a. Verification of ML/Pad interfaces (may involve end-to-end functional checks)
 - b. Verification of ML/vehicle umbilical interfaces
 - c. End-to-end verifications insuring proper control and monitoring of ML, pad and vehicle functions. (Encompasses portions of a and b)
2. During normal launch pad operations periods all test operations will be coordinated from the LCC Firing Room. When vehicle control and monitoring is required, operators (launch crew or flight crew members) will be stationed in the cockpit to perform the required procedure subtasks at the command of the test conductor station in the firing room.
3. Cryogenic (and hypergolic) loading will be controlled from the LCC Firing Room. All personnel will be cleared from the pad area.
4. Terminal countdown sequence will be jointly controlled by the flight crew and the launch crew in the LCC Firing Room. The launch crew will supplement the flight crew in the monitoring and control of the vehicle and will be totally responsible for operation of the ML and Pad equipment.
5. The launch pad operations will make minimum use of carry-on test equipment or drag-on cables which would interface with the vehicle circuitry. All possible attempts will be made to maintain the vehicle in the flight configuration after rollout from the VAB.



6. Subsystem functional verification performed during the countdown will be top level, end-to-end type checks. Detailed performance checks and redundant path verifications will be performed only when break of configuration is not required during the launch pad flow.

Candidate Ground Control System/Equipment Requirements

1. A ground computer system will be provided for control and monitoring of vehicle, support equipment and certain facility system functions. The ground computer system will:
 - a. Provide an economical means for automating checkout, servicing and countdown sequences.
 - b. Provide display and control versatility such that the Firing Room launch crew can be substantially reduced from that required for today's Saturn vehicle.
2. Existing facility-type systems will continue to use their existing dedicated hardware control systems unless significant cost or operational advantages can be realized by interfacing these systems with the ground computer system.
3. A telemetry ground station will be provided for monitoring and recording of telemetry data.
4. Support equipment units to be used for relatively autonomous and non-hazardous operations will incorporate the necessary control and monitor capability within the unit itself and will not interface with the ground computer system.

4.1.6 Safing Area Operations, Ground Rules and Requirements

The operations, ground rules, and candidate system/equipment requirements considered in this study for the safing area are identified below:

Operations

1. Connect required umbilicals and service interfaces.
2. Offload of flight data
3. Flight crew egress
4. Ordnance safing
5. Offload residual hypergolics
6. Flush, purge and dry hypergolic systems
7. Remove storable pods
8. Purge MPS hydrogen systems
9. Drain and purge PRDS LH₂ and LO₂ systems
10. Safe and/or remove payload if required.
11. Hazardous checkout and troubleshooting (if required)

Ground Rules

1. No checkout will be performed in the Safing area except:
 - a. Verification of support equipment interfaces prior to initiating hazardous operations.
 - b. Hazardous checkout or troubleshooting operations requiring the use of cryogenic or hypergolic propellants or pneumatic pressures above personnel safe limits.



2. Cryogenic and hypergolic propellant safing operations and other hazardous operations (see above paragraph) will be controlled from a remote control and monitor station. All other operations should be controlled from the cockpit unless cost or schedules dictate otherwise.

3. The onboard data system will be capable of recording all necessary flight data onto a carry-off tape.

4. RF communications between the vehicle and the ground safing crew will be maintained until the hardline OIS link is connected.

Candidate Ground Control System/Equipment Requirements

1. Support equipment units can be connected, put in operation and then continue operation without operator control and monitoring will incorporate control provisions within the subject units.

2. Support equipment units which require control and monitoring throughout the hazardous safing operations will be controlled and monitored from a remote area.

3. Vehicle subsystem functions required for the hazardous safing operations will be controlled and monitored from the same remote area.

4. For operations other than the hazardous safing sequences controlled totally from the remote station, correlation of vehicle and support equipment functions will be accomplished procedurally using the OIS.

4.1.7 Ferry and Alternate Landing Site Operations, Ground Rules and Requirements

The operations, ground rules and candidate system/equipment requirements considered in this study for ferry and alternate landing sites are identified below:

Operations

1. Safing (if required)
2. Offload of Flight Data
3. Payload Removal (if required)
4. Installation of Ferry Kit (if required)
5. Troubleshooting and Repair (as required to support horizontal flight)
6. Load JP fuel, pneumatics and other horizontal flight consumables
7. Horizontal Preflight Checklist

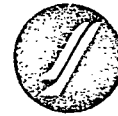
Ground Rules

1. Troubleshooting checkout and repair will be limited to items which constrain horizontal flight.
2. Except for safing, all checkout and servicing operations will be controlled from cockpit.
3. Onboard data system will be capable of recording all necessary flight data onto a carry-off tape.
4. All ground support equipment will be portable and air transportable.
5. Capability will be provided for remote ground control and monitoring of flight vehicle subsystems and support equipment components as required for safing operations.



Candidate Ground Control System/Equipment Requirements

1. Support equipment units will incorporate provisions for local control as practical and consistent with ground rules 2 and 5.
2. Checkout and servicing support equipment will interface with the vehicle via test connection interfaces.
3. Correlation of vehicle and support equipment functions will be accomplished procedurally using the OIS.



4.2 AVIONICS SUBSYSTEM REQUIREMENTS TO FACILITATE CHECKOUT

4.2.1 Operational Checkout

The orbiter and booster avionic subsystem requirements that must be satisfied by the vehicle systems in order to facilitate checkout during the operational phase of the Shuttle Program are described in subsequent sections.

Displays and Controls

Cockpit displays and controls should be designed such that maximum ground checkout can be performed onboard with a minimum of support equipment.

Subsystem Objectives

1. Provide data to enable the crew to evaluate the performance of each functional loop when redundancy is implemented with backup or standby functional loops. This data will be summary in nature to minimize crew evaluations, i.e., Go/No Go type in lieu of multiple measurements.
2. Provide commands to the crew to disable and isolate failed functional loops, energize backup functional loops, put standby units on line, and provide data to the crew to assess the performance of the new active loop.
3. Provide the necessary information which will allow the crew to determine the subsystem status (i.e., active, failed, and available backup loops).

Vehicle/Ground Interface

The vehicle/ground interface shall provide the capability for control and monitor of those vehicle functions which are not required for flight operation, i.e., fill and drain valves, etc., during times when the vehicle is unmanned for safety reasons.

Subsystem Requirements

1. Provide subsystem performance data to a ground control station in cases that onboard subsystem are required to be active during hazardous operations.
2. Provide the capability to reconfigure onboard systems when both ground and onboard functions must be exercised in unison for a particular hazardous operation. Example: ground power to vehicle during fueling, operation of onboard valves in unison with ground valves during chill-down and fueling; countdown sequencer.
3. Provide a means to revert to a safe, static state at any point in a hazardous sequence.



Vehicle RF Subsystems

Vehicle RF subsystems ground checkout will be performed whenever possible by contact with the operational ground facilities, i.e., Comm Lab., ATC, MSFN, etc.

Subsystem Requirements

1. Provide capability for operation using ground re-radiation system to allow testing during M&R activities.
2. Provide antenna hats to minimize range interference during prelaunch operations.
3. Allow maximum utilization of existing RF ground facilities.

Instrumentation

Sufficient instrumentation shall be available so that malfunctioned items can be identified to the LRU level where practical during ground checkout. This capability is not required during flight. Ground checkout to the LRU level may be manual or automated, whichever is most practical. LRU instrumentation points shall be made available without taking the subsystem out of flight configuration.

Subsystem Requirements

1. Provide instrumentation in addition to those required for onboard redundancy management. These points must be accessible to the ground crews for direct management or remotely to an accessible area without disrupting other operating subsystems.
2. Provide the capability to de-energize all support subsystems to each LRU without de-energizing the total support system.
3. Provide a means to revalidate the repaired subsystem with minimum dependency on other subsystems.
4. Provide a means to insert simulation/stimuli signals in cases where it is not practical to functionally retest the invalidated subsystem end to end.

4.2.2 Early Program Phase Checkout

The subsystem requirements to facilitate checkout during the early program phase will include all of the requirements of the operational phase in addition to the following:

1. Provide additional instrumentation to allow detailed subsystem performance evaluation. Additional instrumentation should be a part of DFI. The additional test data is used only to gain confidence in subsystem design.



2. Provide a means to record performance data for comparison of the effects of flight on each subsystem. This data will provide a base for historical trend files.

3. Provide a means to checkout ground interfaces to the vehicle prior to mating with the vehicle.

4.3 CANDIDATE CHECKOUT AND LAUNCH OPERATIONS SYSTEMS

4.3.1 Requirements and Ground Rules

Commensurate with the operational requirements outlined in section 4.1 (Checkout and Launch Operations Requirements), the following set of ground rules are established for the ground system:

1. Not economically feasible to maintain and operate two checkout system configurations.

2. Maximum emphasis will be placed on utilizing existing checkout capabilities.

3. Fueling system is independent of checkout system and is recoverable as is from Saturn V launch complex.

4. Automate launch site checkout to extent practical - High Level Language required.

5. Hold operator stations to minimum through use of exception monitoring.

6. Retain existing servicing interfaces to extent practical.

7. Data will be reduced only when exceptions are noted.

8. Checkout/launch support operator stations will be located in the LCC to maintain close personnel proximity to the pad and maintenance bays and to maintain the launch crew in a common location.

9. Booster and orbiter will utilize the same type checkout system (i.e., the interfaces will be compatible).

10. Orbiter tank checkout will be chargeable to the orbiter and will be accomplished via the orbiter interface.

11. Downlink communications will be via a 96 KB PCM link.

12. Pad uplink communications will be via dedicated hardlines for operational control or a serial link.

13. CIF will support launch operations in a back-up role and DFI measurements exclusively.



14. 1300 OFI measurements will be downlinked via PCM.

15. Cockpit checkout capability exists only to the extent to displays and switches functional for flight (i.e., summary in nature).

16. Capability must exist to support two launch pads and four maintenance bays concurrently.

The two candidate checkout systems selected for analysis are the Saturn Launch Computer Complex (SLCC) and the Apollo Checkout Equipment (ACE). Each of these systems meets the criteria of recoverability in some modified form. The Central Instrumentation Facility (CIF) and the Saturn V propellant loading system are identified as recoverable entities for either checkout system.

Figure 4.3.1-1 portrays the overall system and the interfaces of the functional elements.

4.3.2 Acceptance Checkout Equipment

Description

The ACE checkout system (shuttle application) depicted in Figures 4.3.2-1 and -2 is nearly identical to the system presently in use on Apollo for checkout of the CSM. NOTE: Apollo model number designations. The peripheral ACE hardware is useable without modification. The Control Data 160G has been retained because of the prohibitive costs of integrating a new computer with the unique ACE hardware interfaces. The existing A2A lines are used for data transmission between the MSOB and remote launch facilities. Control of vehicle and support GSE is by computer generated serial digital commands to the DTCS for initiation of relay closures, stimuli generation, etc. DTCS stimuli commands are routed through signal conditioning equipment and back to the computer via the downlink for on-line verification. The vehicle DTCS is capable of providing hardwire uplink commands directly to the vehicle under test. For maximum utilization of the existing propellant loading system, a hardwire interface is shown for propellant system control of onboard valves during the propellant loading operation.

With this system, the capability exists to process unconditioned or conditioned hardwire vehicle or GSE downlink measurements. Conditioned measurements are defined as:

Analog = 0 to 50VDC

Discretes =

Logic "1" level = 4.0 to 33 VDC, and,

Logic "0" level = 0 to 1.5 VDC.

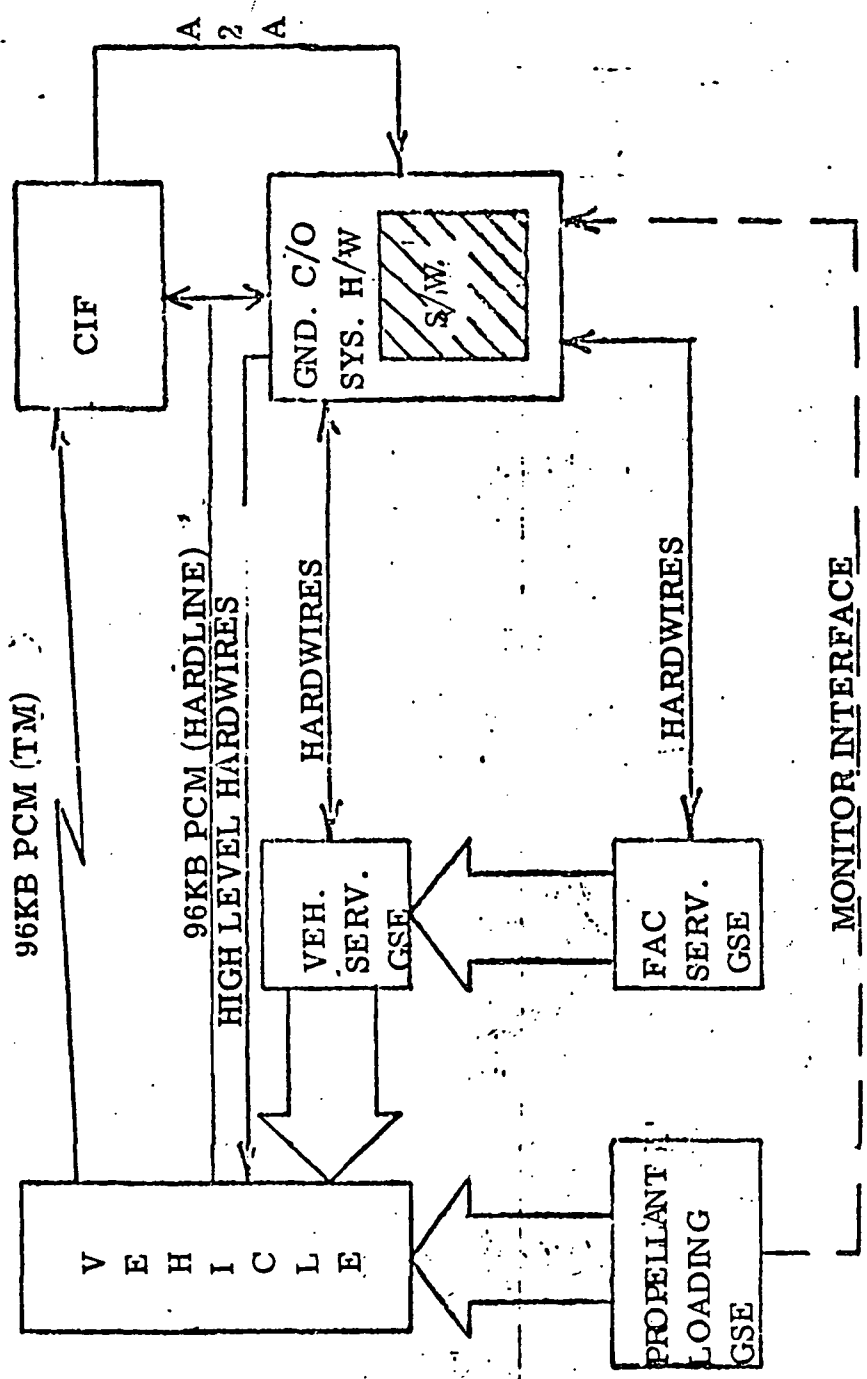


Figure 4.3.1-1 Top Level System Block Diagram

ACE COMPUTER ROOM

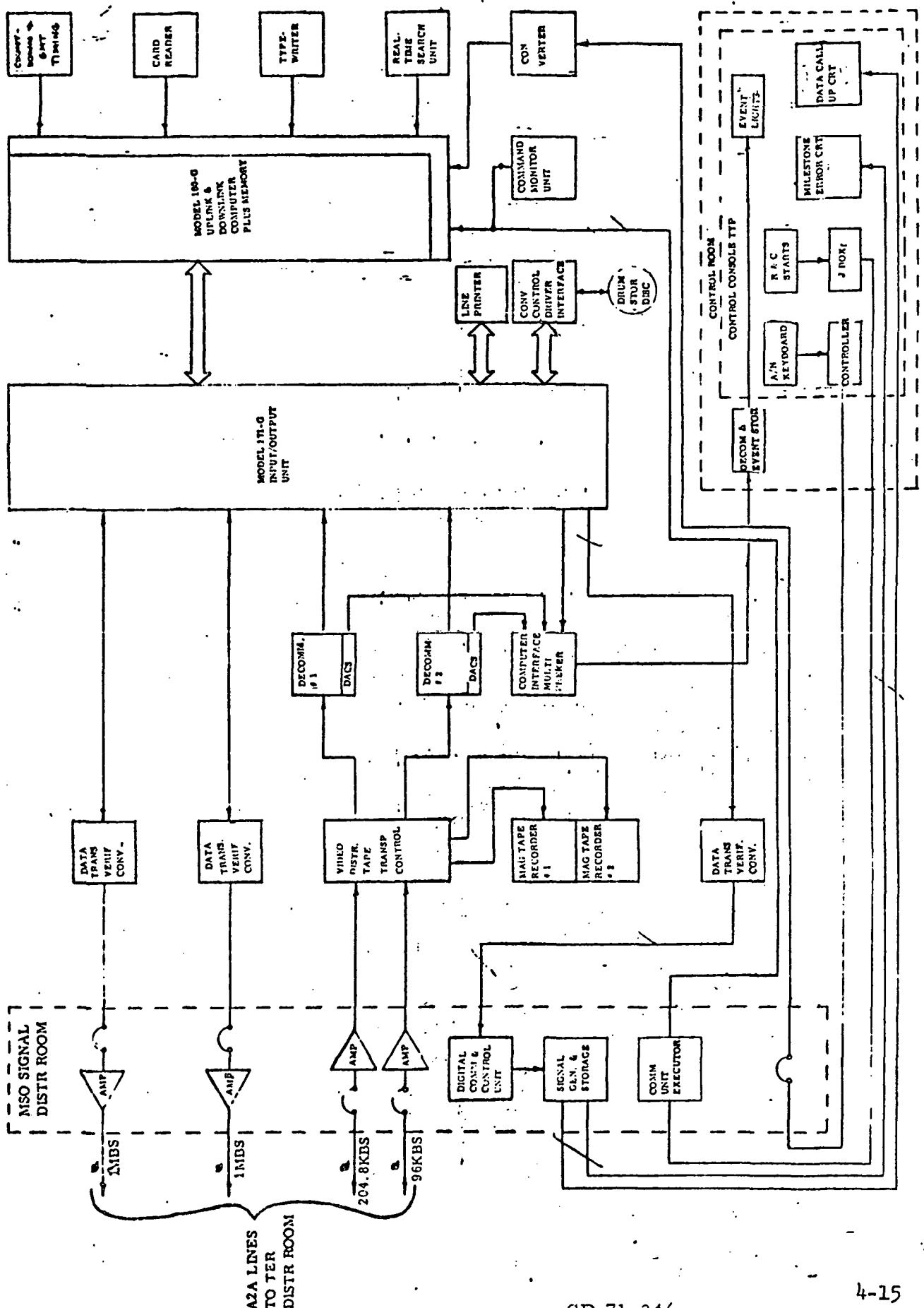


FIGURE 4.3.2-1 ACE CHECKOUT (COMPUTER & CONTROL ROOM)

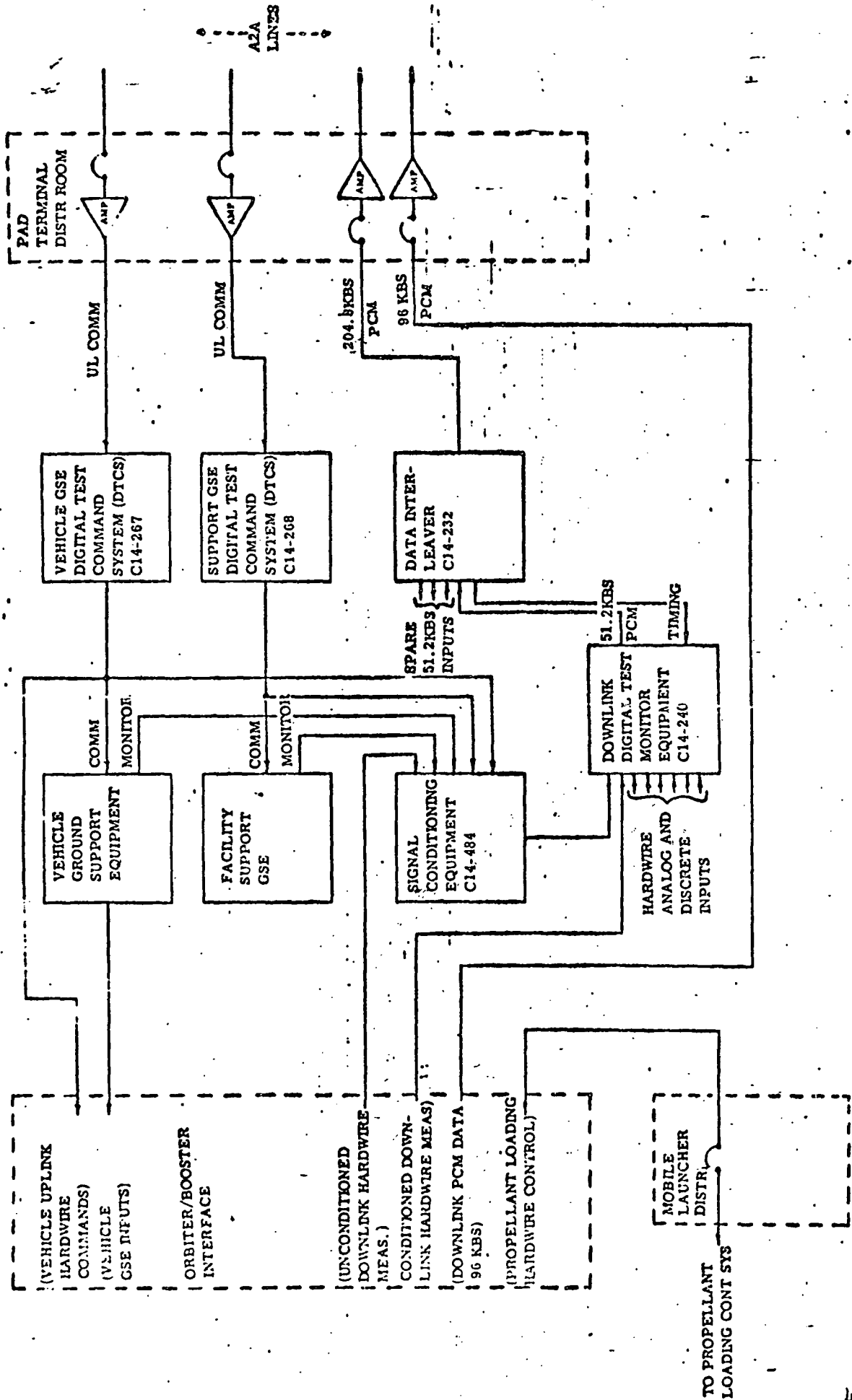


Figure 4.3.2-2. ACE Checkout (Launch Pad)



Downlink analog and discrete measurements are converted to serial 51.2KBS digital data in the DTMS, and interleaved into one of four possible 204.8KBS PCM output channels. The output serial data is routed to computer room decommutator No. 1 for data processing. A separate downlink 96KBS PCM serial data stream is routed to decommutator No. 2 for processing vehicle data.

This system offers the advantage of extreme flexibility using proven Apollo checkout technology. Additional up and downlink capability can be provided by use of the ACE carry-on system (C14-207 and C14-208). Three additional channels of downlink 51.2KBS PCM data can be processed through the data interleaver without modification of the system.

Checkout Philosophy

Control room man-machine interfaces will be minimized. The present ACE checkout concept with man in the loop to initiate commands and evaluate data in real time is not acceptable. A high level test language compiler will be used to translate manual test procedures into automatic machine oriented sequences. Automatic test sequences will be initiated from the test conductor control console alphanumeric keyboard. Real-time display elements will be reduced to a limited number of critical event indications and two CRT displays. One CRT will be used to display milestone and error data, i.e., real-time display of the test being run with status information and exception monitor display of no-go conditions. To supplement this CRT and provide a record of test results, a line printer will be used. Such things as test identification, date and time, test sequence, milestone status, and no-go conditions will be recorded. This will serve as a summary record of the test without maintaining a data file of lengthy manual test procedures. The second CRT is used to display operator requested data by use of the console C-start entry. Console R-start modules are used for manual intervention of automatic test sequences for the purpose of reverting to a safe condition or initiating routine sequences such as power up, etc.

A typical control console is shown with the required computer interface. The number of consoles to support testing is undetermined at this time. All hardware shown may be extracted from existing control consoles and mounted in standard low bay racks. Additional analog meters, event lamps and strip recorders from existing control room inventory may be added to satisfy test requirements.

Modifications/Additions Required

1. Computer

A. Addition of random access mass memory drum and control interface to store automated test procedures.

B. Alphanumeric converter to interface the keyboard with the computer.



2. Control Room

A. Alphanumeric keyboard and controller to call-up automatic sequences from mass memory.

B. Relocate existing control room hardware in standard low bay racks.

3. Peripheral GSE

A. Modify existing facility support GSE interfaces to be compatible with DTCS uplink.

B. Provide downlink monitoring capability for support GSE.

NOTE: This may already exist on a number of GSE items.

4. Software

Develop a high level test language.



4.3.3 Saturn Launch Computer Complex (SLCC)

I. Existing Saturn Launch Computer Complex

- A. Performs Auto Checkout of LV
- B. Operates Test Programs to Check Various Systems
- C. Controls Commands to Vehicle and Receives Responses from

Vehicle.

II. Functional Description of 110A (Reference Figure 4.3.3-1)

A. Core Memory

1. 8 Banks 4K per Bank 32K Total Words
2. LCC uses all 8 banks
3. ML uses 4 banks auto switch to alternate 4 banks if prime core fails.

