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IN-FLIGHT MAINTENANCE STUDY

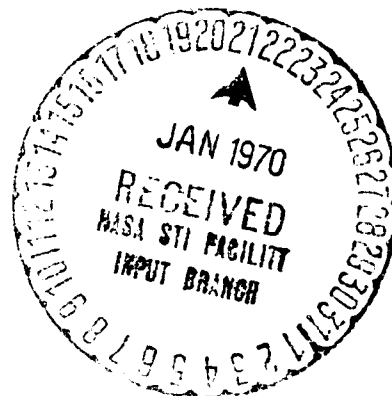
FINAL REPORT

December 1969

Prepared for

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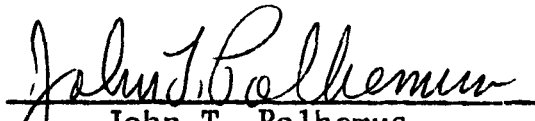
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IN-FLIGHT MAINTENANCE STUDY
FINAL REPORT

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FOREWORD

This report describes the results of an In-Flight Maintenance Study performed under Modification 3 to NASA Contract NAS9-8144, Microfilm Storage and Display.

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I. INTRODUCTION

This study, entitled In-Flight Maintenance (IFM), was performed under Contract NAS9-8144, Mod. 3 for the Information Systems Division, Manned Spacecraft Center, Houston, Texas. R. D. Newlin was NASA Technical Monitor.

The study was concerned with all actions required to obtain maintainable electronics hardware, and subsequently to maintain this hardware in the Space Station - Space Base era from 1975 to 1985. Primary emphasis was placed on processing electronics, as opposed to sensors, controls, displays, actuators, etc. The Block II Apollo Guidance and Navigation, Stabilization and Control, and Telecommunications systems were chosen as hardware examples to work with, making the assumption that similar functions would be necessary in a space base or station.

The study approach and some results are summarized in Section II of this report, with the objectives and approach covered in more detail in Section III. The sample systems analysis, redesign, packaging design, and test and checkout are covered in Section IV and VI through VII respectively. Section V, entitled IFM requirements, is meant to be somewhat autonomous, as it contains some rationale for methods and requirements necessary to achieve successful in-flight maintenance, along with criteria that may be utilized as requirements for future contracts. The conclusions in brief form, and an indication of areas we feel need additional study are given in Section IX.

The bibliography includes a selected list of the most important books and papers that were studied, and all of the references cited in the text.

II. SUMMARY

The In-Flight Maintenance (IFM) Study was initiated in May 1969 by NASA-MSC. The Martin Marietta Corporation performed the work in the Electronics Research Department of the Denver Division.

The study might well have been named "How to make electronic systems last ten years in a Space Station", for fundamentally that is what was studied. The output of the effort is more far reaching however, for the aim was to develop analysis methods, techniques and criteria that will be useful for many programs, not just the space station.

In this brief summary we have described little more than the nature of our findings, methods and conclusions; the details will be found in the body of this report. However, it is worth noting that while criteria, guidelines and methods for achieving maintainable systems were evolving, a parallel effort was in progress to test these tools. This was the analysis and subsequent redesign for maintainability of the Block II Apollo Guidance and Navigation (G&N), Stabilization Control (SCS) and Telecommunications Systems. We delved deeply into all of these systems, but we were particularly fortunate in having personnel available with a comprehensive background in the G&N and SCS, hence the study results in these areas are especially gratifying.

While the primary product of the study was to be criteria and considerations for obtaining maintainable electronics, some of the methods developed during this study have been tested on other programs and have proved to be of value.

IFM Requirements

In order to obtain maintainable systems, a contracting agency must clearly identify this intent at the time of the request for proposal, and reiterate this intent throughout the design phase. However, without design oriented criteria and guidelines to work with, each contractor will develop an independent approach to maintainability, and a contracting agency will find itself with a proliferation of so-called maintainable designs and unable to adequately measure performance. Clearly some standardization constraint on design is desirable, for otherwise we shall surely lose the economies possible in packaging, structure, cooling, test and checkout, sparing, and other factors affecting both non-recurring and recurring cost.

In examining existing criteria relating to maintenance, it was found that they tended to be oriented toward human factors, or were mission guidelines. Certainly these are important considerations for achieving maintenance capability but truly design oriented criteria are needed, relating, for instance, to analysis, packaging, malfunction detection and fault isolation, redundancy, commonality and spares. During this study 53 such criteria were developed. These were applied against design examples to test their validity, then re-examined for generality.

The example systems utilized in this study were the Apollo Block II G&N, SCS and Telecommunications systems. These were re-designed utilizing the mission constraints and considerations of the 1975 space station, assuming that similar functional requirements would exist for that mission. In relating mission function to equipment satisfaction, various existing techniques were examined for usefulness, such as Failure Modes and Effects Analysis (FMEA), Functional Flow and Fault Tree. While these are more or less useful, a simple direct approach was sought that would be useful to personnel at various levels of program involvement. The technique developed utilizes Boolean notation and permits the direct translation from mission function to specific design, and is useful from conceptual phase to equipment acquisition. This analysis technique focuses attention on the critical functions, or "weak links", and can be extremely detailed, which is useful for the designer, or can be greatly simplified to present an easily understandable "quick look" for management. Additionally, the impact of commonality, redundancy or changes in equipment can be readily identified with this technique.

In satisfying the requirements of a mission, the designer will have a certain amount of latitude. Normally many trade-offs are made in arriving at a design, but rarely are these trade-offs documented in such a way that they can be intelligently or effectively reviewed and discussed. To circumvent this, a straightforward method was developed for documenting the rationale for a decision, the IFM matrix. Examples are shown elsewhere, but briefly it consists of a list of independent mission considerations which are weighted as to their importance concerning the problem at hand. The candidate problem solutions (or designs) are examined and scored using each mission consideration in turn. The resulting IFM matrix causes the problem solver (designer) to develop a rationale for his solution, ovviates the omission of important considerations in the trade-off, and furnishes a record and basis for review and discussion. For most trade-off studies the same basic list of considerations are useful, but of course the matrix should be tailored to the problem at hand.

The establishment of the program and mission groundrules, subsequent identification of specific design oriented criteria, systems analysis to relate mission functions to equipment satisfaction and a technique to optimize equipment design are the basic requirements for enhancing IFM. To be effective, these groundrules, criteria and analytical techniques must be developed prior to solidifying equipment specifications and available for usage at preliminary and critical design reviews. The IFM requirements and expected usage or purpose of IFM analysis should be identified in the Request for Proposal (RFP). Further, the RFP should direct the Contractor to identify, in his proposal, how the IFM analysis will be conducted and the unexpected depth of analysis.

Sample System Analysis

An analysis of the Apollo Block II Guidance and Navigation, Stabilization and Control System and Telecommunication System was conducted at the start of this study for the purpose of familiarization as well as providing data on the functional and hardware characteristics, such as redundancy and commonality, pertinent to the subject of in-flight maintenance on future manned space programs.

For the sake of maintaining a relatively straight forward example, the analysis was limited to critical functions of the system; i.e., those functions which in the event of a failure, would impact on the safety of the crew in any way. A most effective tool in accomplishing both the objective of familiarization as well as analyzing the system redundancy was to express each of the functions in a Boolean Algebra equation where each of the terms represents a unique piece of hardware in the system, essential to satisfaction of the function or any redundant paths. The functions analyzed were attitude control, thrust vector control and attitude reference alignment for the G&C (G&N and SCS combined) and spacecraft-to-ground ranging and spacecraft-to-rendezvous vehicle ranging for telecommunications.

It was concluded from the analysis that (1) the Boolean approach was an efficient and effective means of tying hardware to function, particularly for such means as Failure Mode Effects Analysis; (2) there were no single failures in G&C or telecommunications that could result in loss of the crew, and (3) an apparent weakness in the design was use of common switches for redundant paths in the telecommunications system.

In the area of hardware characteristics the major concern was to determine the extent of commonality because of its impact on such

key items as spares weight, the malfunction isolation scheme, opportunity for cannibalization, etc. To illustrate, some SCS system commonality figures are provided and are representative of the other electronic systems. Of approximately 30 different circuit types in the SCS, the following five are the only ones presenting any significant degree of commonality:

Transistor Enable/Disable Circuit	29
AC Amplifier (400 cps)	42
DC Amplifier	12
Demodulator	26
Drive Amplifiers	16

Even greater commonality exists at the smaller piece part level, however, for remove/replace repair, it is practically useless. Even the level of hardware listed above is too small. The interconnections for remove/replace of such components would be prohibitive, as well as the complexity of the fault isolation system. Our trade-offs indicate that providing R/R commonality at such a level would be uneconomic.

Therefore, it seems logical that for the Space Station remove/replace type of maintenance the next larger level of hardware is far more practical; this could best be called hardware commonality at the functional level, such as attitude error display electronics or accelerometer electronics. Commonality rarely exceeds 5 of a kind at this level, nevertheless it does exist and every attempt to maximize it should be made.

To conclude the hardware analysis, the following data is provided for comparison with the redesign characteristics later. The system weights are 582 pounds for Guidance and Control, and 243 pounds for telecommunications while power consumptions are 1052 and 468 watts respectively. The total systems failure rate with all systems operating continuously was on the order of 1 failure per 18 days or 20 failures per year. A rough estimate of spares resupply weight, with an average black box weight of 20 pounds, was 400 pounds per year, or nearly half the weight of the total system.

Redesign

This section takes the systems analyzed earlier and extrapolates their design into an in-flight maintainable system for Space Station '75. A basic assumption was that the same functions are required on the Space Station as were required on Apollo, intentionally reducing the design to more of a restructuring and

repackaging job than a true redesign. The idea was to take Apollo electronics capable of doing the job and reconfiguring it in such a way that changes or additions in function would not impact the basic approach arrived at in the study.

The component selection for the redesign will use essentially today's technology. The key decision to be made was whether or not to extensively utilize Large Scale Integrated Circuits (LSI); the decision was made to use micro logic and hybrid micro circuits, Medium Scale Integrated Circuits (MSI) where they are available, and LSI only as available now for memory and computer applications. LSI is being applied principally where a large number of circuit functions of the same type are needed such as computers. The packaging scheme in this study was chosen to provide for LSI in the future, should it become more extensively used.

Very briefly, the existing Apollo telecommunications is composed of a communications link at VHF frequencies for earth orbital, entry and LM/EVA communication plus backup ranging capability, while a second link at S-Band frequencies provides the same functions as well as lunar phase communications where VHF is inadequate. The S-Band system is also essential to the data transmission function which comprises the remainder of the telecommunications components. For the redesign this system was left unchanged in function.

The G&N on Apollo provides the sole source of guidance and the prime source of vehicle control commands. The SCS provides a backup source of control, all the final-stage driving electronics plus hand controls and displays. For the redesign the backup SCS control capability has been eliminated as well as the displays and controls redundancy. Two redundant "G&N only" systems were provided for the following reasons: (1) the space station RFP requires that a single compartment must be able to be isolated in the event of catastrophe and capability provided in the remaining compartments to safely live and take corrective action, or return to Earth; (2) two identical systems maximizes commonality and cannibalization; and (3) permits complete power down of a "down" system while complete capability of vehicle control is provided from the alternate system.

The major effort in this study for in-flight maintenance was focused on the electronics components rather than sensors, antennas and such. These components will also require maintenance but will be more a problem of securing and aligning rather than packaging.

The initial criteria for configuring or grouping the electronics was optimum fault isolation. This was defined as having minimum test points and minimum fault isolation system complexity. This approach maximized the functional hardware level commonality mentioned earlier. After a first cut using this criteria the resulting configuration was then modified to conform with all other pertinent constraints such as packaging design, environment, cooling methods, interconnections scheme, etc. The final configuration resulted in a total of 61 lowest replaceable units (LRU) per G&C system or a total of 122 for the two systems on the station. Of the 61 total per system, 46 were electronic modules mounted in a cabinet and varying in size from the equivalent of a 4" x 5" board to 10 boards. The resultant total of the telecommunications system was 199 modules, the majority of which were a 2 board size.

The test points required to isolate the 61 components of the G&C were 228 including all means of measurement such as CMC digital word readout, voltage checks and caution and warning test points. Four new stimuli were required. The total per entire space station then was 456 test points and 8 new stimuli. A corresponding estimate of telecommunications test points amounted to approximately 500 test points and 12 additional stimuli required.

One prime candidate for commonality that became apparent in the redesign was power supplies. If the power supplies were standardized, as considerable Martin Marietta study and testing has indicated is feasible, the function of the 25 or so different power supplies in the redesigned systems could be performed by just a few different types of supplies, thereby permitting interchangeability and reducing spares requirements from a minimum of 25 to 2 or 3. These standardized supplies will satisfy a variety of power requirements without compromising efficiency.

The combined failure rate of the redesigned system was estimated at 1 failure per 26.7 days or approximately one per month. From a total weight of G&C and telecommunications of 1231 pounds and a per module weight of 4 pounds, the resultant resupply weight was 48 pounds/year or 1/25 of the total systems weight; considerably better than the 400 pounds for the original system. It was assumed the telecommunications and one G&C system was operating continuously.

Electronic Packaging

In order to facilitate maintenance, the packaging should be modular. To minimize costs at virtually all levels from design through installation and maintenance, the packaging should be standardized.

Trade-off studies indicated that a standardized packaging approach with somewhat flexible module size and forced-air cooling would best satisfy the mission considerations for the space station as weighted in this summary, a weighting that of course is open to discussion. An embodiment of some of these ideas may be seen in the engineering mockup, which represents a truncated cabinet.

A trade-off involving module size indicated that a module may be built having the equivalent of from one to ten printed circuit (PC) boards without losing the advantages of standardization, where a PC board of 4 by 5 inch size will hold up to 50 integrated circuit flat packs. The latter is the nominal size for a module, but the flexibility was found to be important in reducing malfunction isolation complexity.

The cabinet design formulated for housing the electronic modules incorporates features such as easy access for R/R, modular cabling, forced convection cooling, and accommodation for various module sizes, including shielded modules.

Several connectors were selected for consideration on the basis of this study, which encompasses a literature search and a vendor survey. It was felt advisable to choose a packaging and connector approach which would be economic if and when the environmental conditions in the space station or base are no longer adverse. But for the present, the recommendation is to utilize commercial connectors, when available, that employ interfacial seals isolating the contacts from each other. For rack and panel or PC board connectors, the addition of an elastometric squeegee or pin seal is recommended. All of these can utilize silicone grease for additional protection if necessary. Several types of connectors are demonstrated on the mockup.

Test and Checkout

The test and checkout functions are a prime consideration in maintenance and will be found interwoven with virtually every other aspect of the study.

In examining test and checkout requirements and philosophy, trade-off studies were made, the literature was studied, and key personnel from recent or ongoing programs were consulted; notable among these were Computerized Automatic Ground Equipment (CAGE), On-Board Checkout System (OCS), Multi-Parameter Display (MPD) and Built-In Checkout (BIC).

While a test and checkout system must perform the three principal tasks of critical failure monitoring, failure isolation and system checkout, the primary concern was with failure isolation. This study performed an in-depth analysis and redesign of the Block II Apollo G&N, SCS and Telecommunications systems, so the trade-off studies on failure isolation systems deal primarily with those examples.

As it is not practical to design equipment with 100% probability of no failure over a ten-year life span, some degree of system repair will be necessary. For this study, system repair was limited to remove and replace (R/R); therefore, the primary emphasis was on determining the optimum type of malfunction isolation system for the space station.

A very versatile and complex malfunction isolation system will result in a high confidence of isolating a failure, and will place few constraints on the design of the systems to be tested, but there are penalties in the size and complexity of the system, the operator training requirements, and the total system reliability.

A preferable approach is to force systems designs that are testable; for instance, structuring the system and sizing the modules to minimize the number of test points and external stimuli as well as carefully choosing these test points to provide meaningful and understandable data. This design for testability has the dual advantage of simplifying the test equipment while increasing the probability that a malfunction can be isolated; the Space Base Study issued by NASA (1969) indicates that the desired probability is 95% to 98%.

System verification, both for repaired systems and replacement (new) systems, must be provided, and requires some different test procedures, test points and stimuli. Hardware sharing between this function and malfunction isolation is indicated and should be easily realized if the test system is built around a general purpose computer. For new systems, verification in a house spacecraft is recommended before installation in the space station.

A trade-off study was performed to find the optimum malfunction isolation system complexity for a particular system, the redesigned Guidance and Navigation System. This system was redesigned to facilitate test and checkout. Additionally, emphasis was placed on the signal processing electronics, not on sensors, control and display devices, and actuators. For this particular example, a medium complexity malfunction isolation system was chosen as optimum, and the trade-off process utilizing the IFM matrix is documented in the body of this report.

When the totality of electronic systems in the space station is considered, a reasonably sophisticated test and checkout will probably be indicated; however, if all systems are designed for testability, the peripheral equipment complexity will be reduced considerably.

Areas for Additional Study

During the course of our study, it became evident that certain facets of the electronics maintainability problem should be studied, but could not be within this study's scope. Some of these are as follows:

- 1) Sensors - calibration, maintenance and checkout.
- 2) Displays and controls - in-flight maintenance.
- 3) IFM experiment - to test methods, concepts and hardware.
- 4) Checkout and malfunction isolation - further definition of requirements, for the totality of the electronics systems in the space station.
- 5) Interprogram commonality - for instance, Space Shuttle, Space Base electronics commonality.

III. OBJECTIVES AND APPROACH

The primary objective of the In-Flight Maintenance (IFM) Study was to develop criteria and guidelines that could be used to enhance in-flight maintenance as related to electronics systems for Space Station '75. Another objective was to identify typical equipment type and subsystems that are candidates for commonality, and finally, to recommend techniques for controlling those design and development areas that have a significant impact on in-flight maintenance and extended mission life.

Technical Approach

The first step in meeting the objectives set forth was to establish mission ground rules and assumptions for the space station-space base complex. In this, aid was provided by NASA personnel, but in some cases it was found necessary to generate new and specific ground rules for the Study.

Once these ground rules and assumptions were established, it was possible to proceed with an analysis of some existing hardware with the intention of redesigning this hardware to allow in-flight maintenance. This redesign effort, in turn, was utilized as a vehicle to help in the formulation of criteria, and to test and retest these criteria as the program progressed.

The hardware items that were chosen for analysis and redesign were the Block II Apollo G&N, SCS and Telecommunications systems. It is presumed that the bulk of the functions performed by these systems will also be necessary on the space station, therefore utilizing these existing systems gained the advantage of a concrete example and historical data with which to compare attributes of the redesigned system.

During the early part of the selected systems analysis those operational and hardware areas were identified that have the greatest impact on in-flight maintenance and extended mission life. Emphasis was placed on those hardware items that promised the greatest payoff in helping meet the study objectives; examining them for possible packaging levels, commonality candidates and system/mission effect in the event of failure.

In the redesign effort, the intent was to identify what decisions and information are necessary to control design and development in order to ensure the desired result; i.e., the acquisition of a spacecraft electronics system that can be maintained in a space environment. As the various problems were

considered a concerted effort was made to document the thinking, the trade-off study, and all other considerations examined in arriving at decisions. Some of this documentation will be found in later sections of the report.

The redesigned system was used as a vehicle to test the criteria developed, the degree of commonality realizable, and a possible packaging scheme. It is recognized that the list of criteria is not exhaustive, and that the criteria will undergo change with time, just as they did during the study. The method of generating useful criteria is more important than the list itself, and hopefully this will be evident from the results as presented in this report.

IV. SAMPLE SYSTEMS ANALYSIS

An analysis of the Apollo Block II Guidance and Navigation System, Stabilization and Control System, and Telecommunications System was performed at the onset of this study. The objective was two-fold. First, it provided a means of becoming familiar with the electronic systems which later in the study were to be redesigned for in-flight maintenance. Secondly, it provided "guinea pig" flight qualified operational space hardware as close in function to the space programs of the future as is currently available. This second objective entailed analysis of the system from a number of different aspects with the primary purpose of singling out the hardware and functional characteristics especially significant in designing a system that requires in-flight maintenance.

Some of those specific characteristics that were considered pertinent and merited inclusion in this report on in-flight maintenance are: redundancy required and available, and the use and degree of commonality. The hardware weight, power consumption and failure rate data is also furnished for comparison later with the redesign configuration.

Functional Description

This analysis is not concerned with all functions of the three systems. Only those considered of a critical nature are discussed in any detail as they comprise nearly all the hardware in these three systems, including all redundant paths.

A critical function for purposes of this study, as for most Apollo studies (NR, 1968b)*, is a systems function or capability that, if lost, may result in loss of the crew. A single failure that can cause such loss is a Category I, or sometimes referred to as Criticality I, failure. Conversely, a Category II failure is one resulting in abort of the mission but not loss of the crew. To maintain a relatively straightforward example Category II failure effects are not discussed in this report.

The functions of the Guidance and Navigation System and Stabilization and Control system are virtually inseparable. They are: (1) attitude control, (2) thrust vector control, and (3) attitude reference alignment. The telecommunication functions that could be construed as critical are (1) spacecraft to ground communication, (2) spacecraft to ground ranging, and (3) spacecraft to rendezvous vehicle ranging. The criteria for determining

* NR = Abbreviation for North American Rockwell

which functions are critical and which are not was based on a knowledge of the three systems and the Apollo Lunar missions on which they are used. As the mission changes so does the criticality of some of the functions. For this study no attempt was made to finitely determine a given mission. Rather, a conservative approach was taken and any function that could be construed as critical across a range of different missions was included in the analysis as being critical.

Considerable thought was given to find some way of quickly yet thoroughly becoming familiar with the system and hardware required to perform the above functions. A method was employed which appeared to serve the purpose and had the feature of allowing expression of the system function directly in terms of the actual hardware involved. The method utilizes a single Boolean algebra equation for each of the systems' functions. The terms of the equation represent unique pieces of hardware required to satisfy the function or its redundant paths, if any. The equations for those functions identified earlier are provided in Table IV-1. The base of knowledge for development of these equations were the following references: NR, 1969; Honeywell, 1968b; MIT Schematics; NR, 1969d; Collins Radio, 1966; NR, 1968c, as well as considerable background and experience in the design, development and operational utilization of the Apollo subsystems, particularly Guidance and Control. The explanation of the G&C function equation symbols are found in Table IV-2 and Telecommunications in Table IV-3. The equations of Table IV-1 will be discussed in more detail under Failure Mode Effects.

It is quite possible to employ the same approach for any function of any system desired. It is most useful, however, for critical functions where it quickly illustrates the required redundancy, or lack of it. There is no reason why this Boolean approach could not be utilized at various hardware levels below that used in this example for more thorough tracking of mission to hardware.

TABLE IV-1 FUNCTION EQUATIONS

Attitude Control	$\underbrace{(J_{N1}Q_1 + J_{N2}Q_2)}_{\text{Jets \& Drivers}} \underbrace{(I_1 + I_2)}_{\text{Displays}} \underbrace{[BA_U Q_B S_G (K_1 + K_2) + Q_S S_M^E C_2 (C_1 + C_2) (G_1 + G_2) (G_1 + G_2) (BA_U + D)]}_{\text{G\&N Control}} \underbrace{[BA_D U + D]}_{\text{Att. Ref.}}$
Thrust	$\underbrace{G_M (V_1 + V_2)}_{\text{SPS TVSA Eng \& Act.}} \underbrace{[BA_U A_S^E + E_C E_G G_A S_M (P_1 + P_2) + (A_S D + BA_U) S_M^E (I_1 + I_2) (G_1 + G_2) (C_1 + C_2)]}_{\text{G\&N Control}} \underbrace{[BA_D U + D]}_{\text{Att. Ref.}} \underbrace{[BA_D U + D]}_{\text{MTVC Control}}$
Att. Ref Alignment	$\underbrace{(Att. Cont.) [BA_U (K_1 + K_2) (OE_0 + O) + DM(O + OE_0) (G_1 + G_2)]}_{\text{G\&N Att. Determ.}} \underbrace{[BA_D U + D]}_{\text{SCS Att. Determ.}}$
Voice Comm.	$\underbrace{[A_{NV1} A_{NV2} S_{AV} (T_{V1} + T_{V2}) (S_{VA} + S_{VB}) (T_{XV}^{R_{CV1}} + T_{XV}^{R_{CV2}}) + P_{MP} S_S (T_{XS}^{R_{CS1}} + T_{XS}^{R_{CS2}}) S_{PA} (P_{A1} + P_{A2}) (T_{S1} + T_{S2})]}_{\text{S-Band}} \underbrace{[S_{AS} (A_{NS} + A_{NO})] \underbrace{[(M_1 + M_2) (E_1 + E_2) A_{CP1} (A_{C1} + A_{C2})] + []}_{\text{VHF}} + []}_{\text{Radio Link}} \underbrace{[]}_{\text{S-Band}} \underbrace{[]}_{\text{CMP}}$
Ranging, SC ↔ GND	$\underbrace{S (T_{XS}^{R_{CS1}} + T_{XS}^{R_{CS2}}) S_{PA} (P_{A1} + P_{A2}) (T_{S1} + T_{S2}) S_{AS} (A_{NS} + A_{NO}) (S_R + R_{UDL})}_{\text{S-Band}} \underbrace{[]}_{\text{Ranging Sw's}}$
Ranging, SC ↔ LM	$\underbrace{S_{VR} A_{NV1} A_{NV2} S_{AV} (T_{V1} + T_{V2}) S_{VA} S_{VB} X_1^{R_{CV2} D} + R_{LM} T_{RT}^{S_{ARR}}}_{\text{CM VHF}} \underbrace{[]}_{\text{LM Rendezvous Radar}}$

TABLE IV-2 - G&C SYMBOL EXPLANATION

A_D	= A/D & D/A logic with associated timing and control (CL)
A_G	= PGNCs accelerometer and associated timing and control
A_H	= EMS accelerometer and associated electronics
B	= Command module computer (CMC)
$C_{1,2}$	= Rotation Controllers 1 and 2, respectively
D	= Functions of Gyro Display Coupler (GDC) and Attitude Set Control Panel (ASCP)
$E_{C1,2,3}$	= Functions contained in the Electronic Control Assembly (ECA)
E_O	= Optical Subsystem related electronics
G_M	= SPS Engine
$G_{1,2}$	= SCS Gyro Assemblies 1 and 2 (GA's)
$I_{1,2}$	= FDAI 1 and 2 and related electronics in Electronic Display Assembly (EDA)
J_N	= Normally used Jet Solenoids
J_N'	= Redundant Jet Solenoids
$K_{1,2}$	= Display and Keyboard Units 1 and 2 (DSKY's)
M_N	= MSFN and ground computer facilities
O	= Optical Subassembly
$P_{1,2}$	= Function of Gimbal Position Display (GP/FPI) and assoc. electronics in EDA
Q	= RCS driver functions of the Reaction Jet and Engine On-Off Control Assembly (RJEC)
Q_B	= Logic in RJEC related to PGNCs attitude control
Q_S	= Logic in RJEC related to SCS attitude control
$S_{B,M}$	= PGNCs and SCS mode switching, respectively
$V_{1,2}$	= Redundant functions of the Thrust Vector Servo Assembly (TVSA) and related SPS engine actuators

TABLE IV-3 TELECOMM SYMBOL EXPLANATION

$A_{C1,2,3}$	=	Audio Center Electronic Modules
$A_{AP1,2,3}$	=	Audio Control Panel (Vol. switch, etc.)
A_{NO}	=	S-Band Omni Antenna
A_{NS}	=	S-Band High Gain Antenna
$A_{NV1,2}$	=	VHF Antennas (180° apart)
A_{RR}	=	Rendezvous Radar Antenna
D_{RG}	=	Digital Ranging Generator
$E_{1,2}$	=	Earphones
$M_{1,2}$	=	Microphones
$P_{A1,2}$	=	S-Band Power Amplifiers
P_{CM}	=	Pulse Code Modulation Equipment
P_{MP}	=	Premodulation Processor
R_{LM}	=	Rendezvous Radar - LM
R_T	=	Rendezvous Radar Transponder - CSM
R_{UDL}	=	Uplink Link Ranging Select
S_{AS}	=	S-Band Antenna Switch
S_{AV}	=	VHF Antenna Switch
S_{CE}	=	Signal Conditioning Equipment
S_{PA}	=	S-Band Power Amplifier Switch
S_R	=	Ranging Select Switch
S_{RT}	=	Rendezvous Radar Transponder Switch
S_S	=	S-Band Transponder Switch
$S_{VA,B}$	=	VHF AM Trans/Rcv. Switches
S_{VR}	=	VHF Ranging Switch
$T_{S1,2}$	=	S-Band Triplexer
$T_{V1,2}$	=	VHF Triplexer Bandpass Filters
$T_{XS}^R CS1,2$	=	S-Band Transmitter/Receiver Electronics
$T_{XV}^R CV1,2$	=	VHF Transmitter/Receiver Electronics
USBE	=	Unified S-Band Equipment
VHF	=	Very High Frequency Comm Equipment

Failure Mode Effects Analysis

In the course of this study a review of several Failure Mode Effects Analyses (FMEA) on Apollo was conducted (NR, 1968b; NR, 1969a, NR, 1969b; Honeywell, 1965). No significant conclusions could be drawn from them other than that there were no basic disagreements. Therefore, a detailed part by part FMEA is not discussed further in this report. What appears more useful and instructive is a brief but similar analysis at a functional level commensurate with the Boolean expressions in Table IV-1. Before looking at any functions in particular it is important to recognize that since all the equations represent crew safety functions, complete redundancy must be available to guarantee crew safety in the event of a single failure. Examining the first equation of Table IV-1, attitude control; it is possible, for the purpose of determining adequacy of redundancy, to simplify that equation by grouping together all terms ANDED together in an expression and assigning to it some general letter designator such as a, b, or c. The simplified equation becomes:

$$\underbrace{(a + b)}_{\text{RCS Jets}} \underbrace{(c + d)}_{\text{dis-plays}} \underbrace{(e + f)}_{\text{control elect.}} \underbrace{(g + h)}_{\text{att. ref. subsystems}}$$

It is apparent that redundancy for this function is quite adequate. What is not apparent from the equation is the availability, in terms of time, of the redundant paths. This requires more detailed investigation of the operational use of the systems involved. On Apollo where no maintenance was planned, alternate or redundant circuits were, for all practical purposes, immediately available upon failure of the path in use. Switchover was performed manually by the crew.

The thrust vector control equation reduces to:

$$\underbrace{a}_{\text{SPS Eng.}} \underbrace{(b + c)}_{\text{TVC Elect.}} \underbrace{[d + e + f (g + h)]}_{\text{Control Elect.}}$$

In this equation some inconsistencies are apparent. The weak link is "a", or G_M in the original equation, which represents the SPS Engine thrust chamber. This is, in fact, a single point failure on Apollo and is recognized as such. Also of interest in

the above equation is the triple redundant control electronics for the TVC loop. This appears unnecessarily redundant in light of the single point failure in the engine nozzle. It would be premature, of course, to eliminate a control path without considering first the operational aspects of so time-critical a function as TVC. The value of the Boolean expression, however, is that it points immediately to an area of possible over-design and consequently possible simplification.

Reducing the attitude reference alignment equation of Table IV-1 results in:

$$(att.cont.) \underbrace{(b + c)}$$

Att. Determination

The attitude control function required for attitude reference alignment has already been found adequate, therefore, the entire function is redundant. It should be pointed out, in looking back to the original equation, that the SCS attitude determination capability is dependent on one term, M, which is not always available. From Table IV-2 this is the Manned Space Flight Network and computing facility. This path would not provide adequate redundancy for a program without continuous or at least frequent ground communication.

The ground to spacecraft communication function equation can be simplified several times resulting in:

$$\begin{array}{ccc} (a + b) & & \underbrace{(c + d + e)} \\ \downarrow & & \downarrow \\ VHF & S- & Audio Link \\ & Band & \end{array}$$

which appears sufficiently redundant.

The spacecraft-to-ground ranging equation is essentially the S-Band equipment:

$$a(b + c)d(e + f)(g + h)i(j + k)(1 + m)$$

which has three single point failures, a, d and i. All three of these, looking at the original equation and Table IV-3, are switches. This points out a characteristic weakness in telecommunications. Switches were originally neglected in redundancy design on Apollo as being non-fail devices. Later in the program the failure possibility of a switch was recognized and some redesign was performed

to alleviate the more critical redundant circuits but a number of weak links still exist.

The final equation is the spacecraft-to-rendezvous-vehicle ranging capability, which reduces very simply to $(a + b)$. The two redundant paths are the Command Module VHF ranging and the Lunar Module Rendezvous Radar system.

Hardware Analysis

One important area in examining the hardware of the three subsystems was the degree to which commonality existed. In designing for optimum maintainability the greater the number of common items used in a system the simpler will be the malfunction isolation scheme, the fewer different spares required, and the more cannibalization can be utilized.

The analysis has shown that there are certain general types of circuits which occur repeatedly in either the PGNCs or SCS. Those occurring most commonly and the approximate quantities are:

SCS

Transistor Enable/Disable Circuit	29
AC Amplifier (400 cps)	42
DC Amplifier	42
Demodulators	26
Drive Amplifiers	16

G&N

AC Amplifier (800 cps)	53
Transistor Gate Circuits	184
Schmitt Trigger	16
Output Drivers	16
Buffer Amplifiers	13
Push-pull Amplifiers	9
Center-Tapped Transformers	22

These figures do not consider the computer, IMU or individual NCR gates used in the logic. Commonality is found to occur in quantum type steps rather than distributed in any continuous manner. To illustrate, note that there are 184 similar transistor gate circuits used but the next highest level of commonality is in the region of 40-50 AC amplifiers, then a jump to roughly 30 enable/disable demodulation circuits. Beyond this point circuit commonality drops abruptly and is typically a maximum of 3 for the

SCS and a maximum of 5 for the PGNCS, both of these correspond to the number of identical or nearly identical channels in a system. Out of a total of 30 circuit types for the SCS then, only the 5 given above exhibit a high degree of commonality. Similarly for the PGNCS, only 7 of 44 circuit types exhibit a high degree of commonality. Also, 7 SCS circuits are unique as well as 7 of the PGNCS circuit types used only once. Further, many of the circuits which have been grouped together are actually associated with unique coupling networks not common to other circuits in the group.