B. Arithmetic and Control Unit

1. Processes all data
2. Controls test programs
3. Controls operation of IODC devices
4. Performs error checking on all operations
5. Clock 936KC
6. Machine cycle 28.9 usec
7. Memory access cycle 3.5 usec
 - a. CPU uses second cycle
 - b. I/O device uses first cycle and can use second cycle if CPU not using it.
8. Instruction word layout
Command words

C. Magnetic Drum

1. 256 tracks - 128 words per track
2. 32K capacity
3. Maximum access time - 17 millisecc
4. Consecutive words - 135 usec rate.

D. IODCs and Function

1. Each IODC controls own operation after being commanded by the CPU.
2. Upon completion of operation, the CPU is notified by interrupt.
3. During operation, any errors are sensed and the CPU takes appropriate action
4. IODC controls data flow to and from memory
5. IODCs are operated on a priority basis
6. Data from simultaneous IODC operations is interleaved.

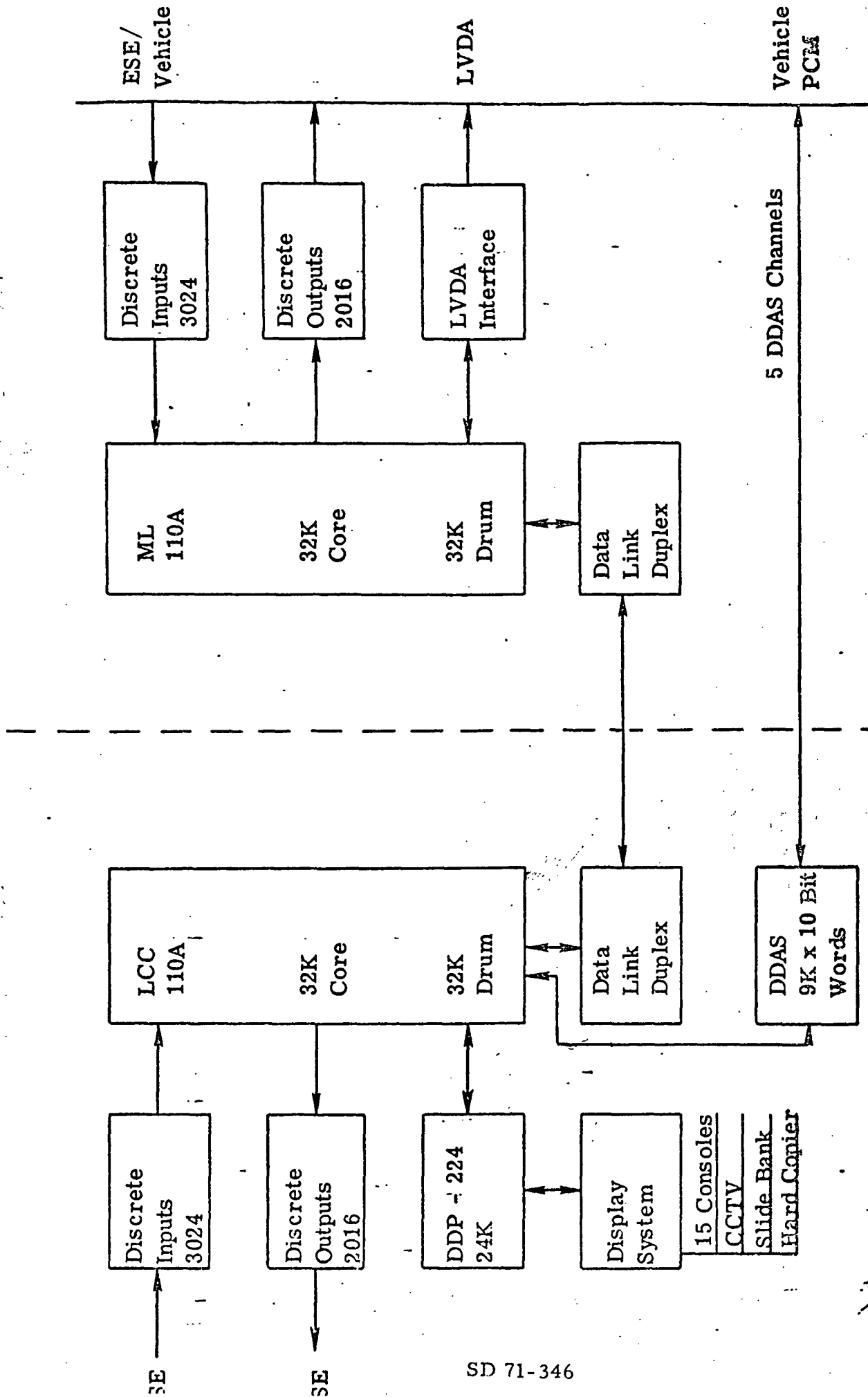


Figure 4.3.3-1. Simplified SLCC Block Diagram

Figure 4.3.3-1.

LCC ML



E. Priority Interrupts

1. Eight (8) lines - 4 levels

- a. Line 8 - error detection (activity monitor)
- b. Line 7 - ML operations
- c. Line 6 - Data link command word requests
- d. Line 5 - Termination interrupts
- e. Line 4 - DI changes
- f. Line 3 - Display requests
- g. Line 2 - Interval timer
- h. Line 1 - Interval timer

F. Error Detection

1. Memory parity
2. All IO DC errors
3. No instruction alarm

G. Alarm Jump Operation

1. ML has alternate memory capability

- a. Jump on memory parity or no-instruction
- b. Inhibits computer and runs short diagnostics
- c. Switchover time requires 300 millisechs to several secs.
- d. ALT core checked every 2 secs.
- e. After a jump, the failing memory data can be retrieved by test program and failed memory can be reloaded for jump back. Double error stop.

2. LC has reload capability

- a. Jump on memory parity or no instruction
- b. Auto reload from magnetic tape (comp. inhibit)
- c. Runs several diagnostics
- d. Gives option to reload action table patches and rerun switch scan.
- e. Switch scan is automatic
- f. Requires 6 min. (variable if DI conflicts).

III. Computer System Vehicle Interface

A. General Description

LC 110 Controls ML 110 (slave)

B. DCE

1. TMR
2. Feedback to 110A
3. Battery backup
4. Manual capability from firing room
5. Can operate with 2 power supplies inop. One in first three channels; one in second three channels.



C. Data Link Terminals

1. Duplexed with auto switch
2. Transfer rate of 250 kilobits per second
3. Serial transfer
4. Possibility of bad word being accepted is 1 in 3.5 billion
5. If errors detected in a word transfer, 2 additional retransmissions are made.
6. Data link can time out and generate an inoperative condition if available information is processed in 59 millisecc. (59 to 120)

D. Discrete Inputs

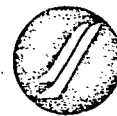
1. Discrete input changes generate interrupts
2. Changes are logged with time of occurrence in relative time.

E. Display System

1. DDP-224 Controls:
 - a. Transfers to and from 110A (110A master)
 - b. Information to and from display consoles
2. Display Hardware includes:
 - a. 15 consoles
 - b. 10 channels of CCTV
 - c. Slide bank (256 slides)
 - d. Hard copier
 - e. TV transmit channel.
3. Display Console Flexibility
 - a. Move memories
 - b. Move consoles
 - c. Look at another console's display
 - d. Look at CCTV or slides
 - e. Call format selects
 - f. Call and monitor test programs.

F. LVDC/LVDA Interface

1. 110A to LVDC
 - a. Load output register
 - b. Set sync
 - c. LVDC reads information and resets sync.
2. LVDC to 110A
 - a. LVDC sends command to generate line 7 interrupt
 - b. 110A sets sync
 - c. LVDC starts transfer and resets sync
 - d. 110A reads information



4.3.4 Central Instrumentation Facility (CIF) (Reference Figure 4.3.4-1)

The CIF will be utilized during development flights and early operational flights for monitoring and analysis of subsystem measurements. Extensive limit checking capabilities exist presently in this facility and will be used to supplement the launch and checkout system, which will possess the capability to display out of limit conditions with respect to selected measurements only. In effect, the CIF will serve as a "kit" to supplement the real time display capability of the operational ground system during the early program time frame.

In addition to the real time display capabilities, CIF has the capability to record raw data and to post process recorded data. These capabilities will be utilized throughout the program to support the establishment and maintenance of a central data base and to enable retrieval of data for required analysis.

Measurements will be transmitted to the CIF Data Core by existing A2A lines and by RF link in a serial PCM format. In addition, existing wide band lines will be used to enable the display of selected CIF data in the LCC. The CIF facility and associated transmission lines are essentially recoverable in their present configuration.

4.3.5 Propellant Loading Control System

The existing propellant loading control system at KSC is shown in Figure 4.3.5-1. The system is almost totally recoverable for support of shuttle propellant loading. The system could be simplified in order to reduce maintenance and operations costs by deleting certain existing elements whose function is to provide redundancy or extreme precision which may not be required by the shuttle system.

The propellant loading system can operate autonomously, with no interface to ground computer systems, and will still provide a capability to load propellants automatically via the networks relay logic.

Although the Saturn Launch Computer (RCA 110A) interfaces with the system today, its role is that of monitor only and no more active role on the part of ground support computers is required for shuttle.

Presently, the manning for propellant loading consists of about 24 men of which about 10 are operators. Simplification of the system could reduce this number by an undetermined factor. The above figures do not include redundant monitoring by the stage propellant personnel.

4.3.6 Rationale for Selection

Table 4.3.6-1 represents a comparison of the ACE and 110A capabilities. Due to time and schedule constraints, completion of a formal trade study was not possible. Inclusion of a ground rule which placed maximum emphasis on common checkout systems at the launch site, and factory acceptance sites led to a decision to drop the SLCC system from

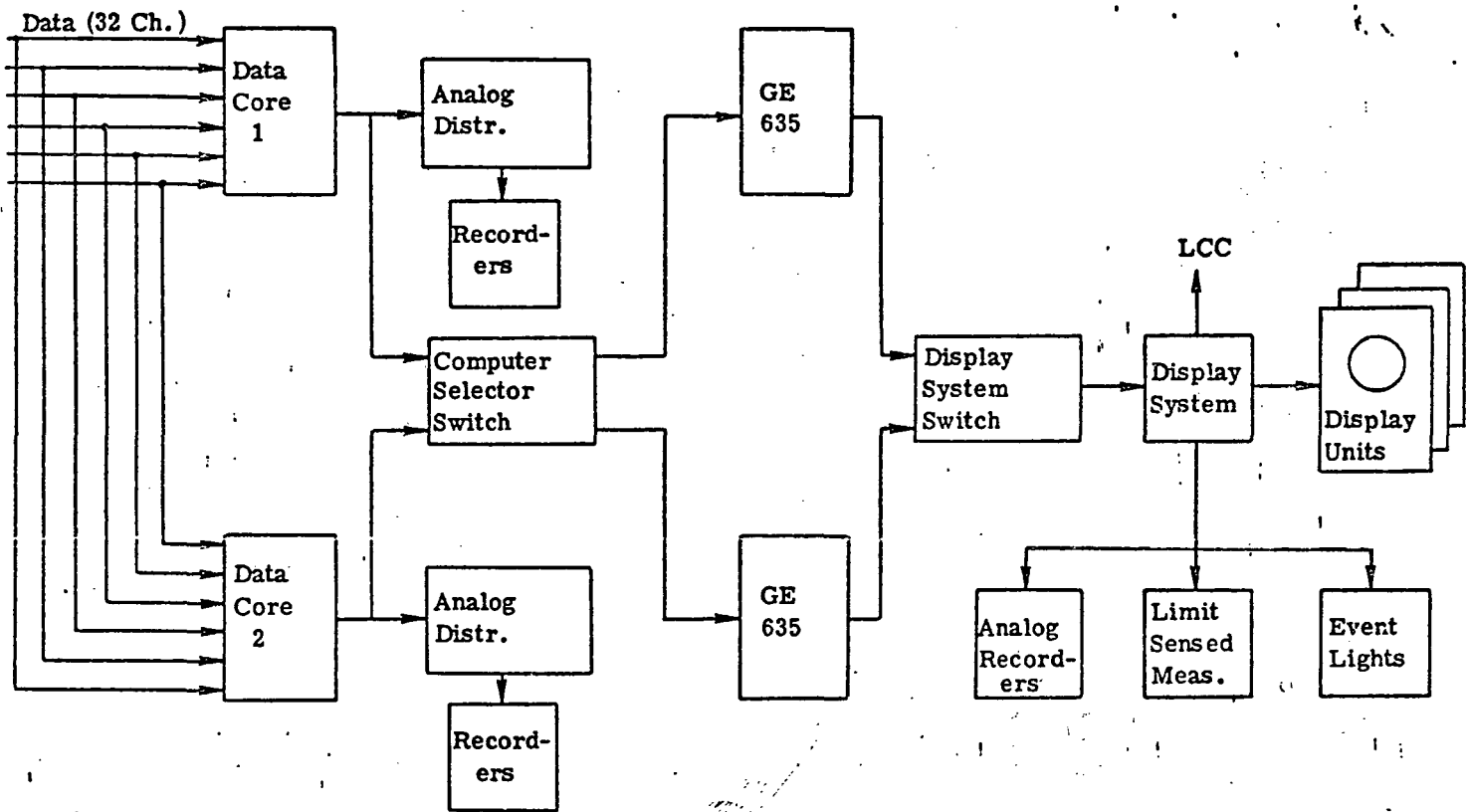


Figure 4.3.4-1 Central Instrumentation Facility (CIF)

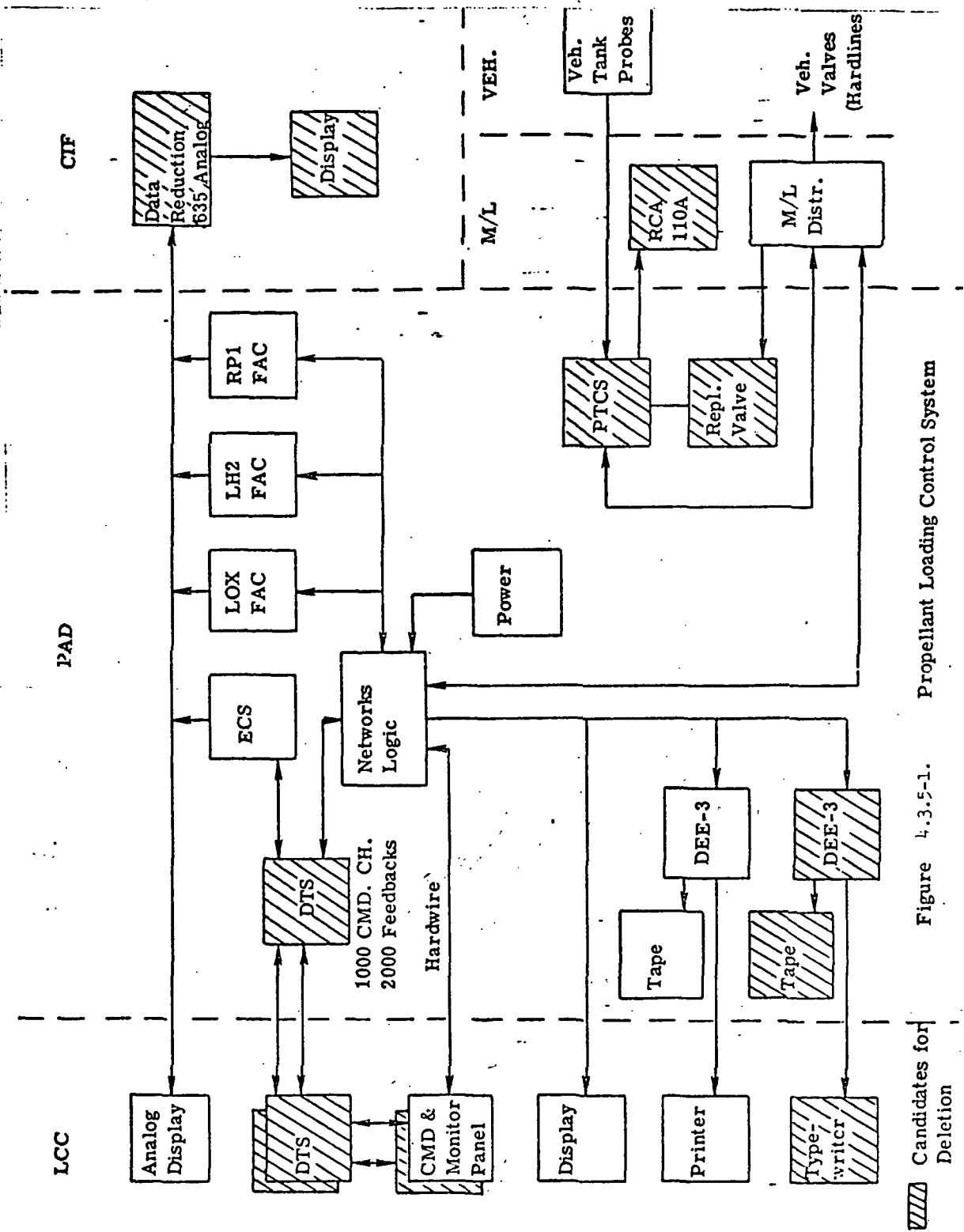


Figure 4.3.5-1. Propellant Loading Control System



	SLCC (RCA 110A)	ACE (CDC 160G)
Word Size	24 bit	13 bit
Cycle Time	10.5 us (3.5 usec Memory Access Time)	1.5 us
Core Size	32K	48K Shared; 8K uplink; 8K downlink
Mass Storage	32K Drum	None
Peripherals	Tape, Card Reader, Printer, DDP 224 Sanders Display	Tape, Card Reader, Printer, CRT's
Recoverable Software	OPS. SYS (partial), ATOLL, Diagnostics, Post Processing and System Tape Generation	Very little, large re-programming effort
Recoverable Hardware Interfaces	Servicing, Countdown Clock, Dedicated panels DI, DO, Analog, Data Link, Signal Conditioning	DTCS, signal conditioning, e.g., existing data interleaver for servicing equipment, vehicle data (use dedicated vehicle PCM downlink)
Reliability	High (98.6% availability on Saturn 506, 508 and 510)	High (99.8% test availability Apollo support). Note: This assumes one active, and three backup computer rooms.
Potential Application	Launch site, labs	Launch site, labs, horizontal flight test, factory
Special Features	Fail safe sys. via DCE equipment TMR logic Duplex redundancy in M/L Switchable Displays, Duplex Data link	Uplink and downlink shared memory. Direct access from PCM decommutator to core memory.
Required Modification	None	Add core and mass storage

Table 4.3-6-1. Preliminary Checkout System Comparison



further consideration and concentrate on a modified ACE system as the selected candidate.

4.4 SELECTED CHECKOUT, ACCEPTANCE, AND OPERATIONS SYSTEMS

Checkout, acceptance and operations systems have been selected for the checkout and launch operations systems, manufacturing acceptance, horizontal flight and ferry sites, and for bench maintenance. These are described below.

4.4.1 ACE

Figures 4.4.1-1 and 4.4.1-2 are presented as overall block diagrams of the selected checkout and launch operation system. The selected system makes maximum use of existing hardware and associated interfaces.

The proposed shuttle ACE checkout system is identical to the system presently in use on Apollo with the exception of a new display and control system. Table 4.4.1-1 is a cost comparison between the minimum hardware modification and the upgraded system modifications. The estimated cost comparison supports this selection.

System Description

The Control Data 160G has been retained because of the prohibitive cost of integrating a new computer with the unique ACE hardware interfaces.

The existing A2A lines are used for data transmission between the control rooms and remote facilities. (Launch pad, checkout bays, safing areas, etc.).

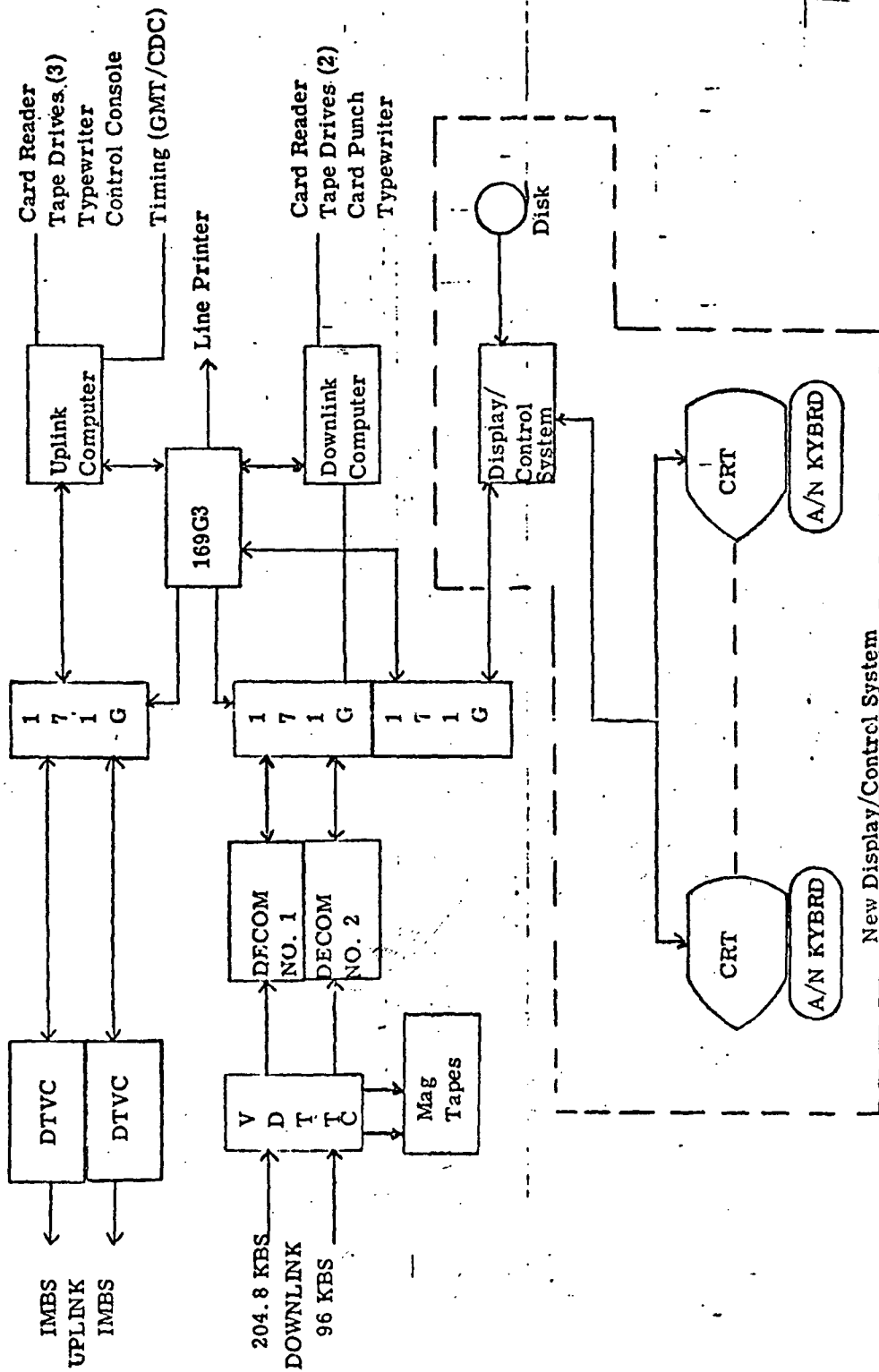
Control of vehicle and support GSE is by computer generated serial digital commands to the DTCS for initiation of relay closures, stimuli generation, etc. DTCS stimuli commands are routed through the signal conditioning equipment, and back to the computer via the downlink for on-line verification. The vehicle DTCS is capable of providing hardware uplink commands directly to the vehicle under test.

For maximum utilization of the existing propellant loading system a hardware interface is shown for propellant system control of onboard valves during the propellant loading operation.

With this system the capability exists to process uncondition or conditioned hardware vehicle or GSE downlink measurements. Conditioned measurements are defined as:

ANALOG = 0 to 5VDC
DISCRETES = Logic '1' = 4.0 to 33VDC
 Logic '0' = 0 to 1.5 VDC

Downlink analog and discrete measurements are converted to serial 51.2 KBS digital data in the DTMS, and interleaved into one of four possible 204.8 KBS PCM output channels. The output serial data is routed



SHUTTLE/ACE SYSTEM CONFIGURATION

FIGURE 4.4.1-1

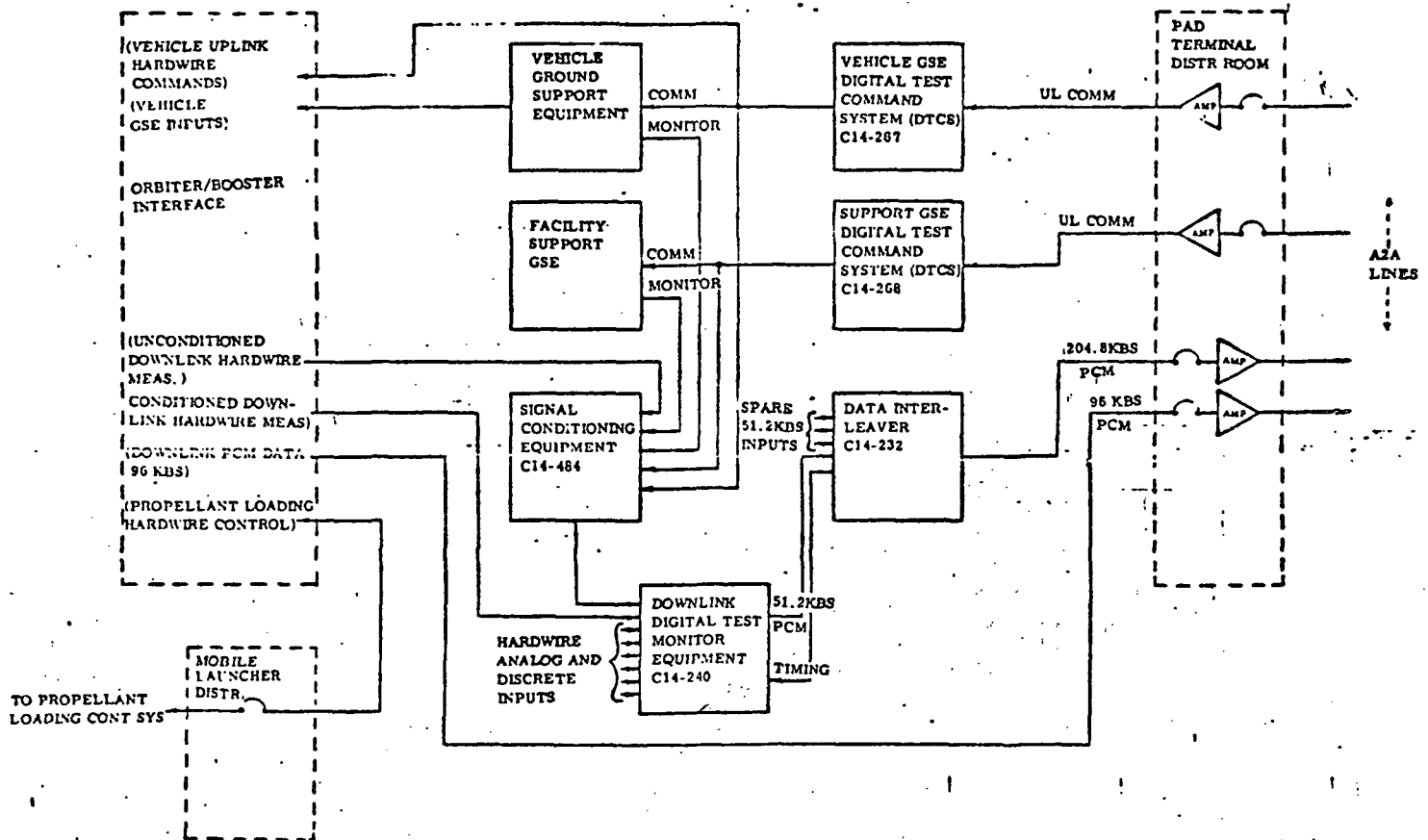


FIGURE 4.4.1-2. ACE CHECKOUT SYSTEM (LAUNCH PAD)



Table 4.4.1-1

TOTAL COST COMPARISON



THE TWO PROPOSED CHECK-OUT DISPLAY & CONTROL SYS (BASED ON OUTFITTING TWO LCC FR, & FOUR MAINTENANCE BAYS)

COST ITEMS	FIG 2A - MIN MOD DISPLAY & CONTROL SYSTEM (IN DOLLARS)	FIG 2B - UPGRADED DISPLAY & CONTROL SYSTEM (IN DOLLARS)
1. UPGRADED 21 IN. COLOR CRT, & KEYBOARD DISPLAY & CONTR SYS A. TWO LCC FR'S WITH NINE CONSOLES PER ROOM B. FOUR MAINT BAYS, TWO MFG ACCEPT (4 EACH)	NA	2,397.2K
2. ADD MASS MEMORY STORAGE, & INTERFACE UNITS TO EACH SYSTEM	676.0K	332.8K
3. ADD A/N KEYBOARD & INTERFACE UNITS TO EACH SYSTEM (40 TOTAL)	2,444.0K	NA-INCLUDED IN ITEM NO. 1
4. MODIFY EXISTING ACE CONTROL ROOM LOW-BOY CONSOLES (40 TOTAL) A. ADD TWO CRT'S PER CONSOLE B. INSTALL A/N KEYBOARD	126K	NA
5. DELTA MAINTENANCE PER YEAR OF THE TWO SYSTEMS A. SYS FIG 2A USES EXISTING CODIM, DTVC, DCCU, SGS, DD/ESDU, & CUE; WHEREAS SYS FIG 2B DOES NOT	48K	NA
6. RELOCATION COSTS TO MOVE ITEMS 4, & 5A	720K	NA
7. INTEGRATE DISPLAY & CONTROL SYSTEM WITH THE COMPUTER	NA	24K
TOTALS	4,014.0K	2,754.0K

CONCLUSION: MORE COST EFFECTIVE TO USE UPGRADED DISPLAY & CONTR SYSTEMS

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to computer room decommuton No. 1 for data processing. A separate down-link 96 KBS PCM serial data stream is routed to decommuton No. 2 for processing vehicle data.

Upgraded Display and Control System

The selected system is augmented by the addition of a display and control system to the 160G configuration.

The display and control system will add the following features to the ACE system:

1. Improve man/machine interface in the control center by the addition of alphanumeric keyboards in the consoles to replace the numerous R START, C START, and K START units located on the existing ACE consoles. The A/N keyboards provides an optimum, standardized device for communication between the console operator and computer system for call-up and execution of automated test sequences stored on the disk file.
2. The ACE checkout system is enhanced by the added flexibility of the display interface by using the increased processing capability of the computers, disk storage of preformatted messages, display of milestones, requested and error data on the CRTS. This concept will replace the existing ACE techniques of direct drive lights, meters, recorders, etc. Display of error data, etc., enhances and man/machine interfaces by reducing operator error and interpretation of problems associated with evaluation of different types and high quantities of measurements.
3. Consolidation and simplification of the control room configuration can be accomplished using a standardize control and display console. The recommended display control console quantity is discussed in subsequent paragraphs.
4. Replacement of the existing manual test initiation and evaluation in favor of automated testing under program control assures test repeatability and reduces overall test flow time and test cost. The improved display and control system also allows for a reduction of test personnel, which result in additional cost savings.

Data Transmission and Processing

The selected data transmission and processing system consists of the following elements:

- DTVC - Data Transmission and Verification Converter
- VDTTC - Video Distribution and Tape Transport Control Unit
- Magnetic Tape Recorders
- DECOM - PCM Decommutators
- 171G - Input/Output Modules
- 169G - External Memory Modules



Display and Control System

The added display and control system will replace the following subsystems in the existing ACE system:

- . START Modules
- . J Box
- . Communication Unit Execution (CUE)
- . Normal Channel (CUE interface for uplink commands)
- . DTVC for downlink computer
- . Symbol Generation and Storage Unit (SGS)

Figure 4.4.1-3 is an overall block diagram of the existing ACE command/monitor system. Figure 4.4.1-4 reflects the deletion of the above units and the addition of the display/control system with associated color CRTS, A/N keyboards, and DISK File.

The following features are available in the proposed display control system:

- . Alphanumeric Keyboard Unit
- . Seven Color, 64 Character Graphic Set Display
- . 40 Character Lines, 24 Lines per Page
- . Flashing Video Image to Alert Operator
- . Expandable to 12 Displays per Station
- . Console up to 2000' Cable Distance from Control Unit
- . Key Switch to Lock Input Controls
- . Built-in System Diagnostic Capability
- . Large Screen Display (21")
- . Discrete Inputs (Switches) 172 Available
- . Discrete Outputs (Lights) 80 Available
- . Built-in Interval Timer
- . Memory to Memory Transfer with 160G - Shared Memory
- . Visual Check of all Keyed Messages
- . Software Protection against Operator Errors and Malfunctions
- . Micro-based Assembly Capability for Software Preparation
- . Disk Storage Interface
 - 2.4 Million 16 Bit Words
 - Average Access Time 126 Milliseconds

4.4.2 Manufacturing Acceptance (ACE Checkout System)

The recommended ACE system described in paragraph 4.4.1 will be installed at the manufacturing acceptance facility. This installation will provide commonality to the manufacturing facility with the launch sites, M&R bays, and safing areas.

As shown in Figure 4.4.2-1 the ACE system will interface with the orbiter vehicle utilizing the standard ACE hardware - (DTCS, DTMS, Data Interleaver, Signal Conditioning Equipment). The PCM downlink will be routed directly to the downlink decom.

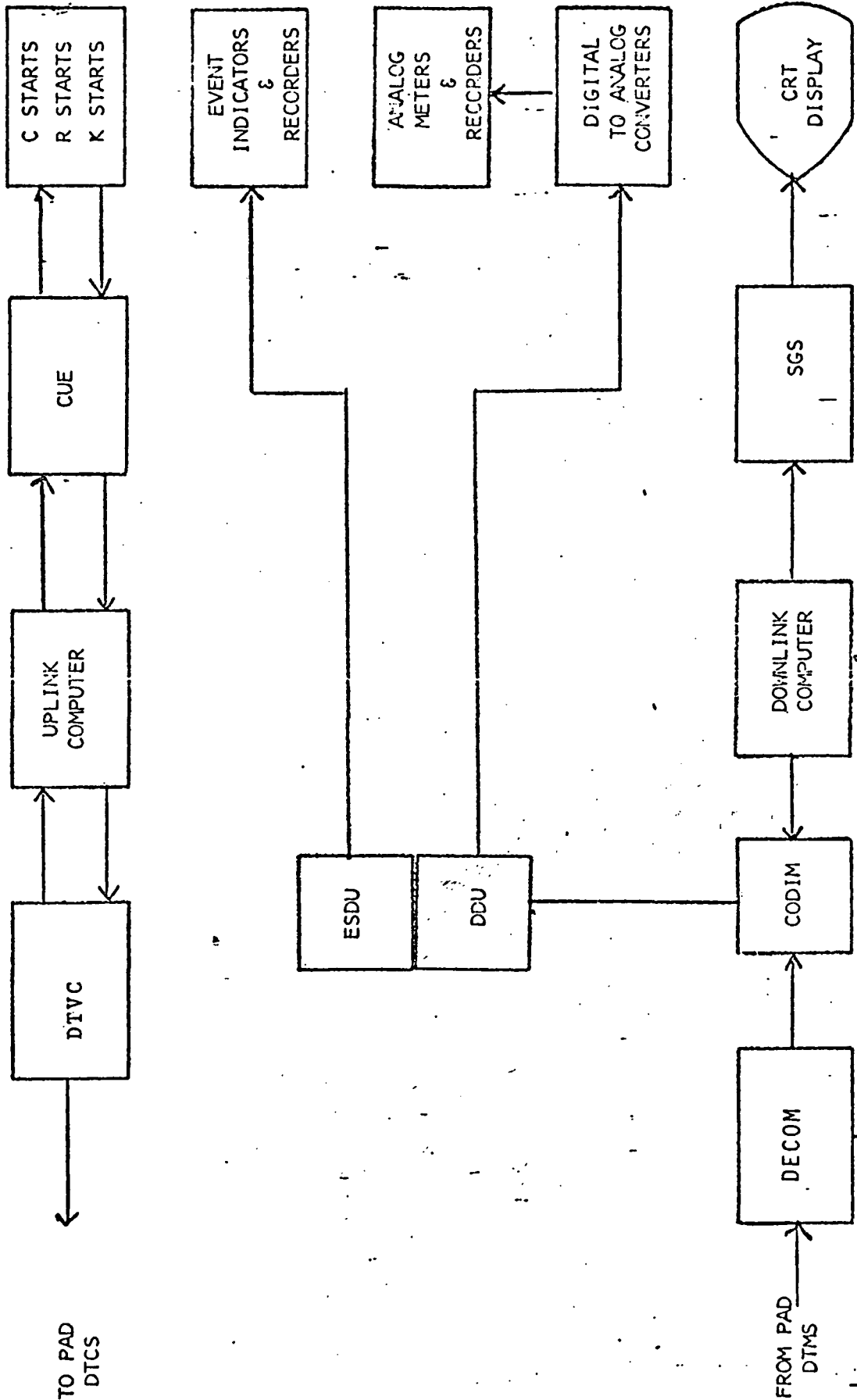


FIGURE 4.4.1-3. EXISTING ACE COMMAND/MONITOR SYSTEM

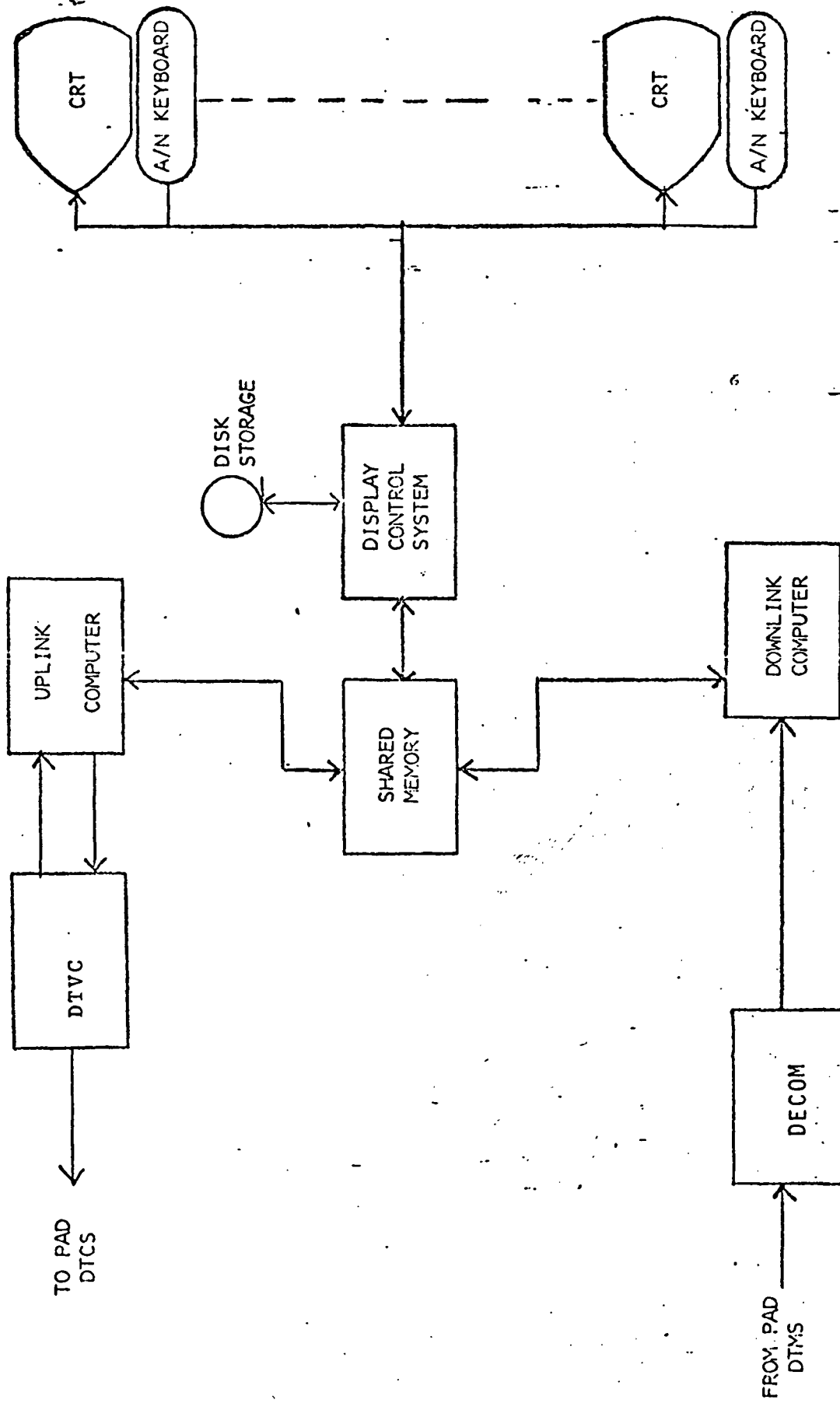


FIGURE 4.4.1-4. MODIFIED ACE COMMAND/MONITOR SYSTEM

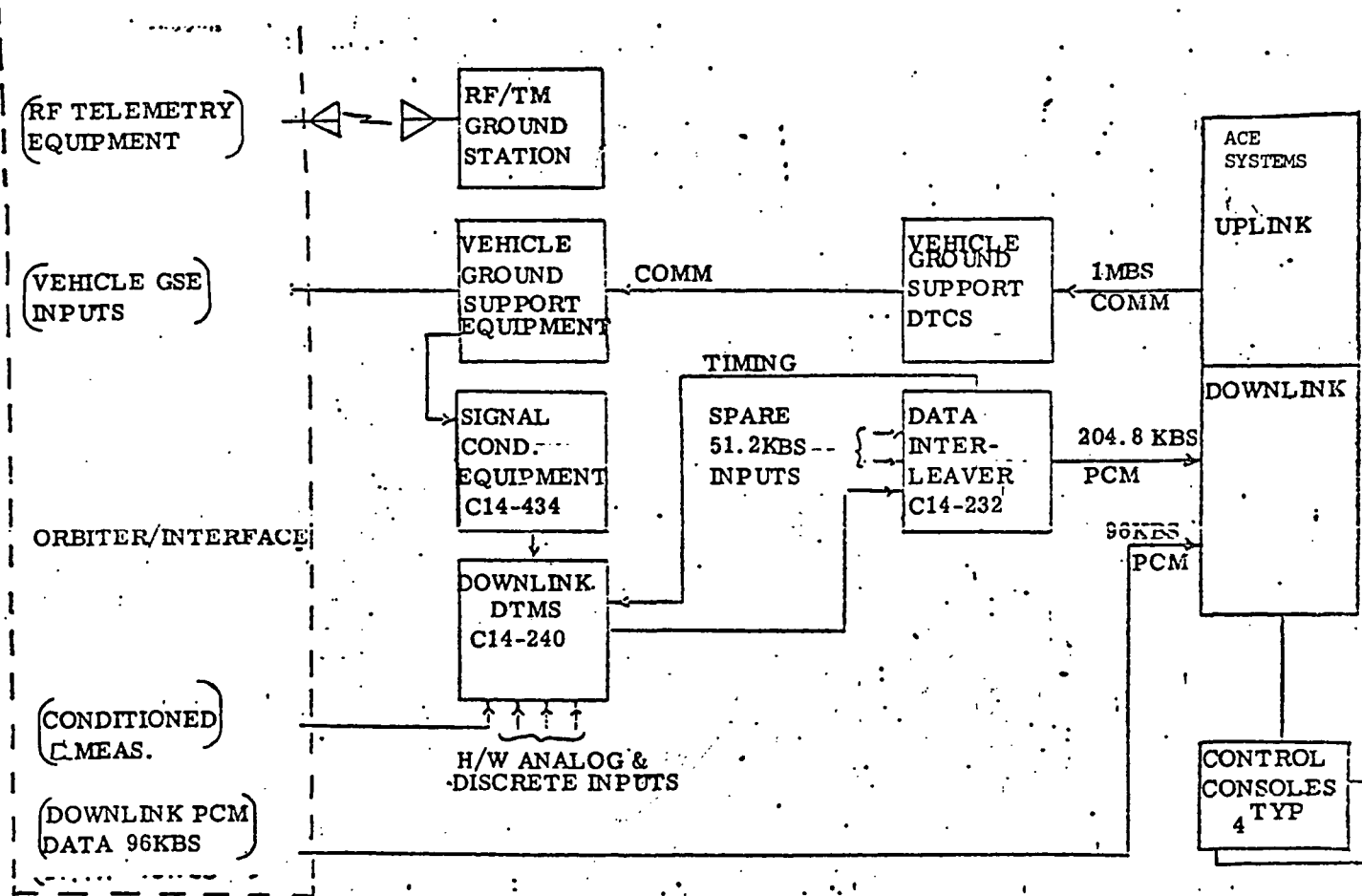


FIGURE 4.4.2-1. MANUFACTURING ACCEPTANCE (ACE C/O SYSTEM)



It is anticipated the manufacturing facility will require 4 display/control consoles.

4.4.3 Horizontal Flight Test (No ACE Checkout System)

The horizontal flight test program will not require an ACE type checkout system. Correlation of vehicle and support equipment functions will be accomplished procedurally by OIS.

All checkout and servicing operations will be controlled from the cockpit except for hazardous operations.

A GFE telemetry ground station will be available at the horizontal test site for receiving flight data in support of subsystem test evaluation.

During horizontal flights for kit 1 and kit 2 the onboard computer will be loaded by a carry-on Field Operational Unit (FOU).

4.4.4 Launch Site (ACE Checkout System)

The recommended ACE system described in paragraph 4.4.1 will be utilized for the firing room/pad complex, the four maintenance and repair areas and the two safing areas.

The overall launch site ACE computer system is shown in Figure 4.4.4-1.

Firing Rooms 1 and 2 are connected to Pads A and B via the computer facility switching matrix located in the LCC terminal switching room.

The VAB maintenance and repair bay will contain an ACE system for each bay. Four bays are required (two orbiter, two booster).

The two safing areas will be controlled from a selected maintenance bay via the facility switching matrix. (See Figure 4.4.4-2).

Figure 4.4.4-3 reflects a typical maintenance bay configuration. It is estimated that 4 display/control consoles will be required.

Figure 4.4.4-4 outlines the ACE interfaces at the launch pad. These interfaces are described in paragraph 4.4.1.

Nine display/control consoles are estimated for launch operations.

4.4.5 Ferry/Alternate Landing Sites (No ACE System)

During operations at ferry and alternate landing sites no ACE system will be required.

Post flight and pre-flight checkout will be performed from the cockpit and will be limited to items which constrain return flight to home base.

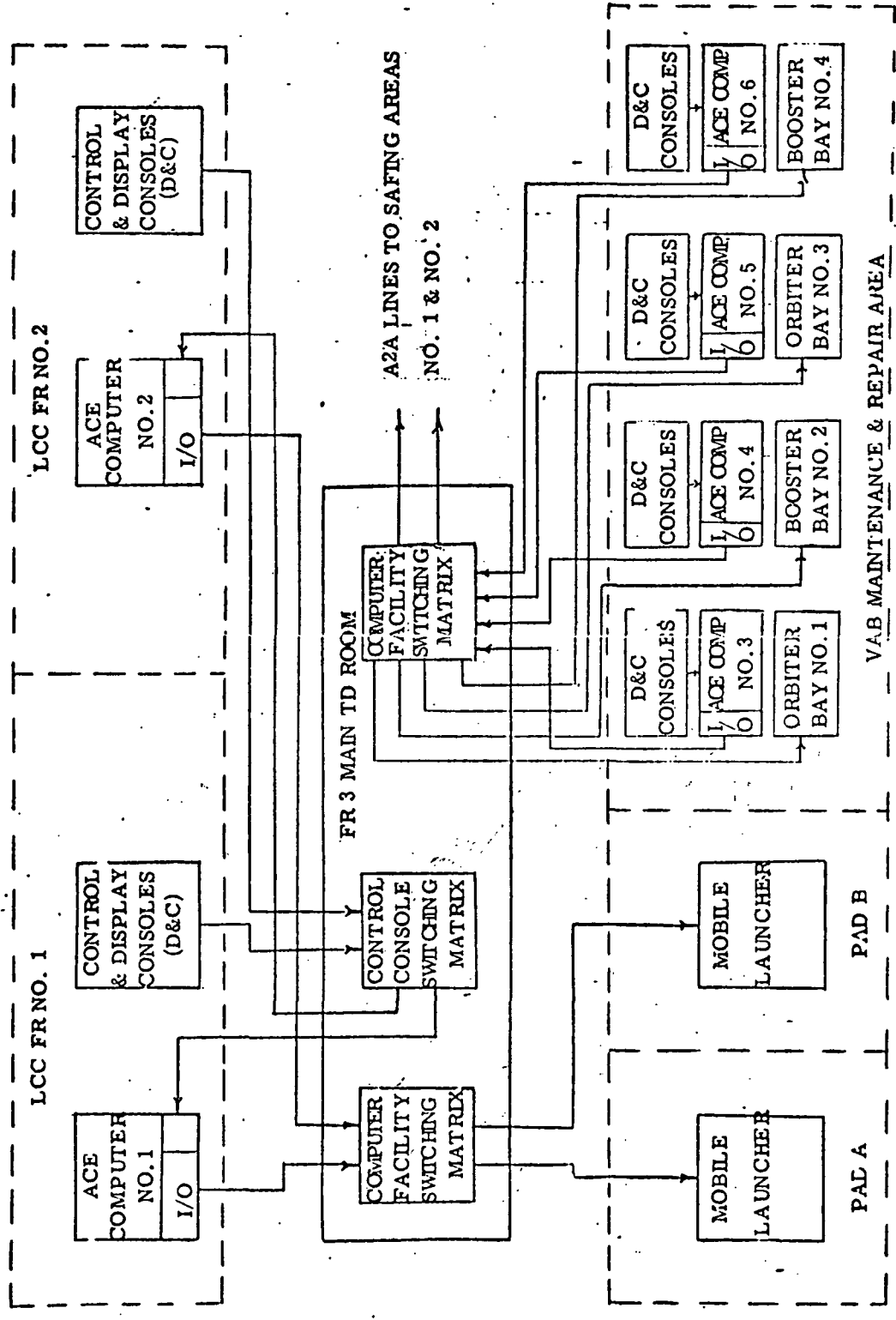
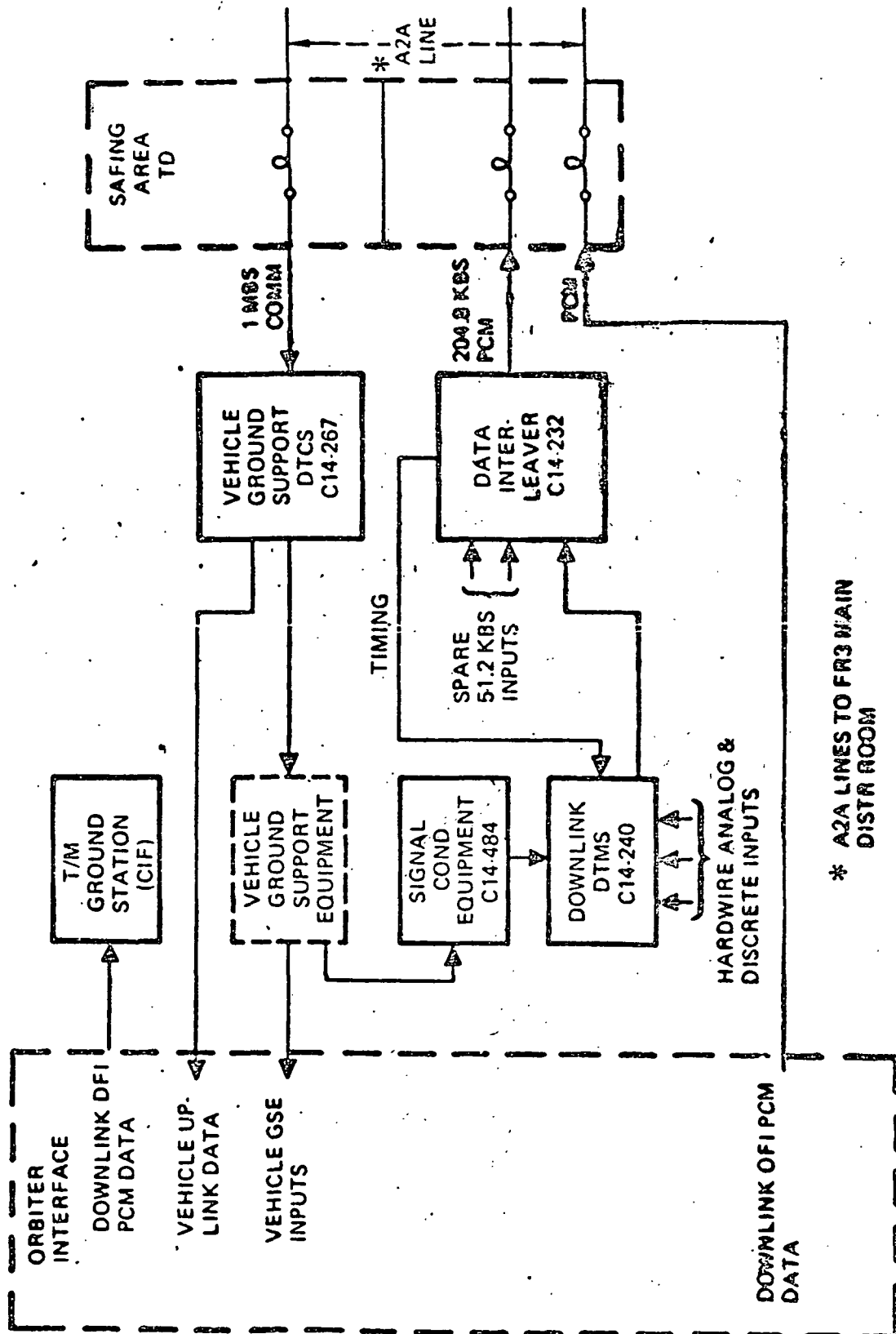