The commonality in the communication system has not been discussed in detail because in general, commonality is exhibited to an even lesser degree than the other two systems. One obvious reason for this is that the "black boxes" in the system, though contracted by Collins Radio, were each built by a different vendor which automatically reduces commonality. In addition, the circuit paths that are potentially common are generally RF circuits and consequently tuned to specific and different frequencies.

The analysis was carried to a level of the smallest reasonable module level, typically a single amplifier circuit. It is at this level, and lower, that substantial commonality can be found for some circuits. However, this level is considered to be unsuitable for modularization for the following reasons:

1. This level of commonality would result in a module that is physically too small and may be only a portion of a chip in 1975.
2. Such a low level of modularization would result in a proliferation of interconnections/connectors severely degrading reliability.
3. Troubleshooting and fault isolation is impractical to this level and would result in unnecessarily complex checkout equipment.
4. Low level commonality constraints would impact design efficiency and result in increased power usage, compromise of operating characteristics, etc.
5. Commonality would not be achieved to a significant degree; there would still remain many small, unique circuits.
6. It would be impractical to place low level commonality constraints on multiple contractors. Exceptions would tend to become unmanageable.

In summary then, current Apollo hardware commonality is not very practical for remove-replace at the level at which it is most prevalent. Another approach to commonality is presented in Table IV-4. This could best be described as functional commonality, or the grouping of electronic components associated with a specific system function where that function is repeated several times in the system. For example, attitude display electronics, which are identical for all three attitude axes; pitch, yaw and roll. It is apparent from the table that there is a commonality at this level throughout all three systems. It does not prevail to the degree that the lower level hardware commonality does, but several factors should be considered. The package is inherently larger, therefore, is easier to handle and easier to remove-replace. Repair at this level could easily be restricted to remove-replace rather than the soldering techniques required for repair or replacement of the lower level components. Packaging for functional commonality also minimizes the fault isolation circuitry required. With function-oriented packages a few test points can generally isolate all elements of a particular function, as illustrated later in Section VI.

To conclude the summary of existing equipment, the subsystem failure rates are submitted in Table IV-5 and weight and power data in Table IV-6. The failure rate figures are specification values and do not reflect actual system failure history. What little data is available at present would indicate that these figures are conservative and system reliability has proved to be better than expected. No attempt was made to document probabilities of success or crew safety ^(P)s from these failure rates since it depends entirely on the duty cycle. Assuming the worst case; all systems operating continuously, which is not far from Apollo Lunar mission conditions, the system failures can be briefly summarized for G&N as .901 Failure/1000 hours or 1 Failure/46 days; for SCS 1.287 Failures/1000 hours, or 1 Failure/32.5 days; and for telecommunications .14 Failure/1000 hours, or 1 Failure 297 days. For all three systems together the failure rate is 1 Failure/18 days or 20 Failures/year.

Looking to Table IV-6 and using the total weight of 865 pounds for all three systems, dividing it by the total number of components, 44, the per component weight averages out to 20 pounds. Based on the failure rate of 20 per year this results in a resupply weight of 400 pounds per year. This is nearly half the weight of the system. The estimate is gross, of course, in that it assumes all components of like reliability and no change in reliability with time. Nevertheless, it does provide a ball-park figure for the maintenance penalties associated with Apollo Block II Electronic systems.

TABLE IV-4 - FUNCTIONAL PATH COMMONALITY

IV-11

<u>Component</u>	<u>G&C</u> <u>Functional Path</u>	<u>Quan.</u>
<u>G&N</u>		
PEA	Accel. Electronics	
	Binary Switch	4
	DC Diff. Amp. & PVR	4
	AC Diff. Amp. & PVR	3
	PIPA Cal. Module	3
PSA	Gimbal Servo Amps	3
	OSS Motor Drive Elect.	4
	ISS 800 Hz, 1% Pwr. Supplies	2
	ISS 800 Hz, 5% Pwr. Supplies	3
CDU	Quadrant Selector	5
	MSA & Quad. Rejection	5
	Read Counter	5
	D/A Converter.	5
	Error Angle Counter & Logic	5
	Coarse System	3
DSKY		2
<u>SCS</u>		
EDA	Rate Display Elect.	6
	Att. Error Display Elect.	5
	Att. Display Elect.	6
	Gimbal Position Display Elect.	2
RJEC	RCS Driver Elect.	16
TVSA	TVC Elect.	4
GA	Gyro Elect.	6
FDAI		2
Rot. Cont.		2
<u>TELECOMMUNICATIONS</u>		
HEADSETS	Earphone	3
	Microphone	3
AUDIO CENTER	Microphone Amplifiers	3
	Earphone Amplifiers	3
	Voice Operated Switch	3
PREMOD PROCESSOR	Biphase Modulator	2
	Bias Amplifier	2
VHF AM	Transmitter (less osc.)	2
	Receiver (less osc.)	2
POWER AMP	Power Supplies	2

TABLE IV-5
FAILURE RATE SUMMARY

		Failure Rate (·) x 10 ⁻⁶	
		<u>Oper.</u>	<u>Standby</u>
<u>GGI</u> (MIT, 1967; Carpenter, 1969)			
ISS			
IMU & PIPA Elec. Assy.		129.0	10.2
PSA		110.0	6.3
CDU		155.0	
OSS			
CDU		97.0	
PSA		77.0	
Optical Assy.		94.0	
Misc.			
DSKY		2.3	
D&C		2.3	
Computer		<u>235</u>	<u>60.5</u>
Total		901.6	77.0
<u>SCS</u> (NR, 1968a)			
RJEC		41.5	
ECA		54.7	
TVSA		57.0	
EDA		201.0	
GDC		144.0	
GA 1		197.5	
GA 2		197.5	
FDAI 1		143.6	
FDAI 2		143.6	
GP/FPI		48.8	
ASCP		24.3	
Rot. Cont. 1		15.0	
Rot. Cont. 2		15.0	
Trans. Cont.		<u>3.5</u>	
Total		1287.0	
<u>TELECOMMUNICATIONS</u> (Carpenter, 1969)			
VHF			
VHF Triplexer		.1	
VHF AM Trans/Rcv		15.2	
Digital Ranging Gen.		17.8	
S-Band			
USBE PM Transponder		19.4	
Power Amplifier		12.1	
Premodulation Processor		6.6	
Misc.			
FM Transmitter		6.8	
PCM Telemetry		1.1	
Data Storage Equip.		45.0	
Audio Center		5.1	
UDL		10.7	
CTE		1.0	
SCE		N/A	
Rend. Radar Transponder		N/A	
Total		<u>140.5</u>	

TABLE IV-6

WEIGHT & POWER

SCS (NR, 1968a; Honeywell, 1968a)

Component	Weight (lbs)	Max. Power (Watts)	
		115 VAC	28 VDC
GA 1	22 1/2	35	60 (30), after
GA 2	22 1/2	35	60 (30), warmup
GDC	25	60	50
EDA	24	35	50
ECA	16	14	14
RJEC	20 1/2	4	2
TVSA	12 1/2	22	13 (peak-60)
ASCP	3 1/2	-	-
TC	5 1/2	-	-
RC 1	10	-	-
RC 2	10	-	-
FDAI 1	11	-	-
FDAI 2	11	-	-
GP/FPI	3	-	-
Total	197 lbs	205 watts	249 watts

G&N (MIT, 1967)

Component	Weight (lbs)	Operate Pwr-28 VDC	Stdby Pwr - 28 VDC
CMC	70	110	10
CDU	36		
PSA	50		
IMU	42 1/2	287	43
Optics	56	123	-
Misc. (12 items)	130	25	-
Total	385 lbs	545 watts	53 watts

TELECOMMUNICATIONS (COLLINS RADIO, 1966)

Component	Weight (lbs)	Max. Power	
		115 VAC	28 VDC
Audio Center	7.9	-	14.0 watts
PMP	11.3	-	8.5
VHF AM Tx/Rx	13.5	-	28.0
VHF Triplexer	1.7	-	-
S-Band Equip.	28.0	34	2.4
S-Band Pwr Amp	32.0	180	5.0
PCM	44	21	2.2
CTE	20	-	20
Up Data Link	21	-	9
SCE	33	-	35
R.R. Xponder	14	-	84
DSE	40	35	-
Dig. Rnging Gen.	7	-	25
Total	283.4 lbs	270 Watts	233.1 Watts
TOTAL ALL 3	865 lbs	1.5 KW	

V. IFM REQUIREMENTS

Approximately fifty studies and reports related to maintenance, reliability and electronic design were reviewed during the first six weeks of this study. It was determined that existent criteria pertaining to maintenance either tended to be human factors oriented or merely constitute mission constraints and considerations. Historically, equipment designers have often ignored human factors criteria. In addition they generally are not mission oriented. They need design oriented criteria. A forcing function is required to assure that performance requirements developed in equipment specifications are initially compatible with mission requirements and remain compatible as equipment design and mission definition are refined. The experience of our study team corroborates these statements.

In order to develop specific design criteria for In-Flight Maintenance (IFM) it was necessary to first develop and refine the mission constraints and operational assumptions. General assumptions and constraints such as the resupply interval of 90 days and a high humidity shirtsleeve environment were identified at contract go-ahead. More specific items such as remove-replace only, and no powered disconnection of modules were identified during the first three months of the study. A complete list of the ground rules and assumptions is provided later in this section.

The development of mission ground rules not only permitted the development of criteria, but also served to identify and define salient points to study. For example, establishing that maintenance would be limited to the remove-replace level identified the need to concentrate on fault isolation at that level. It was concluded that program and mission ground rules should be documented and continuously refined while serving as a baseline requirements document until the equivalent of a mission flight plan is generated.

At the time the flight plan and resulting operational requirements are identified, design oriented criteria can be developed that will assist in translating from a mission function to specific equipment requirements. However, it is still imperative that the document used as a baseline for requirements, whether called Mission Ground Rules or other, be maintained and reviewed to assure compatibility between the ultimate equipment definition and the functional purpose of the equipment, namely, mission satisfaction.

As the design criteria was developed it became apparent that packaging design is a key element in achieving an in-flight maintenance capability; another key element is the capability to fault isolate. The significance of commonality and redundancy at remove-replace level and at module element level was also recognized. Criteria for each of these four areas was developed. These criteria summarized below are documented later in this section.

Packaging - 21 items

Malfunction Detection, Isolation
and Verification - 20 items

Redundancy - 9 items

Commonality - 3 items

In establishing the ground rules and criteria discussed above two types of analysis were found necessary to enhance in-flight maintenance and extend mission life: 1) A systems analysis that relates mission function to specific equipment satisfaction, and 2) A design analysis to optimize design within given constraints, such as electrical power and resupply weight. Various techniques for documenting a systems analysis were examined. It was found necessary to first identify, in depth, the purpose or objectives associated with a system analysis. Hence, criteria for analysis were developed. A complete list and discussion follows later.

With this criteria as a guide a method for the systems analysis identified above was developed. This technique, called a Functional Loop and Alternate Ways (FLAW) analysis, is discussed in detail later, together with an example.

The FLAW analysis is intended to identify equipment requirements and establish a design priority for equipment constraints. The individual equipment designer, however, will still have a degree of latitude within system constraints and program design criteria that will permit many alternate ways to satisfy requirements. The rationale for finally selecting one specific design requires justification. For this purpose a matrix was developed during this study to permit rapid examination of why a specific approach was developed. It permits a sensitivity analysis to be performed while varying the way in which each consideration is satisfied. It readily illustrates the weighting considerations that were used. An example of the initial matrix is given in Figure V-1. More specific applications are given in Sections VII and VIII of this report.

Consideration	Packaging, No. Modules		Common-ality		Maintenance & Spares Level		Planned Cannibal-ization		Redundancy	
	Min	Max	Min	Max	Min	Max	Min	Max	Min	Max
System power	#								L*	H*
System Weight	L	H								
System Volume	L	H								
Malfunction Detection/Checkout	L	H	H	L					L	H
Interconnections	L	H								
Thermal Radiation	L	H								
Training, Procedures, & Tools	L	H	H	L						
Qualification Tests	L	H	H	L						
Engineering, Manufacturing, & Tests	L	H	H	L						
Spares Inventory	L	H								
Reliability of Functional Path	H	L							L	H
Impact on Redesign	H	L	L	H						
Resupply Weight & Volume	H	L							H	L
Reliability of Module	L	H							L	H

*L = low, H = high; e.g., max. redundancy in system requires more power than minimum redundancy.
#Blanks indicate effect is insignificant.

Figure V-1 Initial IFM Matrix

It was found advisable to permit variation in the content of each matrix for items such as weight, electrical power, re-supply weight, etc. For instance, a matrix for choosing a particular design to satisfy a malfunction detection or fault isolation requirement will not be traded off against the same discrete elements that are involved in choosing a design to satisfy the requirement for voice communication. Permitting this latitude also simplifies each individual matrix by eliminating considerations that are not germane to the specific trade-off.

In summary; the establishment of the program and mission ground rules, subsequent identification of specific design oriented criteria, systems analysis to relate mission functions to equipment satisfaction and a technique to optimize equipment design are the basic requirements for enhancing IFM. To be effective, these ground rules, criteria and analytical techniques must be developed prior to solidifying equipment specifications and available for usage at preliminary and critical design review. The IFM requirements and expected usage or purpose of IFM analysis should be identified in the Request for Proposal (RFP). Further, the RFP should direct the Contractor to identify, in his proposal, how the IFM analysis will be accomplished.

IFM Ground Rules & Assumptions

The following ground rules and assumptions were developed during the study, and formed the base upon which all results rest.

1. FMEA will be accomplished at lowest reaction level.
2. Constraining envelope is the Space Station not the CSM.
3. Assume shirtsleeve environment.
4. Identify EVA requirement, if it exists, for maintaining subsystems.
5. Consider active and passive (trends and attendant characteristics) checkout.
6. Classify effects. Category I, injury or mission abort; Category II, unscheduled resupply; Category III, mission degradation.
7. Identify single malfunctions and those causing a condition of one malfunction removed from Category I or II.
8. Guidelines and criteria developed are intended for use at PDR's and CDR's. Not the detailed level of knob size, toggle switch throw angle, etc.

9. Assume single depth mounting.
10. Assume functional requirements and operational times for Station are same as CSM.
11. MSC will provide expected life on existing equipment at R/R level.
12. Analysis of subsystems (sort capability) will identify most critical effect, associated critical mission phase, and permissible downtime for the most critical phase.
13. Only dynamic failures will be considered. No hardware failures.
14. No powered disconnection will be made.
15. No design margins will be identified on existing equipment.
16. All hardwiring exposed to cabin atmosphere is covered and not available as test points. Switches cannot be jumpered.
17. Treat a switch as a plug-in unit with hi-humidity connector.
18. Block permissible and expected downtimes.
19. Families will be identified consistent with packaging concepts.
20. Trade-off Priority: Power, Weight, Volume.
21. No repair will be considered: No soldering, no welding, etc.
22. Consider concept of cannibalizing "once-used" equipment as part of commonality.
23. Displays and Controls will be studied to the extent necessary to determine how or if a malfunction is detectable via one of them. Failures of the D&C's will not be examined.
24. Mission phases to be analyzed for effects are: rendezvous, dock, maneuver, critical experiment activity, operating station (OS) manned, OS unmanned and EVA activity. Recommended redesign will recognize the need to survive the ascent phase.
25. A P of .999 will be used in establishing expected time to failure for Category II malfunctions.
26. One-away from a Class II condition or severe degradation will require a resupply.
27. Identify the best solution considering the system, not just the end item.
28. Tolerances will identify NEED, not capability.
29. Assume 1972 Technology for redesign.

30. Maintenance equipment identification will be limited to tools for R/R.
31. Skill/experience requirements will not be identified. Maintenance is limited to R/R.
32. Display and controls (D&C) requirements will be identified as related to maintenance for the areas of redesign. Any D&C criteria will be discussed under monitoring and checkout criteria.
33. Accessibility requirements will be discussed under packaging criteria.
34. Checkout interfaces will be discussed under monitoring and checkout.
35. Checkout procedures will not be identified. Maintenance is limited to R/R and fault isolation capability is at R/R level.
36. Recommended test planning, qualification and checkout will be limited to the high humidity connector discussion.
37. The engineering breadboard/mockup will not be functional but will demonstrate principle of interconnection and the size of module that can be expected.

Packaging Criteria

Packaging is an extremely important aspect of the system design. The packaging approach may or may not affect the realization of the required electronic performance specifications, but will impact system cost, weight, volume, reliability and repairability. The following criteria should be used for selecting a packaging approach:

1. Provide direct access to R/R items, cabling, and back panel wiring without electrically disconnecting other equipment. This may be accomplished by pulling out, slide out or multi-hinged devices.
2. Provide the extended volume necessary for remove-replace activity considering the volume occupied by the operator as well as the volume required by the equipment and of its moving parts such as doors and slides.
3. Use captive fasteners or captive locking devices for all remove-replace items. Preferably one type of fastening device will be used for all such items. Finger operation is desirable, and many operational cycles must be possible without significant degradation.

4. Provide containment devices necessary to survive the ascent phase that can be removed during activation of the orbiting vehicle, or at first failure, permitting R/R without tools for the duration of the orbital phase.
5. Provide identification or coding of modules to assist in rapid and correct removal and replacement.
6. Provide asymmetric mounting features, keys or guides at each level of assembly for equipment requiring specific installation orientation. This technique must also preclude the possibility of interchanging electrically non-interchangeable modules.
7. Permit remove-replace of a failed item without interrupting a redundant functional loop or another mission-critical function.
8. Permit direct access to all adjustments and controls necessary for system operation, malfunction detection and isolation, and checkout. If it is not feasible to mount the control or adjustment on the front panel or module container, the access path will be protected to preclude equipment damage when inserting an adjustment tool.
9. Permit plug-in module replacement without disconnecting fluid couplings in coolant lines.
10. Provide storage and containment for anticipated portable test tools. Placement of such devices will consider operator orientation and location of usage.
11. The coolant system will be capable of maintaining the temperature of the electronic equipment within operating requirements under all conditions between A) peak load requirements, all redundancy being used and maximum operating equipment load, and B) minimum load requirements, all redundancy failed or off and minimum operating equipment load.
12. Provide redundant, or prime, and backup cooling capability for mission critical equipment. Recognize that the coolant system serves the same function as support electronics: It supports the operating electronic equipment and its failure can result in significant equipment loss or inoperability for the period required to effect repair.
13. Provide connectors and circuitry capable of withstanding a high humidity environment, with its attendant contamination, and identify a satisfactory method of purging a contaminated

item to permit restoration to useful service.

14. Consider expected mission life to failure when determining acceptable leakage rates for hermetically sealed modules. Recognize that packaging for minimum leakage rate possible opposed to leakage rate allowable may have significant impact on cost and weight.
15. Provide containers or enclosures, to be located within pressurized compartments, that will structurally be capable of withstanding the greatest decompression rate expected, considering normal operation as well as emergency conditions such as meteoroid penetration.
16. Provide for a flexible maintenance capability, recognizing that maintenance activity for long-duration missions will be evolutionary.
17. Provide packaging and interconnection design, such as modular cabling, that will permit in-flight modification or retrofit as the technology and mission objectives change. Technology advancement tends to obsolete electronic equipment in three years, and while this does not necessarily demand retrofit or replacement, it is quite likely that retrofit will be desirable after five years.
18. Provide separate packaging for equipment or electronics that have significantly lower predicted or proven reliability than the other equipment in a functional loop. Package weak links separately and provide redundancy or sparing as necessary.
19. Provide integral lighting so that all R/R activities can be accomplished without auxiliary or portable lighting.
20. Module size selection will consider reliability, malfunction detection and isolation complexity, and resupply weight and volume.
21. Selection of control vs. local spare storage or combination thereof will consider system weight and volume, inventory control, system downtime, and handling techniques.

Malfunction Detection Isolation and Verification (MDIV) Criteria

For the long life space station application, repair will be necessary, creating a requirement for MDIV. The following criteria will be applied in selecting the MDIV system:

1. Provide MDIV to the remove-replace module level.
2. Provide verification that stimulus or a command has been received by the equipment interrogated. Monitoring or measuring only the equipment response will not provide adequate fault isolation when test stimuli are used for checkout.
3. Prevent error propagation or damage to the MDIV system from monitored or interrogated equipment and vice versa.
4. The MDIV system shall be capable of verifying that the system will not damage a replacement module.
5. Provide self-check capability in the MDIV equipment. Self-check will also be considered for operational equipment instead of using external test stimuli.
6. Provide verification of all paths where redundancy is used internal to a module.
7. Be capable of monitoring all critical functions concurrent with performing malfunction detection of a failed item or when performing periodic checkout. This may result in an independent MDIV technique being used for caution, warning and detection/isolation of those failures that can jeopardize mission or cause an abort.
8. Provide the capability to test driving electronics up to an output device, such as a thruster or engine, without actuating the output device.
9. Provide for by-pass of a failed item where possible to preclude complete loss of a mission function; such as power amplification in a communication transmission system where by-pass will result in weak transmission but not preclude transmission.
10. Provide limit-testing to isolate potential failures prior to their occurrence.
11. Minimize the number of different stimuli used or generated by the MDIV system.

12. Where possible, use loop or functional path stimuli to permit functional checkout. MDIV equipment can provide simple signals, such as a DC voltage, to trigger an operational loop with a built-in stimuli and a built-in or centralized indication of satisfactory loop performance.
13. Utilize operational checks wherever possible for MDIV. Consumables should not be used if avoidable.
14. Use operational signals rather than special stimuli for check-out whenever possible. Signal loop gains and normal outputs with given input signals can be computed and tested prior to a mission and circuit logic; comparator devices or procedural information can be used to check out equipment or functional loops.
15. Consider monitoring attendant characteristics (for instance, temperature) for qualitative measurement. This technique can be used to sense trends and impending failure as well as conditions that tend to effect multi-signals or outputs.
16. Provide MDIV capability that is independent of ground participation. Ground participation should only be considered with regard to resupply capability at which time additional information, equipment, consumables and skilled personnel may be made available.
17. Consider the use of trend analysis. Where economically attractive, trend analysis can anticipate failures.
18. Crew safety, where applicable, shall be the overriding consideration in MDIV system design.
19. Provide a means of recording malfunction detection data to assist in failure analysis at the convenience of the mission.
20. Establish the MDIV requirements, taking into account the difference in purpose and reaction time required for the three functions of caution and warning, malfunction isolation, and system verification.

Redundancy Criteria

Redundancy implies more than one independent path for satisfying a particular function. This technique is usually applied for increasing the reliability of a system during a time when repair is not allowable. The following criteria should be used in determining the appropriate redundancy for a given application:

1. Use component redundancy when a particular component has significantly lower reliability than the other components in a functional path. Redundancy verification at this level is not required.
2. Use functional path redundancy for critical paths such as those involving crew safety. It will usually be required to verify functional redundancy prior to undertaking a critical maneuver, and this provision should be provided.
3. Provide separate remove-replace modules for each redundant path wherever practical.
4. Insure permissible reaction time in the event of failure by choosing manual vs. automatic switching or standby vs. in-line redundancy. When permissible reaction time is short, automatic switching and active redundancy are requirements; however, a manual mode can back up the automatic redundant mode with an operational constraint to "standby-for-manual" after the initial failure.
5. Provide manual takeover for emergency mode. Protect the emergency-mode device to preclude inadvertent operation and do not rely on automatic emergency-mode takeover.
6. Consider providing functional redundancy using different technologies: electrical and fluids, electrical and mechanical, etc.
7. Provide analysis of the effect of failure on support electronics, such as power supplies, and on signal amplitude or signal distortion to identify failure effect on the functional loop.
8. Provide redundant support systems such as power and cooling for critical functions.

Commonality Criteria

Commonality will be employed to reduce Qualification Tests, enable cannibalization of non-critical equipment and reduce the quantity and inventory of spares required for ground and in-flight maintenance. Candidates for commonality will be identified as part of the systems analysis, using the following criteria:

1. Use commonality when performance and environmental requirements are identical. Prime candidates are items such as power supplies, signal conditioning and signal processing (circuitry and modules) units, and displays and controls.

2. Use when performance requirements are similar and system penalty is insignificant in critical constraint areas such as power. This may result in non-use of elements of electronics within a package for one or more of the applications but will not degrade functional loop reliability and permit more extensive qualification testing on the common item because of fewer items in the system.
3. Use the following considerations when making commonality trade-offs:
 - (a) Cannibalization may be used to partially satisfy sparing requirements. This is particularly true where systems are used for limited periods of time or where critical systems can be spared by cannibalizing a non-critical system.
 - (b) Qualification costs can be greatly reduced.
 - (c) Spares inventory can be reduced.
 - (d) Checkout system complexity may be minimized.

Criteria for IFM Systems Analysis

One system analysis can serve many purposes and, just as design criteria can be customized for a particular set of mission equipment, early identification of those purposes will permit a program oriented systems analysis. Further, establishing the objectives of analysis permits identification of source data required for analysis, analytical working data, deliverable data and the data sorts necessary to obtain visibility and enhance in-flight maintenance. The analysis documentation technique should be capable of translating from working paper to data sorts and to computer application, if necessary. The outputs or results of analysis should be separated into discrete packages so that technical management, systems integration or individual design personnel can rapidly review analysis germane to their specific area. The analysis technique must be capable of translating from a conceptual phase to equipment acquisition, and relate equipment and program changes to the effect on mission functions.

The IFM Systems analysis will identify mission-critical functions and provide the following analysis information for each function.

1. Identify any mission constraints that limit or preclude using a specific conceptual way to satisfy the mission function.

2. Identify alternate ways to satisfy the function.
3. Analyze each alternate way and permit rapid translation to equipment requirements.
4. Identify the effect on the mission as a result of the loss of each one of the equipment functional requirements. Identify the mission phase associated with the mission effect, permissible downtime (if any) and categorize the effects.
5. Identify equipment, elements of equipment, functional redundancy or other means that will preclude the loss of the critical function caused by a single failure.
6. Trade-off nonfunctional requirements (operational requirements, availability, cost, etc.) and identify the technical approach to be developed.
7. Re-examine the selected approach to assure mission critical function satisfaction.
8. Perform detailed failure analysis of critical equipment elements.
9. At remove-replace level identify the types of equipment used to satisfy each functional loop and for each type of equipment identify the following:
 - a. Subsystem identification.
 - b. Cross reference to mission function.
 - c. Program identifier (i.e., part number, drawing number, control point).
 - d. Quantity.
 - e. Expected mission life to failure.
 - f. Failure/effect criticality.
 - g. Permissible downtime.
 - h. Technique for malfunction detection and fault isolation.
10. Provide the capability for arranging the information developed in items eight and nine into the following data sorts.
 - a. Name of module - Alpha
 - b. Mission function - Alpha

- c. Mission function - numeric
 - d. Program identifier - Numeric
 - e. Expected mission life to failure - Min., Max.
 - f. Failure/effect criticality - Cat. I, II, III.
 - g. Critical mission phase - ascent, rendezvous, etc.
11. Compile a list of recommended candidates for functional commonality and a list of candidates for equipment commonality at recommended remove-replace level.
 12. Compile a list of recommended candidates for equipment commonality and equipment redundancy at levels beneath recommended packaging.
 13. Subject commonality candidates, redundancy candidates and equipment whose failure will result in one-away from Cat. I or II effects to in-depth failure mode analysis.
 14. Identify how all failures are detected at remove-replace level.
 15. Identify how the failure of redundant circuits or piece-parts, resulting in one-away failure from Cat. I or II effects, is detected.

Functional Loop and Alternate Ways Analysis (FLAW)

The systems analysis technique developed during this study was called a Functional Loop and Alternate Ways Analysis to avoid pre-conceived ideas associated with Failure Modes and Effects Analysis, Fault Tree Analysis, etc. It is basically a failure effects analysis identifying ways to preclude or minimize failure effect on the mission, modules or elements requiring detailed failure modes analysis, unavoidable weak links. The technique was developed because no available or known techniques satisfied the systems analysis criteria discussed above. Boolean notation was used to permit rapid translation from top mission function to detailed hardware satisfaction, to provide data sorts to satisfy specific and different purposes, and to provide ultimately for the use of a computer.

Listed below are the steps, corresponding to the criteria discussed earlier, to be followed in a typical analysis. An example is provided wherever appropriate to illustrate the details of the technique. The sample function used for the examples is 2-way noise communication between Extra Vehicular Activity (EVA), Space Station (SS) and ground (MSFN).

1. Identify any mission constraints that limit or preclude using a specific conceptual way to satisfy the mission function.

Example -

No EVA • SS umbilicals preclude simple audio communication.

2. Identify alternate ways to satisfy the function or any critical loop.

Example -

$$*C_{EVA \cdot SS \cdot MSFN} = EVA \cdot SS \cdot MSFN \cdot SS \cdot EVA + EVA \cdot MSFN \cdot SS \cdot EVA$$

$$\text{Critical Loop} = EVA \cdot SS \cdot MSFN$$

3. Analyze each alternate way. Subdivide into sub-functions as necessary to simplify understanding and permit rapid translation to equipment requirements.

Example -

No true alternate exists. The two loop permutations will be examined by analyzing sub-functions.

$$C_{EVA \cdot SS} = \underbrace{(\text{audio} \cdot \text{TxRx}) \cdot (\text{Audio} \cdot \text{TxRx})}_{\substack{\text{a} \quad \text{EVA} \quad \text{b}}} \text{SS}$$

$$C_{EVA \cdot MSFN} = a \cdot \underbrace{(\text{Audio} \cdot \text{TxRx})}_{\text{c}} \text{MSFN}^{**}$$

$$C_{SS \cdot MSFN} = b \cdot c$$

4. Identify the effect on the mission-critical function resulting from the loss of each one of the equipment functional requirements. Identify the mission phase associated with the mission effect, permissible down-time (if any) and categorize the effects.

Example -

$$a. \text{ Loss of EVA: Audio} \neq C_{EVA \cdot SS} \cdot C_{EVA \cdot MSFN}$$

$$b. \text{ Loss of EVA: Rx} \neq C_{EVA \cdot SS}$$

$$c. \text{ Loss of EVA: Tx} \neq C_{EVA \cdot SS} \cdot C_{SS \cdot MSFN}$$

$$d. \text{ Loss of SS: Audio} \neq C_{EVA \cdot SS} \cdot C_{SS \cdot MSFN}$$

$$e. \text{ Loss of SS: Rx} \neq C_{EVA \cdot SS} \cdot C_{SS \cdot MSFN}$$

* Example uses Boolean notation.

** MSFN capability is mandatory and can serve as backup for EVA·SS communication in the event of spacecraft EVA Rx failure. Separate analysis is required.

f. Loss of SS : $Tx \neq C_{EVA.SS} \cdot C_{SS.MSFN}$

Mission Critical Phase: EVA

Effect: Terminate EVA

CAT II, may result in early resupply.
Permissible D.T. Time to next EVA

5. Identify equipment, elements of equipment, functional redundancy or other means that will preclude the loss of the critical function caused by a single failure.

Example -

$$\text{Audio}_{EVA.SS} = (\text{Mike}_1 + \text{Mike}_2) \cdot (\text{Earphone}_1 + \text{Earphone}_2) \cdot (\text{P.S.}_1 + \text{P.S.}_2)$$

$$\text{TxRx}_{EVA.SS} = \text{UHF}_1 + \text{UHF}_2 = \text{VHF}_1 + \text{VHF}_2 = \text{UHF}_1 + \text{VHF}_1$$

6. Trade-off nonfunctional requirements (operational requirements, availability, cost, etc.) and identify the technical approach to be developed.

Example -

Only SS equipment is analyzed. EVA equipment requires separate analysis.

$$\text{Comm}_{EVA.SS} = \text{VHF}_1 + \text{VHF}_2 = (\text{Rx}_1 + \text{Rx}_2)_{\text{USBE}} (\text{Tx}_1 + \text{Tx}_2)_{\text{VHF}}$$

7. Re-examine the selected approach to assure mission critical function satisfaction or critical elements and mission effect.

Example -

USBE available only when in contact with MSFN.

VHF is primary mode and the VHF Tx function is required for each loop.

$$\text{VHF}_1 + \text{VHF}_2 = \text{Ant}_1 \cdot \text{Ant}_2 \cdot \text{Ant Swt} \cdot \text{Triplex} \cdot (\text{SwtA} + \text{SwtB}) \cdot (\text{Rx}_1 + \text{Rx}_2) \cdot (\text{T}_{X1} + \text{T}_{X2})$$

$$\text{ACP}^*_1 (\text{AC}^{**}_1 + \text{AC}_3) + \text{ACP}_2 (\text{AC}_1 + \text{AC}_2) + \text{ACP}_3 (\text{AC}_2 + \text{AC}_3)$$

- - -
* ACP = Audio Control Panel • Dual mikes & Earphones

** CP = Audio Center Channel

Energy Sources and buses are triplicate.

Critical Elements: Ant₁•Ant₂•Ant Swt•Triplexer

Loss of:

a. Ant₁ ≠ Ant₂ ≠ Comm_{EVA}•SS

May require attitude maneuver to restore.

b. Ant Swt ≠ Comm_{EVA}•SS

Terminate EVA phase

c. Triplexer ≠ Duplex Comm_{EVA}•SS

Complete EVA Task & Terminate EVA.

8. Perform failure analysis of critical elements.
9. At remove-replace level identify the type of equipment used to satisfy each functional loop and for each type of equipment identify the following:
 - a. Subsystem identification.
 - b. Cross reference to mission function.
 - c. Program identifier (i.e., part number, drawing number, control point).
 - d. Quantity.
 - e. Expected mission life to failure.
 - f. Failure/effect criticality.
 - g. Permissible downtime.
 - h. Technique for malfunction detection and fault isolation.

10. Provide the capability for arranging the information developed in items eight and nine into the following data sorts.
 - a. Name of module - Alpha
 - b. Mission function - Alpha
 - c. Mission function - Numeric
 - d. Program identifier - Numeric
 - e. Expected mission life to failure - Min., Max.
 - f. Failure/effect criticality - Cat. I, II, III
 - g. Critical mission phase - ascent, rendezvous, etc.
11. Compile a list of recommended candidates for functional commonality and a list of candidates for equipment commonality at recommended remove/replace level.
12. Compile a list of recommended candidates for equipment redundancy at levels beneath recommended packaging.
13. Subject commonality candidates, redundancy candidates and equipment whose failure will result in one-way from Cat. I or II effects to indepth failure mode analysis.
14. Identify how all failures are detected at remove-replace level.
15. Identify how the failure of redundant circuits or piece-parts, resulting in one-away failure from Cat. I or II effects, is detected.