FIGURE 4.4.4-1. LAUNCH SITE ACE COMPUTER & CONTROL CONSOLE UTILIZATION



MODIFIED ACE CHECKOUT SYSTEM SAFING AREA

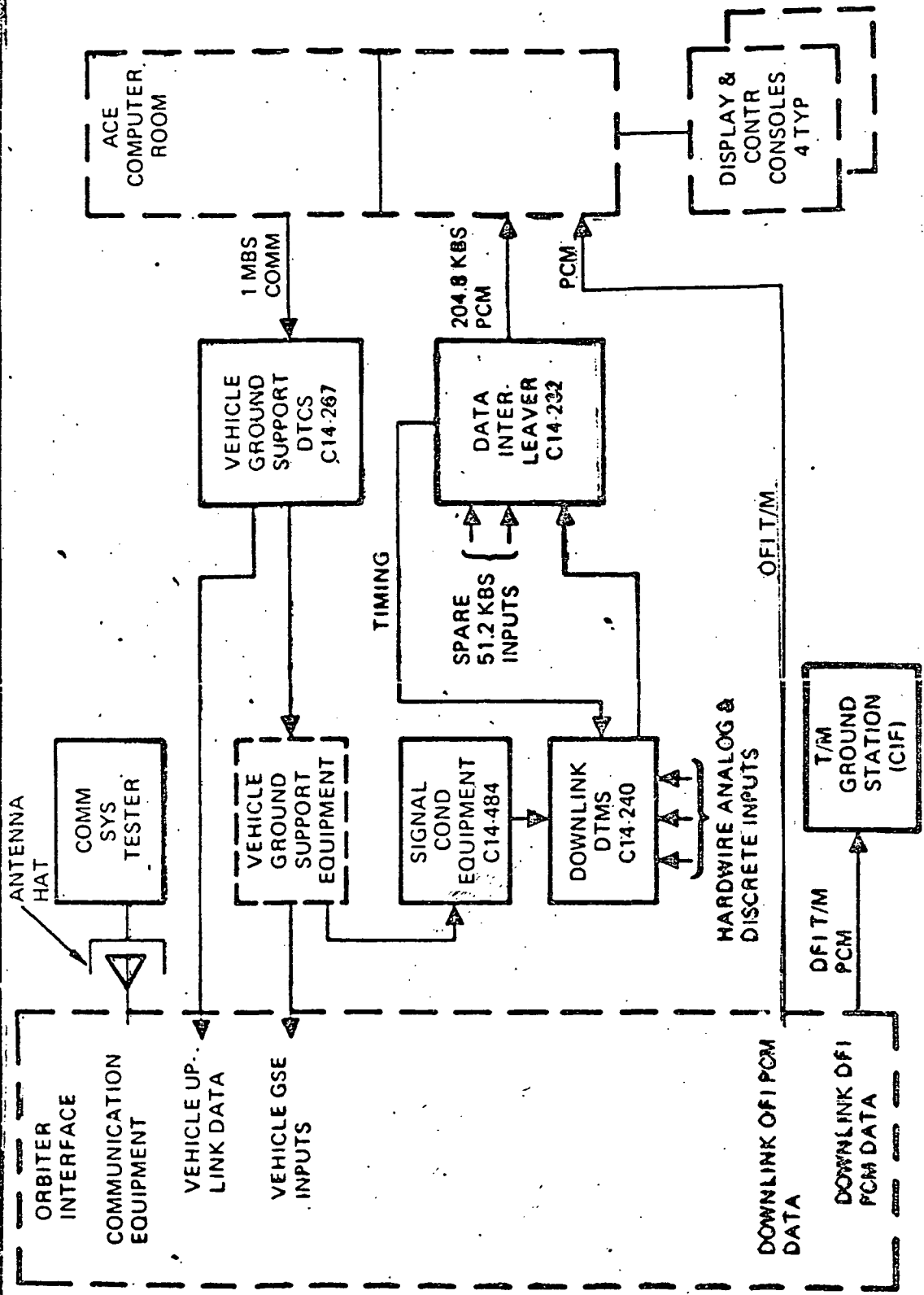


* A2A LINES TO FR3 MAIN
DISTR ROOM

Figure 4.4.4-2. Ace Checkout System (Safing Area)



MODIFIED ACE CHECKOUT SYSTEM MAINTENANCE BAYS



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FIGURE 4.4.4-3 ACE C/O SYS. (MAINTENANCE BAYS)

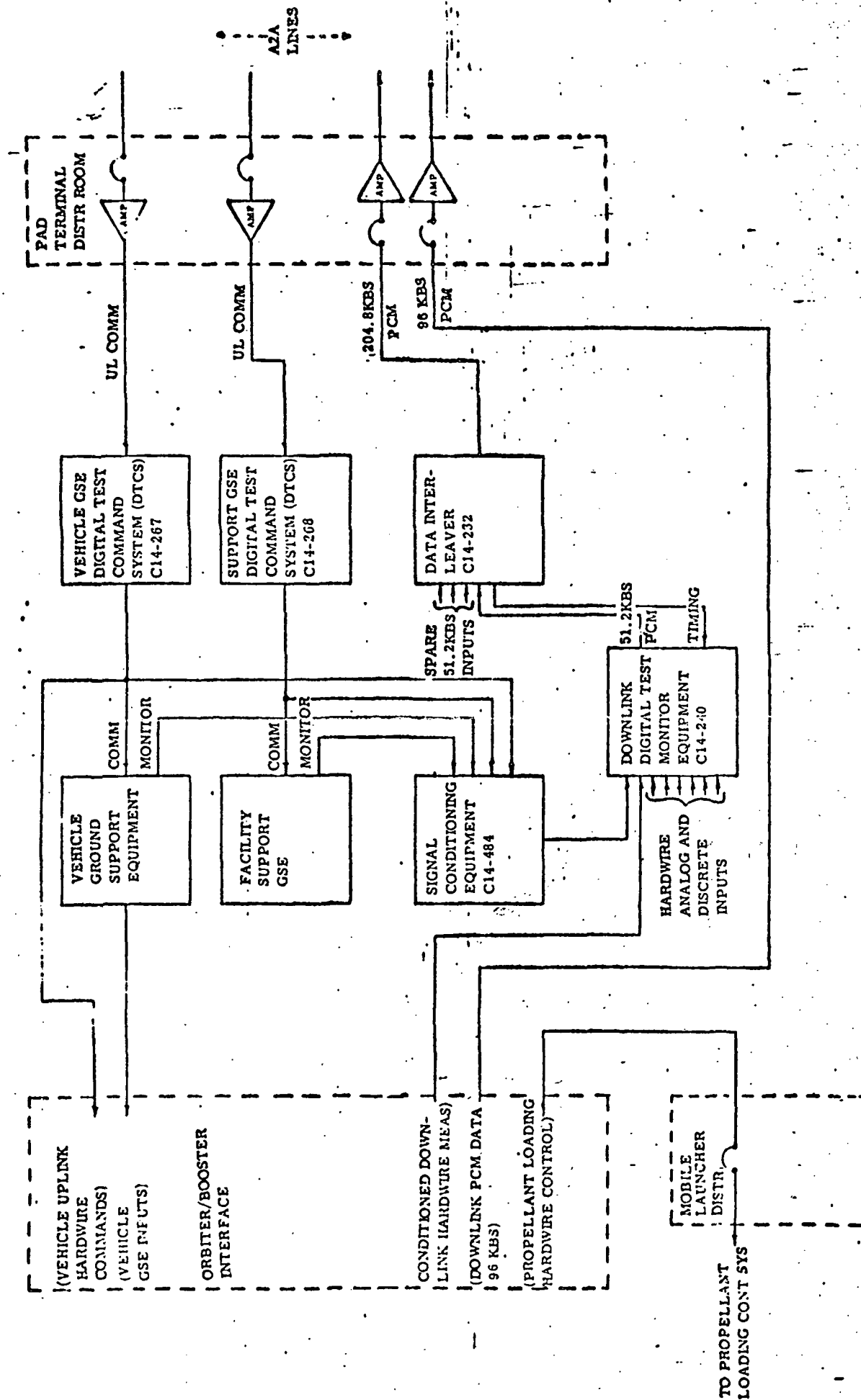


FIGURE 4.4.4-4. ACE CHECKOUT SYSTEM (LAUNCH PAD)



The onboard data system recorded data will be returned to maintenance and repair facility for processing.

4.4.6 Bench Maintenance Support Equipment

The Philosophy defined for LRU repair in the shuttle system logistics plan is to return LRU's to the supplies for repair during the early part of the program when the supplier production lines are in operation. Later in the program when the suppliers have completed their delivers the plan is to set up a complete bench maintenance facility at the operational site.

There are different approaches possible to satisfy the requirements for bench maintenance support. Three basic approaches will be studied in order to determine the most cost effective way to support the bench maintenance task. There three approaches are as follows:

1. Procure all dedicated BME testers for operational support. These units would be the same designs as those used by the supplier for manufacturing acceptance checkout of the LRU's. This will require a definition of the LRU's by part number and supplier in order to establish test equipment availability.

2. Procure an existing multi-purpose BME design of the type used by some of the major airlines at their primary maintenance facility. This computer controlled test set checks out a number of different avionics black boxes. This would be supplemented with dedicated units of BME necessary to support those LRU's which are not compatible with the multi-purpose tester.

3. Design and develop a new multi-purpose BME unit tailored specifically to the shuttle avionic systems. This unit would also have to be supplemented with dedicated testers for any LRU's which could not be cost effectively handled by the multi-purpose design.

A trade-off of these approaches will be performed and the most cost effective approach will be selected considering design and fabrication costs, operating manpower, software costs, and maintenance costs.



4.5 GROUND SOFTWARE

4.5.1 Purpose and Scope

The purpose of this section is to describe the major elements of the Checkout and Launch Operations Software and present software designs, sizes, cost data, and development schedules. The design reflected here has been influenced by the low cost configuration which specifies recovery of as much existing Apollo/Saturn hardware and software as practical.

For the purpose of discussion, these systems are divided into three functional groups, Ground Computer Checkout System, Post Flight Maintenance Data Evaluation, and Telemetry Data Real Time and Post Test Evaluation for Fault Isolation. The various checkout configurations toward which the ground checkout and Launch Operations software effort is directed is identified in Table 4.5-1-1.

4.5.2 Software Functional Description

Ground Checkout Computer System

The Ground Computer Checkout System is composed of various functional elements for the purpose of providing the necessary data processing and control to support the integration, testing and fault isolation of the space shuttle vehicle. The software elements of this system are defined by the configurations of the hardware, procedure and management systems which apply to the various phases of Space Shuttle program testing and repair.

The basic philosophy of the Ground Checkout System is to allow the responsible system engineers to define, implement and execute each testing function without direct programming personnel assistance. This will be accomplished by providing the system engineer with the tools necessary to control the individual area of responsibility through a man machine interface which will be familiar to the system engineer. This interface will be described in the following paragraphs as a High Level Test Oriented Language (HLTL). Figure 4.5.2-1 depicts the operational elements of the Ground Computer Checkout System which requires software support.

Display Control Processor

This element of the Ground Computer Checkout System provides the human interface and performs those tasks necessary to accept command inputs and present results and status of tests being performed.

The software executing within this element provides three basic functions.

. Converts the data which has been generated by the system under test and presents this data to the system engineer in recognizable forms. This data may represent the results of an action initiated to measure and

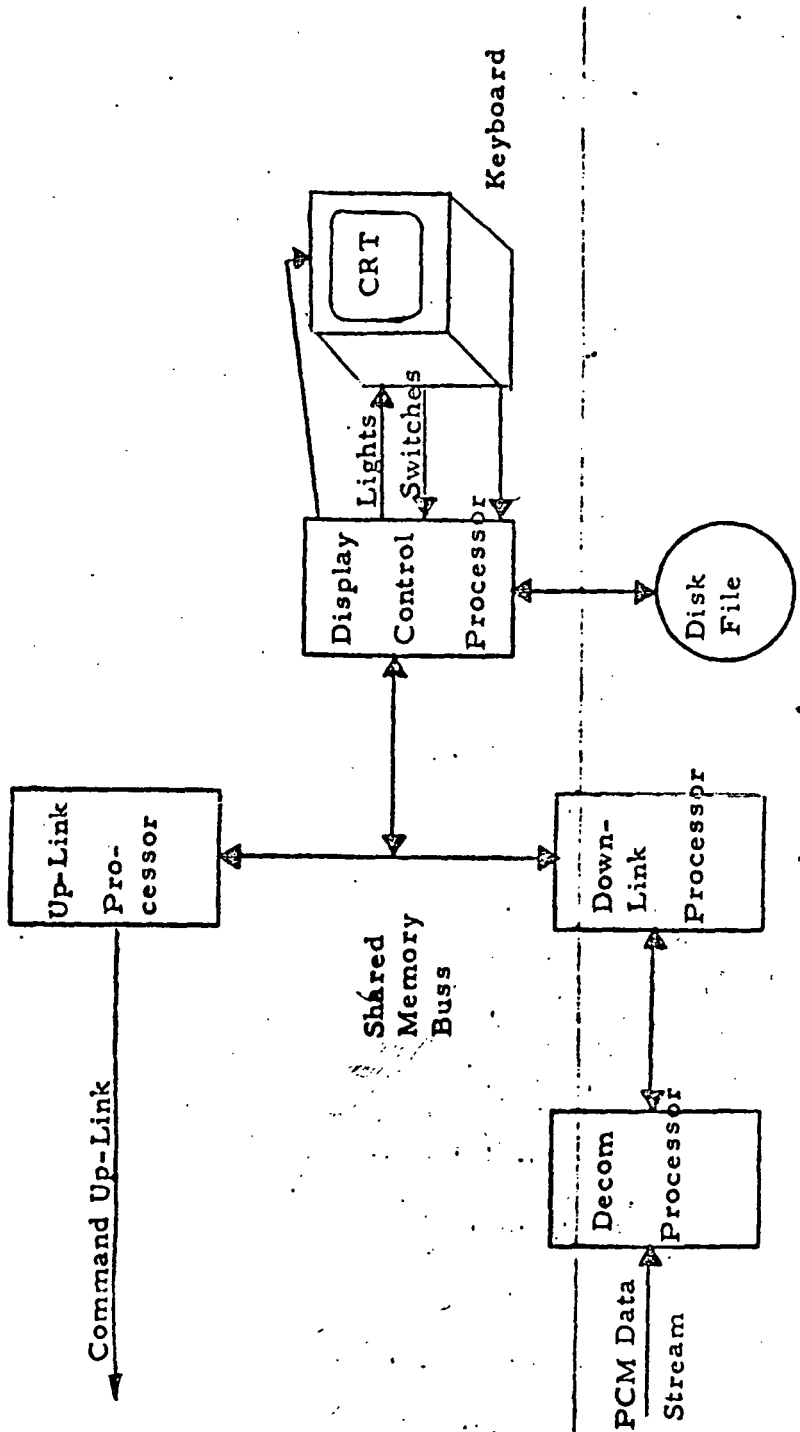


Figure 4.5.2-1. Space Shuttle Ground Computer (Functional)





Table 4.5.1-1. Checkout Facilities Computer Support

	160 G	Display Processor	PCM Decomutator	PCM Processor	GN&C AP-1	Display SP-1
Manufacturing Acceptance	*	*	*		Kit-1	Kit-2
Horizontal Flight Test				*	Kit-1	Kit-2
Launch Site Safing Areas	*	*	*	*		
Post Flight Analysis	*					
Checkout Bays	*	*	*	*	*	*
Prelaunch Support	*	*	*	*	*	*
Ferry and Alternate Landing Sites					*	*
Bench Maintenance						



evaluate the response of a system under test. The data may represent a parameter which requires monitoring to insure its stability within a given range or the data may indicate status or quantity of a constantly varying parameter.

. Processes commands initiated by the system engineer, verifies that the request was initiated within allowable rules (notifies the system engineer of any violations) and routes the request to the appropriate software element elsewhere for processing.

. Maintains control of the mass storage device (DISK) and performs those functions as required to transfer data to and from core memory and perform checks to insure validity of the data transfer.

Up-Link Processor

The Up-Link Processor element has the responsibility of converting the system engineer's command request to the vehicle under test into the machine commands to produce the desired action. This conversion may require multiple actions, critical timing and/or initiation of response verification. Capabilities must be provided to accept these command requests from either the direct system engineers interface (display keyboard, switches), or from predefined procedures which have been stored in the mass storage media of the system.

Data Decomutator

Measurement and status data is constantly being presented to the Ground Computer Checkout System interface. This data must be stored and identified for processing. The Data Decomutator element will perform those functions necessary to prepare the data for processing. These functions will include smoothing to reduce the effect of transit noise, perform limit checking of specified measurements and insert indicators to identify those measurements which are not within the specified limits. The Data Decomutator element will also provide a record, on magnetic tape, of all PCM data received at the interface.

Down Link Processor

The Down Link Processor element has the responsibility for verifying that the vehicle under test has reacted to specific commands within allowable limits of time and quantity. This element must also contain the logic necessary to react to alternate procedures when the expected response falls outside these limits. These alternates may include the initiation of an indication to the system engineer via the Display Control Processor or the initiation of an alternate pre-stored procedure via the command request function within the Up-Link element.



In addition to the response action function, the Down Link Processor element will provide the ability to compare an undetermined number of specified measurements to expected limits. In the event that any of these measurements exceed the limits, initiate an indication of the systems engineer and present the current value of all such out-of-limit measurements.

Ground Checkout Computer Diagnostics

The Ground Checkout Computer Diagnostic programs are those elements used to aid maintenance personnel in the maintenance, fault isolation and repair of the Ground Checkout Computers and peripheral equipment. The programs will be executed during periods of non-test support and will be used to insure that the Ground Checkout Computer equipment is functioning normally prior to test support.

4.5.3 High Level Test Oriented Language

The High Level Test Oriented Language includes those software elements required to compile, edit and validate automatic tests which are generated by the systems engineers to satisfy specific testing requirements.

Test Language Compiler

The Test Language Compiler CPC will convert the systems engineer's test procedures into a format acceptable to the computer. The Compiler will also provide a method to allow the systems engineer to edit and update previously assembled tests. The basic input to the Compiler will be via punched cards and a printer listing will be provided during the compilation. This program will be designed to execute in a non-test support environment.

Test Language Validator

The Test Language Validator will be used to verify all test procedures after compilation in order to detect logic and syntax errors. The Test Language Validator will also verify the branch paths of the procedure to insure that program loops and logic errors do not exist. The Test Language Validator will also be used to check and correlate the test procedure verification data which will be used to execute the procedure for qualification on the simulator system.

4.5.4 Post Flight Maintenance Data Evaluation

During the duration of the flight, several critical measurements will be recorded on the onboard Maintenance Recorder. This data, recorded on magnetic tape, will be removed after the flight and will be processed by a ground computer. The information obtained from this data will be processed in a form to aid in the isolation of items which indicated malfunction or anomalies during the flight. The data from successive flights will be accumulated and trend patterns analyzed to identify



marginal or failure tendencies within the onboard systems.

4.5.5 Telemetry Data Real Time and Post Test Evaluation for Fault Isolation

During the development flight phases of the Space Shuttle Program, it is necessary to evaluate the performance of the flight equipment. The Telemetry Data Real Time and Post Test Evaluation for Fault Isolation System will be implemented in a general purpose computer system which is capable of receiving telemetry data from the Space Shuttle vehicle. The purpose of this system is to develop engineering plans for fault isolation techniques to be used during the operational phases. When the development kit telemetry system is removed, the Telemetry Data Real Time and Post Test Evaluation for Fault Isolation System can be removed with no effect on other Ground Checkout and Launch Operations Systems.



4.5.6 Software Sizes, Costs, and Development Schedules

This section summarizes the checkout and launch operations software sizes, development, manpower and support assumptions and rationale.

The assumptions and ground rules which have affected the costing are itemized.

- o Due to the cost reduction effort, the basic ground checkout computer was recovered from the Saturn/Apollo Program. It is assumed that 20% of the existing Saturn/Apollo software for the ground checkout computers is also recoverable (see Table 4.5.6-1).

- o A new computer/display and disk storage element will be added to the ground checkout computer system.

- o System analysis for the initial delivery of the basic ground checkout computer software is assumed due to minor hardware configurations between the supporting sites. This system analysis is computed to be 20% of the initial development.

- o Two updates to the initial delivery of the High Level Test Oriented Language was also assumed. Each of these updates were assisted at 5% of the initial development effort.

- o In addition to the universal test language implemented in the basic software package, several undefined CPC's will be required for equipments with unique interfaces. The CPC's are assumed to be 20% of the initial basic delivery and five updates have been anticipated with the following increments.

- (1) Basic special CPC are 20% of initial effort for ground checkout computer software.

- (2) First update to special CPC's is 10% of above (1).

- (3) Second update is 5% of above (1).

- (4) Third update is 15% of above (1).

- (5) Fourth update is 10% of above (1).

- (6) Fifth update is 10% of above (1).

Percent of perturbations and penalties for analysis, definition, engineering and special requirements were assessed in the initial development effort and will not be reflected in the update percentages.

- o Development of the High Level Test Oriented Language reflects an effort which is 80% of the Saturn/Apollo ATOLL system. Studies which define much of the analysis for a High Level Test Oriented Language have been performed under NASA budgets and are usable (see Table 4.5.6-2).



Table 4.5.6-1 Existing ACE/SC Apollo Software Sizes

<u>Memory Module</u>	<u>Function</u>	<u>Buffers</u>	<u>Tables</u>	<u>Instructions</u>
U/L Bank 0	System Executive	(None)	164	8028
D/L Bank 0	Special Processing	541	(None)	(7651)*
Bank 2	Messages-U/L Parameters	571	7621	(None)
Bank 4	Relocation-Calibration, etc.	5783	2409	(None)
Bank 6	D/L Parameters	4206	3817	169
Bank 10	System Monitor-G/N Parameter	1345	2721	4126
Bank 12	Scope Output	770	7422	(None)
Bank 14	Scratch Pad-Adap, etc.	3388	2384	2420
Overlay	Utility/Print			11304
	TOTAL	16604	26538	26047

*Special purpose processing programs which are not considered recoverable.



Table 4.5.6-2 Existing Saturn/Apollo ATOLL Software Sizes

<u>FUNCTION</u>	<u>INSTRUCTIONS</u>	<u>TABLES</u>
Compiler		
Executive Control	2500	-
Input Syntax Validation	7500	3750
Output Format	3750	1250
Edit Processor	4100	-
Input/Output Control	2600	-
Utility Support	1900	-
Validation Routines		
Branch Validation	10000	-
Sequence Parameter Validation	6250	3750
Executive	8000	2000



- o Software deliveries are scheduled six months prior to hardware verification complete.

- o No effort has been included to cover cost of directed or unplanned updates.

- o Software sizes, costs and development schedules for checkout support which is executed within the onboard GN&C and display processor computers are included in the Functional Description Document Volume I Onboard Software.

- o The following instruction and table complexity guidelines were utilized:

 - 44 instructions per man-month identifies complex instructions, hardware diagnostics, I/O routines and complex equation problems.

 - 88 instructions per man-month identifies less complex logic, data manipulation, etc.

 - Table generation was calculated on a 500 per man-month basis.

- o Development of the individual system checkout test which will be implemented in the High Level Test Oriented Language is assumed to be a checkout procedure generation effort and has not been included in the checkout and launch operations software effort.

- o The diagnostic programs for the Ground Checkout Computer reflects a modification effort of existing software and is assumed to be approximately 10% of the existing software.

4.5.7 Program Sizes

The following table, 4.5.7-1, summarize the sizes associated with the checkout and launch operations software.



Table 4.5.7-1 Ground Checkout Computer Software Sizes

<u>FUNCTION</u>	<u>INSTRUCTIONS</u>	<u>TABLES</u>
Up-Link Processor		
Executive Control & Timing	500	-
Test Language Executive	1000	200
Up-Link Control	1300	-
Command Interrupter	1000	1000
Command Logging	700	-
I/O Control	200	-
Utilities	1000	-
Self-Test	200	-
TOTALS	5900	1200
Down-Link Processor		
Executive Control & Timing	500	200
Test Language Response Processor	4100	-
Exception Monitor	4600	3000
Response Logging	500	-
I/O Control	1000	-
Utilities	1000	-
Self-Test	200	-
TOTALS	11900	3200



Table 4.5.7-1 Ground Checkout Computer Software Sizes (Con't)

<u>FUNCTION</u>	<u>INSTRUCTIONS</u>	<u>TABLES</u>
Display Processor		
Executive Control & Timing	200	-
Display Output Format and Control	3000	-
Keyboard Processor	2000	-
Discrete Processor	200	-
Disk File Control	300	-
I/O Control	300	-
Utilities	1000	-
Self-Test	200	-
TOTALS	<hr/> 7200	<hr/> -
Data Deocomutator		
Data Smoothing	N/A*	
Limit Checking	N/A*	
Data Buffers	N/A*	
TOTALS	<hr/> 8192	
*The software presently in use on the Apollo/Saturn Program is considered 100% recoverable.		
Ground Checkout Computer Diagnostics		
Command Processor	830	-
Display Processor	5000	-
Data Entry	1000	-
Input/Output	200	-
Light Check	200	-
TOTALS	<hr/> 7230	<hr/> -



Table 4.5.7-2 High Level Test Oriented Language
Software Sizes

<u>FUNCTION</u>	<u>INSTRUCTIONS</u>	<u>TABLES</u>
Compiler		
Executive Control	2000	-
Input Syntax Validation	6000	3000
Output Formatter	3000	1000
Edit Processor	3300	-
I/O Control	2100	-
Utilities	1500	-
TOTALS	<u>17900</u>	<u>4000</u>
Validator		
Branch Executive	8000	-
Sequence Parameter Validation	5000	3000
TOTALS	<u>13000</u>	<u>3000</u>



Table 4.5.7-3 Post Flight Maintenance Data Evaluation
Software Sizes

<u>FUNCTION</u>	<u>INSTRUCTIONS</u>	<u>TABLES</u>
Executive	200*	-
Option Processor	300	-
Data Decomutation & Conversion	1000	-
Output Formatter	1000	4000
I/O Control	100*	-
Data Processing for Fault Isolation	2500	-
Trend Analysis	3200	-
Preventative Maintenance Scheduling	1000	-
TOTALS	<u>9300</u>	<u>4000</u>

*Percentage modification to baseline Ground Checkout Computer
Executive and I/O Routines.



Table 4.5.7-4 Telemetry Data Real Time and Post Test
Evaluation for Fault Isolation

<u>FUNCTION</u>	<u>INSTRUCTIONS</u>	<u>TABLES</u>
ACPS Test Site	4000	-
Cluster Firing Site	4000	-
Horizontal Flight Test	4000	-
TOTALS	<hr/> 12000	<hr/> -



4.5.8 Ground Software Cost Summary

The following table is a summary for the total ground software development cost. The key factors that were used in developing this cost summary are as follows:

- . Vehicle Contractor
 - Manage Software Program
 - Prepare Software Requirement Specifications (not including test language procedures)
 - Prepare Formal Software Verification Specifications
 - Perform Formal Verification
 - Provide Software Support during Operations for Software related problems.

- . Software Subcontractor
 - Interpret Requirements
 - Code and Debug
 - Assemble and Link Edit
 - Assist in Formal Verification
 - Prepare Software Documentation
 - Maintain and Control Software Configuration and Status
 - Compile and Assemble Test Language Procedures and assist in verification (syntax and branching logic)
 - Provide software support during operations for software related problems, maintenance and update.

- . Test Language Procedures are not a part of the software cost.



Table 4.5.8-1 Cost Summary

Ground Checkout Software

- . Vehicle Checkout at NR, GD and KSC
- . High Level Test Language Compiler and Procedure Verif Software
- . All Utility Support Software
- . Verification Software
- . Data Reduction at NR, GD, WSMR, EAFB, and CIF

Orbiter (S/W Sub)	=	\$15.8M
Orbiter (Prime)	=	<u>4.2M</u>
		\$20.0M

Orbiter Total + 40% = \$28.0M
(Growth and Uncertainty)

Booster (S/W Sub)	=	\$ 4.7M
(30% Delta)		
Booster (Prime)	=	<u>1.7M</u>
		\$ 6.4M

Booster Total + 40% = \$ 8.9M
(Growth and Uncertainty)

Total Development Ground Checkout Cost
= \$36.9M



4.6 BOOSTER IMPACT ON GROUND OPERATIONS, LOX/RP AND PF

The booster avionics configuration will not significantly impact the operations and test of the shuttle. The operation of the booster and the test program are different, depending on which booster is ultimately selected, but there appears to be very little impact due to the avionics. There are two configurations of the booster under study; the LOX/RP flyback booster and the LOX/propane pressure fed ballistic booster. There are two versions of the pressure fed; recoverable and expendable. The recoverable pressure fed booster uses brakes and big ribbon drag chutes to reduce the impact velocity to a tolerable level for safe recovery and then through the use of inflatable barges the vehicle is towed back to the launch site. The expendable pressure fed booster is a low cost, drop-in-the-ocean-and-forget-it booster.

Certainly the differences between the flyback and the pressure fed booster will affect the test planning and schedules and the operations. However, with respect to the Alternate Avionics Study there appears to be no significant differences that are driven by the avionics. Section 3, the booster description, discussed the dollar impact and the conclusion was that the booster avionics configuration is not a significant factor in the support equipment configuration.



4.7 COST IMPACT AND SCHEDULE

4.7.1 Checkout Hardware Costs - Selected System

The cost of the ground checkout hardware modifications and new parts and relocations for the selected system for both the orbiter and booster is itemized in Table 4.7.1-1. The costs listed assumes the availability of existing ACE systems at no cost to the Space Shuttle Program. Also, spares and maintenance costs for supporting the use of the system have been omitted.

4.7.2 Operations Costs

The operations costs have been estimated proportional to the number of operators manning the checkout consoles during any program period. Figure 4.7.2-1 presents a ten-year program span with checkout operator requirements (quantity of console stations to be manned) for each of the sites on an annual basis. Total operators per year has then been used as a guide for computing operator costs annually. An increased efficiency has been projected for the later program period.

Hardware and software costs have been listed for the activation periods of the program to provide a fiscal summary of program checkout costs.

4.7.3 Phase B/B' Cost Comparison

Table 4.7.3-1 presents a reduction in overall checkout costs for the avionics and checkout approach taken during Phase B'. This reduction is principally in the large reduction in checkout hardware costs that is realized by modification of equipment presently in the NASA inventory. This approach has become feasible by the Phase B' change to a serial PCM data system instead of the multiplexed data bus that was integral to the Phase B avionics system.

The reduced capability for data processing on board the vehicles has required that data processing software be provided that was not necessary with the Phase B system. The cost of this software has been compensated by a recovery of approximately 20% of the Apollo CSM ACE software technology.

The operations manpower for the Phase B' system is considered substantially higher than that required for the previous system. This is a result of both the decreased onboard data processing capability transferring the task to a ground responsibility, and the estimated reduced efficiency from a fixed format data system, onboard and to the ground, and the resultant less flexible Phase B' checkout system.

It should be noted that although the ten-year projected cost for the Phase B' checkout system, \$76.1 million, is \$17.5 million less than the Phase B costs, and that early development costs and risks have been reduced, the anticipated cost of maintenance for sustaining the basic system that will be approximately 20 years in service by the end of the period is high.



Table 4.7.1-1

TOTAL GROUND CHECKOUT HARDWARE COSTS



INCLUDES:

2 LCC FR, 4 MAINT BAYS, 3 MOBILE LAUNCHERS, 2 LAUNCH PADS,
2 SAFING AREAS, & 2 MFG ACCEPTANCE CHECKOUT AREAS

COST ITEMS	TOTAL COST X 1,000
1. RELOCATE 8 ACE COMPUTER ROOMS - 6 AT KSC, & 2 MFG ACCEPT (ORB & BOOSTER) A. INCLUDES FACILITY PREPARATION, DEACTIVA- TION INSTALLATION, & CHECKOUT VERIFICATION	2,959.360
2. UPGRADE DISPLAY & CONTROL SYSTEM WITH 21 IN. COLOR CRT'S, ADD MASS MEMORY STORAGE, & A/N KEYBOARD ENTRY SYSTEM	2,754.000
3. REWORK REQUIRED TO INTERFACE FACILITY & VEHICLE GSE A. NEW CABLE REQUIREMENTS (305 CABLES) B. MH REQUIRED TO MODIFY, REROUTE, & PATCH TO EXISTING CABLES	793.000 247.000
4. GSE REFURBISHMENT	2,490.000
TOTAL HARDWARE COSTS	9,243.36

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71	72	73	74	75	76	77	78	79	80	81	82	83	84
----	----	----	----	----	----	----	----	----	----	----	----	----	----

SUBSYSTEM DEVEL &
FLT TEST SUPPORT

MFG C/O
(24) (24) (24)

MFG C/O ORB & BOOST
(24) (24) (24) (24)

GND TEST - HORIZ FLT TEST

M&R KSC
(24) (24) (24) (48) (48) (48)

ACTIVATE 2 ADDITIONAL BAYS

ACTIVATE 2ND LCC

STATIC FIRE LAUNCH C/O KSC
(21) (43) (43) (43) (86) (86) (86)

TOTAL OPERATORS		(24)	(45)	(67)	(67)	(67)	(91)	(158)	(158)	(158)	
OPERATOR COSTS	-	0.9	1.7	2.4	2.4	3.2	5.7	5.7	5.7	5.7	\$20M
HARDWARE COSTS	1.4	2.5	(1.4)		(3.9)						9.2M
SOFTWARE COSTS	7.6	6.6	(7.6)		(15.1)						36.9M

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Figure 4.7.2-1 Program C/O Costs



Table 4.7.3-1 ϕB / $\phi B'$ Checkout Cost Comparison

	ϕB	$\phi B'$
CHECKOUT HARDWARE	\$30.8	\$ 9.2
CHECKOUT SOFTWARE	41.8	36.9
OPERATIONS MANPOWER	21.0	30.0
TOTAL	\$93.6	\$76.1

101SV15519A



4.8 SUMMARY

The hardware and software definition and cost will be defined in greater detail during the Phase B' extension. The tasks which have been identified to accomplish this task are as follows:

1. Checkout Systems Definition: Perform studies and prepare the following documents for support of the horizontal flight test, vertical flight test, and operational turnaround checkout.

(a) Vehicle/Ground ICDs: Include up and down data links, RF, servicing, vehicle controls from ground, ground control and interlocks from vehicle. Required for each test/operations site. Specific functions will be identified by ICDs. Operator/machine interfaces will be defined.

(b) Functional Diagrams: Identifies the equipments required for vehicle interface, data transmission processing, display and control. Available equipments and new procurements, designs and modifications are identified by an associated listing.

(c) System/Subsystem Checkout Descriptions: A basic checkout description from the subsystem and operator viewpoint, including measurement and control lists indicating cockpit and/or ground functions. The level of LRU malfunction isolation will be assessed for each functional path identified. This study is to be approached from the ground checkout standpoint and coordinated with a similar concurrent vehicle study.

2. Bench Maintenance Requirements: Analyze and prepare a plan for bench maintenance checkout support for the HFT, VFT, and operational shuttle program. The plan will include the identification of cost effective checkout equipment for both the RDT&T and operational program phases for the appropriate level of maintenance. This study will be based upon specific LRU configurations selected as subsystem elements.

3. Software: Evaluate the impact of checkout study tasks 1 and 2 on the software estimates of the B' study. Analyze the impact of ongoing checkout language studies. Define in more detail the software elements.

4. Cost Data: Generate detailed cost data relative to retrofit, procurement and installation of the hardware/software system recommended. Include plans for the development of the system.



5.0 AVIONICS DEVELOPMENT PLAN

5.1 AVIONICS DEVELOPMENT PLAN DRIVERS

The main features of the Phase B' Avionics configuration and approach which affect the development plan are: (1) maximum use of off-the-shelf hardware, (2) decentralization of Avionics, and (3) early horizontal flight test vehicles phasing into orbital flight capability.

Most of the Avionics hardware is off-the-shelf (e.g., most communications equipment), a minor modification of off-the-shelf hardware (e.g., stability augmentation electronics), or a new assembly of off-the-shelf modules (e.g., D&C panels, relay and distribution boxes). A minimum number of new development equipments (e.g., buffer and decoder) are used. A comparison and discussion of the number of off-the-shelf (or minor modification) and new design (or major modification) equipments is presented in Section 1.0 of this report.

Maximum use of off-the-shelf hardware results in significant reduction in the requirement for component development and qualification testing with associated cost and schedule improvements. The shorter lead times associated with the off-the-shelf hardware approach also permits earlier acquisition of equipment for interface verification.

The effects of decentralizing Avionics (and Non-Avionics) subsystems are:

1. Conventional parallel Avionics (and Non-Avionics) subsystems development can be accomplished without reliance on a central DCM system for performance and operations evaluation.
2. More subsystem integration and verification can be accomplished at major subcontractor facilities, providing earlier design confidence and reducing duplication of test hardware.
3. The requirement for a large, complex Avionics/Non-Avionics test facility for the purpose of total system integration is reduced or eliminated.
4. Conventional total Avionics/Non-Avionics integration may be accomplished in the vehicle with a higher degree of confidence.

Separation and phasing of the Airplane Avionics from the orbital flight Avionics results in a corresponding separation and phasing of the respective development and test activities.



5.2 AVIONICS DEVELOPMENT APPROACH

The Avionics development logic encompasses the requirements analyses and definition, the design and procurement definition, the development test and simulation requirements definition, facility planning/acquisition/check-out activities, performance of development tests and simulation, and the associated reports and revisions. The basic Avionics development logic is depicted in Figure 5.2-1.

Several development test beds will be utilized to accomplish the component, assembly, subsystem, partial combined systems, and finally the total Avionics/Non-Avionics systems integration. These test beds will include vendor and subcontractor facilities, dedicated in-house subsystems laboratories (Avionics and Non-Avionics), a simulation facility coupled with crew station D&C mockups and a capability to insert GN&C hardware in the loop (comprised of hydraulic systems, actuators, control cables, and a partial pilot station), an Iron Bird, and the flight vehicles (ground and flight tests). The utilization of an existing antenna test range is also planned. The development test logic showing the planned utilization of the various test beds is shown in Figure 5.2-2.

5.2.1 Component Development Tests

Since a high percentage of the Avionics equipment is off-the-shelf or assembled from off-the-shelf modules, the requirement for component development testing and the associated development test hardware is minimized. Development tests on new design "procured" components or assemblies will be performed by the appropriate vendor or subcontractor. New design "make" components and assemblies will be developed, evaluated, and integrated into the subsystem in the appropriate dedicated subsystem laboratory. This includes D&C panels, relay boxes, electrical distribution boxes, etc.

5.2.2 Subsystem Development Tests

The COMM, INSTR, & EPCP subsystems will be developed, integrated and evaluated in parallel in the appropriate subsystems laboratories. Significant or critical interfaces between Avionics equipment (or subsystems) and between Avionics and Non-Avionics equipments (or subsystems) will be developed and/or verified by "portaging" equipments between laboratories (e.g., alternators will be taken to the various laboratory areas and compatibility of the equipments with the characteristics of a.c. power verified as early as possible).

The early GN&C subsystem development and integration will be accomplished by the GN&C subcontractor. This early development will be supported by early flight control system analytical development using the simulation Mission Evaluator.

Displays and controls compatibility with D&C interfacing electronics and other subsystems will be developed and verified in the in-house D&C laboratory. The "portage" technique of carrying equipment between laboratories for intersubsystem interface compatibility testing will be employed where necessary.

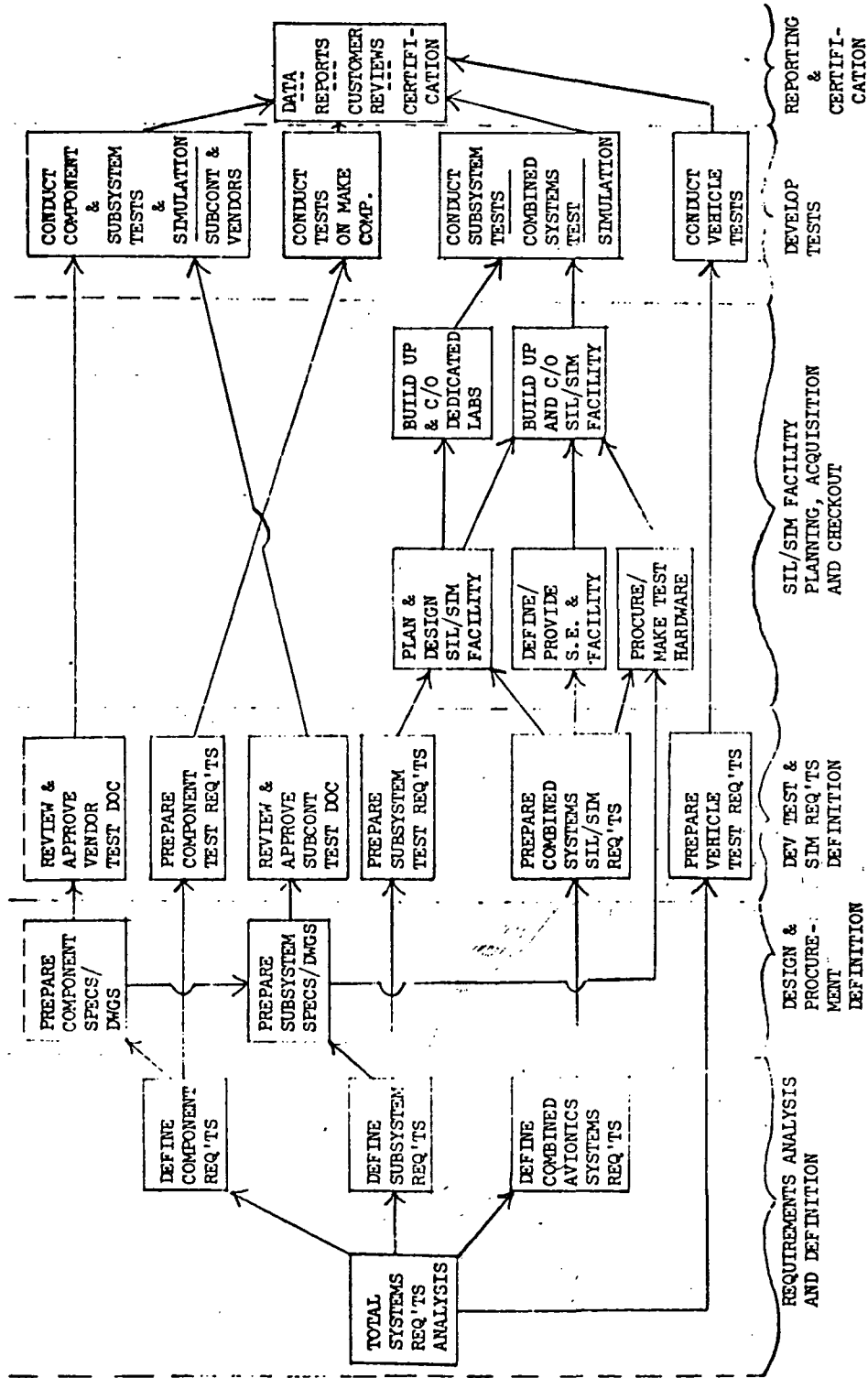


Figure 5.2-1. Avionics Development Logic

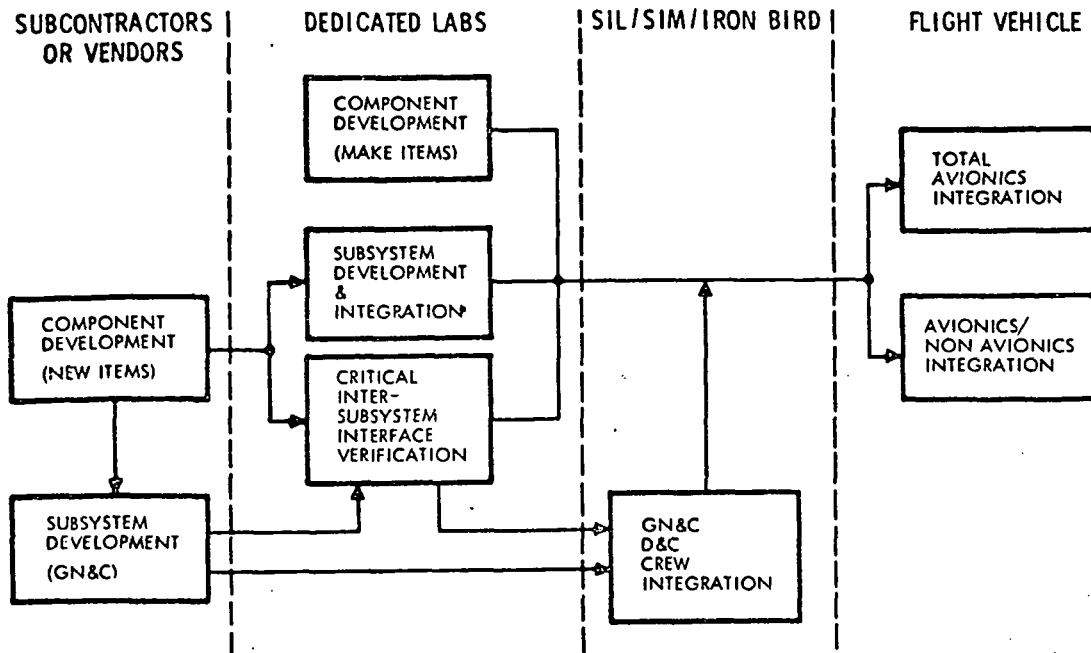


Figure 5.2-2. Development Test Logic

The GN&C and D&C laboratories will be located adjacent to the simulation facility and Iron Bird to permit hardware and support equipment to be shared with the combined systems testing which is defined in the following section.

5.2.3 Combined Systems Testing

Combined systems development, verification, and validation of the GN&C, associated displays and controls, and the crew will be accomplished on the Mission Evaluator, the Hardware Evaluator, and the Iron Bird test beds.

The Mission Evaluator (ME) and Hardware Evaluator (HE) test beds will be combined as shown in Figure 5.2.3-1. The Mission Evaluator will be activated early and will consist of a crew compartment hot mockup containing all GN&C related controls and displays, projected visual scenes, and a hybrid computer complex to simulate vehicle and hardware characteristics. A flight computer and associated test software and flight software will be added to the Mission Evaluator when available.

GN&C sensors and electronics will be added to the test bed when available. A patching arrangement will provide the capability to perform simulations with or without hardware in-the-loop.