SECTION VI - REDESIGN

Our redesign utilizes the systems analyzed earlier, extrapolating their design into an in-flight maintainable system for Space Station '75. A very important point which should be a guidepost for any effort on this subject was made in D. A. Barnes' paper (Barnes, 1968), In-flight Maintenance Study for Space Stations and Spacecraft; "Extrapolation is more effective than pure prediction; therefore, we must investigate the possibility of using existing capabilities and resources "as is" or modifying them before we attempt a totally new capability." Apollo Block II systems are obvious candidates for such an approach.

A basic assumption for this redesign was that the same functions are required on the Space Station as were required on Apollo. This intentionally reduces the redesign to more of a repackaging and restructuring job than a true redesign. Some reasons are: (1) it was not only difficult but inappropriate to provide any really meaningful functional redesign for the space station within the scope and visibility of this contract, and, (2) by maintaining the same functions, many of the existing Apollo circuits and schematics could be utilized, thereby providing greater confidence in estimates of module size, shape, quantity, failure rate, fault isolation requirements, etc.

Some minor changes, which will be discussed in more detail later, were made in the hardware configuration to provide compatibility with the Space Station but were not intended to optimize it. The idea was to take a cross section of Apollo electronics capable of performing Space Station functions and reconfiguring it for in-flight maintainability in such a way that any changes in function, or any new functions added to the basic systems, such as CMG, rendezvous or balancing electronics, would not impact the basic approach adopted in the study.

Component Selection For Re-Design

In the 1975 time period, the hardware to be utilized will use essentially today's technology. Concerning components, the key decision to be made was whether or not to extensively utilize Large Scale Integrated Circuits (LSI). The decision reached was to use micro-logic and hybrid micro-circuits, Medium Scale Integrated Circuits (MSI) where they are available, and LSI only as available now for memory and computer applications.

The reason for this decision is the concern in industry today that improper application of LSI Techniques will result in extremely costly systems, hence this technology is being principally applied where a large number of circuit functions of the same type are needed, i.e., in computers, but even here a recent survey of eight aerospace computers now in development showed only two committed to LSI, and no LSI used in 56 existing aerospace computers (Baechler, 1969). Adding weight to this decision is the lack of intellectual commitment to LSI on the part of systems designers (Feth and Smith, 1968).

The start-up cost to build a typical LSI circuit has been stated as \$10,000 to \$50,000 (Leeds, 1969); this provides the masking and necessary tooling to produce one circuit. Beyond this the circuits are likely to be inexpensive, but a change of any sort in the circuit function or detail will require another start-up cost.

MSI is in somewhat better shape, wherein several commonly used functions, e.g., shift registers, adders are built into a single package and offered commercially at present. These are principally digital functions.

In linear circuits, various types of amplifiers are packaged as micro-circuits, but the more complicated or less used functions are not.

While this outlook might seem pessimistic, the packaging scheme chosen allows the incorporation of LSI should it be more broadly utilized than anticipated. This would allow more functions to be included in each replaceable module.

Where linear circuits are involved, it is often found that none of the above integrated circuits will satisfactorily fulfill the needs, hence we must utilize ordinary resistors, transistors, capacitors and the like. Capacitors in particular are space consuming, and have been found to seriously compromise the advantages offered by certain exotic packaging schemes, for instance the hybrid scheme described elsewhere in this report.

One way around the difficulties created by linear circuits is to convert all analog functions to digital as early as possible. Many functions are now treated this way at some point in the signal path, but in most instances the digitizing can only be carried back to the proximity of the sensor or data source, hence how much is digital and how much is analog becomes largely a matter of degree. Break-through in digital sensors will help, such as the laser gyro (Minor, 1969; Schappel, 1969).

There are attendant difficulties in digitizing that must be traded off, not the least of these is the fact that some computations that have been historically handled in an analog fashion are somewhat cumbersome to handle digitally and require either disproportionate amount of hardware or computer power to solve; closed loop servos and IF amplifiers for example. However, if pressure is brought to bear, many system functions could perhaps be redesigned to take advantage of MSI or LSI, notably repeater amplifiers, converters, filters, frequency synthesizers, phased-array antennas and microwave transmitter circuitry (Leeds, 1969).

A trade-off shows that we will probably wish to continue using analog circuits for at least part of the necessary functions in a spacecraft, hence any standardized packaging scheme should be able to accommodate either without compromising the effectiveness of the approach.

Present Apollo Design

A block diagram of the Block II Guidance and Control is depicted in Figure VI-1 and Telecommunications in Figure VI-2. The components, or "black boxes", comprising these three systems are provided in Figures VI-3, IV-4 and IV-5.

The Apollo Guidance and Control provides the electronics for driving the active elements of vehicle control, namely the Reaction Control jets and Service Propulsion System thrust vector and on/off control. These electronics are in the form of 2 SCS "black boxes" which accept inputs from either the Primary Guidance and Navigation System, the SCS "backup" control system, or direct manual commands from the pilot. The G&N can determine its inertial orientation and maintain it as a reference for generating control system commands derived via the G&N computer. The SCS backup control system cannot determine its orientation without ground information, but can maintain a reference for purposes of holding vehicle attitude during RCS and SPS maneuvers. It does not have a steering capability nor does it have the performance capability of the G&N. The SCS provides redundant displays essential to guidance and control as well as the electronics necessary to drive these displays.

Summarizing then, the G&N provides the sole source of guidance and the prime source of control. The SCS provides a backup source of control, all the final-stage driving electronics, plus peripheral equipment such as displays and hand controls.

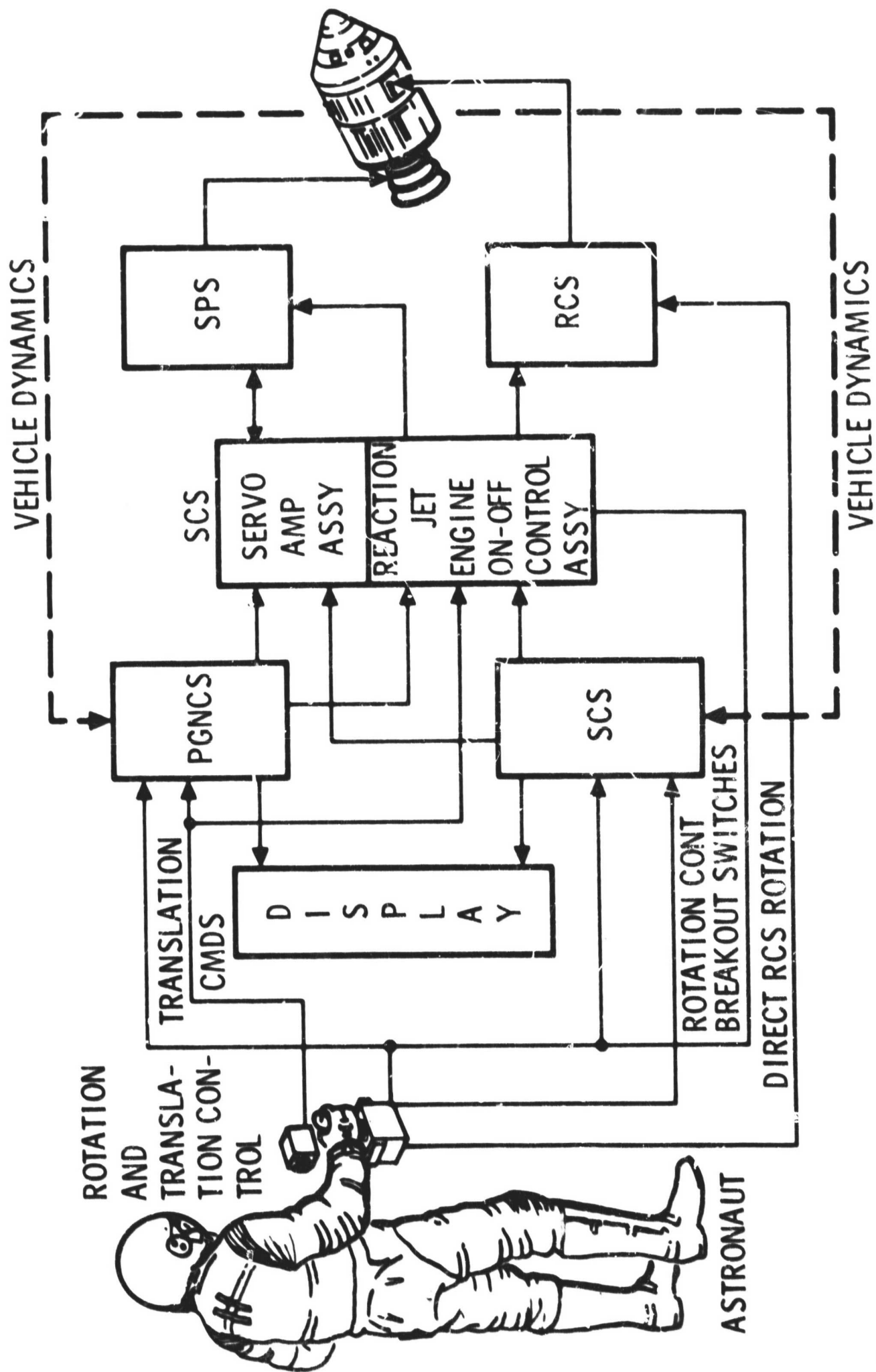


Figure VI-1 CSM Guidance & Control Block Diagram

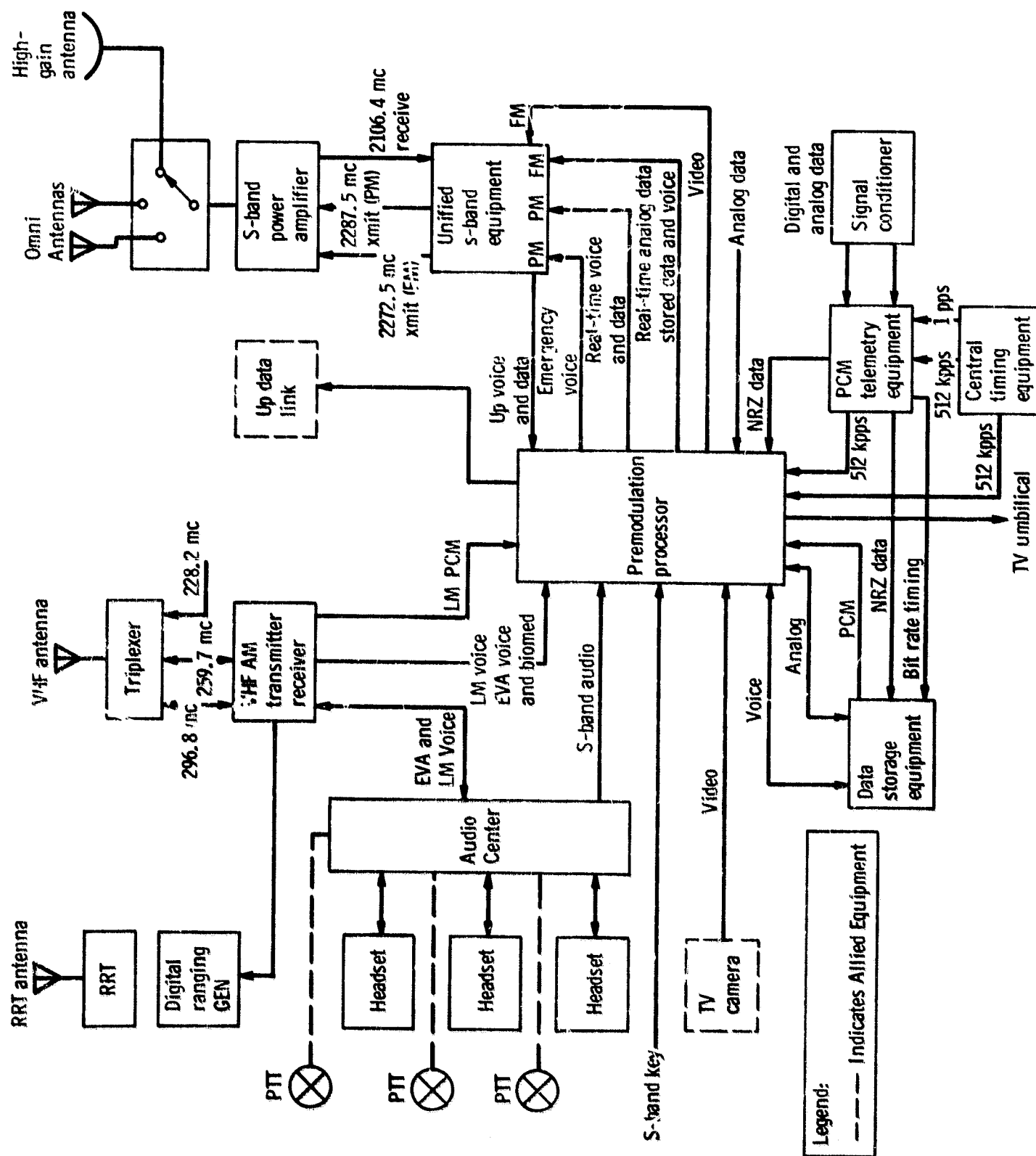


Figure VI-2 CSM Telecommunications Block Diagram

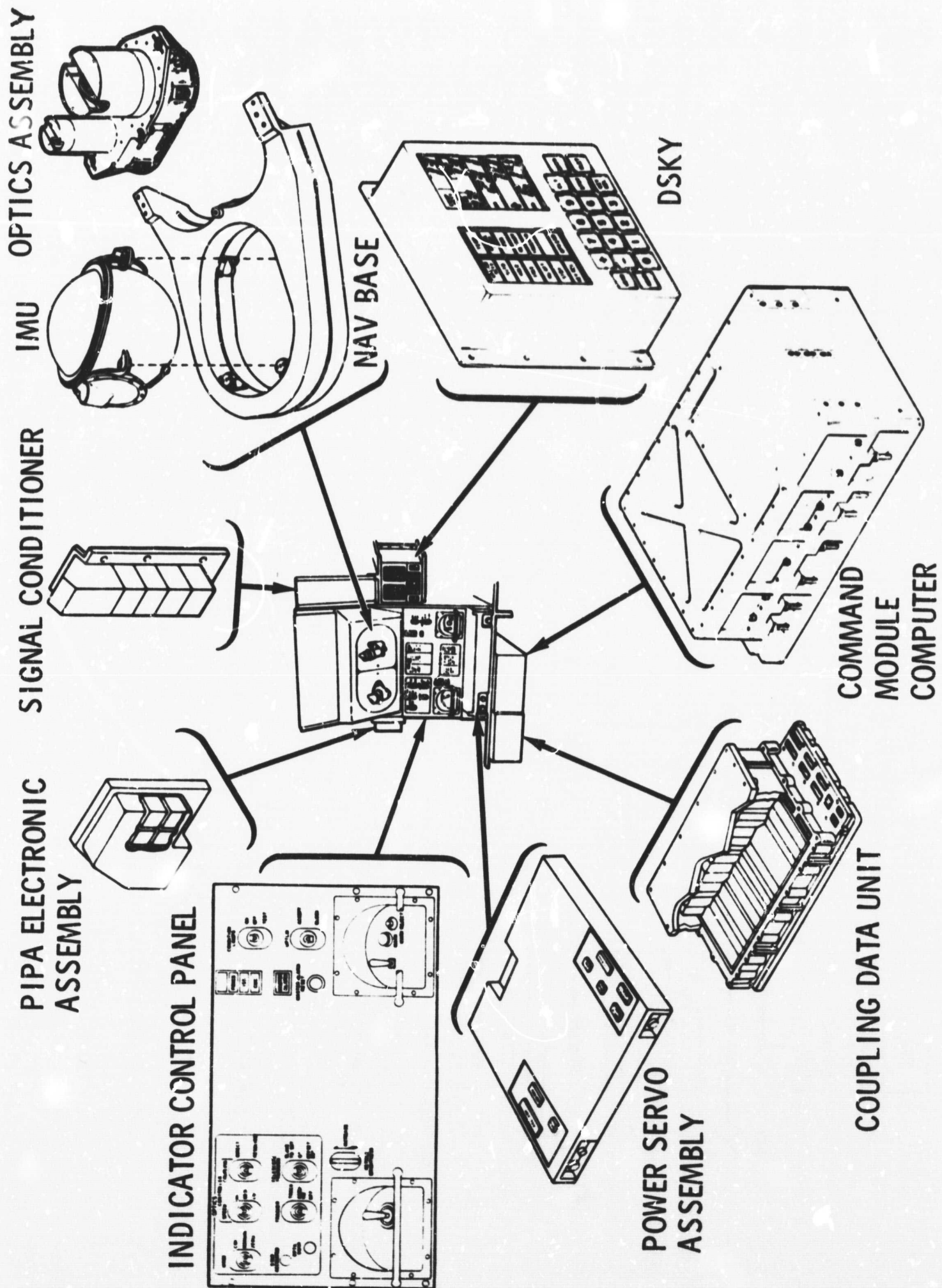


Figure VI-3 PGNC S Equipment

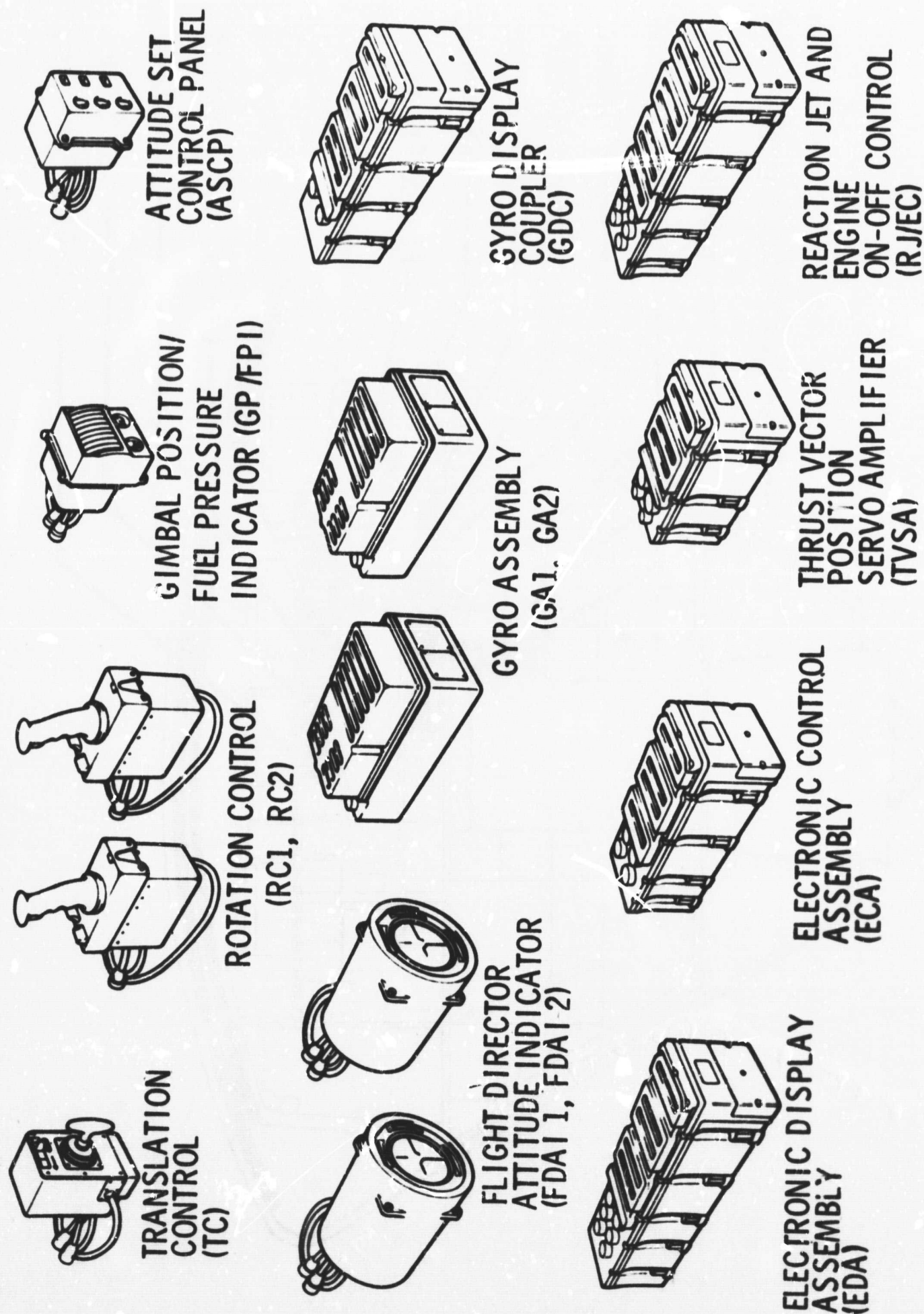


Figure VI-4 SCS Flight Hardware Block II

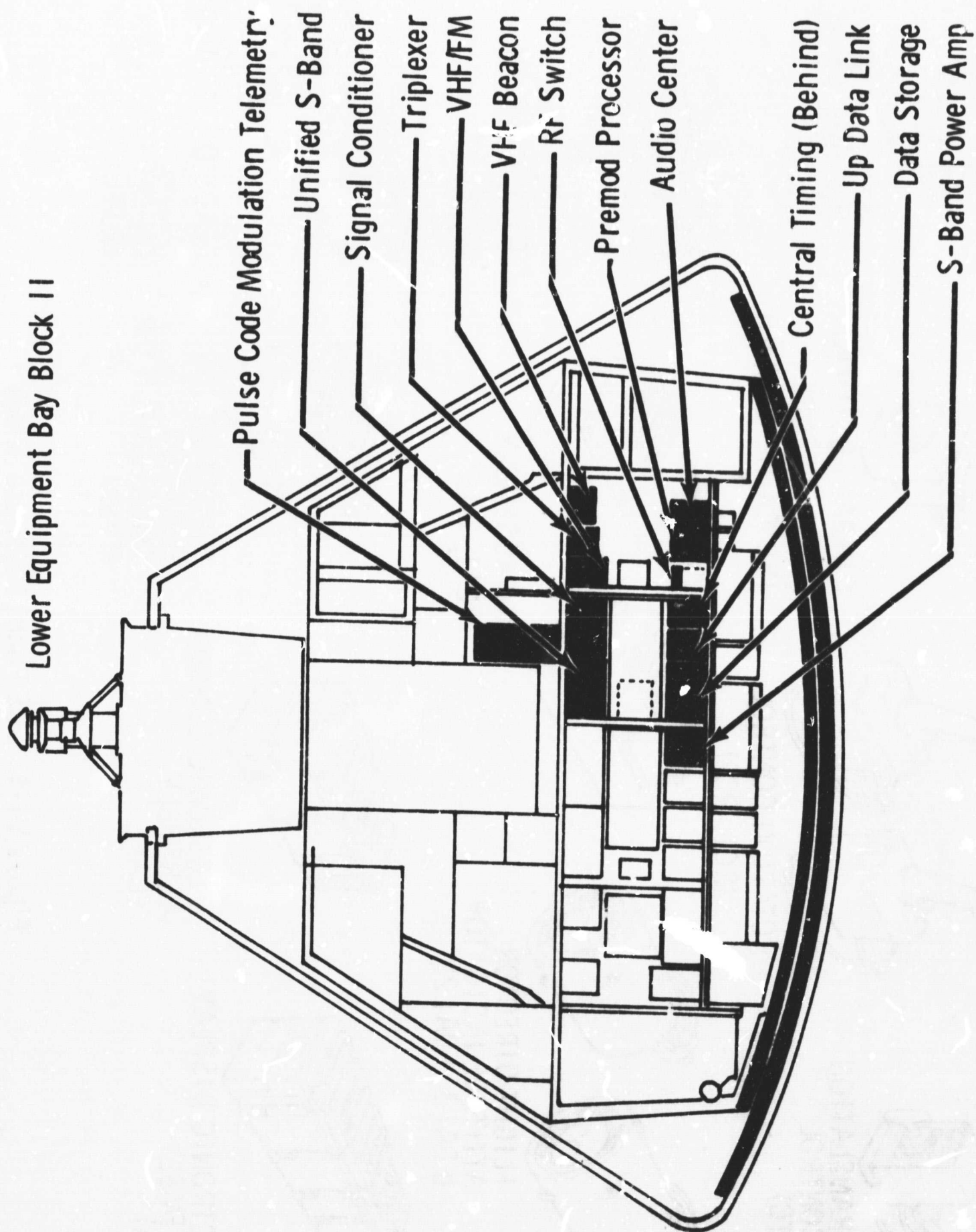


Figure VI -5 Telecommunications Equipment

The telecommunications system in Figure VI-2 is composed of a communications link using VHF frequencies for earth orbital entry communication LM/EVA communication, and a backup ranging capability. Another system at S-Band frequencies provides the same communication functions in addition to the lunar phase communications where VHF is not adequate. The S-Band system is also essential to the data transmission function. This is a major part of the Telecommunication System, involving most of the PCM, Signal Conditioner, Data Storage, Uplink and Premodulation Processor Equipment. The remaining equipment is the audio amplifiers and headsets.

Redesign

It has already been pointed out that the Space Station functional requirements for G&C and telecommunications are assumed to be the same as the Apollo functions described above. Nevertheless, some changes to the G&C were felt necessary to make it at least compatible with the Space Station program. The first was to plan on a requirement for two independent and isolated G&C systems. The Space Station RFP (NASA-1969a) stipulates in Paragraph 3.3.4 that any single compartment must be able to be isolated in the event of a catastrophe within it. "The remaining compartments will be equipped and provisioned so that the crew can safely live and take correction action or return to earth." There undoubtedly are several ways of satisfying this requirement, however, for this study we assumed two identical systems, an approach which has distinct advantages such as maximizing commonality and cannibalization. Also, the performance of the backup system is not degraded from the primary, and maximum power down of a "down" system is permitted during repair.

It would be reasonable to assume that even with the redundancy of two identical systems, each system must have some manual means of backup control for critical or crew safety operations. As on Apollo, critical failures must be reacted to with virtually no down-time. Aside from these special cases a down-time for repair on the order of an hour could be tolerated and repair completed within an hour as well. In addition, the availability of an identical system for operation while one is down would seem to indicate that down-times of considerably more than an hour would not create any serious impact.

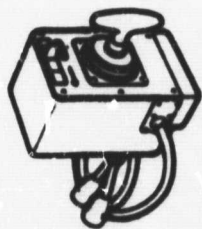
With these considerations in mind it was a relatively straight-forward process to sort out Apollo circuitry that would perform the required functions within the envelope of Space

Station '75. The SCS backup control capability was deleted, since sufficient redundancy exists in the two independent systems. Each of the two systems includes a complete G&N system plus the functional equivalent of the SCS components pictured in Figure VI-6; SCS driver electronics (RJEC, TVSA, 1/2 of the EDA), a rotation and translation control, displays (FDAI & GP/FPI) and a gyro package for rate sensing and display. A block diagram of the new system is provided in Figure VI-7.

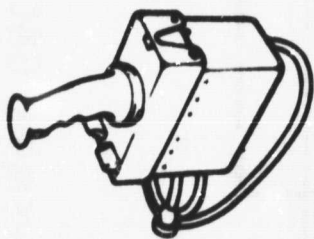
In Figure VI-7 an executive computer has been shown at the bottom of the figure. This was provided because of the likelihood of such a device being required on space station in some form or another. Its function would primarily be to monitor the system performance and vehicle response for the purpose of transferring to a backup or alternate mode of operation in the event of a failure. Its output might be no more than a switching function such as is represented in simplified form on the figure. It is, however, a function that could be performed by an alert and continuously on-station crew, therefore, for the purposes of this simplified redesign, the executive computer was not considered.

The Telecommunication System was considered unchanged from Apollo though the electronics location could be split up to insure that loss of a module does not lose all communication. A factor which was ignored in this redesign is that the data processing functions on Space Station will be considerably increased. On the other hand, they will very likely become part of a separate Data Management System; therefore, details on that part of the telecommunications have been minimized in the redesign.

The next step was to place the components of each of the systems in proper perspective with respect to location. In Figures VI-8 and VI-9 all those components which are finitely defined in size and shape as well as location, such as antennas, center-of-gravity oriented sensors, and controls are shown on the exterior of what is referred to as the electronics bay. It is the equipment within this bay which is highly flexible, not only in how it can be packaged, but in location as well. Consequently it was the real subject of this redesign for in-flight maintainability. This is not to imply the sensors, controls and antennas located on the structure are not remove/replace maintainable, however, it is not an electronics packaging redesign as much as a redesign in the mechanical securing and aligning of the components. The problems in this area are not few in number and in all fairness warrant a study by itself.



TRANSLATION CONTROL
(TC)



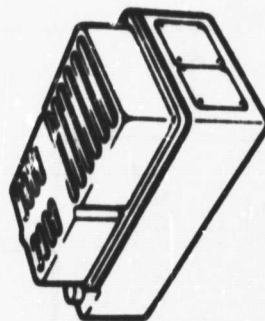
ROTATION CONTROL
(RC1)



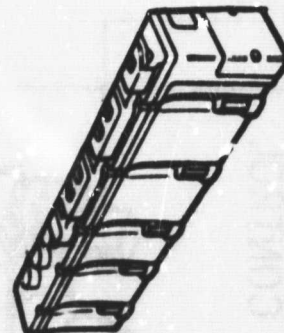
GIMBAL POSITION/FUEL
PRESSURE INDICATOR
(GP/FPI)



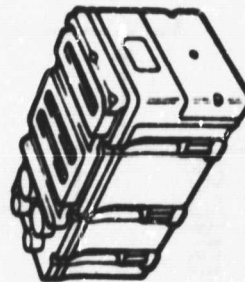
FLIGHT DIRECTOR ATTITUDE INDICATOR
(FDAI 1)



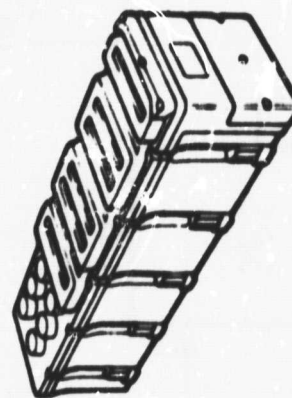
GYRO ASSEMBLY
(GAI)



ELECTRONIC DISPLAY ASSEMBLY
(EDA)



THRUST VECTOR POSITION
SERVO AMPLIFIER
(TVSA)



REACTION JET AND ENGINE
ON-OFF CONTROL
(RJ/EC)

Figure VI-6 SCS Flight Hardware (Redesign)

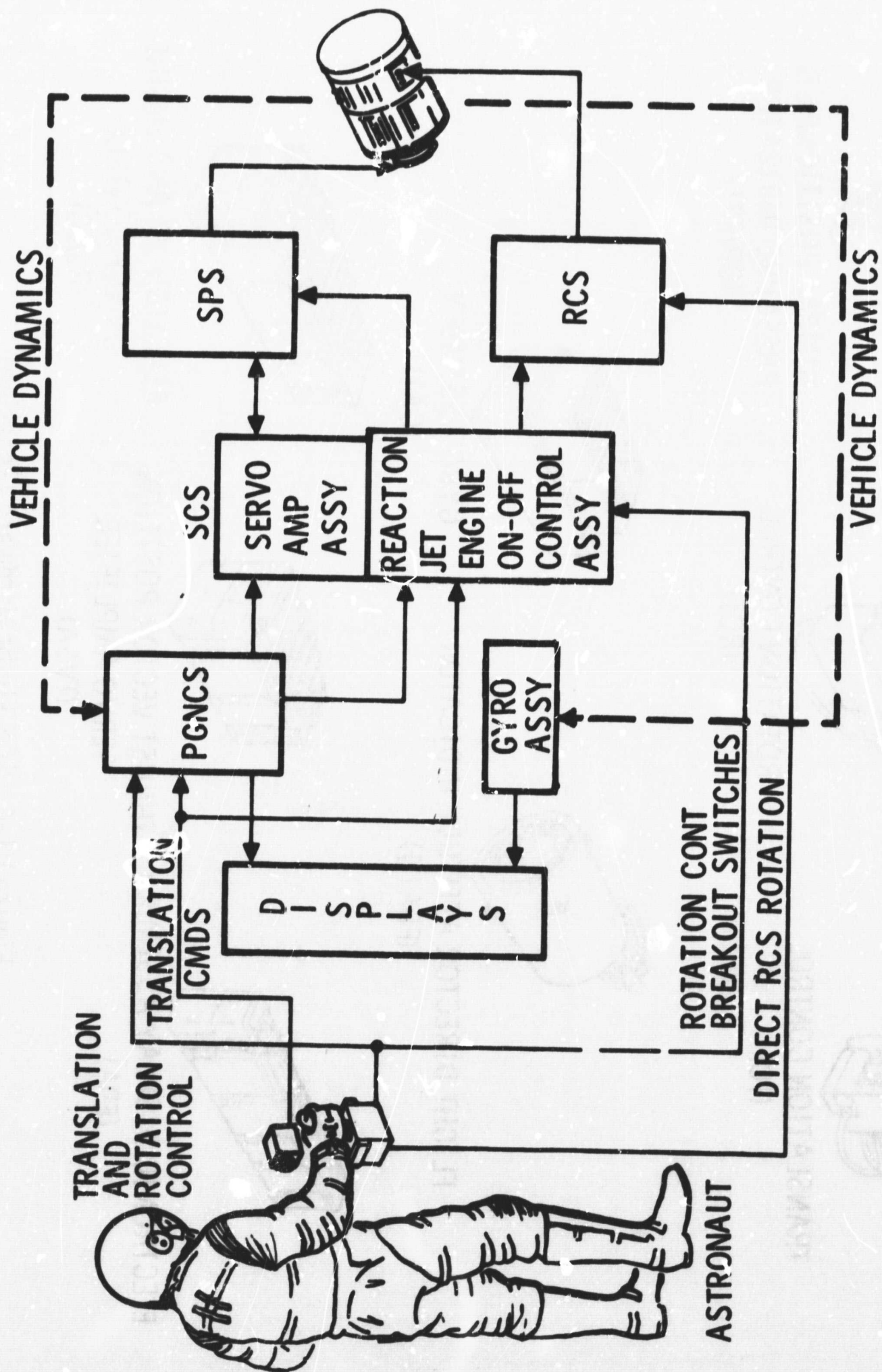


Figure VI-7 Redesigned G&C System (1 of 2)

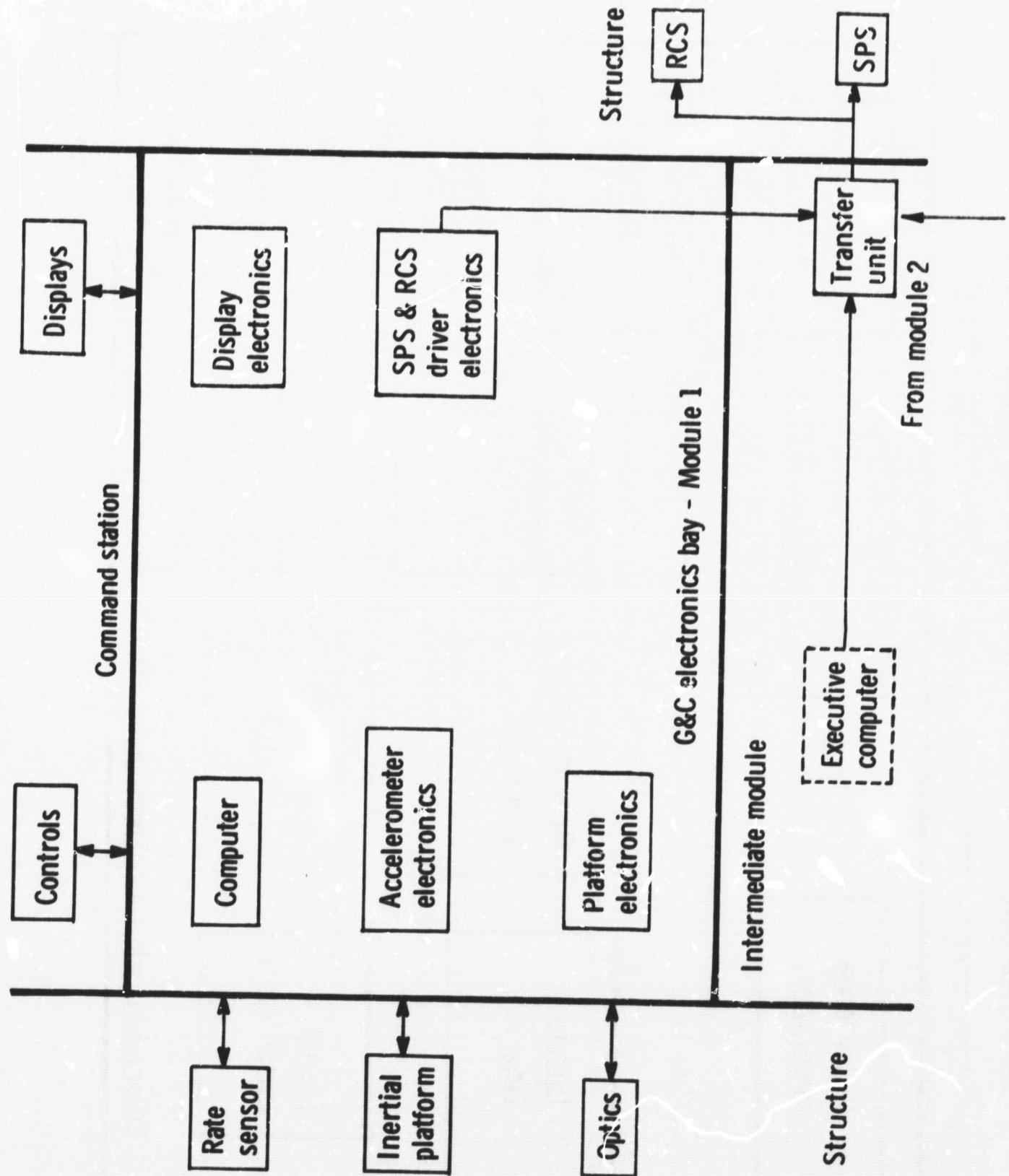


Figure VI-8 G&C Equipment Location (1 of 2 Systems)

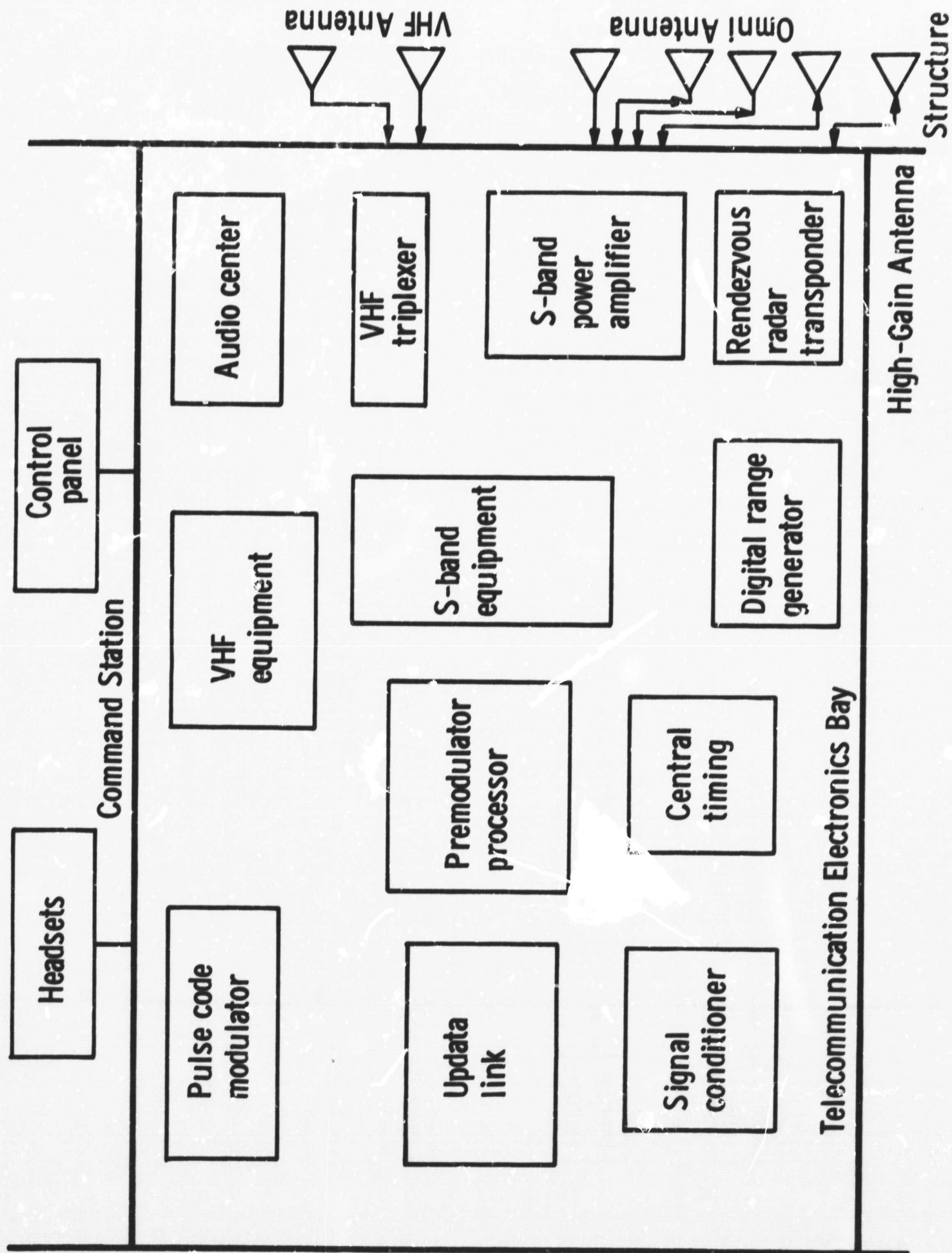


Figure VI-9 Telecommunications Equipment Location

The final decision required to complete definition of the redesign was - Just how should the electronics in those respective electronic bays be configured to optimize maintenance? There are many different approaches feasible such as packaging for a standard module size, or minimum weight and volume, or minimum resupply weight, or maximum environmental protection as on Apollo. With respect to in-flight maintenance, however, it was decided a key consideration in layout of the electronics was optimum fault isolation, therefore it served as the principal criteria for the first cut at hardware configuration. Optimum is defined as a fault isolation scheme that utilizes a minimum of fault isolation test points and a minimum complexity in the fault isolation equipment yet still isolates any failure to a remove/replace level with a high degree of confidence. Another guideline followed that is directly related and compatible with optimum fault isolation was to maximize the functional path commonality tabulated earlier in Table IV-4. Wherever these criteria still afforded a choice the package was defined as that size that was most convenient for single-hand remove/replace activity. This is discussed in more detail later in this section.

Having performed this first cut at packaging based on fault isolation the resulting configuration was then traded off and modified to comply with all other packaging constraints in the IFM packaging matrix, Table VII-1, such as environment, cooling and connections. The result of this iterative process is typified in Figures 10 and 11. Each box represents a single electronic module. Each module is described in more detail in the Replaceable Parts List, Tables VI-1 and VI-2. The original Apollo G&C Circuitry included in each respective module is indicated in the remarks column of Table VI-1. There are a total of 47 electronic modules in G&C which, together with the sensors, etc., come to a system complement of 61; or 122 per space station. The corresponding total for Telecommunications is 199.

At this point a final attempt was made at increasing commonality in the system. One method is to find similar but not identical modules in the respective component list that can be packaged as a common item. Those parts of the circuits which are unique to the one location and not the other are wired to the connector such that they are active only when in the proper location. It seems reasonable that this common packaging is practical only when the common elements of the circuits approach something more than 50% and ideally more than 75%. Only one commonality candidate in the G&C system was identified in this category. The Rate/Att. Error Display Electronics module (Component #12) of which 2 are required, perform 2 different functions yet are interchangeable.

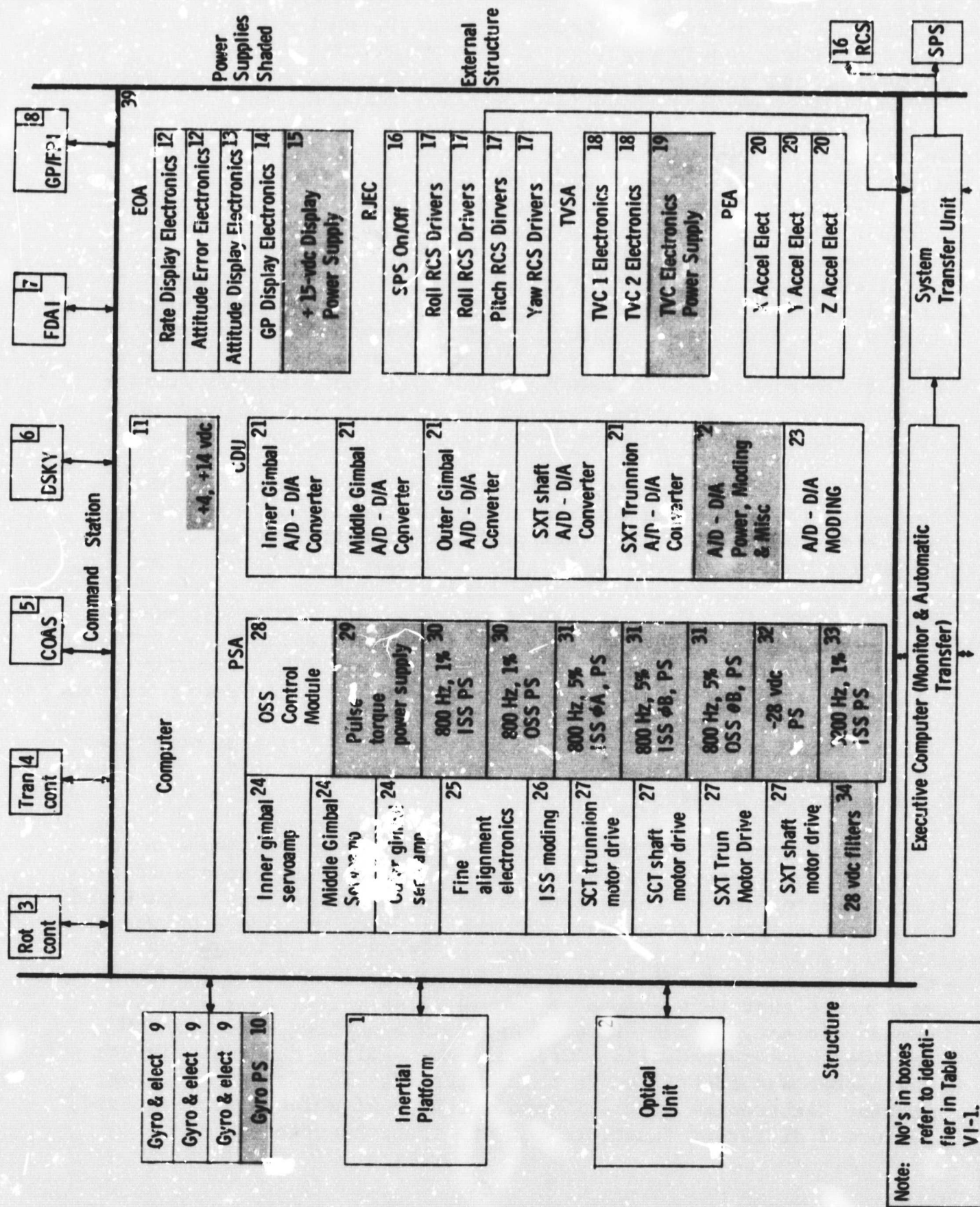


Figure VI-10 Guidance & Control R/R Modules

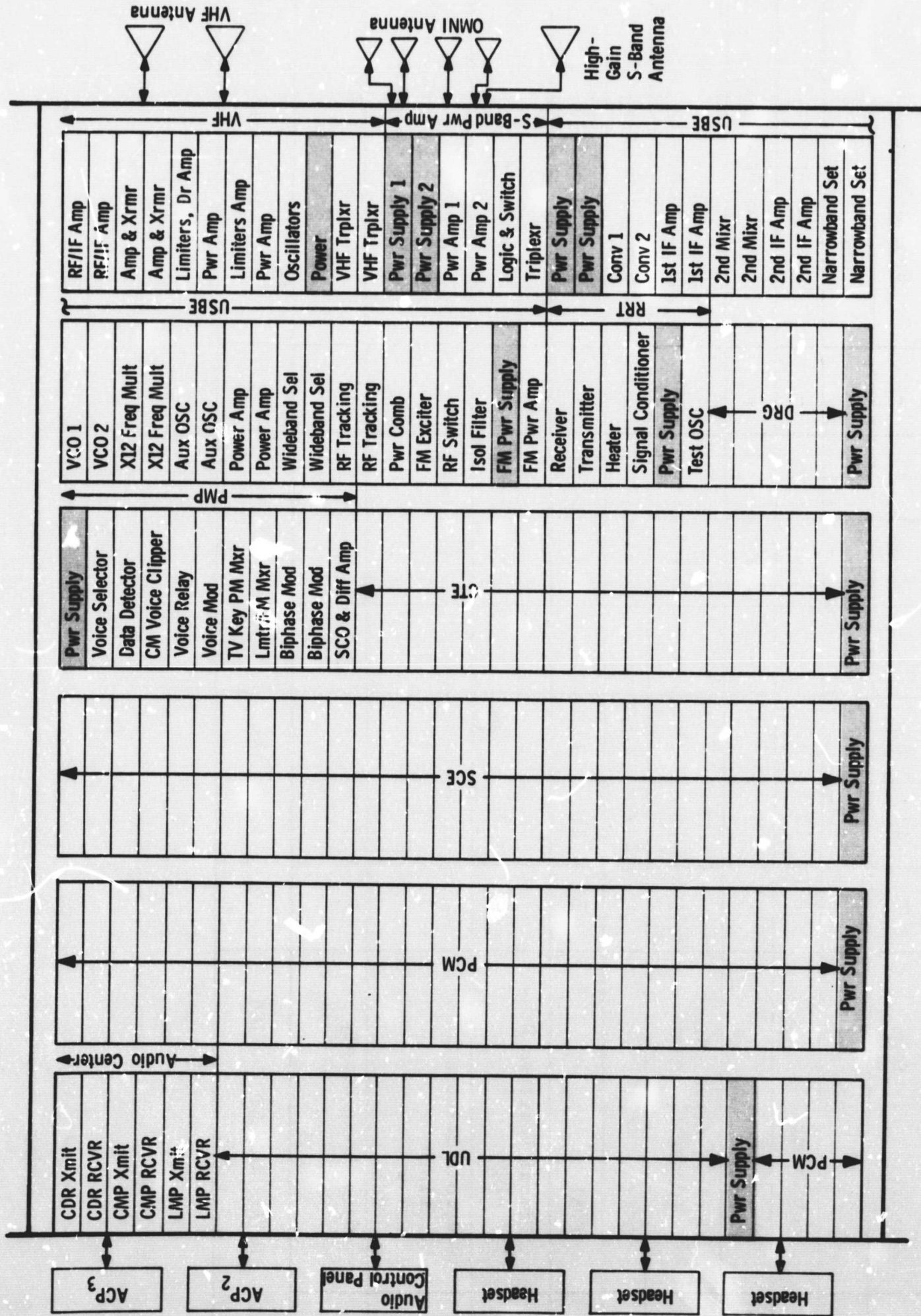


Figure VI-11 Telecommunications R/R Modules

TABLE VI-1

G & C REPLACEABLE PARTS LIST - NON-ELECTRONICS

VI-18

NO.	COMPONENT/MODULE	QUANT.	WT. EA.	WT. TOT.	BDS/ MOD.	TEST POINTS / MOD.			REMARKS
						CMC	ACT.	PASS.	
1.	Inertial Platform	1	42	42	-	-	18	-	Same as Apollo
2.	Optical Unit	1	56	56	-	-	42	-	Same as Apollo
3.	Rot. Cont.	1	10	10	-	2	-	-	Same as Apollo
4.	Trans. Cont.	1	5	5	-	2	-	-	Same as Apollo
5.	Optical Align. Sight. (COAS)	1	11	11	-	-	-	-	Same as Apollo
6.	DSKY	1	17-1/2	17-1/2	-	-	-	-	Same as Apollo
7.	FDAI	1	11	11	-	1	18	-	Same as Apollo
8.	Gimbal Position Display	1	3	3	-	-	8	-	Same as Apollo
9.	Gyro & Elect. Pitch Yaw Roll	3	6	18	-	-	3	-	Each Gyro packaged with its own electronics
10.	Gyro Power Supply	1	1	4	1	-	-	1	+32, +20, -15 VDC
TOTAL/NON-ELECTRONICS		12	-	166	-	-	-	-	Avg. wt./module = 14 lbs.

TABLE VI-1 - continued: G & C REPLACEABLE PARTS LIST - ELECTRONICS

NO.	COMPONENT/MODULE	QUANT.	WT. EA.	WT. TOT.	BDS/ MOD.*	TEST POINTS / MOD.			REMARKS
						CMC	ACT.	PASS.	
11.	Computer	1	70	70	100	24	-	-	Has self-test capability
12.	Rate/Att. Error Display Elect. Rate Elect. Att. Error Elect.	2	1	2	1	-	3	-	1/2 of EDA's A3, A5, A7, A9 modules. Through functions are not identical, the modules are made so by engaging different pins in the connector for one function opposed to the other.
13.	Att. Display Elect.	1	1	2	1	1	6	-	1/2 of EDA's A4, A6, & A8 Modules.
14.	Gim. Position Display Elect.	1	1	2	1	-	12	-	EDA's A11 module
15.	+ 15 VDC Display P.S.	1	2	4	2	-	-	2	1/2 of EDA's A 13, A14, & A15 Modules
16.	SPS ON/OFF Cont.	1	1	2	1	1	2	-	RJEC's A7 Module
17.	RCS Drivers A/C Roll B/D Roll Pitch Yaw	4	1	2	1	2	16	-	RJEC's A5 Module RJEC's A4 Module RJEC's A2 Module RJEC's A1 Module
18.	TVC Electronics Pitch/Yaw 1 Pitch/Yaw 2	2	1	2	1	-	4	-	TVSA's A2 & A3 Modules
19.	TVC Elect. Pwr Supply	1	2	4	2	-	-	6	TVSA's A5 Module
	SUB-TOTAL	14	-	90	-	-	-	-	

VI-19

* 1 bd = 20 sq. in. or 10 ir.³

TABLE VI-1 continued

G & C REPLACEABLE PARTS LIST - ELECTRONICS

NO.	COMPONENT/MODULE	QUANT.	WT. EA.	WT. TOT.	BDS/ MOD.	TEST POINTS / MOD.		REMARKS
						CMC	ACT.	
20.	Subtotal carried fwd. Accel. Electronics Y Y Z	14 3	2 2	90 6	2	1	3	Composed of PEA Modules: a) Binary current SW b) Diff. Amp. & PVR c) PIPA Cal. Module d) AC Diff. Amp
21.	A/D - D/A Converters, etc. Inner Gimbal Middle gimbal Outer gimbal OSS Shaft, SXT DSS Trun. SXT	5	C	30	10	3	16	Each module contains the following CDU modules: a) Quadrant selector b) NSA & Quad, rejection c) Read counter d) Coarse System e) Error Angle Counter & Logic f) D/A Converter g) Digital Mode (partial) h) Moding (partial) i) PSA relay Ass'y
22.	A/D-D/A Power, Timing Interrogation	1	2	2	4	2	3	Contains following CDU Modules: a) Power Supply b) Interrogate c) Digital Mode (partial)
23.	A/D-D/A Moding	1	2	2	3	2	20	Contains following CDU Modules: a) Digital Mode (partial) b) Mode (partial)
24.	Gimbal Servo Amp. Inner Middle Other	3	1-1/2	4-1/2	2	-	7	Same as PSA Servo Amps
	SUB-TOTAL	27	-	134-1/2	-	-	-	

TABLE VI-1 continued:

G & C REPLACEABLE PARTS LIST - ELECTRONICS

NO.	COMPONENT/MODULE	QUANT.	WT. EA.	WT. TOT.	BDS/ MOD.	TEST POINTS / MOD.			REMARKS
						CMC	ACT.	PASS.	
	Subtotal carried fwd.	27		134-1/2					
25.	Fine Align. Elect.	1	3	3	4	1	-	-	Comprised of PSA Modules: a) Gyro Cal. b) DC Diff. Amp. & PVR c) Binary Current Sw.
26.	ISS Moding	1	1	1	1	2	2	-	Comprised of PSA Modules: a) Relay Module (solid state) b) IMU Aux. Module
27.	OSS Motor Drive SCT Trun. SCT Shaft SXT Trun. SXT Shaft	4	1-1/2	6	2	-	7	-	Same as G & N PSA Modules
28.	OSS Control Module	1	4	4	5	1	25	7	Made up of PSA Modules: a) 2 speed Sw. b) Anti Creep Ass'y. c) CSC Gen. d) Optics Auto Relay e) Sct. Moding f) Ret. Lite Dimmer g) Relay Ass'y.
29.	Pulse Torque Power Supply	1	3	3	4	-	-	4	Same as G & N PSA Module
	SUB-TOTAL	35	-	151-1/2	-	-	-	-	

TABLE VI-1 continued: G & C REPLACEABLE PARTS LIST - ELECTRONICS

NO.	COMPONENT/MODULE	QUANT.	WT. EA.	WT. TOT.	BDS/ MOD.	TEST POINTS / MOD.			REMARKS
						CMC	ACT.	PASS.	
	Subtotal carried fwd.	35		151-1/2					
30.	800 Hz, 1% Amp.	2	1-1/2	3	2	-	-	1	Made up of PSA Modules: a) 800 Hz, 1% Amp b) 800 Hz, AAC filter & Multi-vibrator c) 1% load compensation (IMU & optics)
31.	800 Hz, 5% Amp. 1SS ØA 1SS ØB OSS	3	1-1/2	4-1/2	2	-	-	1	Made up of PSA Modules a) 800 Hz, 5% Amp. b) 5% load compensation
32.	-28 VDC Power Supply	1	1-1/2	1-1/2	2	-	-	1	Same as G & N PSA Module
33.	3200 Hz, 1% Amp.	1	1-1/2	1-1/2	2	-	-	1	Same as G & N PSA Module
34.	28 VDC Filters	1	1	1	1	-	-	3	Same as G & N PSA Module
	TOTAL/ELECTRONICS	43	-	163	-	-	-	-	Avg. wt./Module = 4 pounds

TABLE VI-1 continued: G & C REPLACEABLE PARTS LIST - HARDWARE

NO.	COMPONENT/MODULE	QUANT.	WT. EA.	WT. TOT.	BDS/ MOD.	TEST POINTS/MOD			REMARKS
						CMC	ACT.	PASS.	
35.	Interconnect Harness-TVC	1	2	2	-	-	-	-	3' wide x 4' high (180" of 5" high slide-in rack = 6 rows, 30" long)
36.	Interconnect Harness-Att. Cont.	1	2	2	-	-	-	-	
37.	Interconnect Harness-Platform	1	15	15	-	-	-	-	
38.	Interconnect Harness-Displays	1	3	3	-	-	-	-	
39.	Interconnect Harness-Computer	1	10	10	-	-	-	-	
40.	Cabinet/Cooling Duct, etc.	1	50	50	-	-	-	-	
Total/Hardware		6	-	82	-	-	-	-	Avg. Wt/Module = 13 lbs.

SUMMARY

Total/Hardware	6	82		
Total/Electronics	43	163		
Total/non Electronics	12	166		
Total/System	61	411		Avg. Wt/Module = 7 lbs
Total/Space Station	122	822		2x System Total
				Commonality = 122-60 - 68Z
				122

TABLE VI-2

TELECOM REPLACEABLE PARTS LIST - NON-ELECTRONICS

NO.	COMPONENT/MODULE	QUANT.	WT. EA.	WT. TOT.	BDS/ MOD.	TEST POINTS / MOD.		REMARKS
						ACT.	PASS.	
1.	Audio Cont. Panel	3	4	12	-	-	-	
2.	Headset	3	1	3	-	-	-	
3.	High Gain Antenna	1	30	30	-	-	-	
4.	Omni Ant.	4	2	8	-	-	-	
5.	VHF Ant.	2	3	6	-	-	-	
TOTAL		13	-	59 (Est.)	-	-	-	Wt./Module = 4-1/2 lbs.

TABLE VI-2 (continued)

TELECOM REPLACEABLE PARTS LIST - ELECTRONICS								
NO.	COMPONENT/MODULE	QUANT.	WT. EA.	WT. TOT.	BDS/ MOD.	TEST POINTS/MOD.		REMARKS
						ACT.	PASS.	
6	Present telecomm electronics within the "black box" case is packaged in 1 and 2 bd modules approx. 4" x 5". This is an optimum size for R/R, and since uniform throughout, an est. total of the modules is presented here based on the module count within the existing Telecomm.	180	1-1/4	225	2	4	-	
90 (est.)								
TOTAL		180	-	225	--	--	-	Wt./Module = 1-1/4 lbs.

TABLE VI-2 (continued) TELECOM REPLACEMENTABLE PARTS LIST - HARDWARE

NO.	COMPONENT/MODULE	QUANT.	WT. EA.	WT. TOT.	BDS/ MOD	TEST POINTS/MOD.		REMARKS
						ACT.	PASS.	
91.	Interconnect Harness- S-Band	1	10	10	--	--	--	
92.	Interconnect Harness- VHF	1	4	4	--	--	--	
93.	Interconnect Harness- CTE	1	3	3	--	--	--	
94.	Interconnect Harness- SCE	1	5	5	--	--	--	
95.	Interconnect Harness- UDL	1	5	5	--	--	--	
96.	Interconnect Harness- PCM	1	10	10	--	--	--	
97.	Electronics Cabinets	1-1/4	50	65	--	--	--	
	TOTAL/HARDWARE	6-1/4		107				Wt./Module = 17 lbs.

SUMMARY

TOTAL/HARDWARE	6-1/4	107	
TOTAL/ELECTRONICS	180	225	
TOTAL/NON-ELECTRONICS	13	59	
TOTAL/SPACE STATION	199	391 lbs.	Wt./Module = 1.5 lbs. Commonality = $\frac{199-97}{199} = 51\%$

Another candidate for commonality with far more potential is the standardization of power supplies. A detailed discussion of standardized power supplies is provided at the end of this section. Standardization of power supplies was not reflected in the redesign represented in Figures VI-10 and VI-11, primarily to illustrate the variance in type and quantity of power supplies that do exist on the current Apollo hardware. A summary for G&C is provided in Table VI-3. Another reason for not standardizing the power supplies in the redesign is that in standardization some impact on circuit redesign is inevitable as pointed out on page VI-47. Rather than try to reflect this impact on the electronics design the power supplies were left "as is"; however in the eventual design for Space Station it is recommended that standardization of power supplies be a primary design goal.

To provide further insight as to what was considered optimum fault isolation and just how each of the modules are isolated refer to Figures VI-12 thru VI-23. A summary of total test point requirements is provided in Tables VI-4 and VI-5. Only a few representative examples of telecommunications fault isolation circuits are provided, Figures VI-22 and VI-23.

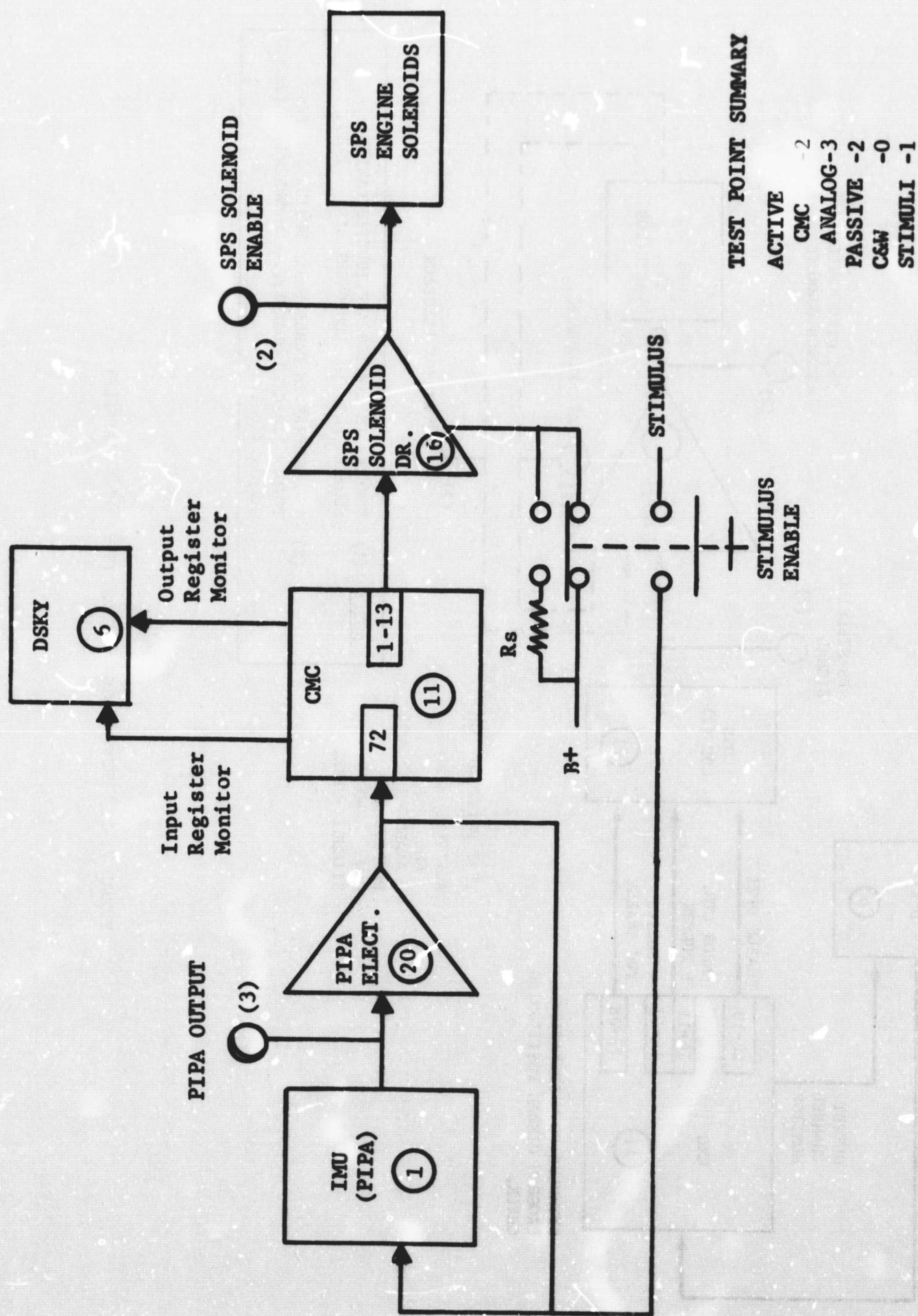
Some definition of terms used in the Test Point Summary and on Figures VI-12 thru VI-23 is warranted at this time. "STIMULI" or "ADDITIONAL STIMULI" refers to stimuli required which is not mechanized on Apollo or available through any operational means. "PASSIVE" test points are temperatures, power supply voltages, oscillators and a select few events all of which can be monitored at any time without interfering in circuit operation. "ACTIVE" test points are those that are actively varying during system operation, therefore require removal of the system from line operation and monitored under a given stimuli condition for proper assessment of systems operation. "C & W", or Caution and Warning, test points are those test points that are routed to some audible or visible caution and warning system. Those test points are limited to (1) critical system parameters, that, if failed, are not immediately obvious to the crew in other visual cues or because a crewman is not at station; or (2) those system parameters that, if failed, are not obvious to the crew in vehicle operation until the next periodic check, and may result in a degraded performance that can jeopardize mission objectives.