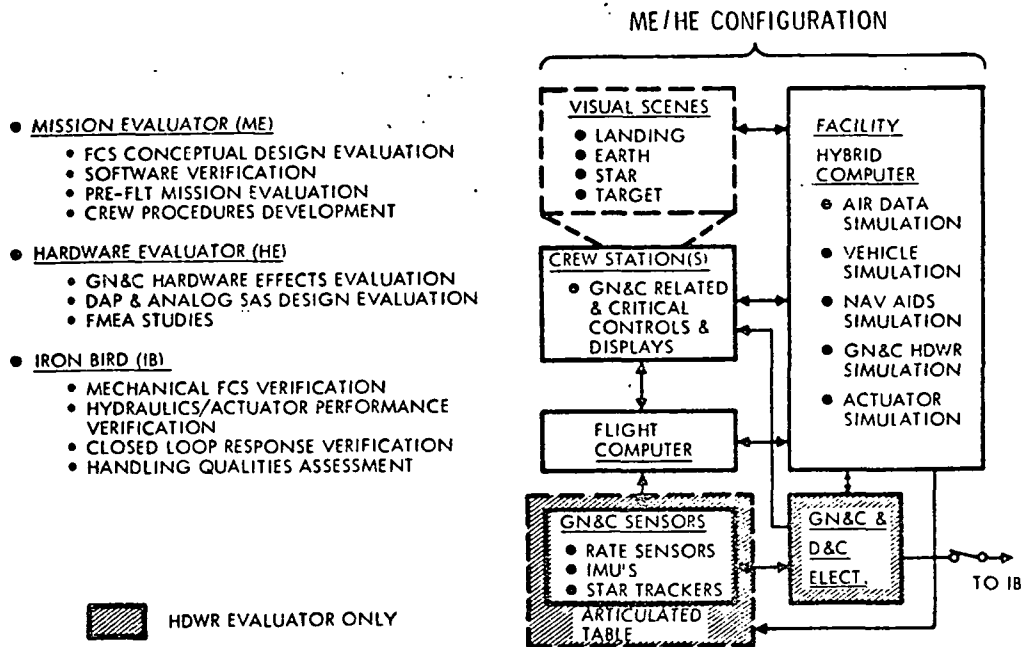
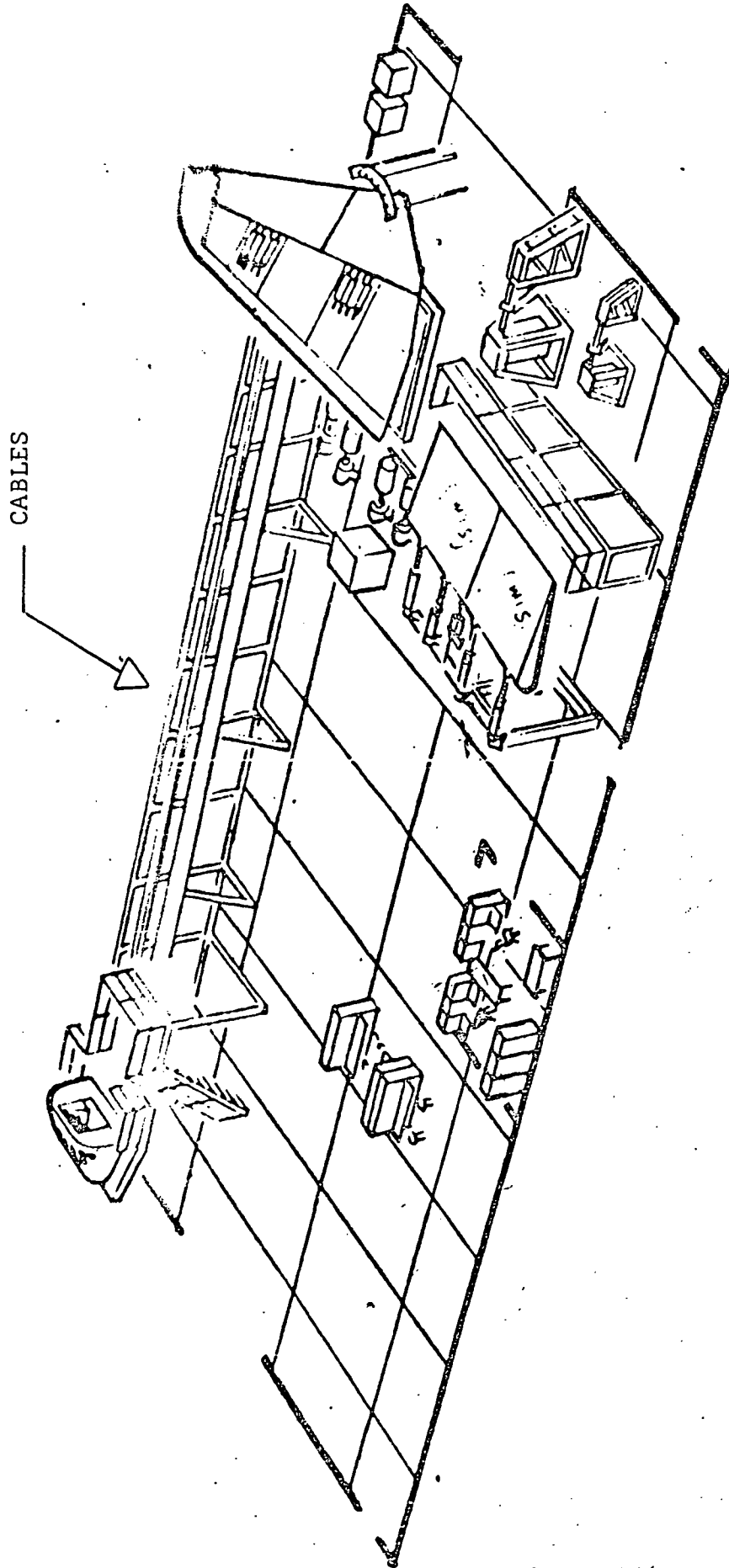


Figure 5.2.3-1. Combined Flight Systems Integration

The ME/HE test bed will be initially configured with the airplane Avionics only (e.g., no computer, IMU, star tracker, etc.). The orbital Avionics will be phased into the test bed with a Kit #1. A hardware tie-in to the Iron Bird test bed which will be located adjacent to the ME/HE test bed will be provided.

Separate Orbiter and Booster ME/HE test beds will be provided at a single facility. The capability to run either an ME or an HE on each test bed simultaneously would be provided. A single facility provides the capability to develop and verify combined Orbiter/Booster separations and aborts with crew participation and hardware in the loop.

Since the Phase B' baseline utilizes a mechanical flight control system for backup, the requirement for an Iron Bird for flight control development and verification is emphasized. The Iron Bird will now require a full scale layout and installation of the mechanical flight controls rigging, as well as the hydraulic subsystems planned in Phase B. A minimal crew station mockup is planned, with landing field scene generation only. Aero surface drivers, stabilization augmentation hardware, and rate gyros would be "portaged" to the Iron Bird to develop and verify the closed loop dynamic characteristic of the combined systems. The hardware tie-in to the ME/HE test bed can be utilized to perform hardware evaluations in fly-by-wire and automatic modes with the hydraulic hardware in the loop. A layout of the proposed Iron Bird is shown in Figure 5.2.3-2.



"IRON BIRD"
HYDRAULIC SIMULATOR LAB
FIGURE 5.2.3.2



The Mission Evaluator will be used for flight control system conceptual design evaluation, software verification, preflight mission evaluation, and crew procedures development. The Hardware Evaluator will be used for hardware effects evaluation, digital autopilot and analog stability augmentation design evaluation, failure mode effects studies, and software validation.

The Iron Bird will be used for mechanical flight control system (FCS) development and verification, hydraulics/actuator performance verification, closed servo loop and SAS loop dynamics verification, and mechanical FCS handling qualities assessment.

5.2.4 Integrated Systems Tests

Total integrated Avionics/Non-Avionics systems verification and validation will be accomplished on the flight test vehicles during both ground and flight tests.

Partial validation of ground checkout equipment and checkout software in the ME/HE test bed was considered but was not felt to be cost-effective. If the ME/HE test bed could be located in the near vicinity of factory checkout equipment, a tie-in could be made and the partial verification accomplished without requiring additional support equipment for the ME/HE.

5.3 SCHEDULE EVALUATION

Several combined vehicle configurations and schedule variations are being carried as alternate options. The concurrent schedule and the manned fly-back LOX RP booster configurations were used for basic schedule evaluation. The effects on schedule of alternate configurations such as the expendable LOX RP booster, the expendable pressure-fed booster, and the recoverable pressure-fed booster were not evaluated in detail.

Figure 5.3-1 shows the concurrent schedule for the two Mark I Orbiters and the first Mark II Orbiter, key PDR and CDR periods, and design activate and operate schedules for the ME/HE and Iron Bird. The test bed schedules are designed to permit over 1-year operation of the HE in the appropriate configuration prior to that respective vehicle starting final checkout.

Figure 5.3-2 assumes a four month delay between prime contractor authority to proceed (ATP) and major subcontractor ATP, and clarifies the spans of time allowed for delivery of engineering model hardware and flight hardware. Figure 5.3-1 also shows the GN&C subcontract development, design, procure, fabrication, and test schedules required to support both engineering and flight hardware deliveries. A similar evaluation was made for other Avionics subsystems.

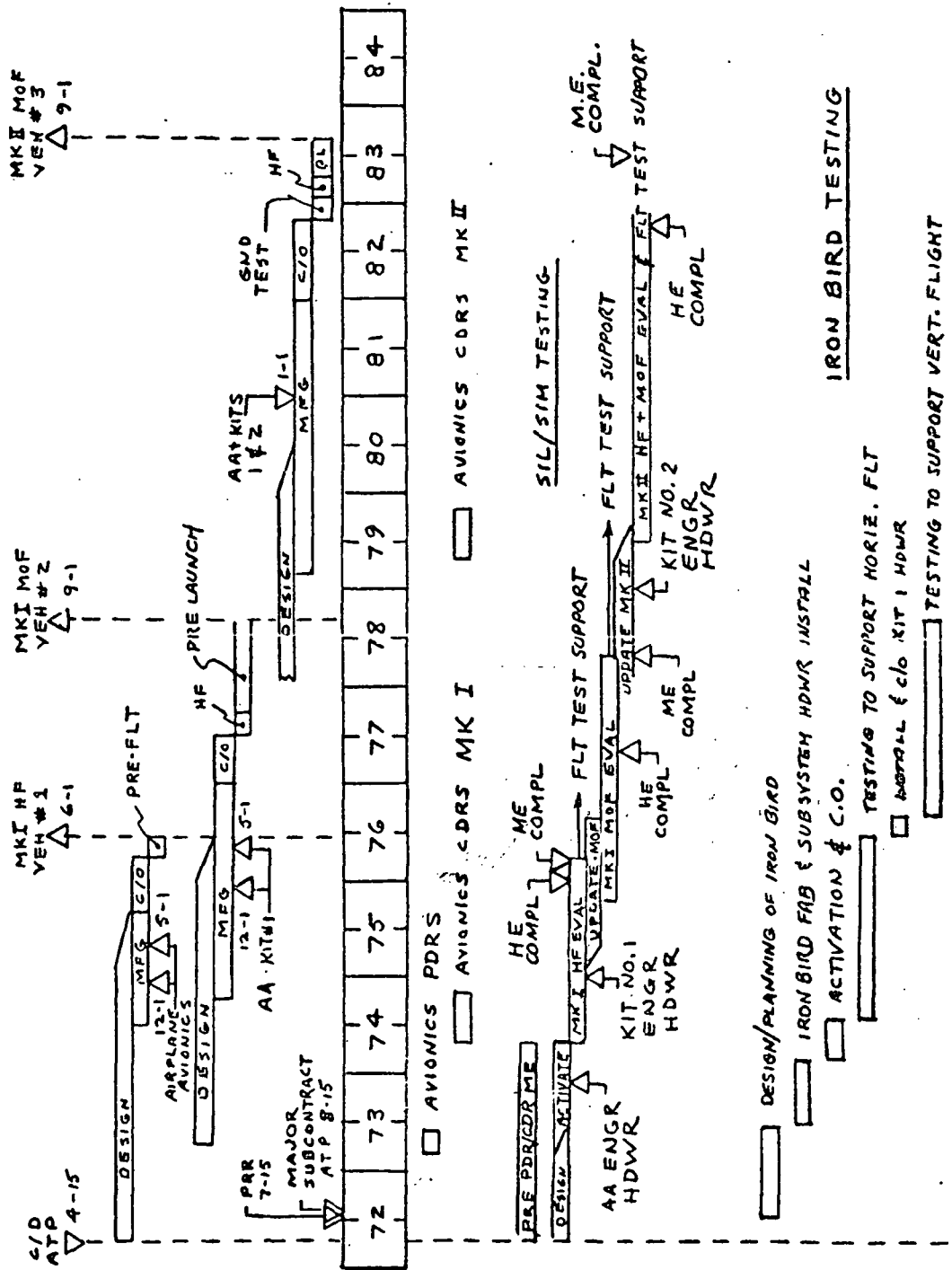


Figure 5.3-1. Combined Systems Test Bed Schedules

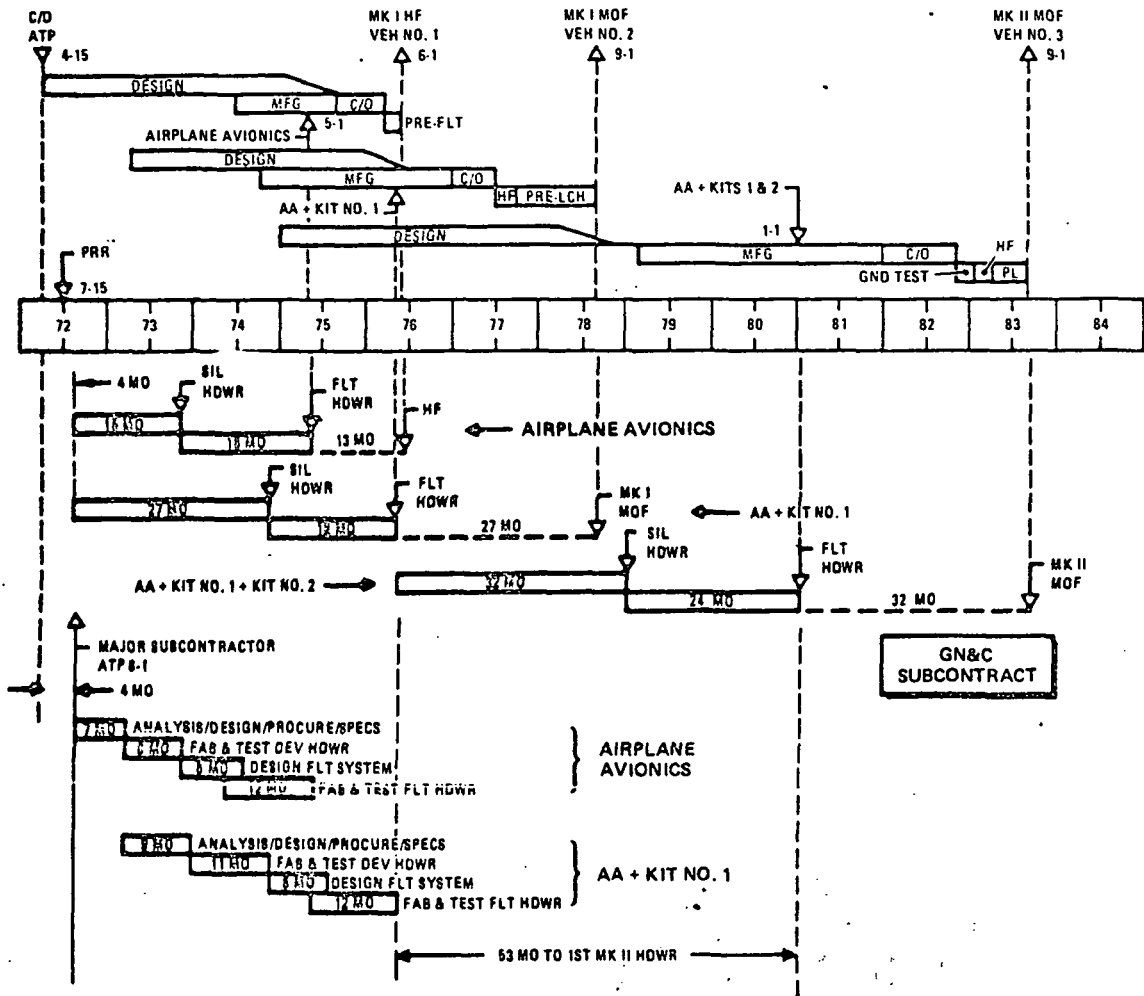


Figure 5.3-2. Avionics Schedule

The following conclusions were drawn from the schedule evaluation:

1. No schedule problems were identified with the development, design, fabrication, test, and delivery of major Avionics subsystems hardware.
2. Serial rather than parallel development, design, and production for first airplane Avionics followed by the orbital Avionics is achievable even when a reasonable waiting period between prime contractor ATP and subcontractor ATP is planned.
3. The long span of time between the second Mark I vehicle flight hardware delivery date and the first Mark II flight hardware delivery date may require that Mark II flight hardware be purchased early and stored. Evaluations of phased procurement vs. early procurement and storage are recommended.



COST EFFECTS

The cost effects of the change to the new baseline avionics as concerns the development testing, can be identified by the following test bed utilization comparisons.

<u>Old Baseline</u> <u>(Phase B)</u>	<u>New Baseline</u> <u>(Phase B')</u>	<u>Estimated</u> <u>Deltas</u>
. Subsystem Labs	. Subsystem Labs	None
. Mission Evaluator	. ME/HE Combined & Expanded	+\$22M
. Hardware Evaluator	.	
. SIL	. Replaced by Expanded ME/HE	-\$76M
. Iron Bird - No Mechanical System	. Iron Bird - With Mechanical System	+\$ 5M
		----- -\$49M



6.0 UNMANNED TEST FLIGHT ANALYSIS

To provide the capability of unmanned flight for the new avionics baseline configuration, additional ground and airborne equipment must be provided as shown in Figure 6-1. Logic functions would be mechanized into the GN&C dedicated computer by software to provide for the redundancy management of all critical systems and functions in response to failure detection signals from the Caution & Warning system. Hardware would be provided in the airborne equipment to perform the actual switching and valving required under the control of the computer. Wherever possible sequential operations will be mechanized into a programmer.

All critical operational and redundancy management functions initiated by the GN&C computer would be backed up through the MSFN command link. To provide this additional on-board capability another Up Data Link (UDL) would be added to both the booster and the orbiter. Finally it would be necessary to add the airborne portion of a drone control system to provide the steering and discrete commands necessary during the approach, landing and roll out portion of the flight.

The additional support equipment needed for unmanned flight would be the ground portion of the drone control system for approach, landing and roll out; mechanization for the proper orbital command functions through MSFN and at least one additional commissioned ship to provide MSFN coverage during the deorbit phase of the mission. The ground control portion of the Command Guidance System used to control the booster and orbiter through approach and roll out would be also installed in 2 or 3 escort aircraft. This would provide control capabilities along with visual observation in case of anomalous operation that would either take the booster and orbiter from line of sight of the Command Guidance System or require collision avoidance corrections or crash site selection.

A destruct system of the proper design to interface with the existing destruct transmitter system at KSC must be provided in both vehicles.

It should be noted here that it has been assumed that the decision to go unmanned is made before the start of Avionics wire harness fabrication to obtain minimum cost and scar weight for the vehicles.

The increased costs between this system and the baseline unmanned system are due to the following:

1. The deletion of the data control management system, complete with data bus and associated interfacing hardware, will require additional hardware and software development. All possible housekeeping and redundancy management control logic will be mechanized by software changes into the



GN&C dedicated computer. Any logic functions in excess of the computer memory capacity will have to be mechanized into the unmanned system hardware. Switching amplifiers, contactor drivers and solenoid valve drivers must be added to provide these type of functions under the control of the computer and/or sequencers.

2. The unmanned system defined in the Phase B baseline proposed a remote control system for use both on the ground and in the escort aircraft as backup to the autoland system. The deletion of the autoland system from the new baseline configuration makes the remote control system primary and more sophisticated.

3. It is anticipated that the horizontal flight test costs would remain constant. Additional flight test hours would be required to prove out the primary remote control landing system and to provide additional training to the ground controller. These should be offset by the deletion of any autoland flight testing as was planned for the Phase B baseline.

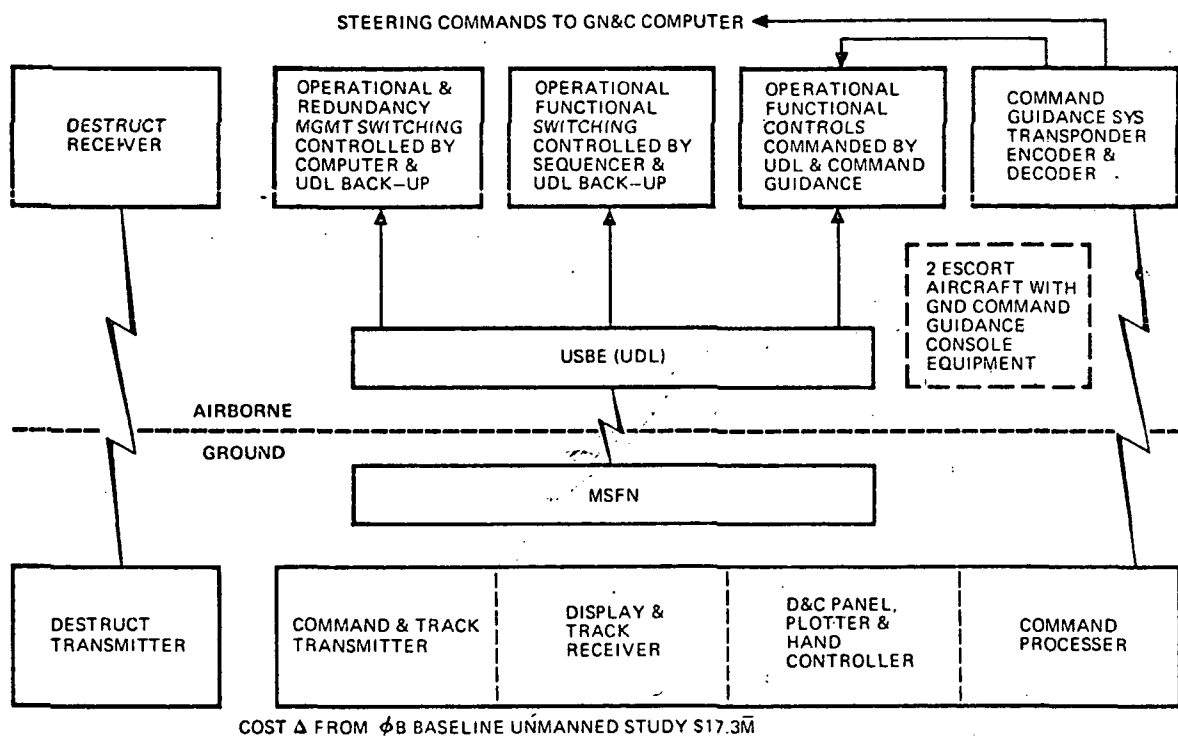


Figure 6-1. Phase B' Unmanned Concept



7.0 COST ANALYSIS

This section reports the cost estimates made for the selected Avionic System, compares them with the costs for the baseline Phase B baseline Avionic System, and provides the backup cost data required by the study directive, Alternate Avionics System Study, Space Shuttle Program, SOW, Exhibit H, dated September 23, 1971. Costs are presented in the formats prescribed in NASA TD No. NR-5, Contract NAS9-10960, October 1, 1971.

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	Power Generation	
	Actuators	
	Unmanned First Vertical Flight	
	GSE	
	Requirements Accomplished by MCC/MSFN	

Avionics cost estimates for the present system and for the Phase B baseline system are shown in the following three summaries, Tables 7.0-1, -2 and -3.

Figure 7.0-1 shows the avionics cost estimate for the orbiter and booster, spread by year. Shared development is shown separately as are individual orbiter and booster development costs.



Table 7.0-1. Summary Total
Avionics Cost Estimate Comparison
(\$Millions)

	360 DAY PHASE B	PHASE B' 10-21-71
ORBITER	676	509
BOOSTER	570	404
TOTAL	1246	913

- AVIONICS COSTS REDUCED BY APPROXIMATELY ONE FOURTH
- EFFECTS ON OTHER SYSTEMS MINOR

Table 7.0-2. Avionics Cost Estimate Comparison (Subsystem)
(\$ Millions)

SUBSYSTEM	φB	φB'	CHANGES
GN&C	155	173	DELETED AUTOMATIC FUNCTIONS UPDATED ANALYSIS ESTIMATE
D&CM	157	67	DATA BUS REMOVED GN&C COMPUTER ONLY
D&C	160	155	REDUCED AUTOMATIC FUNCTIONS ADDED CREW FUNCTIONS
COMMUNICATIONS	111	33	MINIMUM ESSENTIAL ITEMS
INSTRUMENTATION	104	84	MINIMIZED SIGNAL LIST
FLIGHT SOFTWARE	220	120	GN&C FUNCTIONS ONLY
GROUND SOFTWARE	137	137	INCREASE MISSION SUPPORT OFF-SHELF CHECKOUT SOFTWARE
POWER DISTRIBUTION	175	134	CHANGE TO OFF-SHELF ITEMS SHORTER WIRE RUNS
SYSTEMS INTEGRATION LAB	27	12	LESS EQUIPMENT LESS INTEGRATION
TOTAL	1246	913	

Table 7.0-3. Avionics Costs Orbiter - Booster

	360-DAY REPORT						PRESENT SYSTEM									
	DDT&E		PROD		OPNS		TOTAL		DDT&E		PROD		OPNS		TOTAL	
1.3.1 GUIDANCE, NAV, 3.3.1 AND CONTROL	O	86.6	11.2	5.1	102.9	96.6	7.0	3.2	106.8							
	B	49.7	2.3	0.4	52.4	59.3	4.1	0.8	64.2							
	T	136.3	13.5	5.5	155.3	155.9	11.1	4.0	171.0							
1.3.3 DATA CONTROL 3.3.2 AND MANAGEMENT	O	68.3	17.6	7.1	93.0	26.3	8.1	3.8	38.2							
	B	57.0	5.7	1.5	64.3	22.0	5.6	1.0	28.6							
	T	125.3	23.3	8.6	157.3	48.3	13.7	4.8	66.8							
1.3.3 DISPLAYS AND 3.3.3 CONTROLS	O	45.6	17.1	7.9	70.7	49.7	20.2	11.7	81.6							
	B	63.6	22.0	3.8	89.4	51.7	16.2	5.6	73.5							
	T	109.2	39.1	11.7	160.1	101.4	36.4	17.3	155.1							
1.3.4 COMMUNICATIONS 3.3.4	O	41.2	18.1	10.4	69.7	10.6	5.5	5.1	21.2							
	B	27.3	11.6	2.0	40.9	9.3	2.3	0.4	12.0							
	T	68.5	29.7	12.4	110.6	19.9	7.8	5.5	33.2							
1.3.5 INSTRUMENTATION 3.3.6	O	37.2	4.5	2.1	43.9	34.1	5.2	3.5	42.8							
	B	54.8	4.2	0.8	59.9	35.9	4.8	0.8	41.5							
	T	92.0	8.7	2.9	103.8	70.0	10.0	4.3	84.3							
1.3.6 SOFTWARE 3.3.5 (FLIGHT)	O	87.9	-	39.6	127.5	46.5	-	20.9	67.4							
	B	63.9	-	28.8	92.7	36.8	-	15.5	52.3							
	T	151.8	-	68.4	220.2	83.3	-	36.4	119.7							
1.10.10 SUPPORT SOFTWARE	O	47.3	-	21.3	68.6	56.9	-	25.6	82.5							
	B	47.3	-	21.3	68.6	37.8	-	17.0	54.8							
	T	94.6	-	42.6	137.2	94.7	-	42.6	137.3							
1.4.3 POWER 3.4.3 DISTRIBUTION	O	57.3	20.1	7.5	84.9	50.8	7.3	4.8	62.9							
	B	72.9	11.8	5.3	90.0	50.8	15.8	4.9	71.5							
	T	130.2	31.9	12.8	174.9	101.6	23.1	9.7	134.4							
TOTAL ORBITER		471.4	88.7	101.0	661.1	371.5	53.3	78.6	503.4							
TOTAL BOOSTER		436.6	57.6	64.0	558.1	303.6	48.8	46.0	398.4							
TOTAL		908.0	146.3	165.0	1219.2	675.1	102.1	124.6	901.8							
SYS INTEGRATION & LAB TOTAL		27.0			27.0	12.0			12.0							
		935.0			1246.0	687.0			913.0							



ANNUAL COST
(MILLIONS \$)

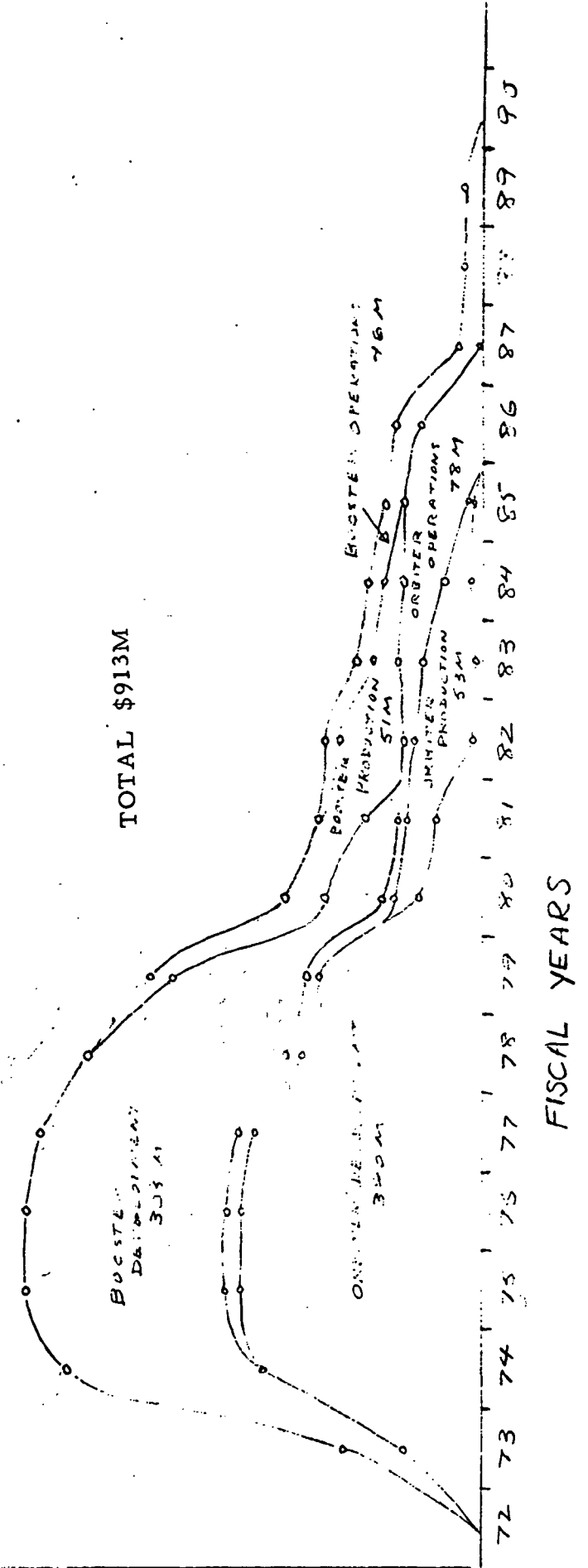
150

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SD 71-346

Figure 7.0-1. Shuttle Avionics Cost Orbiter & Booster
DIT&E + Production + Operations





7.1 APPROACH AND METHODOLOGY.

Estimates were prepared by the same procedures and according to the same rules in Phase B and Phase B-Extension, as nearly as possible.