The total number of test points required for both of the identical G&C systems, including passive, active and C&W are 456. Other estimates have ranged from 450 (Ligocki, 1968) to 1050 (Jorgenson 1969) for G&C on Space Platforms. The total Telecommunications requirements were 505 test points. (Jorgenson 1968)

TABLE VI-3
G&C POWER SUPPLY SUMMARY

MOD. NO.	COMPONENT/MODULE	QUAN	VOLTAGES	PWR-WATTS	REG.	SOURCE	REMARKS
10	Gyro P.S.	1 1 1	+32 VDC +20 VDC -15 VDC	25		115 VAC	derived from an unfiltered +40 VDC supply
11	CMC P.S.	1	+14 VDC + 4 VDC	100		28 VDC 28 VDC	
15	Display P.S.	1	+15 VDC -10 VDC -2.5 VDC	30		115 VAC	
19	TVC Elect. P.S.	1	+14 VDC +15 VDC	22	None	115 VAC	Actually 2 sep. P.S. on 1 bd.
22	A/D-D/A P.S.	1	+14 VDC + 4 VDC	27 12	±1%	28 VDC 28 VDC	2 sep. supplies
28	Pulse Torque P.S.	1 1 1 1	+120 VDC + 28 VDC + 20 VDC - 20 VDC	80		28 VDC 28 VDC 28 VDC -20 P.S.	ref. to 12.8K pps in C,C
31	-28 VDC P.S.	1	- 28 VDC	17		28 VDC	
29	800 Hz, 1%	1 1	28 VDC 28 VDC	24, OSS 12, ISS	1% 1%	28 VDC 28 VDC	Ref. to CMC 800 pps Ref. to CMC 800pps
30	800 Hz, 5%	1 1 1	28 VDC 28 VDC 28 VDC	39, 0A 48, 0B 35, OSS	5% 5% 5%	28 VDC 28 VDC 28 VDC	Ref. to CMC 80 pps Ref. to CMC 800 pps Ref. to CMC 800 pps
32	3200 Hz, 1%	1	28 VDC	10	1%	28 VDC	Ref. to CMC 800 pps





TEST POINT SUMMARY

ACTIVE	
CMC	-2
ANALOG-3	
PASSIVE	-2
C&W	-0
STIMULI	-1

FIGURE VI-13 THRUST ON/OFF CONTROL (Typ. of 2 Channels)

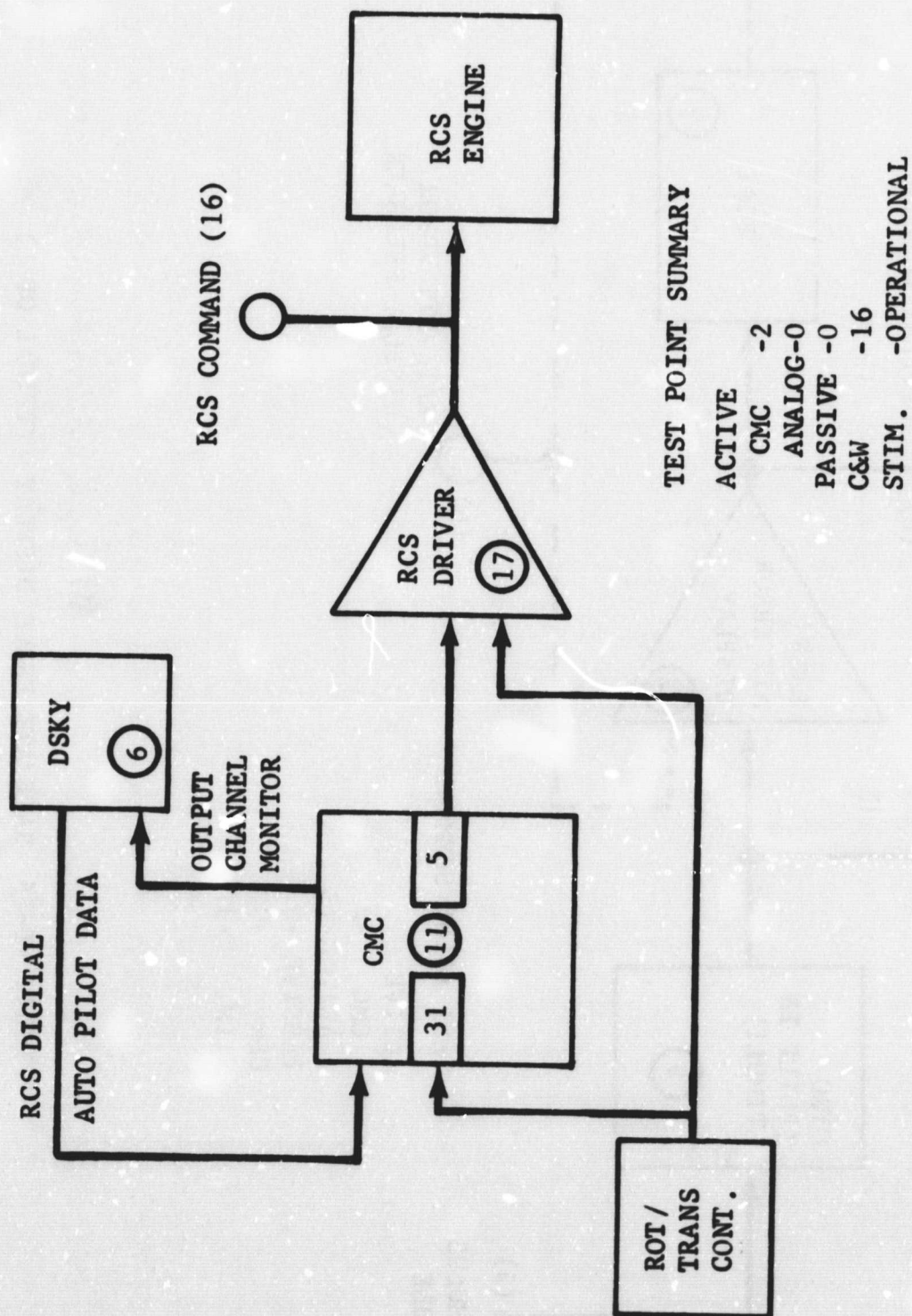


FIGURE VI-14 ATTITUDE CONTROL (TYPICAL OF 16)

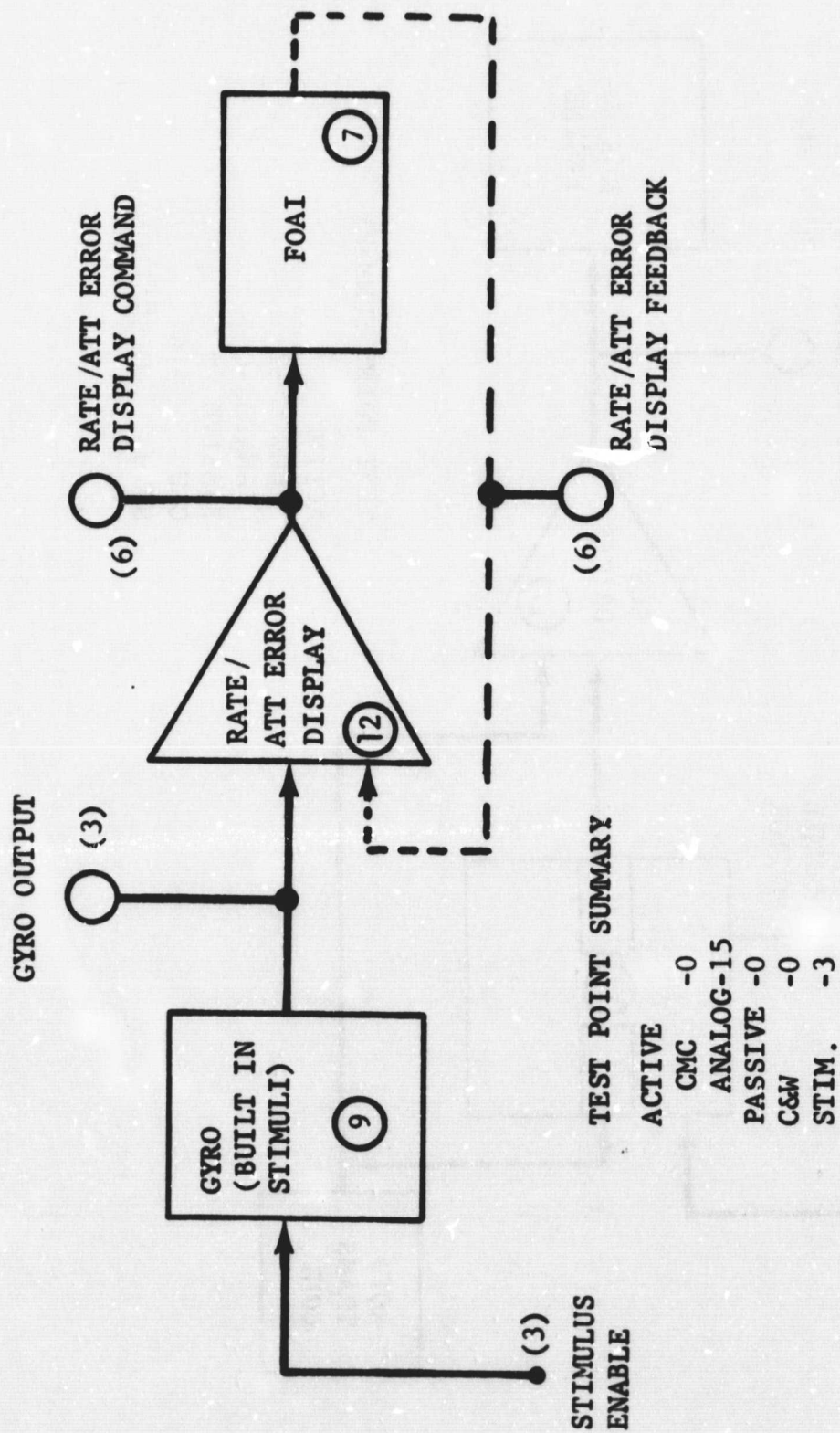
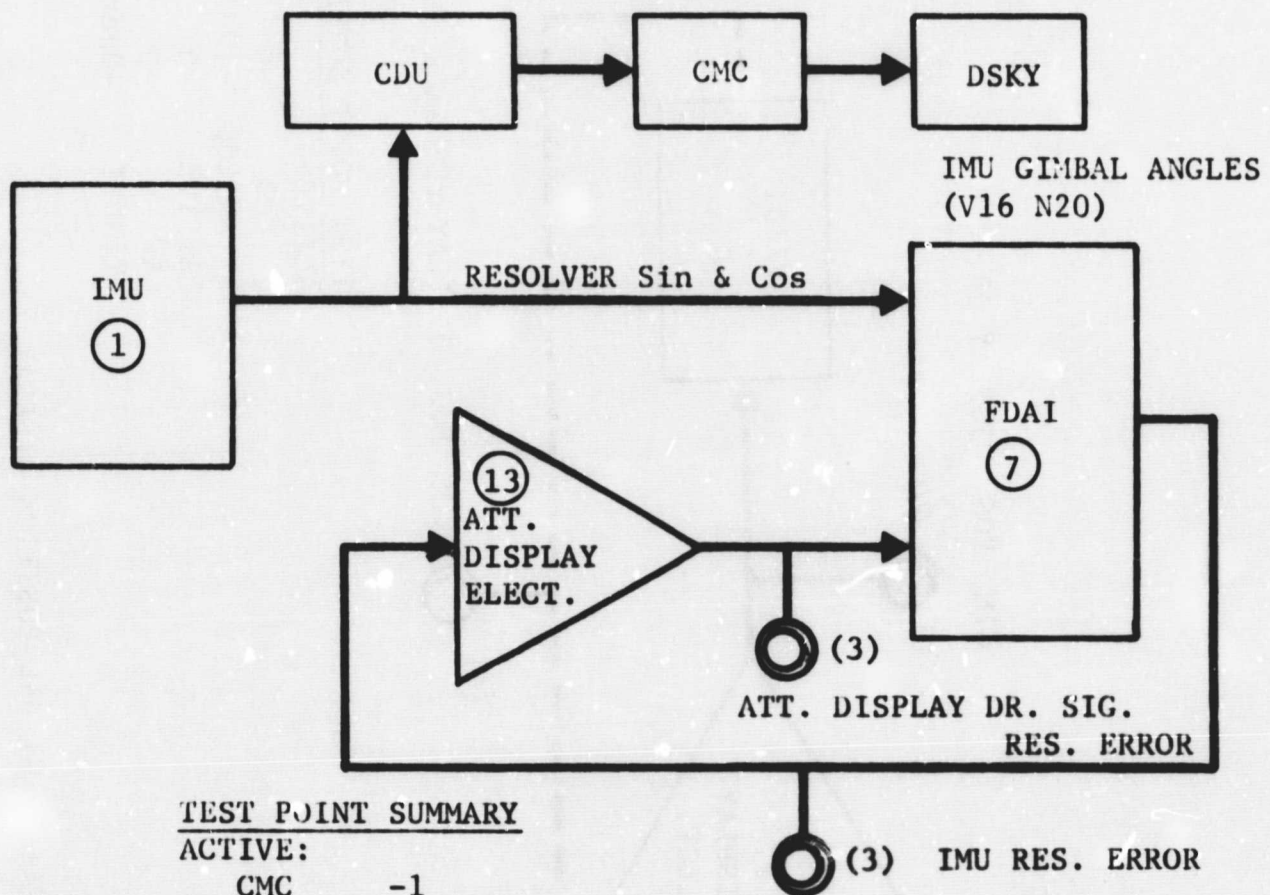


FIGURE VI-15 RATE/ATT ERROR DISPLAY (TYPICAL OF 6)



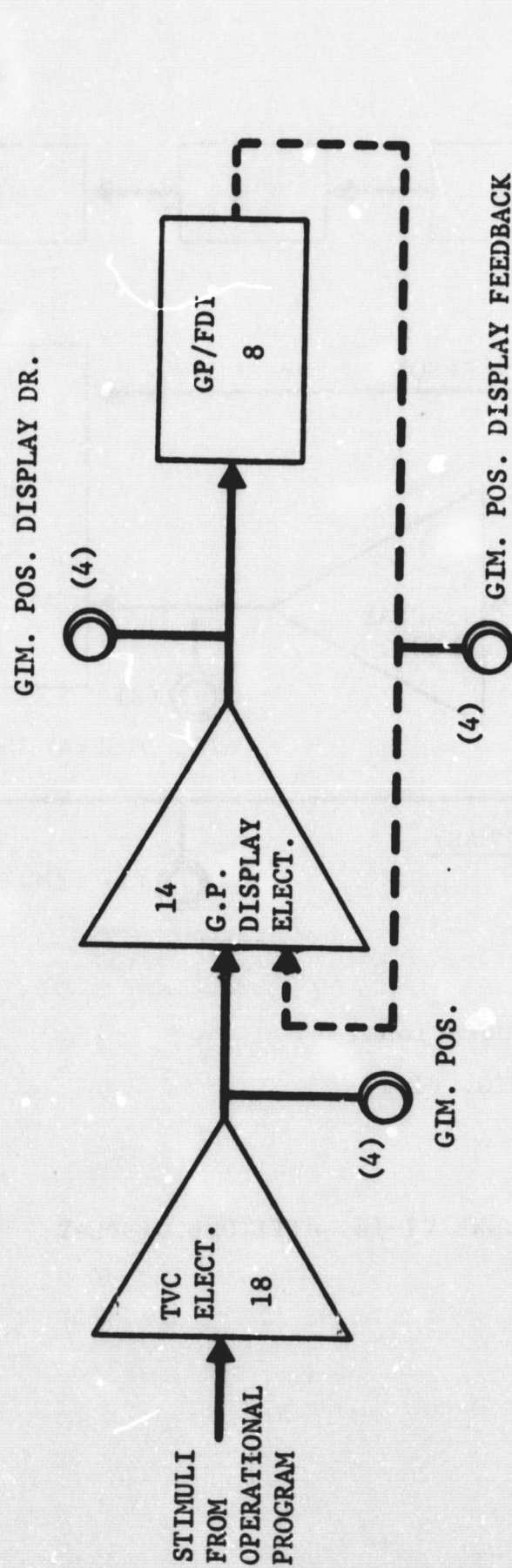
TEST POINT SUMMARY

ACTIVE:

CMC	-1
ANALOG	-6
PASSIVE	-0
C&W	-0
STIMULI	-Operational

SEE NOTES ON FIG. VI-12

FIGURE VI-16 ATTITUDE DISPLAY

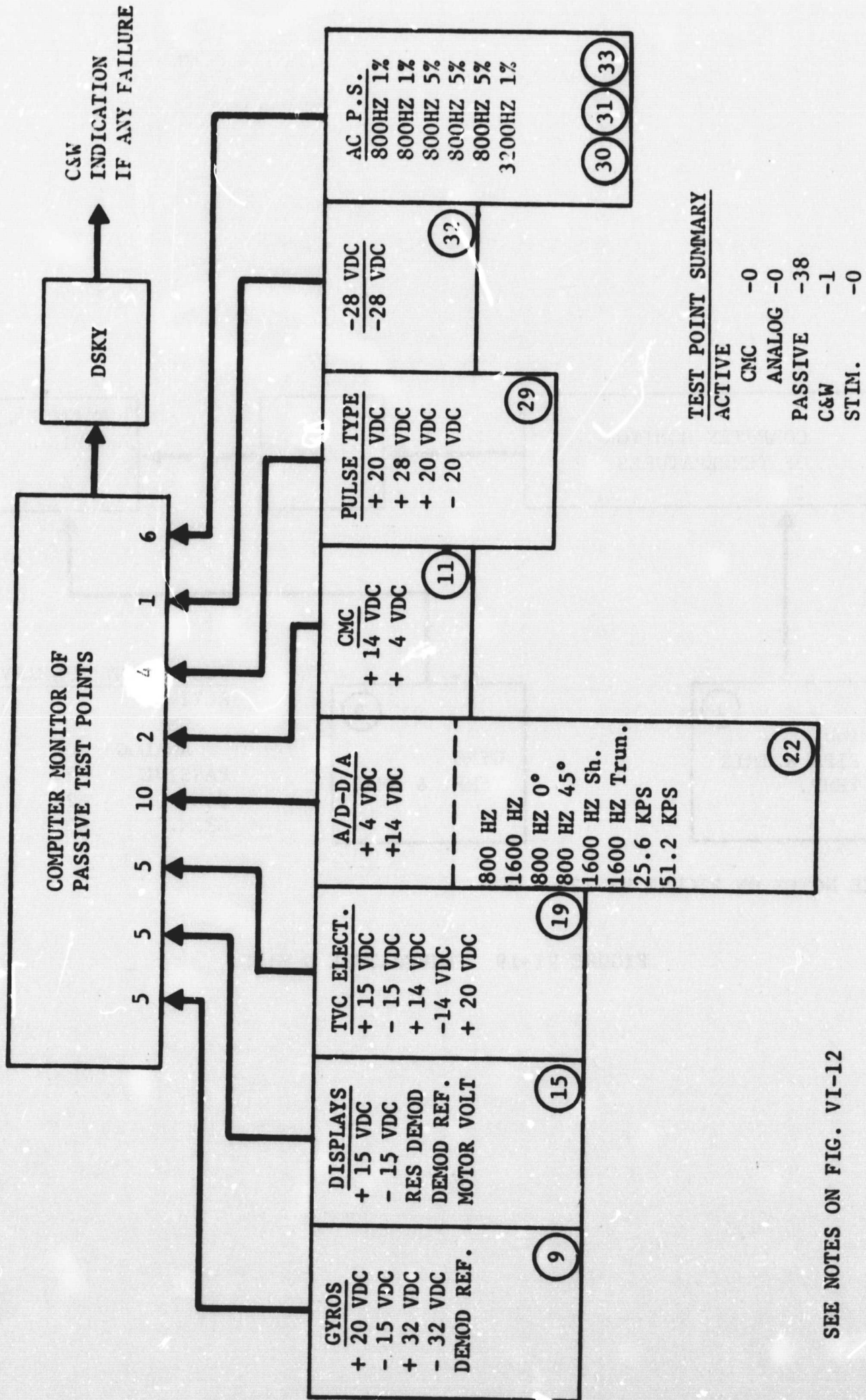


TEST POINT SUMMARY

ACTIVE:	
CMC	-0
ANALOG	-12
PASSIVE	-0
C&W	-0
STIMULI	-Operational

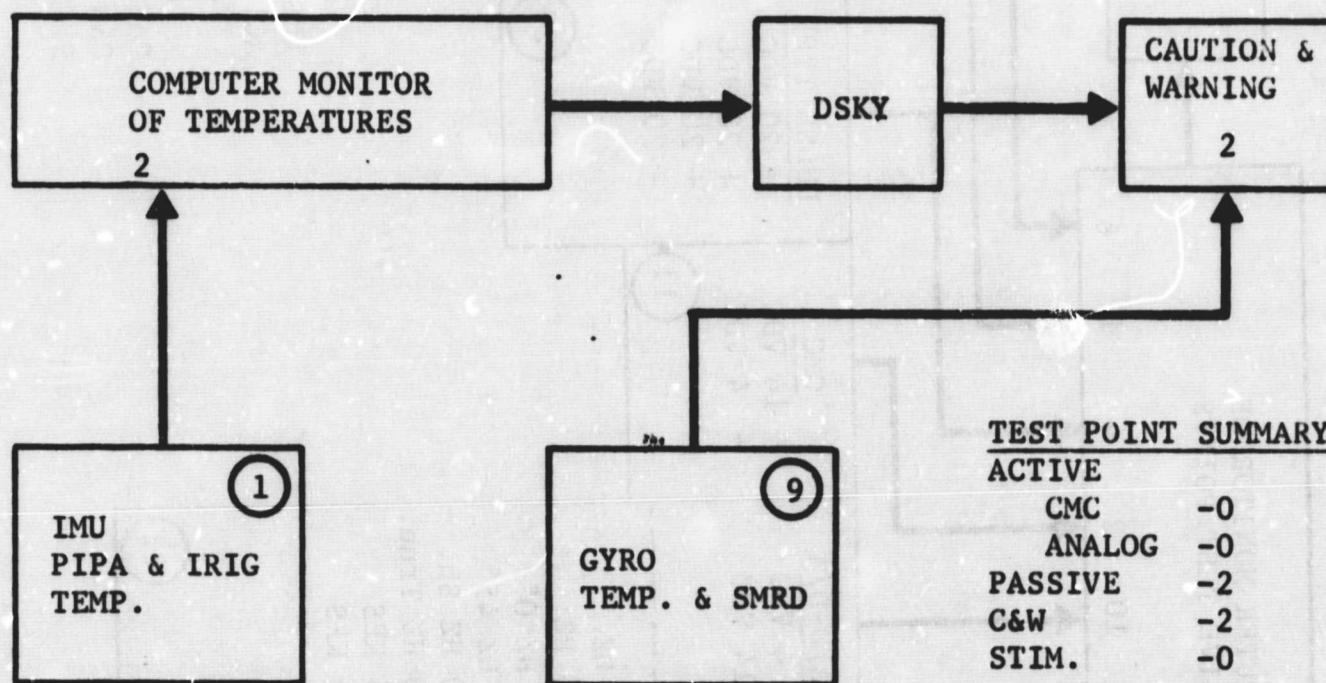
SEE NOTES ON FIG. VI-12

FIGURE VI-17 GIMBAL POSITION DISPLAY



SEE NOTES ON FIG. VI-12

FIGURE VI-18 G&C POWER SUPPLIES

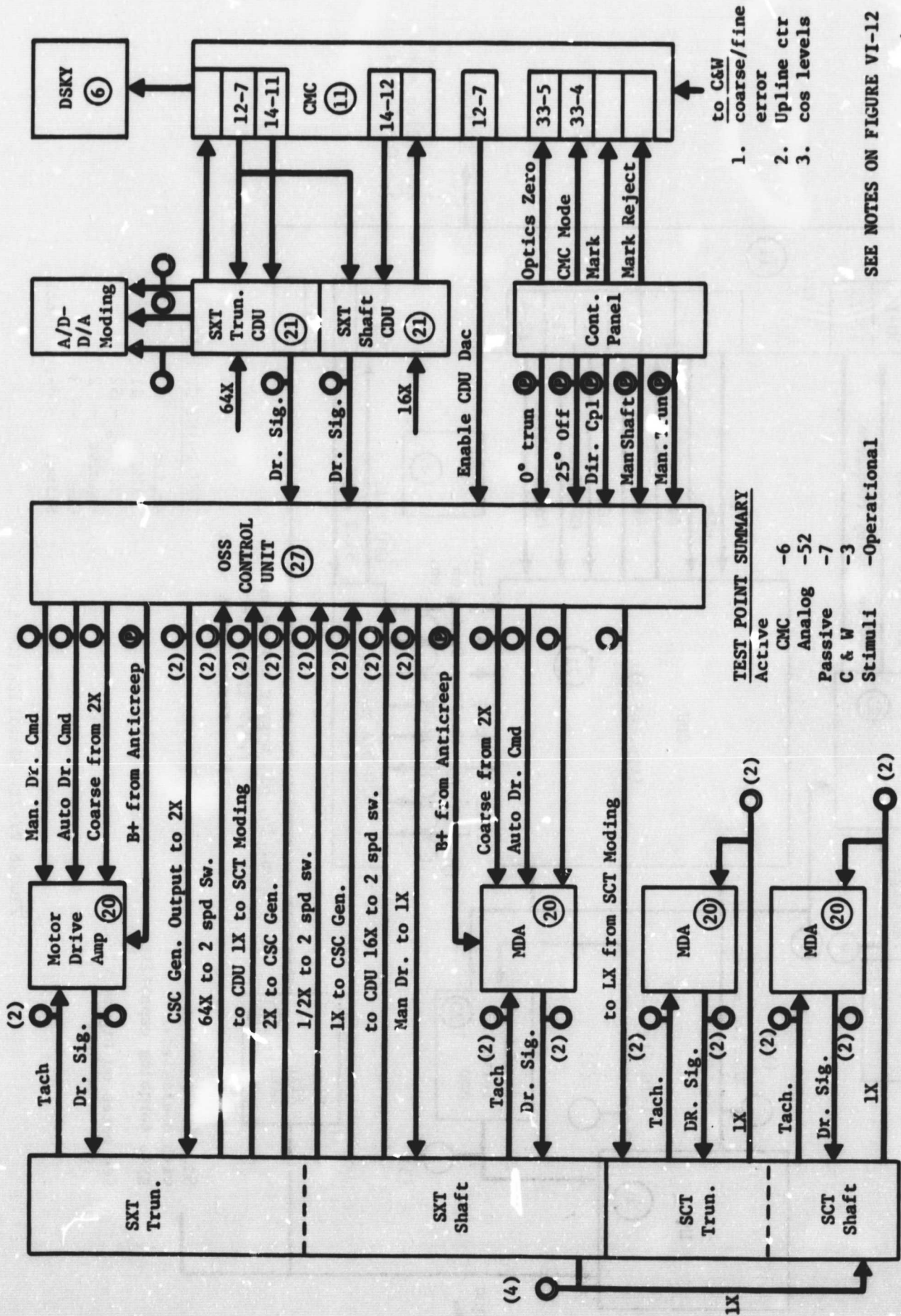


TEST POINT SUMMARY

ACTIVE	
CMC	-0
ANALOG	-0
PASSIVE	-2
C&W	-2
STIM.	-0

SEE NOTES ON FIGURE VI-12

FIGURE VI-19 TEMPERATURE & MISC.



TEST POINT SUMMARY

Active	Stimuli	Operational
CMC	-6	
Analog	-52	
Passive	-7	
C & W	-3	

1. coarse/fine error
2. Upline ctr
3. cos levels

SEE NOTES ON FIGURE VI-12

FIGURE VI-30 OPTICAL SUBSYSTEM

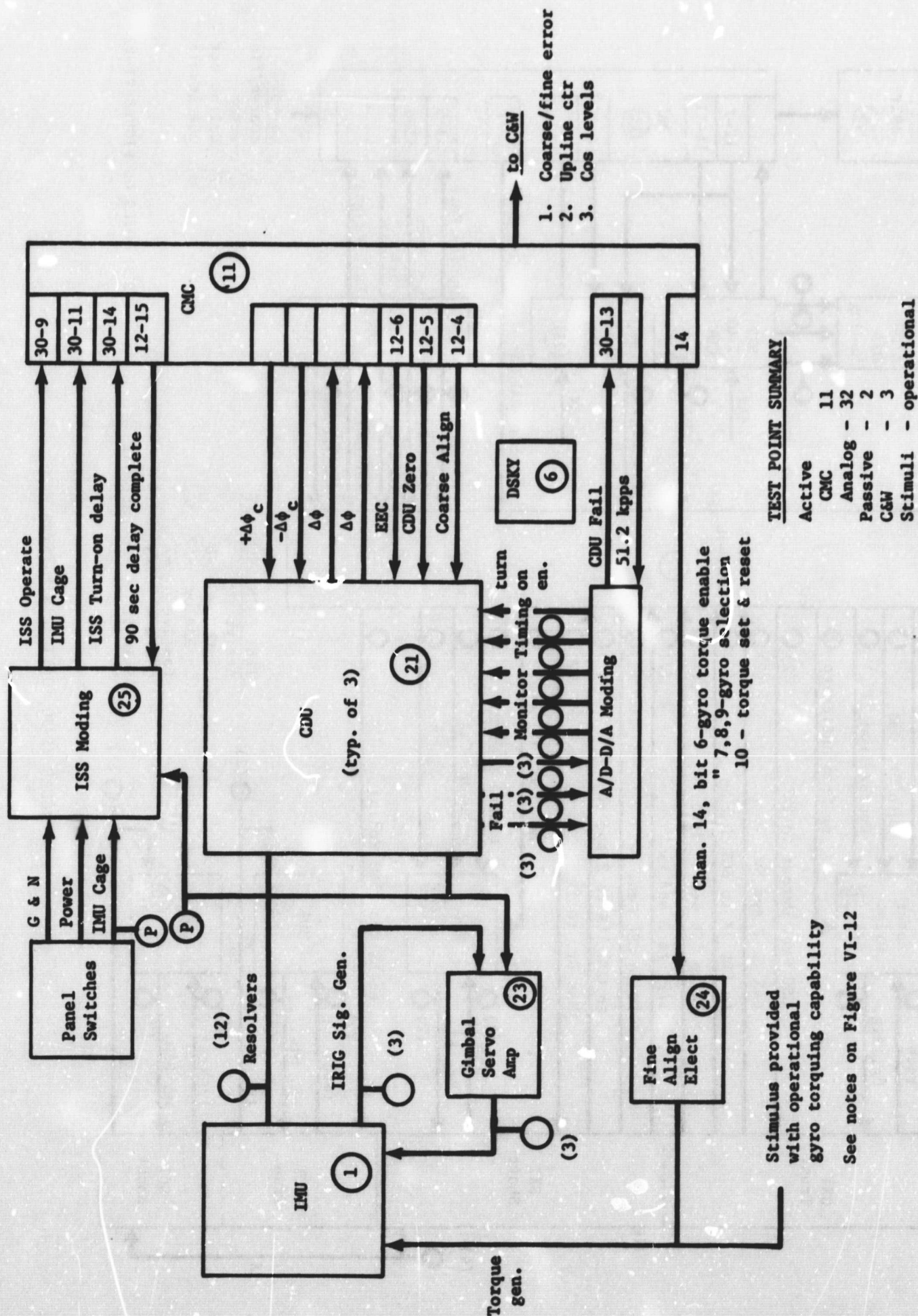
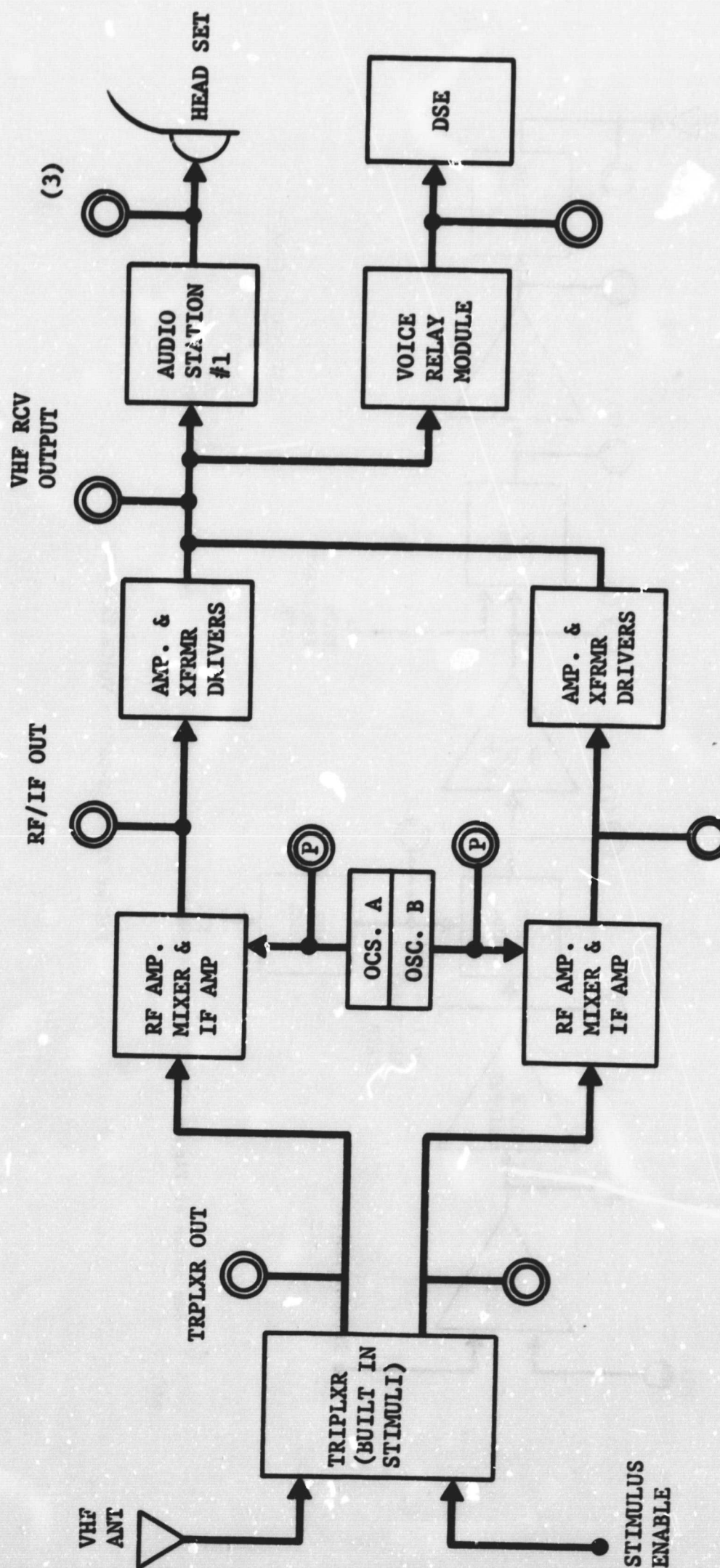


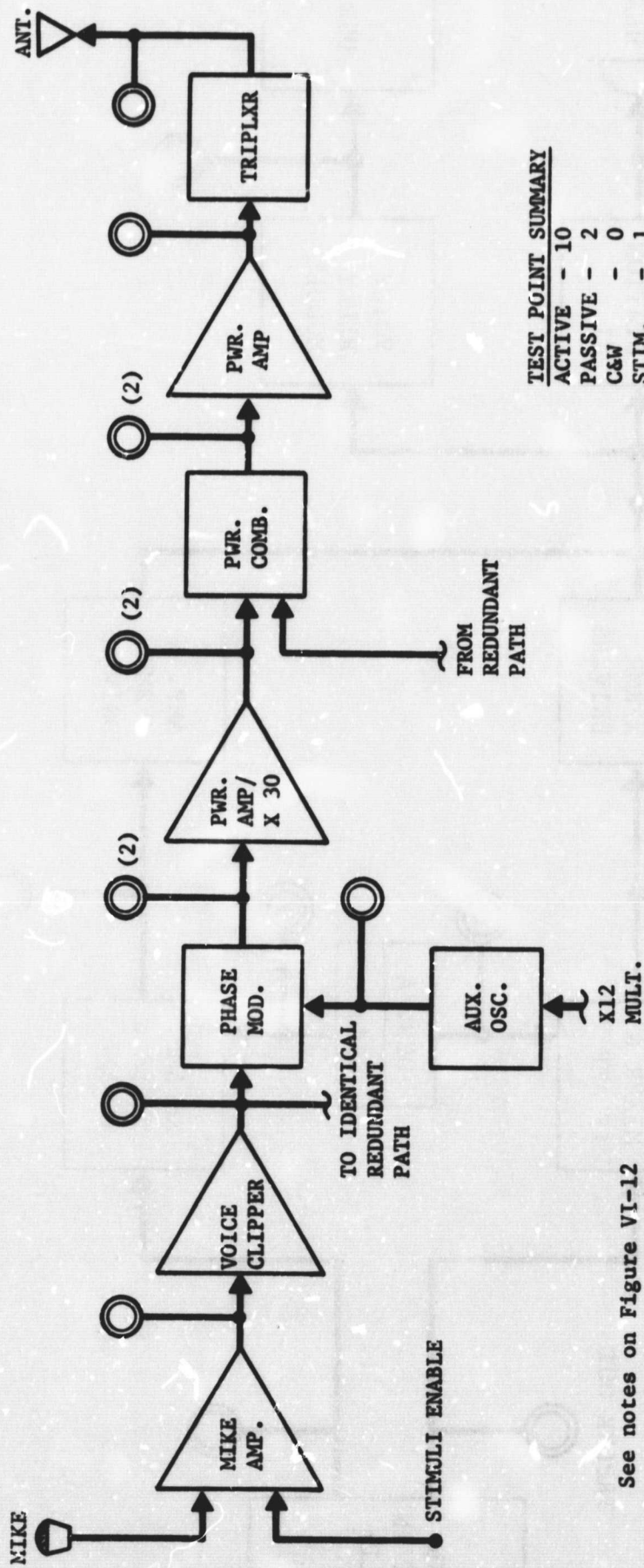
FIGURE VI-21 STABILIZATION LOOP



TEST POINT SUMMARY	
ACTIVE	- 9
PASSIVE	- 2
C&W	- 0
STIMULI	- 1

See notes on Figure VI-12

FIGURE VI-22 VHF RECEIVE - VOICE & DATA



See notes on Figure VI-12

FIGURE VI-23 S-BAND - VOICE TRANSMIT

	Additional Stimuli	Passive Test Points	Active Test Points		Caution & Warning
			Analog	CMC	
Thrust vector control	1	2	11	4	--
Attitude control	--	--	--	2	16
Displays	3	--	33	1	--
Power supplies	--	40	--	--	3
Attitude reference					
OSS	0	7	52	6	3
ISS	--	2	32	11	3
Subtotal			128	24	
Total/Module	4	51	152		25
Total/Space Station	8	102	304	7	50
			456		

Table VI-4 G&C Fault Isolation Test Point Summary

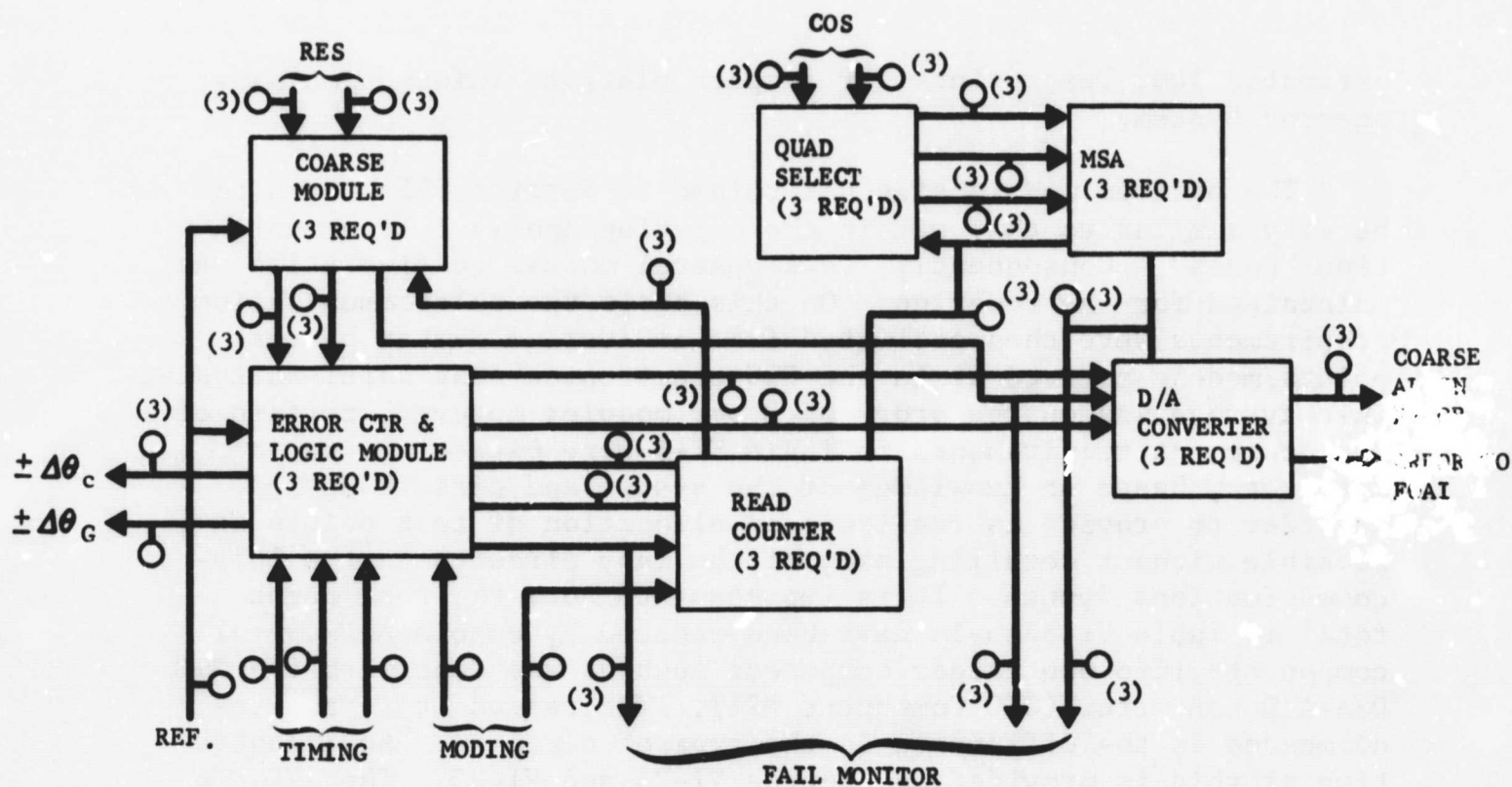
	Additional Stimuli	Passive Test Points	Active Test Points
S-band (including PMP & Audio Center)	8	15	100
VHF (including Digital Range Generator & Audio Center)	3	8	60
Central timing &PMP	-	4	40
Pulse code modulator	-	4	100
Signal conditioning	-	4	90
Update link (& PMP)	-	3	60
Rendezvous radar transponder	1	2	15
Total	12	40 → 465	505

Table VI-5 Telecommunication Fault Isolation Test Point Summary

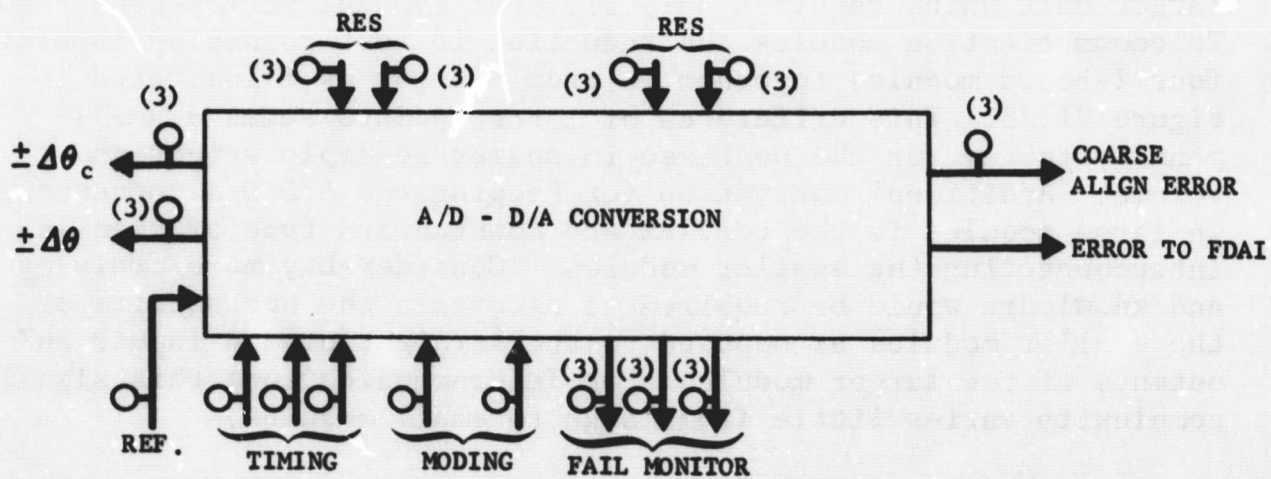
estimated 1000 Test points for a space platform Information Management System.

The optimum module size determined in Section VII proved to be very similar to that within the existing Apollo telecommunication "boxes". Consequently, this general module configuration was maintained for the redesign. On this basis the telecommunication requirements were then estimated from an average number of test points/module arrived at in the G&C electronics test point analysis. This average was on the order of 3 per module; however, certain of the equipment requirements in Table VI-5 were biased one way or the other, based on knowledge of the system and circuit design, in order to provide as realistic an allocation of test points as possible without detailing each of the many circuits in the Telecommunications System. It is important to note the test point total in Table VI-5 could have been reduced by combining several components into one larger component such as was done with the G&C D/A-A/D converter (G&C Component #21). The reason it is not recommended is the difference in the type of circuits. An illustration of this is provided in Figures VI-24 and VI-25. The A/D-D/A converter (Figure VI-24) is a 10 board (4" x 5") module with considerable interrelationship between the 10 boards. Were this module broken into 5 smaller modules the number of test points required for complete isolation is 60, as opposed to the single larger unit which requires only 31. For typical series-connected Telecommunication modules the reduction in test points by combining four 2-board modules together is from 8 to 5, as illustrated in Figure VI-25. This difference of 3 test points seems a small penalty to pay for the decrease in spares resupply weight and volume. Additional motivation for keeping the A/D-D/A converters in large modules is the complex and nonstandard type of signals interconnecting the smaller modules. Considerably more training and knowledge would be required to ascertain the operability of the smaller modules as opposed to the straightforward inputs and outputs of the larger module. For Telecommunications this signal complexity varies little from large to small modules.

An estimate of failure rate of the redesigned G&C is tabulated in Table VI-6. The system total is one failure per 26 days. The Telecommunication remains unchanged at 1 failure/297 days. Neither this figure nor that of the G&C has taken into account failure rates of the additional connections and wiring due to repackaging; however, for the type of comparison to be made this can be neglected. Taking the two failure rates together, assuming one G&C and the Telecommunications on continuously and one G&C in standby, the electronics total is 1 failure per 26.7 days or



(a) 6 SMALL MODULES (2 bds) PER CHANNEL (TOT = 3 ea)



(b) 1 LARGE MODULE (10 bds) PER CHANNEL (TOT = 3)

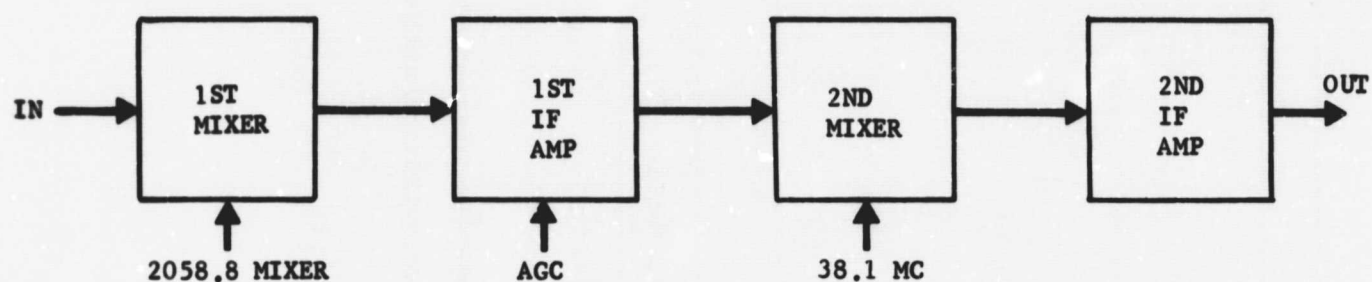
ITEM (a) REQUIRES 72 TOTAL TEST POINTS FOR ISOLATION

ITEM (b) REQUIRES 36 TOTAL TEST POINTS FOR ISOLATION

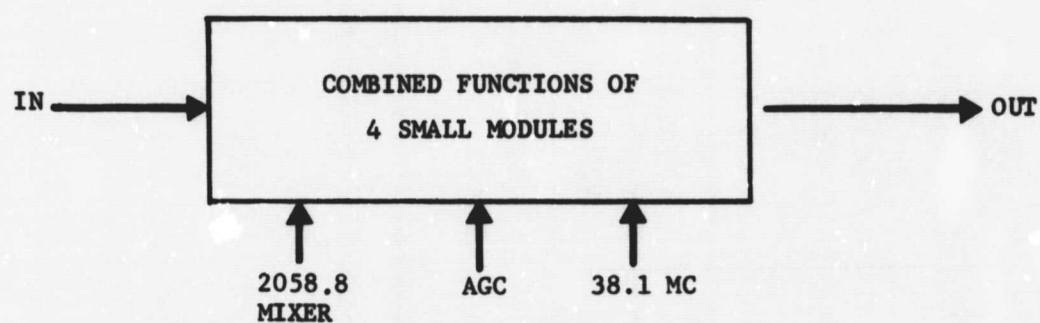
SAVINGS 36 TEST POINTS

NOTE: NUMBER IN PARENTHESIS BESIDE TEST POINT INDICATE QUANTITY. NO NUMBER INDICATES QUANTITY = 1.

FIGURE VI-24 ALTERNATE PACKAGE SIZE FOR G&C A/D - D/A CONVERSION



(a) 4 SMALL MODULES (2 4"x5" BD. EA.)



(b) 1 LARGE MODULE

ITEM A REQUIRES 8 TEST POINTS FOR ISOLATION
 ITEM B REQUIRES 5 TEST POINTS FOR ISOLATION

SAVINGS: - 3 TEST POINTS
 PENALTY: - RESUPPLY WEIGHT OF SPARE - 4 TIMES GREATER

FIGURE VI-25 ALTERNATE PACKAGE SIZE FOR TYPICAL COMM. MODULE
 (PART OF S-BAND RECEIVER)

TABLE VI-6 - G & C FAILURE RATES - PEDESIGN

COMPONENT	$\lambda \times 10^{-6}$	
	OPERATING	STANDBY
G & N	895.0	77
RJEC	41.5	
TVSA	57.6	
EDA (1/2)	100.5	
GA 1	197.5	
FDAI 1	143.6	
GP/FPI	48.8	
ROT. CONT. 1	15.0	
TRANS. CONT. 1	3.5	
SUB TOTAL	1503.0	77.0
TOTAL	1503.0 + 77 = 1580/Space Station *	
	$\lambda =$	1.58 Failures/1000 hours
Or;	$\lambda =$	1 Failure/630 hours
Or;	$\lambda =$	1 Failure/26.3 Days

*Assumes only one system operable at a time except for critical periods which, for this study, are neglected.

approximately one a month. From the Telecommunication weight of 391 pounds, the G&C weight of 822 pounds and a total number of components of 319 (See Table VI-1 and VI-2), the average resupply weight is ≈ 48 pounds/year, or $1/25$ of the total system weight; compared with an Apollo Block II resupply weight of 400 pounds or $1/2$ of the total system weight.

The power requirements for just one of the two G&C Systems was calculated at 750 watts while operating and 83 watts in standby. This is compared with 1000 watts and 115 watts respectively, for the original Apollo G&C (Table IV-6). Assuming one G&C operating and one in standby, the total of all three systems (Telecommunications remains unchanged at 503 watts), is 1.3 KW compared with 1.5 KW on original Apollo (Table IV-6).

Maintenance of Power Supplies

Maintenance of power supplies begins at the design level, for good design represents minimal maintenance. If the good design is standardized for many system components, a failure in a power supply is most readily remedied by replacing the unit with a standardized spare supply. Martin Marietta's Denver Division Electronic Research Department has been utilizing a standardized power supply for various equipment developed under NASA contracts; a supply for power requirements varying from 5 watts to 30 watts. While there are penalties paid in standardization, the advantages, in most cases, far outweigh the shortcomings.

Some of the advantages are as follows:

1. Design time minimized.
2. A minimum number of parts types need qualifying.
3. Qualification testing is reduced through commonality.
4. Maximum experience and failure rate data can be obtained from multiple applications.
5. Commonality of parts and spare power supplies is maximized.

Some of the penalties are as follows:

1. Equipment must be designed with some standardization in voltages.
2. Efficiency varies with load.
3. All voltages in a standard supply will not be required in every application, hence a weight and volume penalty.
4. The power supply configuration may influence the size and shape of the entire electronic package.

The following presents some design philosophy for establishing a standard design. A standard design will be described and actual performance data will be presented for several different applications.

One of the first steps for establishing a good design is to provide realistic and practical specifications. We have found that very few circuits require voltage regulations better than $\pm 3\%$ or ripples under 50 millivolts. These specifications are usable even with highly accurate instrumentation circuits excepting such applications as A/D converter reference voltages. Where the situation requires a highly accurate and stable voltage, additional local regulation should be used. While an exotic power supply may be aesthetically attractive, the unnecessary complexity and constraints imposed on part selections by excessive specifications can do nothing but reduce reliability.

Another important step in good design is to select and use all components well within their ratings, particularly true where power switching devices and tantalum capacitors are involved. A figure of margin recommended as a general goal is 50% of manufacturer's specifications on voltage and power ratings.

There are obviously other factors influencing a good power supply design but the items mentioned above contribute most to good reliability.

Figure VI-26 is a basic schematic diagram of a standard power supply. The design of this supply employs the foregoing design criteria. The power supply is divided into sections as shown by the dashed lines; an input filter, a preregulator, a dc-dc converter, and output filters. These sections are installed in separate compartments in the shielded power supply case to minimize EMI difficulties. A brief explanation of the circuit is included in Figure VI-26 is presented as follows for those readers unfamiliar with switching preregulated power supply operation:

The input filter consists of two EMI filters, FL-LP3 and FL-LP4, and LC filter L1 and C3. These filters decouple incoming EMI from the power supply and also decouple noise generated in the supply from the power bus. Diode CR1 prevents accidental reverse voltage from damaging the supply.

The pre-regulator provides all of the voltage regulation in the power supply and limits the peak voltage applied to the converter. If the peak voltage on the converter were to equal normal power bus fluctuations, selection of fast power transistors for converter application would be difficult. The pre-regulator is a low level switching regulator. Transistors Q1 and Q2 are the series switch and are initially on, allowing C8 to be charged to a fixed voltage thru the primary inductance of T1. The voltage of C8 is determined by the voltage sensing feedback amplifier consisting of transistors Q3, Q4, Q5, and associated components. When C8 is charged to the proper value, Q1 and Q2 are turned off, but energy is stored in T1 and current continues flowing thru CK3 and T1 into C8 and the load. As the load removes energy from the circuit the voltage across C8 drops below the reference and Q1 and Q2 again turn on, repeating the cycle. Thus the voltage across C8 remains constant. The switching regulator is also known as a demand regulator because power is essentially furnished only upon request of the load.

The dc-dc converter is a conventional two transistor - two transformer free running converter. Frequency is determined by a small saturating transformer in the base drive circuits of the transistors. Voltage and power transformation to the loads is accomplished by an unsaturated transformer in the transistor collector circuits. For additional discussion of this switching action refer to Roddam, 1963 and Royer, 1955.

Output filters are provided for the full-wave rectified outputs of the power transformer.

The output voltages suggested for the power supply are based upon the use of both linear and digital integrated circuits in the electronics. A 5-volt output is provided to operate digital integrated circuits. Plus and minus 15 volts are compatible with most linear integrated circuits. Voltages of +40 and -25 are typical voltages for operating multiplexer circuits.

The described power supply design has been used (in some cases with minor modifications that could have been avoided) on the following NASA contracts:

NAS9-8146	PCM Telemetry System
NAS9-8301	PAM Decommutator System
NAS8-20960	Telemetry Programmer
NAS8-25066	Remote Multiplexer
NAS8-24682	Frequency Division Multiplexer.

In addition there have been several other contracts where the basic design was used along with some additional power conditioning for high peak power circuits.

The foregoing list represents 15 power supplies, the majority of which were temperature tested (-55°C to $+75^{\circ}\text{C}$) and vibration tested (sine and random typically 10G). Performance of all supplies was excellent. The history shows no semiconductor failures after correct assembly and only a few capacitor failures, none of which occurred after the first day of operation. None of the parts were actually screened beyond commercial levels prior to installation. While these contracts still do not present a large amount of statistical data, the multiple use of a single design demonstrates a confidence level for a high reliability and minimum maintainability that would not be achieved from five different designs. The penalties that were paid in some of the contract by standardization will now be examined.

When the power supply is operated at 10 watts input, which was the original design goal, and 50% of the load is at 5 volts, the efficiency for a 28 volt input is approximately 73%. When the power input is reduced to 5 watts, the efficiency at 28 volts drops to 67%. When the power input is increased to 30 watts and 90% of the load is at 5 volts, the efficiency is still 60%.

It is obvious there is no severe penalty in efficiency by using a standard supply. This is further emphasized by the fact that where the lowest efficiency occurs (at the high load) almost 4 watts of power is lost in the 5 volt rectifiers. This situation would not change appreciably with a custom design for 30 watts.

Where there is a large change in load distribution as well as a vastly changing load there is some degradation in load regulation. The rectifier voltage drop, in the case illustrated above changes about 0.1 volt. There will also be an increase in the IR drops. Fortunately, digital integrated circuits will operate satisfactorily from 4.5 volts to 5.5 volts.

A problem with the package size could not be avoided in one contract. The standard power supply volume exceeded the available space by 60%. This problem had to be resolved by higher density packaging of the power supply.

The foregoing represent the major disadvantages of a standard design. It is felt that a standardization technique

is still applicable to mos. systems. For example, where maximum efficiency is paramount for all applications in a system, a power supply design can be constructed with voltage and current techniques to proportion switch drive (pre-regulator and converter) powers to the load and proportion losses to loads. This will obviate secondary short circuit protection unless unduly complicated circuits are added. If minimum size is paramount high density standard packaging is possible. Design and construction costs will go up and reliability may go down slightly. Our limited experience with high density packaging does not indicate a substantially reduced reliability.

In conclusion a recommended maintainable power supply is the basic type of circuit described herein. Efficiency, load variations, unused voltages, and packaging density have been considered and deemed acceptable. Short circuit protection in the design would be retained. A reasonable standard package size for powers from zero to 50 watts appears to be about 20 cubic inches. Typical dimensions of a plug in module shielded and filtered for EMI are 4" x 5" x 1". See photo in Figure VI-27.

In the area of EMI, some of the test specifications the standard power supply discussed above is successfully complied with are:

- 1) IESD, EMC-10, Interference Control Requirements Earth Resources Survey System Specification. (NAS-MS).
- 2) IESD, 19-3, Interference Control Requirements for Spacecraft Equipment, (NAS-MS).
- 3) MIL-I-26600.

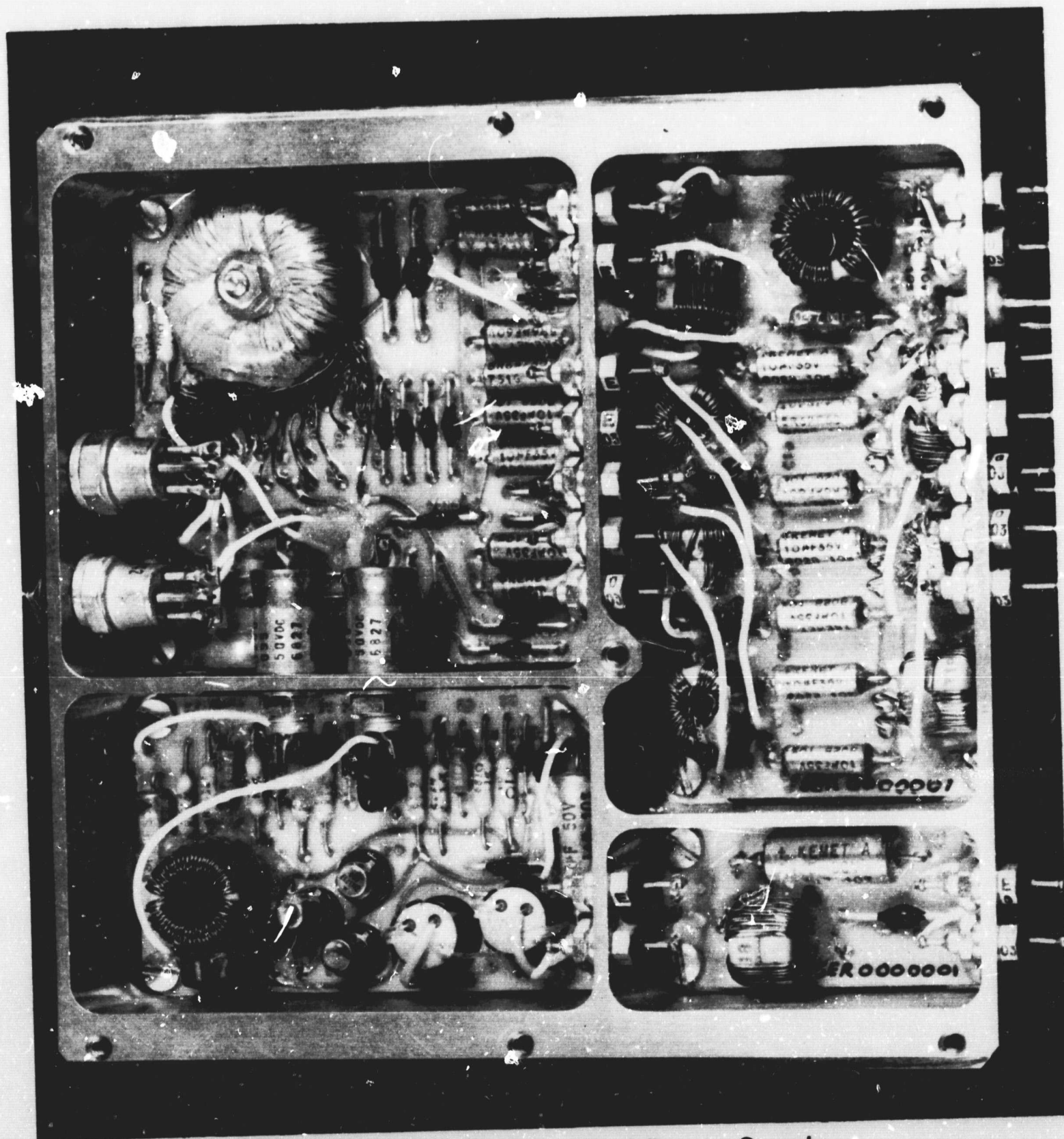


Figure VI-27 Standardized Power Supply

VII. PACKAGING DESIGN

With some limitations, the methods used for packaging are derived from the end item constraints. If, for example, weight and volume minimization are among the most important constraints, then it is reasonable to spend the extra money required to design and build 3D cordwood modules. It may even be necessary to have some circuit functions built in custom designed hybrid or integrated microcircuits. At the opposite end of the scale, where cost is the major consideration and weight and volume are less important, the system could be built using planar component arrays or single or double sided PC boards. For this system, the components would not be crowded on the boards making the boards easier to design and build.

The packaging system for a space station will fall somewhere between the above extremes. The design constraints require definition. Should the systems be individually customized or should they be standardized? What kinds of packaging at the high (system) and low (component and module) levels should be considered allowable?

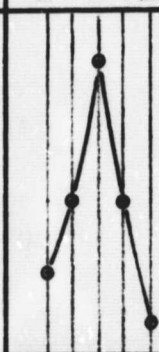
At the high level we might consider standard racks with plug-in modules; or standard, sealed chassis with standard or non-standard interior packaging; or non-standard, sealed chassis with standard or non-standard interior packaging.

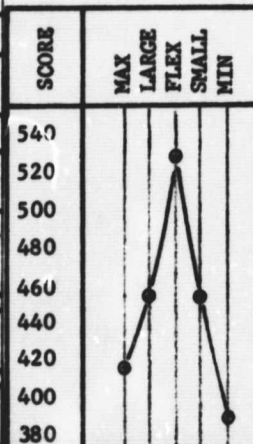
At the low level of electronic packaging, for most components, we can consider planar arrays versus volumetric (3-D) arrays for components.

Packaging Trade-Offs

On what basis are packaging design decisions made? How does one assure himself that these decisions are optimum? A packaging trade-off matrix, Figure VII-1 demonstrates the use of the IFM matrix as a tool for effectively making and documenting the rationale for the decisions. The list of criteria includes important independent factors that should be considered. Some of these criteria items are discussed and clarified below:

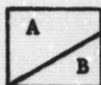
System Weight - When considering standard vs non-standard packaging design this includes the electronics modules or sub-assemblies, wiring, connectors, housings, all levels of mounting structures including those to the spacecraft structure, and all items required for electronics thermal control.

WEIGHTING FUNCTION	CONSIDERATIONS	ITEM										
		STD. PACKAGE DESIGN	NON STD PACKAGE DESIGN	LARGE MODULE	SMALL MODULE	MAX MODULE - STANDARD SIZED FOR MAX FUNCTION	MIN. MODULE - STANDARD SIZED FOR MIN FUNCTION	FLEXIBLE MODULE SEVERAL SIZE OPTIONS	COOLING - CONVECTION	COOLING - COLD PLATE	COOLING - DUPLEX (LIQUID & GAS SYSTEMS)	
7	WEIGHT-SYSTEM	4 28	7 49	7 49	7 49	3 21	6 42	9 63	9 63	5 35	4 28	SCORE 540 520 500 480 460 440 420 400 380 COMPARISON OF MODULE SIZE MAX LARGE FLEX SMALL MIN 
10	RELIABILITY FUNCTION	5 50	7 70	8 80	3 30	10 100	2 20	8 80	8 80	8 80	10 100	
5	RELIABILITY MODULE	7 35	8 40	2 10	10 50	2 10	10 50	8 40	8 40	8 40	10 50	
7	R/R ACCESSIBILITY (TOOL COMPLEXITY)	10 70	6 42	7 49	7 49	7 49	7 49	7 49	10 70	8 56	8 56	
3	SPECIAL FEATURE (HI HUMIDITY & CHANGE	7 21	7 21	8 24	8 24	8 24	8 24	8 24	3 9	9 27	9 27	
4	COST - NON RECURRING	10 40	3 12	8 32	8 32	8 32	7 28	9 36	10 40	1 4	4 16	
6	COST - RECURRING	10 60	3 18	7 42	6 36	7 42	6 36	8 48	10 60	4 24	6 36	
4	SYSTEM VOLUME	4 16	6 24	10 40	9 36	2 8	5 20	10 40	9 36	5 20	4 16	
9	INTERCONNECTIONS	4 36	7 63	10 90	6 54	10 90	3 27	7 63	0 0	0 0	0 0	
7	TRAINING, PROCEDURES, TOOLS	8 56	3 21	4 28	5 35	4 28	5 35	6 42	9 63	5 35	4 28	
6	EQUIPMENT COMMONALITY	7 42	2 12	2 12	10 60	2 12	10 60	7 42	9 54	9 54	9 54	
8	POWER (COOLING)	0 0	0 0	0 0	0 0	0 0	0 0	0 0	7 36	9 72	6 48	
	TOTALS	454	372	456	455	416	391	527	571	447	454	



KEY

BEST = 10
WORST = 1
N/A = 0



A - WEIGHT

B - PRODUCT OF WEIGHT AND WEIGHTING FUNCTION

PACKAGING DESIGN TRADE-OFF MATRIX

FIGURE VII-1

Functional Reliability and Module Reliability - As a system is divided into smaller modules, the reliability of the individual module is increased, however, the system reliability decreases simply because of the increased number of connections.

R/R Accessibility (Tool Complexity) - This was considered in terms of how easily and quickly a small part of a system could be replaced, and how many special tools might be required for this.

Special Feature (High Humidity and Changes) - The special feature of particular interest for this matrix is the effectiveness of the designs for a high humidity environment. Also considered was the flexibility required to permit somewhat less expensive re-supply items, if and when the high humidity problem is relieved.

Non-recurring Cost - This includes all design costs, tooling, and for some items, fabrication.

Recurring Cost - This includes fabrication, spares, and incorporation of design changes.

System Volume - This considers the total volume required for the electronics systems.

Interconnections - Minimizing the number of interconnections was considered desirable. Dividing a system into smaller modules increases the number of connections.