Costs are reported in the three formats which were prescribed in TD number NR-5, Contract NAS9-10960, October 1, 1971. The first format is included as Table 7.1-1. The second and third are identified as Work Sheets No. 1 and 2, Tables 7.1-2 through -13. Work Sheet No. 1 was revised to delete MTBF data (reference NR letter 71MA6151 from B. Hello to NASA/MSC attention H. E. Gartrell, dated November 5, 1971). Work Sheet No. 2 with no revision, is filled out by subsystem.

Cost methodology and cost model structure remain the same as described in detail in the Phase B reports, with following change: A growth factor is now applied to equipment items which require more than 10% design change, and is not applied to those with less than 10% design change. Previously, growth factors were applied to all items.

Details of the subsystem cost buildups were provided to MSC in a splinter conference on November 1, 1971, and are held by MSC Operations Analysis, phone 2741. Therefore these details are not included in the present report.



Table 7.1-1. Subsystem Costs
Orbiter - Booster

SHUTTLE AVIONICS COSTS (MILLIONS)

Subsystem WBS Element	S	Units	Cost Per Unit TH	TH	Units	Fth	DOME	TFU	Units	Prod	Units	Refurb	Prod	Units	Refurb	Units	OP3	Spare	Total
JMC																			
Contractor	38.316		.662	1.059		1.390	40.765	.662				.288	2.284				1.053		44.102
Orbiter Subcontractor	50.334		1.182	2.179	2.1	2.866	55.973	1.362	3.0			.613	2.699				2.166		62.738
Total	88.650		2.034	3.238		4.256	96.739	2.024				.911	4.983				3.219		106.840
Contractor	10.722		.396	.694		.792	12.208	.796				.277	1.069				.198		13.475
Orbiter Subcontractor	47.387		1.133	1.983	2.0	2.266	47.136	1.133	2.0			1.070	3.059				.866		50.761
Total	58.109		1.529	2.677		3.058	59.344	1.929				1.347	4.118				.764		64.236
Contractor	49.038			1.753		2.182	52.973					.575	3.353				1.251		57.977
Orbiter Subcontractor	93.721		4.162	4.162		5.126	103.009					1.406	7.758				2.783		113.499
Total	142.759		5.915	5.915		7.308	155.982					1.981	11.111				4.034		171.476
DCN																			
Contractor	8.017		.643	.579		1.350	9.946	.643				.450	2.379				1.125		13.450
Orbiter Subcontractor	11.721		1.541	1.387	2.1	3.236	16.364	1.541	3.0			1.079	5.702				2.697		24.763
Total	19.738		2.184	1.966		4.586	26.310	2.184				1.529	8.081				3.822		38.213
Contractor	3.697		.536	.938		1.072	5.707	.536				.375	1.447				.268		7.422
Orbiter Subcontractor	10.563		1.533	2.683	2.0	3.066	16.312	1.533	2.0			1.073	4.139				.767		21.218
Total	14.260		2.069	3.621		4.138	22.019	2.069				1.448	5.986				1.035		28.640
Contractor	11.714			1.517		2.422	15.653					.835	3.826				1.393		20.872
Orbiter Subcontractor	22.304		4.070	4.070		6.302	32.676					2.152	9.821				3.464		45.981
Total	34.018		5.887	5.887		8.724	48.329					2.977	13.667				4.857		66.853
DCB																			
Contractor	21.974		2.967	2.670		6.231	30.875	2.967				1.335	10.236				5.934		47.045
Orbiter Subcontractor	10.164		2.892	2.603	2.1	6.073	18.840	2.892	3.0			1.301	9.977				5.784		34.601
Total	32.138		5.859	5.273		12.304	49.715	5.859				2.636	20.213				11.718		81.646
Contractor	6.192		.939	1.642		1.876	9.620	.939				.835	2.711				.938		13.269
Orbiter Subcontractor	24.494		4.687	8.302	2.0	9.374	42.070	4.687	2.0			4.171	13.545				4.687		60.302
Total	30.686		5.625	9.944		11.250	51.690	5.625				5.006	16.256				5.625		73.571
Contractor	28.076			4.312		8.107	40.495					2.170	12.947				6.872		60.314
Orbiter Subcontractor	34.058		10.805	10.805		15.447	60.910					5.472	23.522				10.471		94.903
Total	62.134		15.117	15.117		23.554	101.405					7.642	36.469				17.343		155.217
COM																			
Contractor	7.66		.997	1.211		1.254	3.333	.997				.439	1.617				1.698		6.448
Orbiter Subcontractor	1.121		1.474	3.155	2.1	3.011	7.287	1.474	2.0			1.068	3.680				3.392		10.433
Total	8.781		2.471	4.366		4.265	10.620	2.471				1.507	5.297				5.090		21.307
Contractor	4.612		.331	.579		.662	5.853	.331				.295	.997				.165		6.975
Orbiter Subcontractor	1.671		.481	.942	2.0	3.475	4.81	.481	2.0			.428	1.340				.241		5.106
Total	6.283		.812	1.521		1.624	9.328	.812				.723	2.337				.406		12.081
Contractor	5.378			1.892		1.916	9.186					.714	2.374				1.663		13.423
Orbiter Subcontractor	2.792		3.997	3.997		3.973	10.762					1.434	5.270				3.933		15.450
Total	8.170		5.889	5.889		5.889	19.948					2.148	7.544				5.596		28.873
INSTRUMENTATION																			
Contractor	21.093		.741	1.771		2.371	25.105	.741				.199	1.322				1.041		27.668
Orbiter Subcontractor	1.321		1.777	3.218	3.2	5.686	8.954	1.777	3.0			.475	3.643				2.672		13.089
Total	22.414		2.518	4.989		8.057	34.059	2.518				.674	5.165				3.713		40.757
Contractor	7.661		1.308	1.962		2.616	12.229	.953				.314	1.020				.177		13.436
Orbiter Subcontractor	8.183		4.216	6.524	2.0	8.832	23.639	4.216	2.0			1.360	3.442				.995		27.616
Total	15.844		5.524	8.486		11.448	35.868	5.169				1.674	4.462				.772		41.112
Contractor	29.054			2.703		4.987	37.344					.513	2.542				1.218		41.104
Orbiter Subcontractor	8.401		8.074	8.074		14.513	32.993					1.535	7.085				3.087		42.765
Total	37.455		16.148	16.148		29.057	70.337					2.048	9.627				4.255		84.869



Table 7.1-1. Subsystem Costs
Orbiter - Booster (Continued)

SHUTTLE AVIONICS COSTS (MILLIONS)

Subsystem MS (VEN)	DCE \$	Units GTH	Cost Per Unit \$	GTH	Units GTH	RIR	30% \$	Units Prod	Prod	Units ReCur	ReCur \$	Prof ReCur	Units Spare	Spare	Total	
																30% \$
SOFTWARE (VEN)																
Contractor	16.033	0	0	0	0	0	16.033	0	0	0	0	0	0	0	7.215	23.248
Orbiter Subcontractor	30.512	0	0	0	0	0	30.512	0	0	0	0	0	0	0	13.730	44.242
Total	46.545	0	0	0	0	0	46.545	0	0	0	0	0	0	0	20.945	67.490
Contractor	8.901	0	0	0	0	0	8.901	0	0	0	0	0	0	0	3.738	12.639
Orbiter Subcontractor	27.946	0	0	0	0	0	27.946	0	0	0	0	0	0	0	11.737	39.683
Total	36.847	0	0	0	0	0	36.847	0	0	0	0	0	0	0	15.475	52.322
Contractor	24.934	0	0	0	0	0	24.934	0	0	0	0	0	0	0	10.953	35.887
Orbiter Subcontractor	58.458	0	0	0	0	0	58.458	0	0	0	0	0	0	0	25.467	83.925
Total	83.392	0	0	0	0	0	83.392	0	0	0	0	0	0	0	36.420	119.812
SOFTWARE (GND)																
Contractor	4.200	0	0	0	0	0	4.200	0	0	0	0	0	0	0	1.900	6.100
Orbiter Subcontractor	52.700	0	0	0	0	0	52.700	0	0	0	0	0	0	0	23.700	76.400
Total	56.900	0	0	0	0	0	56.900	0	0	0	0	0	0	0	25.600	82.500
Contractor	1.700	0	0	0	0	0	1.700	0	0	0	0	0	0	0	.800	2.500
Orbiter Subcontractor	36.100	0	0	0	0	0	36.100	0	0	0	0	0	0	0	16.200	52.300
Total	37.800	0	0	0	0	0	37.800	0	0	0	0	0	0	0	17.000	54.800
Contractor	5.900	0	0	0	0	0	5.900	0	0	0	0	0	0	0	2.700	8.600
Orbiter Subcontractor	88.800	0	0	0	0	0	88.800	0	0	0	0	0	0	0	39.900	128.700
Total	94.700	0	0	0	0	0	94.700	0	0	0	0	0	0	0	42.600	137.300
POWER DISTRIBUTION																
Contractor	27.580	0.8	4.340	3.432	13.888	44.940	2.065	4.130	5.452	1.322	4.41	1.206	1.75	3.614	54.006	
Orbiter Subcontractor	1.674	0.8	1.042	8.814	3.374	5.842	.689	1.378	0.64	1.763	7.271	4.820	1.75	8.867	62.873	
Total	29.254	0.8	5.382	4.306	17.222	50.782	2.754	5.508	6.092	3.085	11.683	6.026	3.50	12.481	116.879	
Contractor	2.876	1.5	1.095	1.643	6.709	6.709	1.095	2.190	3.164	.974	3.164	3.896	0.89	.974	10.847	
Orbiter Subcontractor	28.754	1.5	4.377	6.565	8.754	44.073	4.377	2.0	0.89	4.870	15.814	4.870	0.89	3.896	60.619	
Total	31.630	1.5	5.472	8.208	10.944	50.782	5.472	10.944	4.053	8.844	19.628	8.766	0.89	4.870	71.466	
Contractor	30.456	0	0	0	0	0	30.456	0	0	0	0	0	0	0	4.588	64.853
Orbiter Subcontractor	30.458	0	0	0	0	0	30.458	0	0	0	0	0	0	0	5.102	69.486
Total	60.914	0	0	0	0	0	60.914	0	0	0	0	0	0	0	9.690	134.339
SERVOAL WITHCUT GND SOFTWARE																
Contractor	134.679	0	9.834	26.484	170.997	170.997	9.834	19.447	4.023	23.490	4.023	23.490	21.480	21.480	215.947	
Orbiter Subcontractor	107.517	0	11.935	24.200	143.672	143.672	11.935	24.805	4.915	29.720	4.915	29.720	21.667	21.667	205.059	
Total	242.196	0	21.769	50.684	314.669	314.669	21.769	44.252	8.938	53.210	8.938	53.210	43.147	43.147	421.006	
Contractor	44.571	0	7.458	9.208	61.237	61.237	7.458	7.298	3.070	10.368	3.070	10.368	6.458	6.458	78.063	
Orbiter Subcontractor	144.498	0	26.899	33.254	204.651	204.651	26.899	36.804	11.421	38.225	11.421	38.225	22.489	22.489	265.345	
Total	189.069	0	34.357	42.462	265.888	265.888	34.357	44.102	14.491	48.693	14.491	48.693	28.947	28.947	343.428	
Contractor	179.250	0	17.292	35.692	232.234	232.234	17.292	26.765	7.093	33.858	7.093	33.858	27.938	27.938	294.030	
Orbiter Subcontractor	252.035	0	38.834	57.154	348.323	348.323	38.834	51.609	16.336	67.945	16.336	67.945	54.156	54.156	470.422	
Total	431.285	0	56.126	92.846	580.557	580.557	56.126	78.374	23.429	101.803	23.429	101.803	82.094	82.094	764.452	
Contractor	138.879	0	9.834	26.484	175.197	175.197	9.834	19.447	4.023	23.490	4.023	23.490	21.480	21.480	222.067	
Orbiter Subcontractor	160.237	0	11.935	24.200	196.372	196.372	11.935	24.805	4.915	29.720	4.915	29.720	21.667	21.667	281.459	
Total	299.116	0	21.769	50.684	371.569	371.569	21.769	44.252	8.938	53.210	8.938	53.210	43.147	43.147	503.526	
Contractor	46.271	0	7.458	9.208	62.937	62.937	7.458	7.298	3.070	10.368	3.070	10.368	6.458	6.458	80.563	
Orbiter Subcontractor	180.598	0	26.899	33.254	240.751	240.751	26.899	36.804	11.421	38.225	11.421	38.225	22.489	22.489	317.665	
Total	226.869	0	34.357	42.462	303.688	303.688	34.357	44.102	14.491	48.693	14.491	48.693	28.947	28.947	398.228	
Contractor	185.150	0	17.292	35.692	238.134	238.134	17.292	26.765	7.093	33.858	7.093	33.858	30.638	30.638	302.630	
Orbiter Subcontractor	340.835	0	38.834	57.154	437.123	437.123	38.834	51.609	16.336	67.945	16.336	67.945	54.056	54.056	599.124	
Total	525.985	0	56.126	92.846	675.257	675.257	56.126	78.374	23.429	101.803	23.429	101.803	84.694	84.694	901.754	

Table 7.1-2. Guidance, Navigation and Control

BRIEF EQUIPMENT DESCRIPTION	UNIT WEIGHT (LB)	POWER (WATTS)	UNIT VOLUME (FT ³)	UNITS/SHIPSET		DATA SOURCE	COSTS (\$ MILLION)	
				O	B		D&D	UNIT
IMU Platform	25.0	210	.45	3	3 *	Kearfott KT-70 IMU Group	.229	.071
IMU Power Supply	22.0	150	.35	3	3 *	Kearfott KT-70 IMU Group		
Star Tracker	10.0	17/45	.18	3	-	ITT Aerobee	.556	.061
Horizon Sensor	22.0	66	.20	3	-	Barnes Horizon Sensor Group	.126	.126
Stability Augmentation System	13.0	87/126	.29	3	3	HI FL4 Flt. Control Group	4.890	.018
Air Data Package	25.0	266	.52	3	3 *	HI HG280 DC-10 ADP Group	2.220	.040
Rate Sensors (Roll, Pitch, Yaw)	3.5	12	.04	3	3 *	HI FL4 Gyro Group	.290	.0055
Accelerometer Pkg. (Normal Lat.)	1.0	2	.02	2	2 *	HI FL4 Accelerometer Group	.055	.006
Angle of Attack Transducer	10.0	30	.15	3	3		.050	.001
ACPS Logic/Drivers	19.6	3/4	.48	2	12	HI Shuttle/Apollo Group	.371	.020
TVC Driver	5.0	29/42	.10	3	-	HI FL4 Flt Control Group		
TVC/Aero Surface Driver	12.0		.21	-	4	HI FL4 Flt Control Group	1.140	.020
Aero Surface Driver	6.5		.18	-	1	HI FL4 Flt Control Group	Incl. in Stab. Aug. System	
ABES Throttle Driver	2.0		.14	-	10	HI FL4 Flt Control Group	.671	.022
Power Supply Aero	20.0		TBD	-	1		.050	.020
TVC Control, Manual	5.0	10	TBD	1	1 *		.100	.010
Speed Brake Driver	6.4	10/5	.10	1	-			

*Denotes Common Equipment

Table 7.1-3. Data Control Management

BRIEF EQUIPMENT DESCRIPTION	UNIT WEIGHT (LB)	POWER (WATTS)	UNIT VOLUME (FT ³)	UNITS/SHIPSET		DATA SOURCE	COSTS (\$ MILLION)	
				O	B		D&D	UNIT
Computer (w/Aux Mem.) 48K	50	1020	.96	3	3 *	IBM AP-1 F15 Computer	1.128	.187
Buffer	24	300	.30	3	3 *	4K Computer Components Plus Skylab, Power Supplies	1.366	.058
Decoder	27	240	.30	3	3 *	4K Computer Components Plus Skylab, Power Supplies	1.444	.040
Multiplexer	10	60	.08	3	3 *		.250	.020

*Denotes Common Equipment

Table 7.1-4. Displays and Controls

WORK SHEET NO. 1
1 NOVEMBER 1971

BRIEF EQUIPMENT DESCRIPTION	UNIT WEIGHT (LB)	POWER (WATTS)	UNIT VOLUME (FT ³)	UNITS/SHIPSET		DATA SOURCE	COSTS (\$ MILLION)	
				O	B		D&D	UNIT
Primary Flight Display Set								
3-Axis Attitude Dir. Ind.	10	36	.37	3	2 *	HI Mod. of Apollo CM FDI	.840	.1133
Horizontal Situation Ind.	8	28	.13	2	2 *	Bendix E-C, ACA AQU-4A		
Ind. Airspeed/Mach Ind.	3	24	.06	2	2 *	Bendix E-C, ASK-14/A24G-18		
Altitude Vert Speed Ind.	4	32	.05	2	2 *	Bendix E-C, AAK-23/A24G-17A	.780	.090
Cross Pointer Ind.	6	30	.10	2	-	Bendix E-C, Mod. Apollo IM Unit		
Display Coupler Driver Unit	18	150	.44	3	2 *	Prime Contractor Make	.360	.110
Backup Flight Display Set								
2 Axis Attitude Dir. Ind.	4	22	.10	1	1 *	Lear Siegler MS27150-3		
Bearing Dist. Heading Ind.	3.5	22	.04	1	1 *	Kollsman Motor Corp.		
Barometer Altimeter Ind.	1.5	3	.02	1	1 *	Aerosonics, AAU-16/A		
Airspeed Indicator	1.5	3	.02	1	1 *	Aerosonics, AVU-19/A		
Vertical Speed Ind.	1.5	3	.02	1	1 *	Aerosonics, AAU-18/A		
2 Axis G Meter	2	5	.03	1	1 *	Cedar Engrg., TRU-3/A		
Vertical/Directional Gyro Pkg.	10.5	225	.21	3	3 *	CE, Model 8KD9AR3 w/Amp.	.240	.037
Rotational Hand Controller	6	--	.31	3	2	Honeywell, Apollo Type	.240	.037
Translation Hand Controller	5.5	12	.10	2	-	Honeywell, Apollo Type	.180	.025

*Denotes Common Equipment

Table 7.1.4. Displays and Controls (Continued)

WORK SHEET NO. 1
1 NOVEMBER 1971

BRIEF EQUIPMENT DESCRIPTION	UNIT WEIGHT (LB)	POWER (WATTS)	UNIT VOLUME (FT ³)	UNITS/SHIPSET		DATA SOURCE	COSTS (\$ MILLION)	
				O	B		D&D	UNIT
Throttle/Speed Brake Quadrant	9	6	.62	1	1	GE, B-1 Type	.360	.060
Control Encoder/Coupler Unit	12	225	.37	3	2 *	Prime Contractor Make	incl.	incl.
CRT Display Unit	20	80	.71	4	2 *	IBM, FL4 Type	.560	.085
Computer Entry Key-board	7	30	.15	3	2 *	IBM, Apollo Type DSKY	.210	.027
Caution/Warning Unit	28	30	.22	1	2 *	Autonetics, Apollo Type	2.660	.280
Event Timer	1	22.5	.01	1	2 *	Apollo Type	.120	.023
Mission Timer	1.5	10	.02	2	2 *	Apollo Type	.180	.060
Subsystem Display Processor	11	40	.25	1	1 *	IBM, Mod. SP-1A Processor	.680	.090
Pedal Control Assembly	50	24	TBD	2	2	Honeywell	.032	.016
Control Stick Assembly	20	24	TBD	2	-	Honeywell	inc in hand controller	
Panels	20	50	TBD	19	7	Prime Contractor Make	3.240	.068
Interior Lighting	84	565	TBD	SET	SET	Apollo & Std. Aircraft	.380	.120
Exterior Lighting	123	3299	TBD	SET	SET			
Standby Magnetic	1	-	.01		1	Standard Aircraft		

*Denotes Common Equipment





WORK SHEET NO. 1
1 NOVEMBER 1971

Table 7.1-5. Communications

BRIEF EQUIPMENT DESCRIPTION	UNIT WEIGHT (LB)	POWER (WATTS)	UNIT VOLUME (FT ³)	UNITS/SHIPSET		DATA SOURCE	COSTS (\$ MILLION)	
				O	B		D&D	UNIT
Antennas								
L-Band	3.5	-	.024	4	2 *	} AMECOM Apollo/Saturn and Other	.070	.030
UHF	5.0	-	.39	4	2 *		.070	.035
S-Band	2.5	-	.01	8	2		.070	.009
ILS						Standard Aircraft	-	.010
Antenna Selector						AMECOM		
L-Band	2.5	5	.02	3	2 *	} AMECOM Apollo/Saturn and Other	.175	.0025
UHF	2.5	5	.02	2	2 *			.0025
S-Band	2.5	5	.02	2	2 *			.0025
UHF/AM Transceiver	30.0	300	.63	2	2 *	Collins Radio	.050	.015
ATC Transponder	14.0	41	.63	2	-	Motorola	-	.100
S-Band Transponder	24.0	28	.69	1	-	Motorola	-	.100
Signal Processor	14.5	13	.20	2	2 *	Collins Radio	.125	.075
Audio Center	8.0	20	.14	2	2 *	Collins Radio	.110	.0425
TACAN	25.0	360	.45	3	2 *	Collins Radio	.050	.025
Up Data Link	21.0	12	.60	1	2	Motorola	.100	.015
ILS Receiver						Standard Aircraft	-	.015

*Denotes Common Equipment



WORK SHEET NO. 1
1 NOVEMBER 1971

Table 7.1-6. Instrumentation

BRIEF EQUIPMENT DESCRIPTION	UNIT WEIGHT (LB)	POWER (WATTS)	UNIT VOLUME (FT ³)	UNITS/SHIPSET		DATA SOURCE	COSTS (\$ MILLION)	
				0	B		D&D	UNIT
Operational Transducers	.15	-	-	1000	1000	Various	.075	.0003
PCM Remote Unit	6	300	.14	6	4 *	SCI, Teledyne	.220	.040
PCM Master Unit	20	60	.52	2	1 *	SCI, Teledyne	.100	.045
Signal Conditioning Unit	15	210	.22	6	4	Sat/Apollo Autonetics SCE	.050	.015
Maintenance Recorder	50	60	1.85	1	1 *	Sundstrand, Echo Science, Davoll	.010	.025
Loop Recorder	20	40	.29	1	- *	Sundstrand, Echo Science, Davoll	.050	.025
Flight Log Recorder	50	60	1.85	1	1 *	Sundstrand, Echo Science, Davoll	.050	.025
Timing Unit	20	30	.41	1	1 *	Apollo CTE, General Time	.300	.035
DFI Transducers	.15	-	-	3000	2000	Various	.150	.0003
PCM Remote Unit	6	600	.14	12	8 *	SCI, Teledyne	.220	.040
PCM Master Unit	20	60	.52	2	1 *	SCI, Teledyne	.100	.045
PCM Recorder	50	60	1.85	1	1 *	Sundstrand, Echo Science, Davoll	.500	.025
Signal Conditioner	15	595	.28	17	16	Saturn/Apollo Autonetics SCE	.100	.015
Wide Band FM Unit	15	100	.28	2	2 *	Saturn/Apollo	.050	.025
Wide Band Recorder	50	60	1.85	2	1 *	Kinelogic, Ampex, CEC, Motorola	.050	.025