Training, Procedures, Tools - This includes the total of these items at all levels, from the factory through to astronaut operation of the spacecraft.

Equipment Commonality - This is the level to which equipment can be designed such that various functions in the same or even different systems will be performed by identical modules.

Power (Cooling) - A Comparison of Coolant Pumping Power. Each of the above criteria was assigned a weighting function (at the left on Figure VII-1) based on its importance.

The various aspects of packaging that were studied using the matrix were; standard vs non-standard package design, module size, and cooling. Each of the alternatives in a given trade-off, as

listed at the top of the matrix, were given a desirability score with respect to each criteria listed at the left. The criteria's weighting function was then multiplied by each desirability score and finally, all of the scores were added to provide an indication of the respective worth of alternatives. Some additional discussion of the alternatives follow.

Standardization

Non-standardized, or customized, vs standardized packaging was considered first in the Trade-Off Matrix, Figure VII-1. Either method can be used in a spacecraft system, but the standardized approach has a number of advantages. Standardized packaging was defined to mean a restricted number of configurations with some allowance for flexibility, both of size and methods of packaging. One of the features considered as a part of standardization was the use of racks containing several systems and subsystems, roughly equivalent to packaging in a standard 19 inch laboratory equipment rack. The advantages of standardization include:

- 1) Reduced subsystem packaging design effort.
- 2) Reduced design and fabrication costs for mounting structure and coolant system.
- 3) Reduced test tooling costs for qualification tests.
- 4) Reduced qualification testing due to mechanical similarity between modules.
- 5) Reduced design/development for high humidity requirements.
- 6) Improved R/R accessibility.
- 7) Reduced training, procedures, and tools for handling R/R.
- 8) Vehicle cabling between equipment cabinets is simplified.
- 9) Mechanical form factors are well documented, hence interface problems for new systems are lessened.

Some disadvantages of standardization are:

- 1) System weight may be greater.
- 2) Interconnections within a module are probably greater because of the two dimensional size restrictions, affecting modular and functional reliability.

Cooling

The two simplest and least expensive methods for electronic assembly cooling are cold plate (forced liquid) and convection (forced air). In a non-standard package array, the cold plate is the easier to accomplish. In a standardized array, either method may be easily incorporated. Looking beyond the rack structure, however, it becomes apparent that cold plate cooling requires conductive cooling for each heat producing component in each electronic package. This requirement will significantly increase the design and fabrication costs of each package. On the other hand, with convection cooling, few of the components will need special attention. Most of the components mounted on open PC boards will be sufficiently cooled by the forced convection. The "hot spots" can be eliminated by extended surfaces such as fins, or be mounted on thermally conductive structure that contacts the air plenums, providing a heat sinking path.

Shielded modules may be handled the same as open PC boards if shielded openings are provided for air flow through the module. If provision of these openings is not possible, the internal design must provide for conductive cooling to the module mounting brackets. Additional cooling can be obtained by placing external fins on the module.

A high humidity requirement should have little impact on a convection cooled cabinet, because the cooling air can be a closed loop with air conditioning separate from that for cabin air.

The redesigned G&C Electronic System, Section VI, dissipates approximately 750 watts. With all the electronics for that system in one cabinet, the cooling air requirements will be approximately 5.6 pounds/minute. This number is based on inlet air at 70°F and exhaust air at 102°F, a temperature range considered adequate for insuring component reliability. With the ambient cabin air pressure at 10 psia, this cabinet air flow will be about 125 cfm. With the plenums sized about 1 x 6 inches this corresponds to a gas velocity of about 34 mph in the inlet and exhaust plenums.

If convection cooling cannot be used, conversion of a rack to accommodate cold plate cooling is a simple matter. The air plenums can be re-designed to carry fluid ducts, and in effect, become cold plates. The major impact of this change will be the additional design requirements for the modules.

Module Size

After the standardized system packaging approach was selected, one of the decisions to be made was how large should a module be. This was given considerable attention in the Trade-Off Matrix: Figure VII-1.

The initial module size was thought of as a PC board containing 50 flat packs without excessive crowding. This board size was then varied both larger and smaller to accommodate some different sized remove-replace electronic functions identified in the Redesign of Section VI, and studied in the Trade-Off Matrix. Then the option was introduced for varying the board size to best suit the circuits being packaged. The resulting module sizes and their nominal flat pack counts were:

Maximum - 120; large - 80; small - 30; and minimum - 15.

The preference indicated by the totals on the matrix was for the flexible approach.

Regarding modularity, the primary trade-off concerned a rigidly standardized module size as opposed to a flexible module size which we have shown would minimize malfunction detection and checkout complexity. The way in which a module is allowed to vary in size required some additional study. To vary component height on boards, and hence the board-to-board spacing, is not a major problem. However, varying the width and perhaps the length of boards immediately injects several variables into the cabinet design, and while this variability can be helpful at the module design level, it impairs many of the advantages that can be derived from standardization. In particular, the possibility for commonality becomes much smaller. Therefore, the standard PC (Printed Circuit) board was defined as one with a 50 flat pack equivalent size and the module as an assembly consisting of one or more standard PC boards. As module size becomes large, i.e., equivalent to the black boxes of Block II Apollo, the resupply weight and volume penalty become severe, mounting and cooling become inefficient, and the throw-away module becomes very expensive. Assuming that maintenance and malfunction isolation are to be performed at the removable module level, it was found desirable to let the module vary from 50 to 500 micro-circuit flat pack equivalent size, or one to ten standard PC boards. This range is drawn from a consideration of detailed studies on the re-designed Guidance and Control System, covered in Section VI of this report, with minimization of malfunction isolation complexity as the prime consideration in modularizing the system. To achieve module size flexibility over

this range, the only practical dimensional variation is the number of boards in a module.

The trade-off of various module sizes on a total system basis is shown in Figure VII-2 and indicates from the totals of the alternatives that no one size is significantly optimum. Therefore, specific penalties and advantages must be examined. The complexity of malfunction detection and the number of interunit connections are definitely affected by module size and the spread or range of the effect is significant. The capability to use commonality at R/R level is also significantly affected by module size and all three of these considerations are important to enhance IFM. Therefore, the majority of the repackaging was done at the smaller one and two board module sizes.

Some structural schemes and guidelines were generated for the design of modules. A candidate module design is provided in Figure VIII-3, and Criteria for packaging is found in Chapter V.

Module Design

In order to facilitate maintainability, we have shown that the packaging should be modular and standardized. Commonality will be promoted by making functional divisions between identical channels rather than by combining identical functions of the channels.

For many requirements, a simple PC board could be adequate whether the design incorporates discrete components, integrated circuits, MSI, or LSI. With higher density packaging, one of the considerations for functional division is connector pin limitation. With the requirements for moisture-proofing, the board and components would be coated and the connector designed as discussed later in this section.

For planar arrays, the intra-connecting base might be 1 or 2 sided PC boards, or multi-layer PC boards, or even point-to-point wiring. For volumetrically dense component packaging, many schemes are available; flat packs as well as discrete components, and cordwood modules have been used with solder as well as welded intra-connections. In addition a number of designs have utilized a combination of planar arrays with the arrays stacked so as to achieve a high packaging density in modular sub-assemblies; this has been used primarily for flat pack integrated circuits.

Some very good presentations of various packaging techniques are given by J. L. Adams (1965), J. L. Easterday, et al. (1966),

Weighting from 0 to 10 on "Goodness" Scale 0 - Worst 10 - Best	Considerations	Weighting (Importance)	MODULE SIZE				
			1 Board	2 Boards	5 to 10 Boards	Over 10 Boards	
Weighting from 0 to 10 on "Goodness" Scale 0 - Worst 10 - Best	Cost - Recurring	5	6 30	6 30	5 25	5 25	
	- Nonrecurring	5	4 20	5 25	6 30	7 35	
	System Power	--	--	--	--	--	
	System Weight	9	4 36	5 45	7 63	8 72	
	Tool Complexity (Accessibility)	--	--	--	--	--	
	Expendables - Launch	--	--	--	--	--	
	- Orbital	--	--	--	--	--	
	System Volume	2	4 8	5 10	6 12	7 14	
	Malfunction - Checkout Complexity	9	3 27	3 27	7 63	8 72	
	No. of Interunit Connections	3	4 12	5 15	8 24	9 27	
	Cannibalization	2	5 10	4 8	1 2	1 2	
	Training (Procedures) - Time	4	4 16	4 16	7 28	7 28	
	- Cost	4	4 16	4 16	7 28	7 28	
	Reliability - Long-Term	8	4 32	4 32	6 48	6 48	
	- Short-Term	--	--	--	--	--	
	Resupply - Weight, Volume, Cost	9	9 81	8 72	2 18	1 9	
	System Life Expectancy	--	--	--	--	--	
	Commonality, R&R Level	2	7 14	6 12	2 4	1 2	
	Special Features	--	--	--	--	--	
Total		302	318	345	352		

Figure VII-2 Module Size Determination Matrix

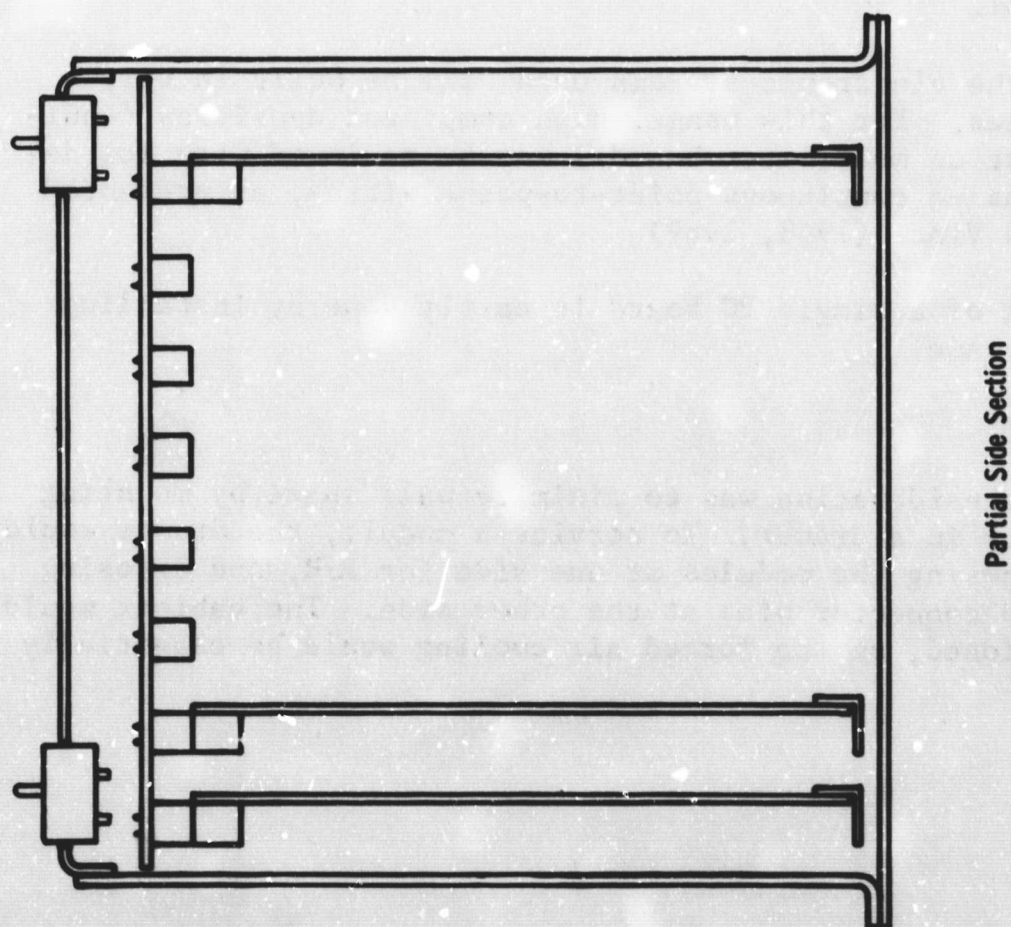
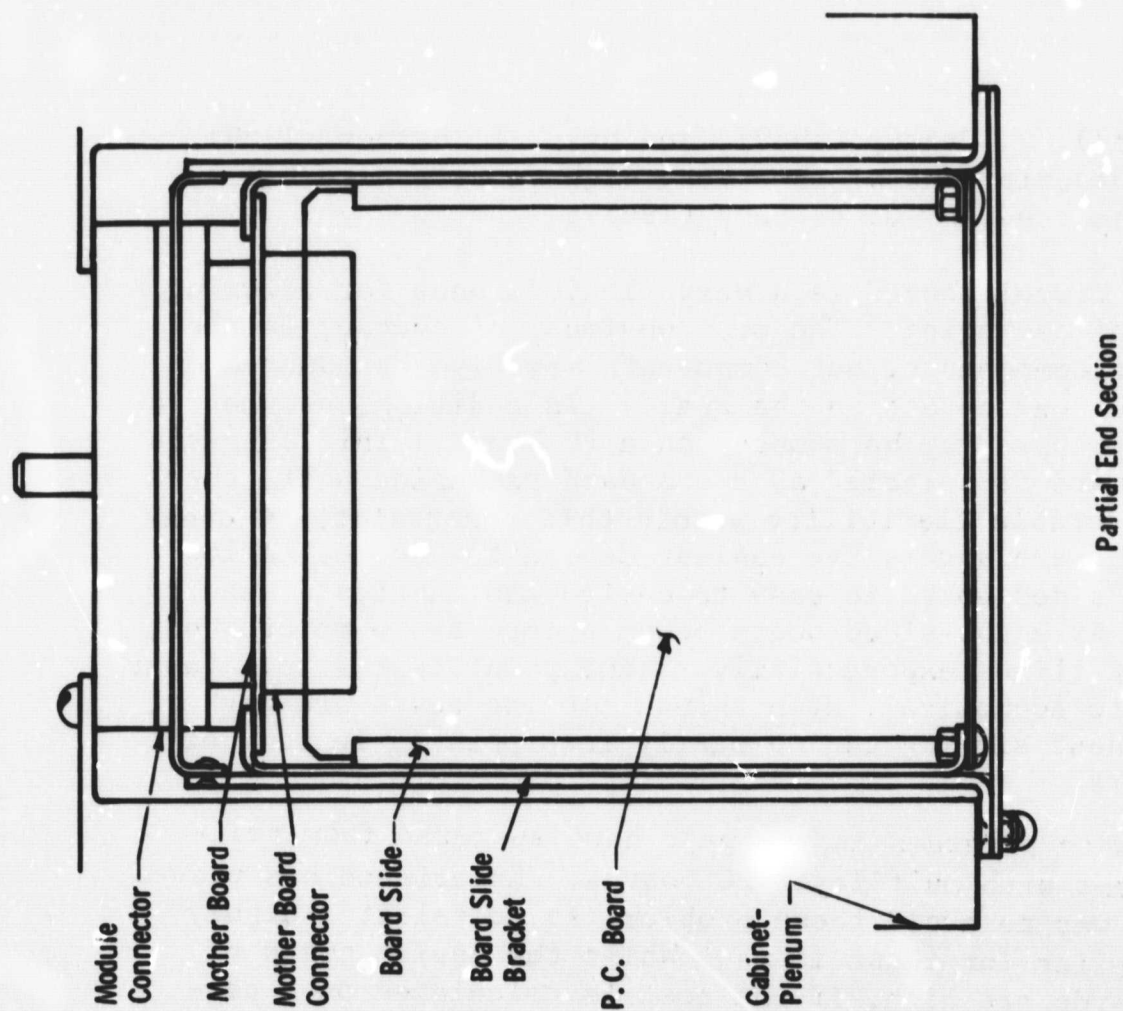


Figure VII-3 Multiboard Shielded Module

G. B. Bunn (1967), S. Drasky (1969), and by C. A. Harper (1969). These include descriptions of some very high density assemblies, particularly the study by G. B. Bunn (1967).

A printed circuit board is a very flexible base for allowing various kinds of packaging. The most obvious, of course, is the planar array of components, but components may also be mounted on end to conserve real estate on the board. In addition, volumetric modules of many types may be mounted on a PC board. This suggests that if a PC board is selected as a standard packaging unit, there is still considerable flexibility within this constraint. Conventional PC boards are surely the easiest design to handle. A low density double sided board is easy to design and fabricate, and has low cost. As a two-sided board becomes more and more crowded, the design cost climbs exponentially to the point that a multilayer PC board is more economical, even though tooling costs are higher. Shielding and heat sinking can be easily incorporated in some of the board layers.

Many hardware programs in the past have suffered from reliability problems with multilayer PC boards. Experience has proved that the best way to avoid these problems is to select a highly competent supplier for these items. While the design costs of multilayer boards are high, if the cost is calculated on a per-connection basis (for example, a resistor is two connections, a transistor is three, and a 14-lead flat pack with all leads active is 14 connections), then the per-connection design cost is lower for a multilayer board than for a considerably less crowded two-sided PC board.

Many of the electronic systems used will be built in very small quantities. For this usage, high component densities (equivalent to those on multilayer boards) may be achieved with low design cost by using continuous point-to-point wiring, as presented by Maxwell and Vinci (1968, 1969).

Shielding of a single PC board is easily done by installing it in a metal case.

Cabinet Design

A first consideration was to minimize wall space by mounting the module rack in a drawer. To service a module, the drawer would slide out, exposing the modules at one side for R/R, and exposing the wiring and connector pins at the other side. The cabinet would normally be closed, so the forced air cooling would be essentially

a closed system. The drawer-to-cabinet wiring (mating with a vehicle cabling connector) would require a service loop, but its design would be relatively simple. The drawer concept has several disadvantages, namely the complexity of the linkages that latch the drawer and cabinet, the requirement for two separate primary structures, and the long service loop in the cable which flexes upon each access.

A review of probable constraints of the system packaging suggested that a severe restriction on front cabinet panel area was unlikely. So an iteration was considered in which the front door of the cabinet was hinged, providing access to the modules for R/R. For additional testing, access to the module connectors and wiring was provided by hinging the rack in which the modules were mounted. Again, some allowance had to be made for cable flexing as the rack swung open, and some of the primary structure had to be duplicated in both the cabinet and the rack.

Another review of the system constraints suggested that since the pressure wall of the crew compartment must be accessible for repair of punctures, equipment should be mounted away from the wall. A study report in Aviation Week (Himmel, 1969) also indicates this as a preferred configuration. So a fixed cabinet with fixed module racks could be used, with hinged doors front and rear providing access to modules and wiring. Doubling up of primary structure would be eliminated and there would be no cable flexing. This cabinet is shown in Figure VII-4. Details of the cooling air flow are illustrated in Figure VII-5. To provide insight regarding the volume requirements for the cabinet and module approach taken here, the redesigned G&C and Telecommunications modules, listed in Table VI-1 and VI-2, are shown installed in electronics cabinets in Figures VII-6 and VII-7.

Modularization of Connectors and Cabling

A long life space station presents some unique and challenging requirements and possibilities, enhanced by the 90-day resupply vehicle providing replacement personnel, parts and supplies.

During the life of the space station, we can expect technologies to change to the extent that we may want or need to replace complete systems, as pointed out by Mueller (1969), and requested in the Space Station Statement of Work (NASA, 1969a). Major modification or replacement will require replacing cabling in the rack, and allowance for making these changes can be made by modularizing the module connectors and cabling. This can be done by adding additional connectors to interface between components of a system that are mounted on different shelves of a rack.

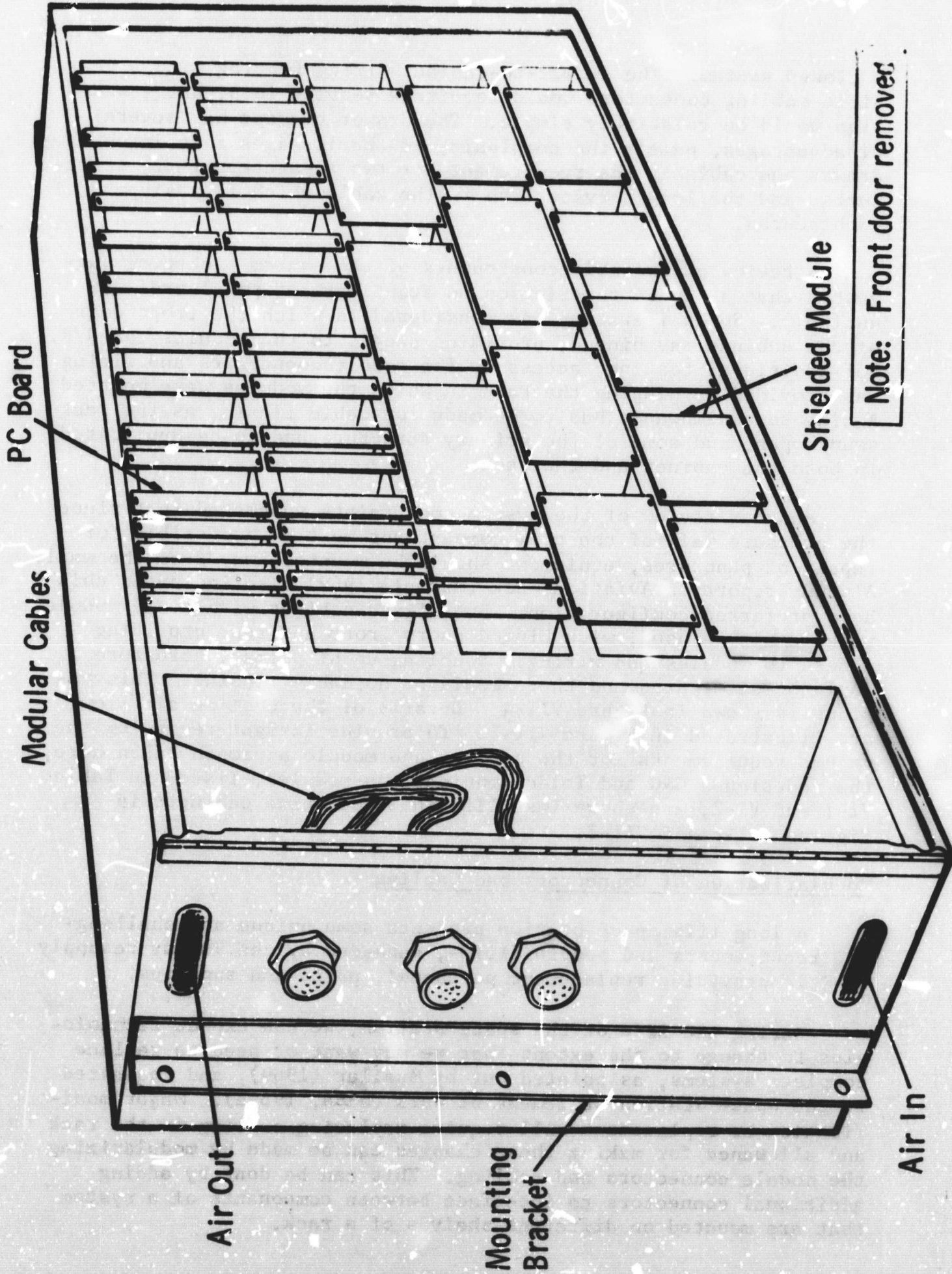
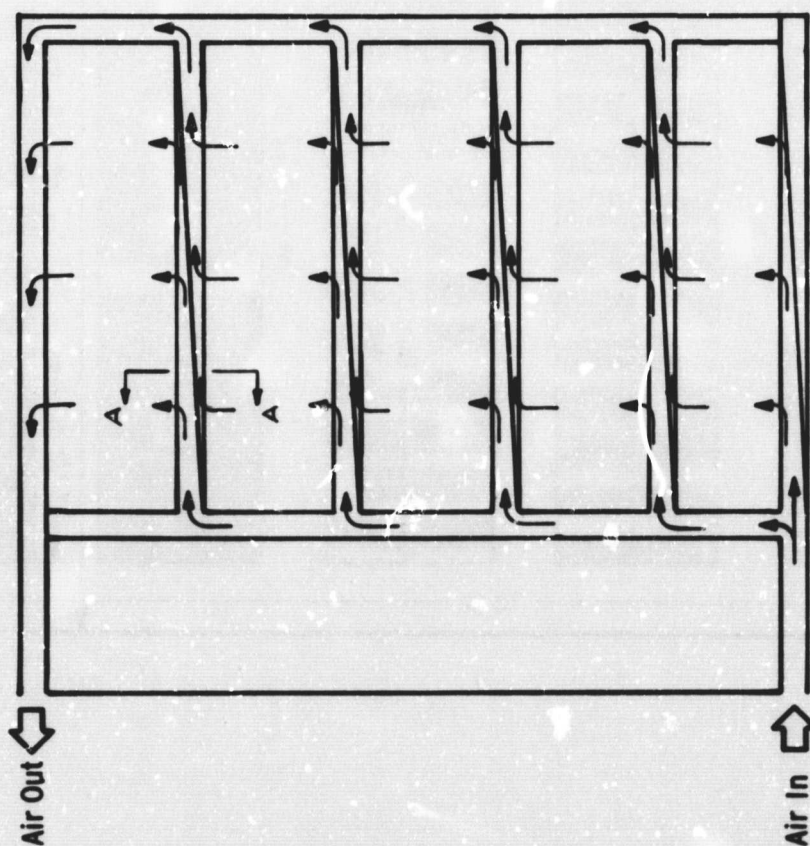
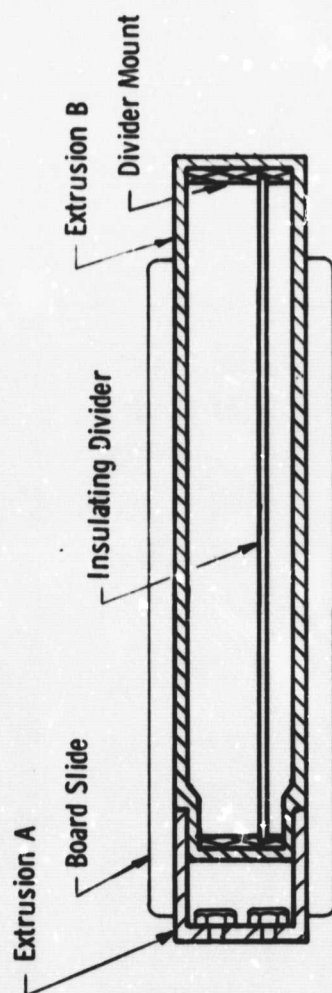


Figure VII-4 Electronics Cabinet



Cooling Air Flow Pattern
In Electronics Cabinet



Detail A-A of Air Pleum (Enlarged)

Figure VII-5

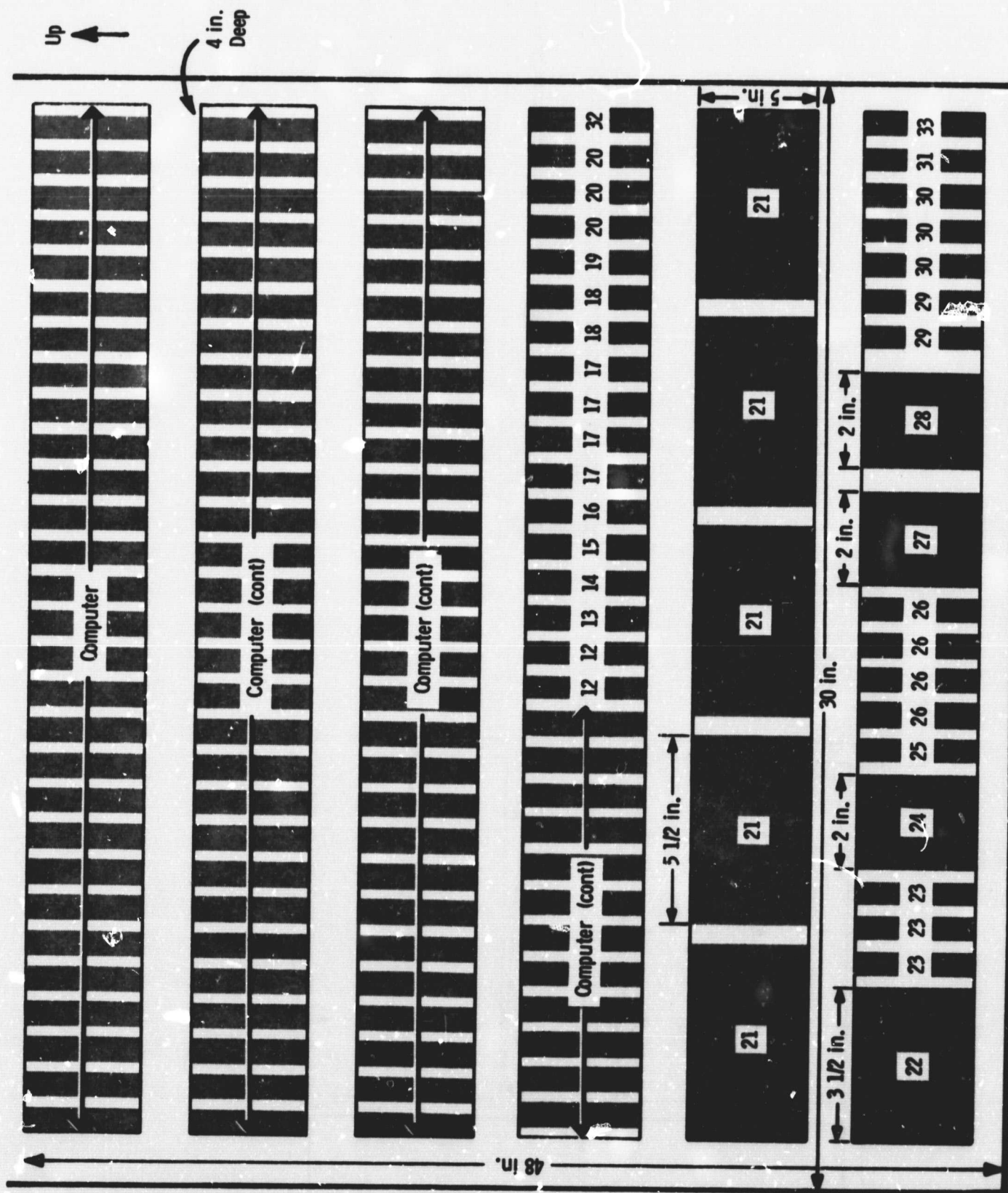


Figure VII-6 G&C Electronics Cabinet

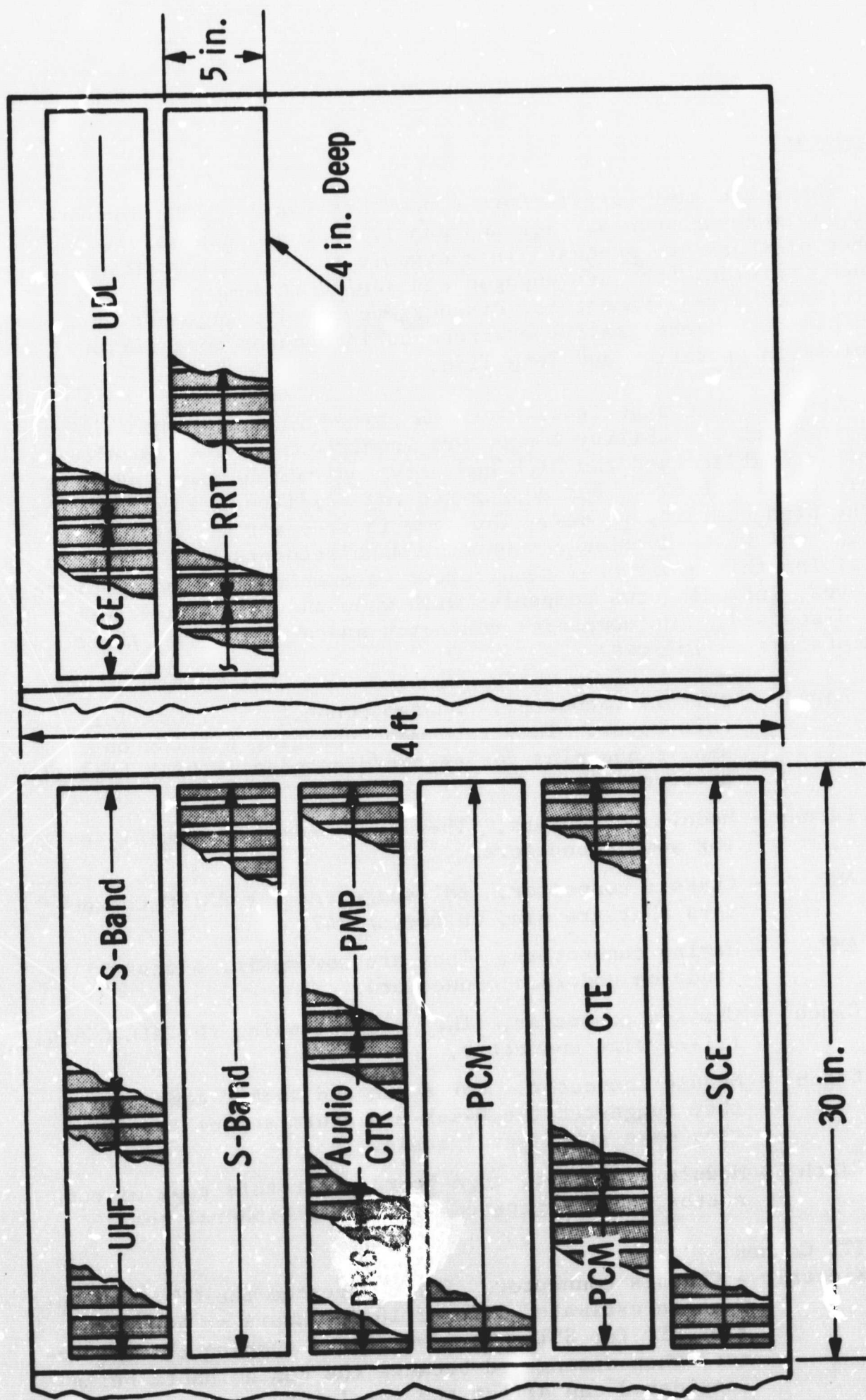


Figure VII-7 Telecommunication Electronics Cabinets (Modules Installed)

Connectors

There are many satisfactory connectors for use in a laboratory environment such as rack and panel, PC board and module. A number of these are adequate in a dynamic environment such as a launch, provided they are engaged and sealed in some way. The addition of a requirement for disengagement and engagement in a high humidity, zero gravity environment introduces some serious problems in operation and long life.