*Derotes Common Equipment

Table 7.1-7. Electrical Power Distribution & Control

WORK SHEET NO. 1
1 NOVEMBER 1971

BRIEF EQUIPMENT DESCRIPTION	UNIT WEIGHT (LB)	POWER (WATTS)	UNIT VOLUME (FT ³)	UNITS/SHIPSET		DATA SOURCE	COSTS (\$ MILLION)	
				O	B		F&D	UNIT
Inverter, 3 Phase 1.25 KVA Sequencer	53.0	Load Dep.	.93	4	-	Apollo Type (Westinghouse)	.105	.0044
Generator Control Unit (on APU)	8.0	30	.21	4	4	New Design, Standard Relays	.233	.022
Generator Current Transformer	3.5	350	.04	3	3 *	Standard (Bendix or Westinghouse)	.221	.0033
Battery Charger	2.3	-	.02	3	3 *	Standard (Bendix or Westinghouse)	.079	.0022
Transformer Rectifier	3.5	96	.08	1	-	Custom Design (N. Cad)	.166	.0044
Relay Box	16.0	Load Dep.	.20	3	6 *	Standard Aircraft (Bendix, Westinghouse)	.271	.00176
Relay Box	10.0	40	.45	2	2	Custom Design, Std. Relays	.140	.0097
Local Power Control Box	10.0	40	.45	2	2	Custom Design, Std. Relays	.140	.0107
Local Power Control Box	18.0	60	.84	3	3	Standard Aerospace Type	.205	.020
Local Power Control Box	18.0	40	.84	2	2	Standard Aerospace Type	.192	.020
DC Distribution Box	56.4	60	2.67	3	3	Standard Aerospace Type	.406	.0667
Main AC Dist. Box Aft	45.4	60	2.08	3	3	Standard Aerospace Type	.406	.0674
Inv. Distribution Box	17.7	60	.89	3	-	Standard Aerospace Type	.242	.0355
AC Distr. Box, Avionics	14.4	30	.74	3	3	Standard Aerospace Type	.075	.0497
Drop Tank/Booster Umb.	10.0	-	-	1	1	Standard Space Program	.085	.0087
Space Station Umb.	11.0	-	-	1	-	Likely to be new design	.232	.0093
GSE Umb.	18.0	-	-	1	1	Standard Space Program	.049	.0133
Overcurrent Trans- former	.3	27	.005	-	9	Standard (Bendix, Westinghouse)	.050	.002

*Denotes Common Equipment



Table 7.1-7. Electrical Power Distribution & Control (Continued)
 WORK SHEET NO. 1
 1 NOVEMBER 1971

BRIEF EQUIPMENT DESCRIPTION	UNIT WEIGHT (LB)	POWER (WATTS)	UNIT VOLUME (FT ³)	UNITS/SHIPSET		DATA SOURCE	COSTS (\$ MILLION)		
				O	B		D&D	UNIT	
DC Distr. Panel - Main	20.0	40	1.74	-	6	Standard Aerospace Type	.500	.0345	
Ext. Power Trans. Bus Panel	5.0	60	.31	-	1	Standard Aerospace Type	.060	.002	
Ext. Power Receptical Wiring and Harness	2.0		.04	-	1	Standard Space Program	.050	.003	
Wiring and Harnesses				1	1	(1)	21.900	{ 4.232 w/o 1.621 w/o	
EH Test Procedures & Support				1			19.450	2.483	
EH Test Procedures & Support					1		.975	-	
							1.005	-	
(1) includes 6.443 Mfg. & Tooling									

*Denotes Common Equipment

EQUIPMENT LIST MODIFICATION NARRATIVE (if significant Mod. to off-shelf)

<p>B 1 2 3 4 5 6 7 8 9 10 11 12 13 14 15 16 17 18 19</p>	<p>GN&C Rate Sensor-Pitch Rate Sensor-Roll Rate Sensor-Yaw IMU Platform IMU Power Supply Air Data Package APS Logic/Drivers Thrust Control Electronics Aero Control Electronics - Pitch (Triple redundant) Aero Control Electronics - Roll (Triple redundant) Aero Control Electronics - Yaw (Triple redundant) Accelerometer Pkg-Y Axis Accelerometer Pkg-Z Axis Computer Computer I/O Star Tracker Star Tracker Encoder Coupler Star Tracker Shutter Assy. Horizon Sensor</p>	<p>Revise scaling of HI GG1027 FL4 Yaw Rate Gyros (triple redundant) Modify scaling of GG1027 FL4 Yaw Rate Gyros Modify scaling of GG1027 FL4 Yaw Rate Gyros ----- Analysis to determine how to adapt as is the KT70 Kearfott Platform (or equivalent) which appears to be lowest cost unit which meets requirements. Use HI HG280 DADS with new memory card. Add angle of attack trans- ducers and electronics. New design based upon Apollo/MOL building block. Use HI MIL std shop practices. ----- Based on HI FL4 and C5A building blocks. Analysis to determine how best to adapt these existing circuits. Use HI MIL-std shop practices. ----- Revise scaling of GG1026 FL4 Accelerometer package. Modify mounting and scaling of FL4 Y Axis accelerometer package. Analysis to determine how to adapt as is HI HG600 or equivalent digital computer and memory with minor interface changes. New design which MUX's and adapts sensors, nav aids, control and display interfaces to computer Use ITT Aerobee Tracker using magnetic instead of electrostatic focusing (accuracy). Pressurize or bleed down. New design of level detector/switch and conditioning circuits. New design based upon SPARS shutter. Requires optical analysis for sun sensor incorporation. Analysis to determine how to adapt as in Barnes 13-156 sensor or equivalent.</p>
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Table 7.1-8. GN&C Subsystem (Continued)

WORK SHEET NO. 2

EQUIPMENT LIST	MODIFICATION NARRATIVE (if significant Mod. to off-shelf)
Requirements Analysis	Based upon Δ to Phase B estimating rationale. Δ includes reduction by 1/3 in F/C, 1/4 in guid., and 1/8 in nav. for deletion of landing and ferry algorithms. Simplified D&C interface reduces human factor by 1/2.
System Integration	Based upon Δ to Phase B estimating rationale. Δ includes no system qual test and 1/5 reduction of effort since system is simplified and uses more off-the-shelf equipment.
Software Development	Based upon Δ to Phase B estimating rationale. Program Design reduced by 1/4 for deletion of landing and ferry mode design. Program coding increased 3 times and verification/validation increased 2 times with change from a supporting to primary responsibility.



Table 7.1-9. D&CM

WORK SHEET NO. 2

EQUIPMENT LIST	MODIFICATION
Computer AP-1	1/2 Word Storage Protection Parallel I/O Channel Memory Address Expansion
Aux. Storage	AP-1 Computer -- Remove CPU New Backpanel New Harness
Input Buffer	SP-1 Computer Structure
Output Buffer	Skylab Power Supply 4 P1/SP-1 Backpanel 4 P1/AP-1/SP-1 Page Hardware 4 P1/SP-1/AP-1 Logic Component
	28 VDC Power Air-cooled MIL-E-5400 Class II



EQUIPMENT LISTMODIFICATION NARRATIVE

3-Axis ADI	Modification of Honeywell Apollo CM FDAI and Electronics, consisting of tailoring displays/scaling and of interfaces modifications.
Aero Flt. Inst. Set HSI IAS-Mach Indicator Altitude-Vert. Speed Ind. Cross Pointer Indicator	ACA-Type AQU-4/A, MIL-I-27848 (No mods.) Bendix E.C. Div. - Type ASK-14/A24G-18, MIL-A-38138 (No mods.) Bendix E.C. Div. - Type AAK-23/A24G-17A, MIL-A-38137 (No mods.) Modification of Honeywell Apollo IM Unit, consisting of interface provisions.
Display Coupler/Driver Unit	New device, to be made by NR, to provide interfaces.
Backup Flight Display Set 2-Axis ADI BDHI Altimeter (Baro.) IAS Indicator Vertical Speed Indicator Z-Axis G Meter	Lear Siegler - MS27150-3, MIL-I-27623 (No mods.) Kollsman Motor Corp. item (No mods.) Aerosonics - Type AAU-16/A, MIL-A-38140 (No mods.) Aerosonics - Type AVU-19/A, MIL-I-38135 (No mods.) Aerosonics - Type AAU-18/A, MIL-I-38139 (No mods.) Cedar Engrg - Type TRU-3/A, MIL-T-27848 (No mods.)
Vert./Dir. Gyro Pkg.	GE - Model 8KD9AR3 with Model 8KEG3AR3 Amplifier (No mods)
Rotation Hand Controller	New item similar to Honeywell Apollo type.
Translation Hand Controller	New item similar to Honeywell Apollo type.
Xducer Subassy. (Center Stick)	Honeywell B-1 item, with new interface provisions.
Xducer Subassy. (Pedals)	Honeywell B-1 item, with new interface provisions.
Throttle Quadrant	GE B-1 item with new interface provisions.
Control Encoder/Coupler	New device to be made by NR, to provide interfaces.



Table 7.1-10. Displays & Controls (Continued)

WORK SHEET NO. 2

EQUIPMENT LIST	MODIFICATION NARRATIVE
CRT Display Unit	IBM - FL4 type head with addition of a read/write refresh buffer, a symbol generator, analog and digital control logic, D/A's, and power supplies
Computer Entry Keyboard	IBM - Apollo type DSKY without display, repackaged for Shuttle task and with new 400 cycle power supply
Subsystem Display Processor	IBM - modified SF-LA Processor with added PCM and selector/buffer MIL-STD-1336 (very little modification)
Caution & Warning	Autonetics - Apollo type (new item)
Digital Event Timer	Apollo type - uncertain whether an off-shelf item will be found.
Mission Time	Apollo type - uncertain whether an off-shelf item will be found.
Lighting Equipment Set	Apollo types plus standard aircraft types.
Panels	New items to be made by NR.



Table 7.1-11. Communications

WORK SHEET NO. 2

EQUIPMENT LIST

MODIFICATION NARRATIVE

Signal Processor	Change of the Apollo Premodulation Processor to fulfill the new horizontal and vertical flight requirements (TACAN, DFI, payload and Space Station data).
L-Band Antennas	For horizontal flights: off-shelf antennas with adapter plates. For vertical flights: modifications for compatibility with TPS.
UHF Antennas	Same kinds of modifications as L-band antennas.
S-Band Antennas	Apollo off-shelf antenna with major changes in the antenna-structure interfaces.
Antenna Selectors	The NASA Houston developed solid state switch selectors will be used with modifications for the UHF and S-band applications.
Transmission Lines & Microwave Passive Devices	Frequency multiplexers, power dividers, and hybrids (i.e., the devices mounted with the coaxial cables) will have to be modified to fit the frequencies and RF-distribution interfaces.

Table 7.1-12. Instrumentation

WORK SHEET NO. 2

EQUIPMENT LIST

MODIFICATION NARRATIVE

Sensors/Transducers	These items of many different types and sizes are mostly off-shelf with some modifications required for connection and mounting provisions, fatigue prevention, and quantity gauging.
PCM Remote Units & PCM Master Units	Off-shelf except for minor modifications for connection and mounting provisions.
Signal Conditioning Units	These are made by packaging existing signal conditioners, and are custom-built items, not off-shelf.
Wide Band FDM	Custom-built, by packaging existing items.
Timing Unit	Modification to existing Apollo CTE for connections and mounting provisions.
Recorders	Minor modifications for connections and controls provisions.



Table 7.1-13. Power Distribution & Control

WORK SHEET NO. 2



EQUIPMENT LIST	MODIFICATION NARRATIVE
Generator Control Unit - AC Current Transformer	Standard aircraft type such as made by Bendix and Westinghouse for 30 KVA AC generators. (No mods.)
AC Generator Contactor & Bus Tie Contactors	Standard aircraft type such as made by Westinghouse and Cutler Hammer for 30 KVA AC generators. (No mods.)
AC & DC Remote Control Circuit Breakers	Electromechanical type, made by Leach Relay & Cutler Hammer, used on DC-10. (No mods.)
Bus Feeder Fuses	Burdny Current Limiters, used on most large aircraft since B-52 (No mods.)
Transformer-Rectifier (T-R)	Air cooled type, as made by Bendix, Westinghouse, etc., for many aircraft.
DC Fuel Cell & T-R Contactor	Aircraft type, as made by Hartman or Cutler Hammer.
Inverter	Surplus and used/refurbished Apollo inverters, made by Westinghouse.
Battery Charger	Custom design for 10 AH nickel cadmium battery (similar to aircraft type except charge rate).
Sequencers & Relay Boxes	Custom design using standard relays.
Wire	MIL spec type Kapton teflon impregnated wire, as used on B-1.
Connectors	NAS-1599, MIL-C-38999, and Rack & Panel connectors as used on Apollo, B-1, and commercial aircraft, respectively
GSE Umbilical, Power	Standard military aircraft external power receptacle, as made by Cannon, Burdny, etc.



7.2 ORBITER & BOOSTER COMMONALITY

Whenever functionally possible and desirable in terms of cost, common equipment elements were utilized for booster and orbiter. The cost impact of this commonality is in the Subsystem Design and Development Engineering (SSDE) area. Cost benefits (savings) from these common items were determined as adjustments to each of the booster and orbiter SSDE cost "build-ups".

Orbiter and booster SSDE subcontractor and/or contractor "make or buy" cost lists were developed for each subsystem. From these basic lists the total SSDE costs of identified (by asterisk in Tables 7.1-2 through 7.1-7) common and peculiar items were determined. Cost commonality factors for each orbiter and booster subsystem were then developed as follows:

$$\text{Factor} = \frac{(\text{Peculiar equip. list SSDE Cost}) + .55 (\text{Common equip. list SSDE Cost})}{\text{Total subsystem equipment list SSDE Cost}}$$

In the cost "build-up" of each subsystem for orbiter and booster the total subsystem SSDE cost was multiplied by the applicable commonality cost factor. This resulted in reduced subsystem SSDE costs according to each subsystem's commonality. Because the amount of peculiar equipment, cost often differs between orbiter and booster, the common cost factors for a given subsystem may not be the same for the orbiter and booster.

It is recognized that this methodology appears to generate cost adjustments which are difficult to track, since reductions to orbiter costs are not readily identifiable as additions to booster costs. Therefore, commonality cost considerations are being analyzed and will be presented in a more traceable manner in a later report.

7.3 COST RISK ANALYSIS

A model was developed for determining the probability of exceeding a given total avionics development cost; input cost risks were identified, and cost risk functions were generated from the model. The elementary input risks were as follows:

- Power Distribution System Cost
- Connector Problems Cost
- Wiring Problems Cost
- Aft Compartment Problems Cost
- Displays & Controls Cost
- Displays & Controls Growth
- Instrumentation Cost
- Sensor Failure Problems
- Data Control & Management Hardware Cost
- Instrumentation Impact on Communication
- GN&C Impact on Communication
- Communication Cost
- Antenna Coverage Cost



Table 7.4-1. Impact of Cost Drivers - 360-Day-To-New Baseline

AVIONICS ELEMENT	COST DIFF. MILLIONS \$	MAJOR COST DRIVERS			FUNCTIONS REQD.
		ONBOARD COMP. DATA PROCESS & MGMT.	EQUIPMENT REDUNDANCY	NEW DEV. VS. OFF-SHELF EQ.	
GN&C	+ 15.7	GN&C Data Processing & control via dedicated computer	FO-FO-FS provided for critical funcs.	No new dev. required	No auto land. Add mech. FCS
DCM	- 90.5	-	-	-	No central DCM GN&C comp. only
D&C	- 5.0	Reduced general purpose	Increased crew mgmt of redun. / fault isolation	More off-shelf ded. equipment	-
Comm.	- 78.0	Reduced auto cont. of functions	Reduced redun. w/o functional degradation	More off-shelf equip. used.	No satellite compatibility. Reduced nav/landing aids
Instr.	- 20.0	Dedicated subsystem meas. permits crew control	-	More off-shelf MUX-PCM	Reduced oper. meas. (3400-1500)
EPDC	- 41.0	Crew mgt. & control	Same	Reduced dev. & dev. risk: More off-shelf hardware	Reduced Oper. meas.
Flight Software	-100.0	No central DCM - Reduces reqd. flight software	-	Reduced software dev.	No gen. purpose redun/config. mgmt
Gnd Software & Hardware	- 17.0	-	-	Reduced new equipment	More manpwr. effort reqd.
Sys. Integ. Lab	- 15.0	-	-	-	Subsys. integ. limited to GN&C



Cost of Power Generation

Study Requirement

Ref. Contract NAS9-10960, Exhibit H, Paragraph 3.0-h
"The Avionics System to be evaluated for the orbiter and booster shall include the following, where required:

(10) Power generation."

Cost Estimate

Costs of the power generation subsystem are reported in WBS categories 1.4.1 (Orbiter Electrical/Mechanical Power Generation) and 3.4.1 (Booster Auxiliary Power Unit). Costs for Orbiter and Booster, Phase B and Phase B-Extension, were reported as follows:

	360 DAY Ø B				FINAL Ø B ¹			
	DDT&E	Prod.	Opns.	Total	DDT&E	Prod.	Opns	Total
1.4.1 Orbiter Power Gen.	144.6	31.1	12.0	187.7	133.4	40.3	52.5	226.2
3.4.1 Booster APU	60.3	2.2	1.8	64.3	18.3	2.9	1.3	22.5
TOTAL	204.9	33.3	13.8	252.0	151.7	43.2	53.8	248.7



Cost of Flight Control Actuators

Study Requirement

Ref. Contract NAS9-10960, Exhibit H, Paragraph 3.0, item h.

"The Avionics System to be evaluated for the orbiter and booster shall include the following, where required:

 (4) Guidance Navigation and Flight Controls (including actuators)"

Cost Estimate

Actuators for the flight controls are part of the Hydraulic Power subsystems (WBS 1.4.2, Orbiter, and WBS 3.4.2, Booster) and their costs are so reported.

Hydraulic system costs for Orbiter and booster, Phase B and Phase B-Extension, were reported as follows:

	ØB 360 DAY REPORT				ØB' FINAL SYSTEM			
	DDT&E	Prod.	Opns.	Total	DDT&E	Prod.	Opns.	Total
1.4.2 Orbiter Hydraulics	32.5	17.0	5.6	55.2	30.6	11.0	15.2	56.8
3.4.2 Booster Hydraulics	143.0	24.2	18.9	186.1	132.0	26.8	16.6	175.4
TOTAL	175.5	41.2	24.5	241.3	162.6	37.8	31.8	232.2

Cost details for the flight control actuators are available for orbiter (have not been estimated for booster) as follows:

Item	No.Reqd	Supplier Unit Cost	Supplier Cost/Veh.	Supplier Dev.Cost	In House Dev.Cost	In House Cost/Veh.
Main Engine Gimbal Actuator	8	.068	.544	.680	.406	.510
Elevon Actuator	4	.037	.148	.785	.469	.139
Rudder/Glide Brake Actuator	4	.031	.124	.621	.371	.117

Cost details for other actuators (landing gear, etc.), servos, valves, plumbing, pumps, coolers, etc., are available on request if desired.



Cost of Unmanned First Vertical Flight

Study Requirement

Ref. Contract NAS9-10960, Exhibit H, Paragraph 3.d.

"For this study, assume the first vertical flight will be manned.
Define delta impact of flying unmanned."

Cost Analysis

1. Motivation. In the event that the early Phase C/D analyses and tests indicate that the first vertical flight will not have adequate assurance of crew safety, then unmanned first vertical flight would be required. Therefore, the delta cost for this eventuality should be estimated.

2. Background. A Phase B study of unmanned considerations was documented in NR Report No. 76-1150-509. Ground rules for unmanned vertical launch were established as follows:

- Both vehicles unmanned
- One flight
- Orbit with once-around free return option
- Conservative trajectory
- Maximum use of baseline automatic onboard systems
- Redundancy levels remain the same as for baseline configuration
- Selected crew function redundancy provided by escort aircraft and ground control
- Software and onboard hardware modification to perform crew dedicated functions
- No additional abort considerations
- No additional tracking requirements over first MOF
- Additional horizontal flight test verification of automation
- Horizontal flight tasks and ferry flights will be conducted manned
- Additional ground tests for integration - software and hardware

The cost elements required for unmanned vertical launch are indicated in the following Phase B table of added functions, added onboard hardware and software, and added ground support requirements.



Cost of Unmanned First Vertical Flight

/ Phase B Study Conclusions

<u>Added Functions</u>	<u>Added Onboard Hardware</u>	<u>Escort Aircraft Requirements</u>	<u>Additional Tests Required</u>
Separation Command Command Destruct Main Propulsion System Shutdown ABES Deploy, Start, and Operate Landing Gear Operations Landing Rollout and Stop Nose Wheel Steering Drag Chute Guidance Program Mode Selection Flight Control Immediate Post Flight Safing	Destruct System Over-ride Receiver ACT Select Buffer <u>Added Onboard Software</u> G&N Sequencing Landing & Roll Out Configuration Management Remote Flight Control Added Checkout <u>Added Support Equip/Facil</u> Ground Command/Control for Approach/Landing. Flt Comd/Control in escort aircraft	Command Destruct Transmitter Over-ride Transmitter Display Receiver Display & Control Panel Hand Controller Command Processed <u>Ground Station Requirements</u> Same items as Aircraft (identical installation)	Horizontal Test Booster 30 hrs. Orbiter 30 hrs. <u>Added Ground Test</u> Supplier's SIL Dev. & Qual Test

Delta cost for unmanned first vertical flight was estimated as follows:

Phase B Study Conclusions

	Orbiter	Common	Booster
Vehicle			
Design and Development	6.8		5.8
Software	2.5		1.7
Hardware	1.0	1.	1.8
ASIL	1.1		1.1
Horizontal Flight Test	12.3		10.1
Ground Station		3.9	
Escort Aircraft Equipment		12.4	
Escort Aircraft Operation		3.0	
Total	23.7M	19.3M	20.5M
Total Cost 63.5M			



Cost of Unmanned First Vertical Flight

3. Phase B' Study. A preliminary survey of the problem produced the conclusion that the Phase B' avionic system requires considerably more augmentation for unmanned first vertical flight than would the highly automated Phase B system. The additional onboard functions and equipments required are as follows:

Deltas to Phase B Requirements
For Unmanned First Vertical Flight

<u>Δ Functions</u>	<u>Δ Hardware</u>		<u>Δ Cost</u>
Receive more command data	Additional Up Data Receiver	2 x .1 =	.2M
Route and address commands	Mission Control Processor	5.6M + (2X.2=.4)=	6.0M
Automatic rollout	Brake actuator) Steering actuator) Electronics drivers)	2 x .5 =	1.0M
Restore COFI	Software Module		4.3M
Restore redundancy management	Software module		4.2M
Escort/control aircraft	Aircraft & ground station equipment		1.6M
		ROM TOTAL Δ	<u>17.3M</u>

Major uncertainties: amount of onboard COFI and automatic redundancy management required.



Costs of Spacecraft Functional Requirements
to be accomplished by MCC/MSFN

Study Requirements

Ref. Contract NASO-10960, Exhibit H, Paragraph 3.j.

"Costs shall be identified for those spacecraft functional requirements that are to be accomplished by the MCC/MSFN".

Cost Estimate

The cost to be incurred is the cost of the Ground Operations and Mission Support Software.

Cost is as follows:

	<u>DDI&E</u>	<u>Prod.</u>	<u>Opns.</u>	<u>Total</u>
Orbiter	28.9	--	13.0	41.9
Booster	28.9	--	13.0	41.9
Total	57.8	--	26.0	83.8

Not all of this is for use by MCC/MSFN. The portion which will be used by MCC/MSFN was estimated to be about 38%.



8.0 CONCLUSIONS

The conclusions for Avionics from the Phase B Extension and Alternate Avionics System Study are briefly described as follows: (1) A new Avionics baseline has been developed which reflects a cost figure of about \$460 million less than the Phase B baseline. In addition, the cost risk appears to be less primarily because new equipment development is considerably reduced. (2) The only functional capability sacrificed in the new approach is automatic landing. (3) The net weight increase over the Phase B baseline is minimal. This includes the weight increase due to the addition of mechanical flight control linkages, and the weight decrease associated with Avionics, ECLSS, and Power Generation as shown in Table 8.1. (4) No significant change in power requirements is seen in the new baseline. (5) No turnaround impact is seen with the new baseline; two weeks remains practical. (6) There will be a slight increase in ground operations activity primarily associated with LRU fault isolation, a part of which must now be done on the ground.

The above conclusions are summarized below in Table 8-1. It is felt that the Avionics configuration defined in this report is a viable one and will support NASA objectives for the Shuttle Program.

Table 8-1. Avionics Change Impact

FUNCTION	• NO AUTO LAND CAPABILITY	
WEIGHT CHANGES	• AVIONICS	-123
	• FLIGHT CONTROL LINKAGE	+600
	• ECLSS	-198
	• POWER GENERATION	-112
		+167 LB
POWER	• NO SIGNIFICANT CHANGE IN PEAK POWER OR ENERGY	
GROUND CHECKOUT	• DECREASED EQUIPMENT DEVELOPMENT	
	• SLIGHT INCREASE IN OPERATIONS	
	• GROUND SOFTWARE LRU ISOLATION	
	• NO TURNAROUND IMPACT	
COST DIFFERENCE	• ABOUT -\$462M TOTAL SHUTTLE AVIONICS	