Considerable experience has been gained in selecting and applying high reliability connectors in airborne and space electronic assemblies and the high humidity environment for a chassis connector is not new. But adding the engage/disengage requirement in the high humidity is new. The same is true for module connectors. To find out what experience connector manufacturers have had with satisfying this additional requirement, a number of companies were surveyed, including two companies with whom undersea connectors were discussed. The companies contacted and a summary of their comments are as follows:

- Burndy - Chassis connector. Bantam connectors are ok for salt water. They recommend changing plating on shells and pins for engage/disengage in high humidity environment.
- Burndy - Module connectors. They have made interfacial seals for some connectors.
- AMP - Chassis connector. AMP suggested the ARINC connectors that are used on Boeing 747.
- AMP - Marine connectors. They are now making a feasibility study on undersea connectors.
- Cinch - Chassis connector. They suggest using the MIL-C-81511 tri-service connector.
- Cinch - Module connector. For a pin and socket connector, they suggested the D-sub-miniature series for which they make interfacial seals.
- Cinch - Module connector, edge board. For this type of connector, they suggested making a peripheral seal.
- ITT Cannon
Electric - Chassis connector. They suggested the CV connector with an estimated life of 10,000 hours at 400°F in free air for SST. They suggested capping the connectors when disengaged because the contaminants being considered can affect the insulators.

- Viking Industries - Marine bulkhead connectors. They have not built undersea make-and-break connectors, but as company funds permit they are currently working on a design for this.
- Amphenol - Chassis connectors. The 348 series connectors have a positive pin-to-pin interfacial seal that should provide positive insulation against water droplets.
- Amphenol - Marine bulkhead connectors. They make two series but not for engage/disengage. For salt water environments they suggested sealing wire-to-solder joint -to-connector using shrinkable tubing over the wire and solder joint. This is then coated with GE Marine Seal. This treatment has lasted for 5 years on marine equipment subjected to ocean waves.
- Deutsch - Chassis connectors. The DBA series connectors have a positive pin-to-pin interfacial seal that should insulate against water drops.

The companies surveyed had several general comments regarding connectors for engage/disengage use in a high humidity environment. They were:

- 1) Power should be off when engaging/disengaging.
- 2) Connectors should be blown off with a dry gas or an absorbent material should be provided to remove moisture droplets before engaging.
- 3) Platings should be selected; possibly chrome for connector shells and heavy gold for connector pints.

Testing and/or development of connectors for high humidity engagement is an extensive undertaking. However, there is a large fund of experience in some aspects of the problem. In a NASA funded IBM Study (IBM, 1965) PC board pin and socket connectors were grease impregnated and sealed with a rubber squeegee. These have been operated and disengaged and engaged in water containing salt and urea and then tested for electrical resistance. Rack and panel connectors have been built that offer positive plastic seals around each individual pin.

The above indicates that a minimum amount of additional testing and development will result in connectors that will perform well

through disconnect-connect, or remove-replace in a high humidity, zero gravity environment.

Even an edge-board connector for a PC board can be developed. A possible configuration is shown in Figure VII-8 wherein a plastic squeegee is bonded to the connector face to seal out moisture droplets when the board is out of the connector. The connector fingers on the PC board could be coated with a high wettability, water-proof oil to provide additional protection.

Shielding

In addition to modular shielding discussed above, subsystem and system shielding may be designed into the cabinet, consisting of shielded cable races and panels, or shrouds, that compartmentalize the racks. These items combined with modular cabling would make it possible for a vendor to supply both system modules and system cabling and connectors. All items including electronic, cabling and connector modules would be checked out before delivery, and can be plugged into the rack.

Idea Model

A model was built that is a truncated mockup of the last iteration for the standardized cabinet. Photos of the model are provided in Figures VII-9, -10 and -11 at the end of this section. It incorporates the provision for modular cabling. A module mounting frame member was fabricated to be a close approximation of the detail involved in a dual air plenum. Several types of modules and PC boards are included in the model, using several kinds of connectors. The major difference in this model and expected design is that the final detail would consist of a pair of simple extrusions rather than a combination of detail parts. The intent in this, as in all other parts of the model is to stimulate ideas - possibly for totally different concepts - in addition to demonstrating the flexibility of this particular concept.

The connectors included in the model are:

Masterite Industries

005AD55-DR-I-D

Printed circuit edge connector

110 contacts (55 per side)

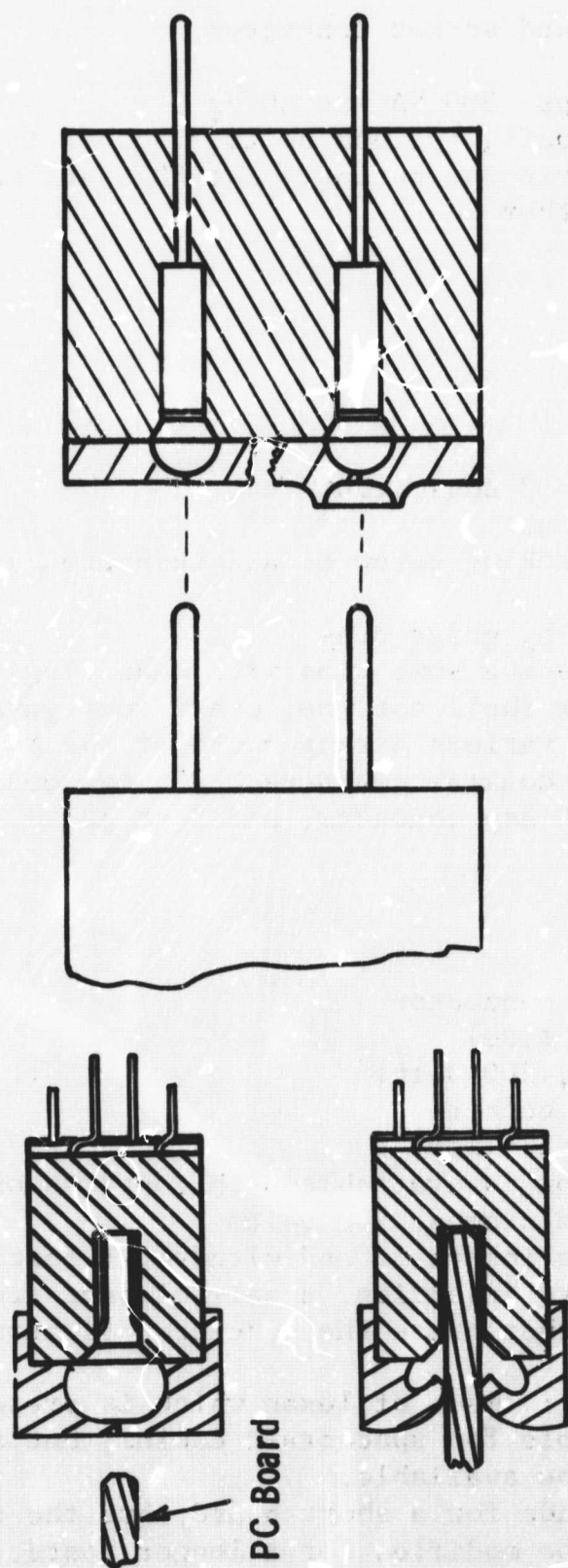
Contact rating -0.5 amp., 250 VAC

contact on .05 inch center

Other configurations available - 120 contacts maximum

May be keyed - each key uses 4 contact positions

Also available in .100 inch contact spacing with 3 amp, 1200 VAC rating.



PC board pin and socket connectors with two alternatives for elastomeric seal

PC edge board connector with elastomeric squeeze seal

Figure VII-8 Candidate Connectors

Burndy Corporation

UPC2A41P-4

UPC2A41R-4

Printed circuit pin and socket connector

41 contacts

Contact rating -3 amps, 300 VAC at 60 Hz

Available in other configurations up to 92 contacts

No provisions available for keying. The PC board may be keyed to the mother board

Cinch Manufacturing

65-608A-1

65-608A-2

Module connector

92 contacts

No. 20 contacts rated 3 amps, 1500 VAC, RMS

Solder pot terminations

Connector contains jacking screw to aid insertion and withdrawal

Shells are polarized by guide pins

May be keyed by replacing some pins with dust plugs

In the same connector shell outline, other configurations are available with various arrangements of the No. 20 contacts, standard coaxial contacts, Plaxial coaxial contacts, 40 and 80 amp contacts, and test jacks.

Cinch Manufacturing

65-721

Printed circuit edge connector

114 contacts (57 per side)

Contact rating 1 amp, 500 volts

Contacts on .05 inch centers

May be keyed using nylon keys

The same configuration is available with contacts on .100 inch centers, rated 3 amps, 500 volts

This design is a zero insertion and withdrawal force connector. After board insertion, a pair of cams are turned actuating devices that press the connector fingers against the board.

This connector shell is made of lexan which is presently considered unsuitable for spacecraft cabins, but a suitable substitute should be available.

This connector was made for a short board, but the design and tooling could be modified for a longer board.

Deutsch

DBA series

RBC74H-19-OPN

DBA77-19-OSN

Cable connector

Push-pull quick disconnect coupling with positive ball-lock coupling mechanism

Contacts are crimp terminated, rear insertion rear removal

Available with up to 61 contacts

Triple seals clamp around each wire at rear entrance and a positive interfacial seal around each pin is provided at the connector interface.

Deutsch

RSM series

RSM5105-12-14P

RSM07-12-14S

A miniature series cable connector

Available with up to 91 contacts

Same push-pull feature as the DBA series

Crimp connections

The jam-nut-mounted panel connector in the model has a hermetically sealed insert. Wire connection to this is with a crimp-on, rear insertion, snap-in connection.

The connector in the model does not have the stepped interfacial seal around each pin that is in the DBS series connector. However new seal molds have been made, and this stepped seal is on new builds.



Figure VII-9 Electronics Cabinet

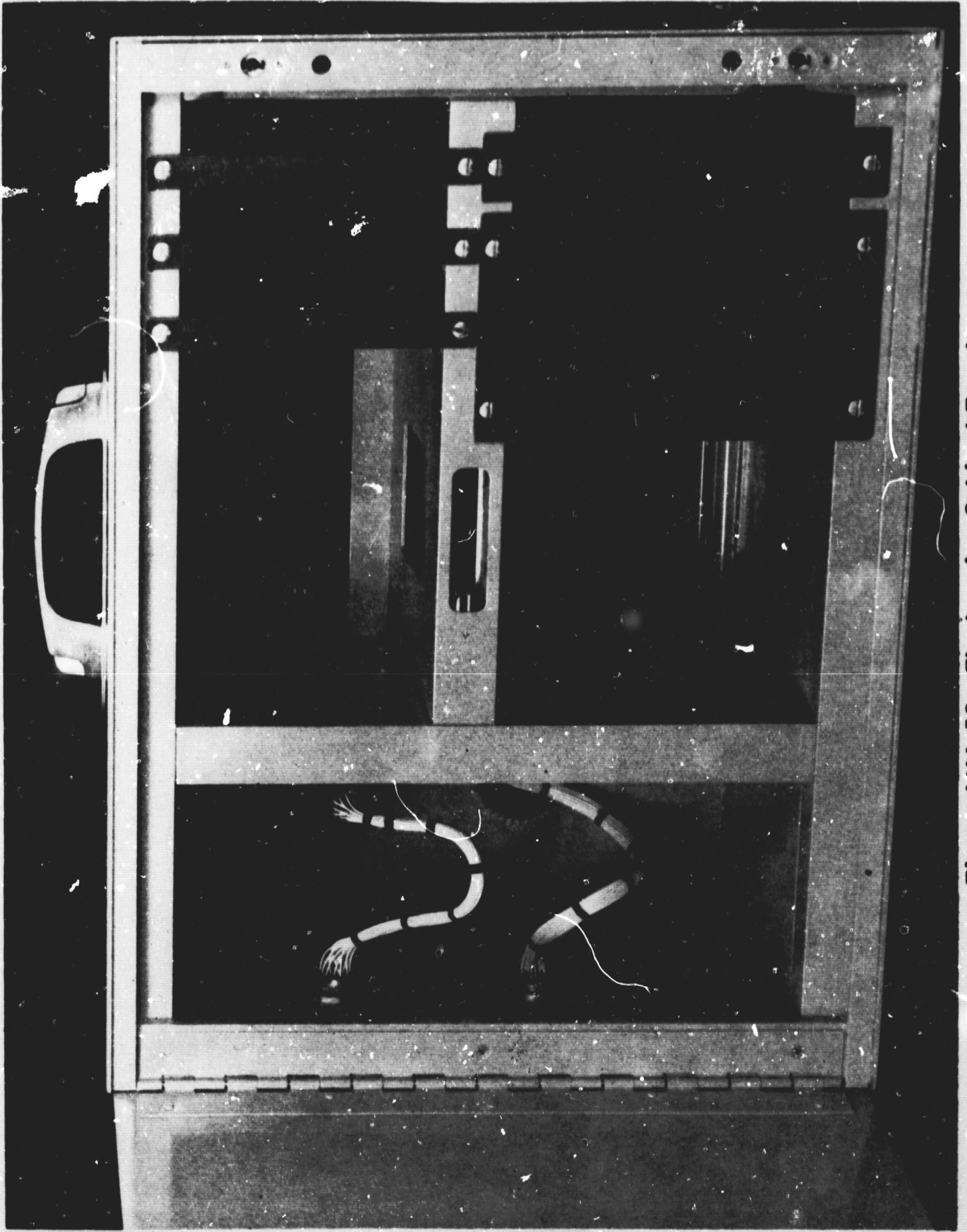


Figure VII-10 Electronics Cabinet-Front



Figure VII-11 Electronics Cabinet-Rear

VIII - TEST AND CHECKOUT

The test and checkout function, as defined later in this section, is a prime consideration for in-flight maintenance, and will be found interwoven with virtually every other aspect of our study. In examining the test and checkout philosophy and requirements, it was considered necessary to seek the counsel of key personnel from the following programs: Computerized Automatic Ground Equipment (CAGE), On-Board Checkout System (OCS), Multi-Parameter Display (MPD), and Built-In Checkout (BIC).

Their thoughts and experience, along with the trade-offs in this study, a research of the literature, and an in-depth analysis and redesign of the Apollo Block II G&N, SCS and Telecommunications System all shaped the final conclusions. Throughout, it should be recognized that the prime concern was with malfunction isolation, with lesser emphasis on operational checks or data gathering.

The test and checkout system must perform the three principle tasks of critical failure monitoring, failure isolation, and system checkout and verification. There is some interrelationship between the three functions, but they are basically three distinct tasks. The following paragraphs will discuss the requirements of these tasks and then develop the rationale for designing a system to meet the overall set of requirements.

Failure Monitoring

The equipment used in failure monitoring is often called the caution and warning (CAW) system. This equipment must continuously monitor critical sub-systems, such as life support subsystems, and issue an audio-visual alarm if abnormal operation is detected. Certain important but non-critical signals such as power supply voltages are often monitored by the CAW system as well. This monitoring function is continuous and automatic and involves a comparison of measured parameters with predetermined tolerances. Any out-of-tolerance condition results in one or more alarms. Since many of the monitored parameters are life critical, redundancy is required.

A goal would be to achieve 100% detection of all critical malfunctions. The degree to which this can be achieved is determined by the ability of the designers of the systems monitored to predict the failure modes of those systems and the methods by which the failures can be detected. For the critical parameters,

it is assumed the ability to detect malfunctions will not be compromised to reduce the complexity of the CAW system. For the non-critical parameters a trade-off will be discussed in more detail later regarding overall test and checkout system design.

The types of signals monitored by the CAW system include analog, bi-level and digital. In order to process these signals, store the tolerances, make the necessary decision, and take the required actions, some special purpose or general purpose computer is indicated. It is possible, but not necessarily desirable, to share this function with some other function such as guidance and control; in fact, Mueller (1969) has stated the desirability of decentralizing the computer functions. This trade-off must be made considering the overall electronics system requirements.

In addition to generating alarms, certain types of failures may require immediate corrective action which cannot wait for operator intervention. An example of this might be fire detection. For those cases the CAW system would provide for an automatic command capability to take the necessary corrective action.

Many of the malfunctions which can occur on the spacecraft need not be monitored, and will be determined by the crew during normal operational activity. Certain communications or vehicle and experiment control operations fall in this category. Manually selectable backup modes may be employed until the malfunction can be corrected. This is typical of present-day aircraft operation.

Failure Isolation

It is not practical to design equipment with a 100% probability of no failures over a ten year life span. Therefore, some degree of repair capability is required, however, in order to repair a system it is necessary to isolate the malfunction. For this study the repair was limited to remove and replace - a very reasonable ground rule, since repair to the component level requires extremely sophisticated and complex test equipment, and a high level of crew knowledge and training. In addition it requires operations such as decapsulation soldering, encapsulation soldering and welding which will be difficult and perhaps impossible in the first space station environments. As experience in space station operations increases, this ground rule will need to be re-evaluated, but it seems likely that economics will favor the remove and replace concept for a considerable period of time.

It is possible to achieve a high degree of confidence of isolating a failure with a malfunction isolation system by designing very versatile and complex equipment capable of providing a wide variety of stimuli and of measuring a multitude of signal types. This approach places no constraints on the design of the systems to be tested, but will usually result in a malfunction isolation system which is more complex than necessary for any particular application. The operator training requirements may also be large.

A preferable approach is to force the electronics system designers to make their systems testable in a relatively simple fashion using a minimum of external stimuli. If the stimuli are peculiar to that equipment, serious consideration should be given to integral stimulus circuits requiring only a discrete or digital command to control them. This design for testability has the dual advantage of simplifying the test equipment and of increasing the probability that a malfunction can be isolated. The malfunction isolation system can be designed to fit the particular application. It should be modular in design so that changes and additions to the electronics systems can be accommodated easily. This ability will be aided by the use of a standard testing philosophy which will be used wherever possible.

The malfunction isolation system can be made to have almost any desired degree of automatic test capability. In general, some operator action will be required, and a trade-off must be made between operator training requirements and test equipment complexity and reliability.

The malfunction isolation system must be able to isolate a failure to the lowest replaceable unit with a reasonable probability of success without any assistance external to the spacecraft. The probability of success is a function of test equipment complexity and operator training, which requires that a trade-off be made for each particular application. The Space Base Study (NASA-1969b) indicates that a 95% to 98% probability of success is desired.

System Verification

It is necessary to verify the electronic system operation prior to critical maneuvers and following any repair actions. The test points for system verification are most likely a subset of those used for malfunction isolation, but the stimuli requirements may be different. Certainly the test procedure will be different for the two functions.

The functions of malfunction isolation and system verification will not be required simultaneously, therefore an effort should be made to share hardware between these two functions. This sharing of hardware is easily realized in a test system built around a general purpose computer.

The system verification function also performs malfunction detection for those non-critical parameters not monitored by the CAW system. A related task which could also be performed is that of trend analysis. By keeping a history of the measured parameters, it is often possible to predict a malfunction before it occurs. This is a particularly important consideration for the life critical systems. Trend analysis requires large data storage capability and moderately large amounts of calculation time, however, need not be done on a real-time basis. Where trend analysis is a requirement, a trade-off needs to be made between on-board and ground processing. This is one area where autonomous operation is not mandatory.

System verification in the event of replacement of a complete system is a still different consideration. As technology advances, certainly some systems will be rendered obsolete, and replacement will be desirable. It is recommended that such new systems be verified in a house spacecraft, duplicating the structure and systems in the orbiting space station. Such a house spacecraft would have the following uses related to this study:

- 1) Insure electromagnetic compatibility.
- 2) Test form, fit and function for new products or product improvement cycles.
- 3) Provide simulation to determine operational limits of systems or subsystems.
- 4) Verify hardware and software for malfunction isolation and detection.
- 5) Verify hardware, software and procedures for on-board support equipment utilization.

System Design Considerations

For the initial discussion of the system design, the CAW system will be ignored. It has specific requirements, many of which cannot be compromised. It is of necessity a centralized automatic monitoring system. After consideration of the malfunction isolation and system verification design tradeoffs, the CAW system will again be considered as to how it might blend with the other two functions.

Consider using the same equipment for malfunction isolation and system verification, even though the test procedures will be different. It is possible to consider something as simple as a multimeter for measuring AC and DC voltages at various test points within the system. An analysis was made of the Block II Apollo systems which indicated that most catastrophic failures and many out-of-tolerance failures could be detected and isolated with this simple testing instrument. The operator training requirements would be quite large because an intimate knowledge of the systems under test is required. The confidence level for either failure detection or isolation is not adequate. In summary, this is not an acceptable check-out system, however it is of interest to consider a trade-off of a centralized vs a decentralized system on this simple testing concept.

Assume for now that all required stimuli are built into the systems under test. The decentralized system would require an operator to move about the spacecraft activating the required stimuli and measuring the necessary parameters. Two or more operators may be required for certain tests, creating some coordination problems. There will have to be many multimeters built into the equipment, or the operator will carry one instrument which he will attach to the test points. The first option requires a large duplication of equipment; the second produces a system hazard from operator error and possible connector contamination. The operator would have to carry the maintenance manuals with him, and these are voluminous. Many of the functional loops traverse a large portion of the spacecraft, and would require extensive operator travel to check them out.

The centralized system would have all test points and stimuli control lines available at a central location. The weight penalty of wiring can be nearly eliminated by the use of multiplexed data bus techniques. The operator need not move about and has all reference manuals conveniently available. There is no requirement for uncapping connectors or probing test points so that operator error and contamination hazard are minimized. There has been some additional multiplexing hardware required, but this is considerably less than the additional hardware required for many built-in multimeters. The conclusion is that a centralized system is desirable for even this simple approach. The decision becomes even more obvious as the test equipment complexity becomes greater.

It is now of interest to consider the deficiencies of the centralized multimeter system. The multimeter cannot interface directly with digital information, particularly where the data is serial and possibly encoded in some manner. It is, therefore, necessary to have some digital data processing capability built into the central unit.

Care must be taken to control the individual system designers so that the variety of digital signal types does not get out of hand. The multimeter does not have the ability to measure transient responses. This type of measurement is often important and therefore should be included. The exact implementation can be fairly simple such as a timer and a sample-and-hold circuit, or complex such as an oscilloscope. The initial consideration was to have all stimuli built into the systems under test. It is more efficient to have the more common stimulus types with multiple usage in the central test equipment and multiplexed to the required points. The unique stimuli should remain built into the systems under test.

The process of increasing the testing capability can go on almost indefinitely, going to computers and automatic testing procedures, etc. This other extreme, a multimillion dollar computerized checkout system, perhaps functionally typified by the ground checkout systems like ACE and CAGE (Titen IIIM system), is also not acceptable.

The problem then is to come to a reasonable compromise. Figure VIII-1 is a graphic illustration of some of the more important considerations. It is desirable to have a large confidence that failures will be detected and isolated. It is evident that the more complex the checkout system becomes, the larger the confidence. It is also clear that the confidence is never 100% and that the curve becomes quite flat beyond some value of complexity. To achieve 100% confidence, special equipment with a team of experts may need to be brought up from Earth on a demand basis, or even kept on the station for some initial activation period.

Another important factor is checkout system reliability. The larger and more complex the checkout system, the lower its reliability and the more stringent are its calibration requirements. The large effort required to keep electronics lab test equipment in proper working order and adjustment provides an insight into the problem.

A third important factor is operator training and intelligence requirements. As previously stated, the simplest checkout system requires of the operator an intimate knowledge of the system under test. He must be capable of making decisions based on very subtle system performance changes. On the other extreme the very complex checkout system requires the operator have an intimate knowledge of the checkout system and its capabilities. At some intermediate point this requirement is a minimum.

The exact shape of the curves in Figure VIII can only be determined for a specific example. The correct point on the curves

Legend:

Checkout System Reliability —————

Confidence of Isolating Malfunction ————

Operator Intelligence & Training ————

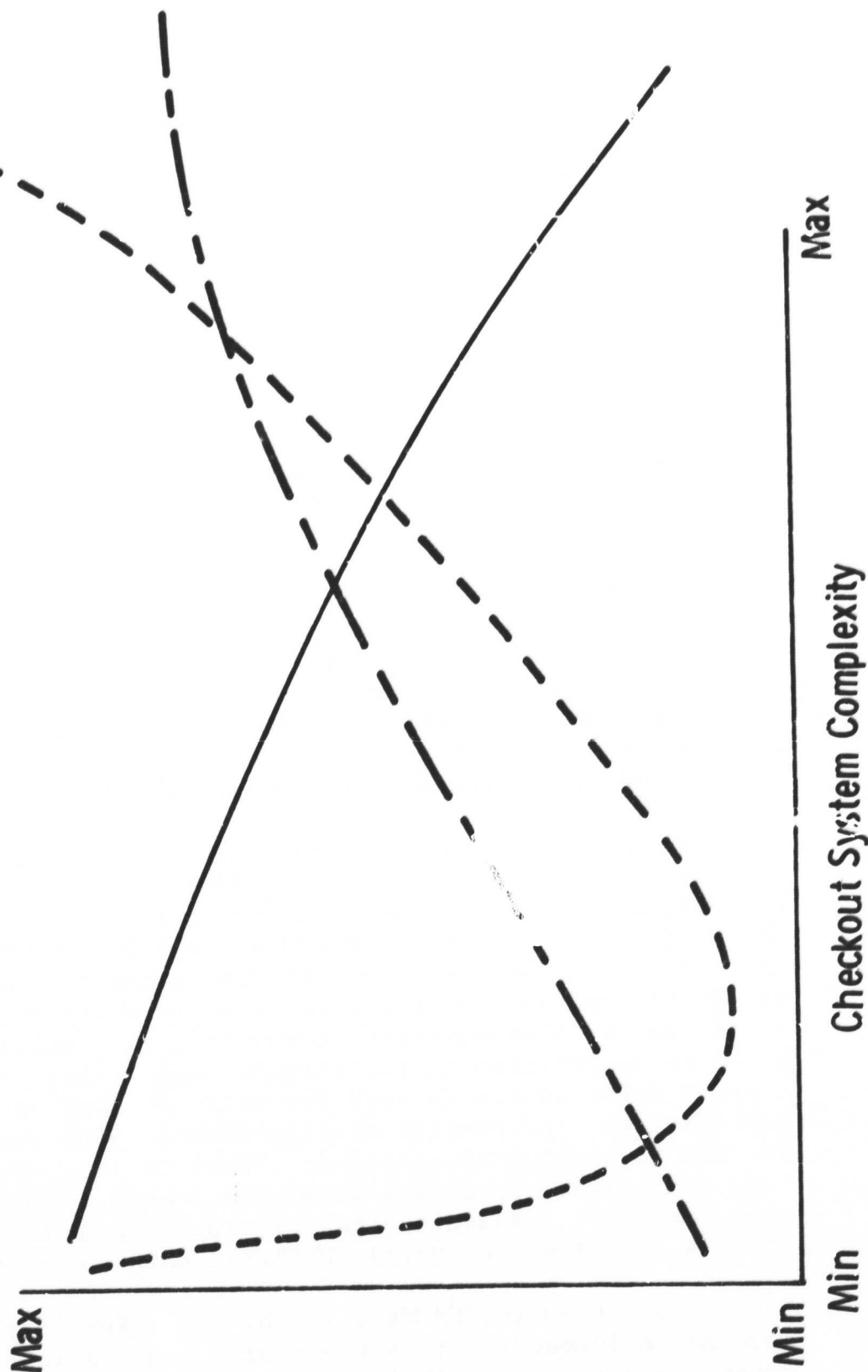


Figure VIII-1 Checkout System Tradeoff Parameters

can be selected by applying the IFM matrix introduced earlier to the checkout system trade-off as demonstrated in the example below.

Once the checkout system of the correct complexity has been selected for malfunction isolation and system verification, the CAW system can be integrated into the system to the extent possible or desirable. The overall checkout system will probably be redundant, with the possibility of having operator consoles in more than one spacecraft compartment. The redundant systems should also have the capability of checking each other.

Example

Three malfunction isolation systems will be considered for the redesigned G&C system. All three are centralized systems, however the sophistication of the systems is widely variant. The IFM matrix of Figure VIII-2 formalizes the trade-off thought processes used in evaluating these systems.

This evaluation is based on only one system, the redesigned G&C (Section VI), and on weighting factors which are subject to discussion; consequently it is intended to serve as an example for illustrative purposes only. Refer to Figures VI-13, SPS ON/OFF circuit or VI-15, Rate/Attitude Error Display, as typical of the circuits considered during this trade-off. The following defines the three systems, and develops the rationale for the weighting factors and matrix entries.

The simplest system consists of a multimeter with manual range selection. The meter can measure AC and DC voltages and currents. The data points are accessed through a remote multiplexed PAM data buss. There are stimuli provided where not already inherent in the G&C systems hardware. The intermediate system has the measuring and stimulus capability of the simple system, and in addition a time controlled sample and hold measurement capability. A central step function generator is an addition to the stimulus capability. A Multi-Parameter Display (MPD) device is used for meter display as well as instruction and tolerance information display (Martin Marietta Corporation, 1966). The MPD is keyboard controlled. There is automatic call-up of display formats for each requested data measurement. The most complex system is a complete On-Board Checkout System (Martin Marietta Corporation 1969) as developed by Martin Marietta Corporation for NASA-MSC.

From the basic IFM Matrix, the following items were judged to have no influence for this trade-off; expendables, number of inherent interconnections, cannibalization, tool complexity, short term reliability, system life expectancy and commonality. Malfunction detection/checkout complexity is not included since that is the trade-off being made.

Assumption: Centralized System	IFM MATRIX			
	Weighting	Simple Meter, Manual Switching	Intermediate Complexity	OCS (Computerized)
Considerations				
Cost - Nonrecurring	3	10 30	6 18	3 9
- Recurring	3	3 9	6 18	9 27
System Power } System Weight } System Volume }	5	9 45	6 30	3 15
Expendables - Launch	—	—	—	—
- Orbital	—	—	—	—
Malfunction Detection/Checkout Complexity	Item under study			
Number of Interunit Interconnections	—	—	—	—
Cannabilization	—	—	—	—
Training, (Procedures) - Time } - Cost }	7	2 14	9 63	5 35
Tool Complexity - (Accessibility)	—	—	—	—
Reliability - Long Term	8	10 80	8 64	5 40
- Short Term	—	—	—	—
Resupply Weight & Volume, Cost	3	9 27	7 21	4 12
System Life Expectancy	—	—	—	—
Special Features - Confidence of Isolating Malfunction	10	3 30	6 60	10 100
Commonality (R/R Level)	—	—	—	—
Total		235	274	238

Figure VII-2 Malfunction Isolation System for Redesigned G&C System

The special feature of confidence in isolating a malfunction was considered the pacing parameter. All other parameter weights were selected relative to this one. The recurring cost parameter is significant in this application, as it includes the cost factor associated with bringing up special personnel and equipment to isolate failures that the checkout system failed to isolate.

As indicated by the totals on Figure VIII-2 the trade-off resulted in the selection of a medium complexity system. It should be remembered that is based on a Guidance and Control system that was redesigned to facilitate test and checkout and is comprised primarily of signal processing electronics plus a few reasonably straightforward sensors. A similar analysis dealing with a system of many relatively inaccessible sensors, controls and actuating devices will undoubtedly modify the conclusions. Nonetheless, the example illustrates what is felt to be a useful technique for evaluating test and checkout requirements and arriving at an optimum system.

The optimum test and checkout system, incidentally, should take advantage of technology being advanced in several systems currently under development, including the On-Board Checkout System (OCS) and the Built-In Checkout System (BIC). (The latter is similar to a concept known as BITE; see Logicki, 1968).

We have concluded from this study that the designers should:

- 1) Design modules for testability.
- 2) Minimize externally generated stimuli.
- 3) Minimize test points for unique stimuli.
- 4) Build stimulus generators into modules.
- 5) Condition unique or unusual signals to normal level within the module.
- 6) Utilize BIT or BIST* techniques wherever advantageous.

It may be noted that all of the above are proposed in an effort to reduce total system complexity (system under test and checkout system) while maintaining or increasing the level of confidence of isolating a malfunction or obtaining the desired data.

* BIT - Built In Test. This concept indicates module status in a passive way - no stimulus required.

BIST - Built In System Test. Stimuli may be required, but are furnished by other system elements.

IX. CONCLUSIONS

The conclusions from this study can be grouped into three major categories; program considerations, system considerations, and areas for future study. The program considerations are those tools and techniques which are important for program management from the top levels down through the detailed system designer, and are considerations which will affect the resulting system design. The system considerations are particular areas of concern which require continued emphasis throughout all phases of the program. The following paragraphs will discuss the three categories.

Program Considerations

The ground rules, constraints and criteria should be identified early in any development program, and kept current by updating them as the program proceeds. It is necessary to ensure appropriate consideration of these items and to enable evaluation of the decisions made using them as a basis. The IFM matrix developed during this study has proven to be an excellent means of accomplishing the above requirement (objective). When the ground rules, constraints and criteria are entered into the matrix with appropriate weighting factors, it becomes a tool for trade-off decision-making at all levels of management and through all phases of the program, and also serves as a quick reference method for documenting the decisions and their supporting analyses. Another tool developed during the study which has proven very valuable, is the Boolean equation approach to failure mode analysis. This approach allows rapid identification of problem areas and overdesigned areas.

System Considerations

The considerations listed below were selected because of their program importance and the lack of emphasis placed on these items on past programs.

1. Standard packaging must become a reality to reduce cost and improve system reliability. The example presented is a rack and panel approach with a closed air cooling system. This is not presented as the optimum solution, but as an illustration that a standard packaging approach is feasible and that difficult problems like cooling and connectors can be solved.

2. The basic system design must include checkout considerations for all long life missions, since complete reliance on redundancy for long life is not feasible, therefore repair capability is a requirement. The checkout system complexity can be affected significantly by designing the system to be testable. Within the ground rules, criteria and limitations of this study, it was concluded that the checkout system should be centralized and of intermediate complexity with a limited amount of built-in stimuli in the system under test.
3. The remove and replace module size should be determined by choosing a system functional block as a module. This simplifies checkout, provides the maximum commonality for modules of appropriate size for remove and replace, and does not unduly complicate the standard packaging approach.
4. Use of standard packaging and common functional modules needs to be accomplished on an interprogram basis.
5. The two specific areas of power supplies and displays are prime candidates for commonality and standard designs which can be applied within a program as well as across several programs.

Areas for Additional Study

The following areas have been identified as being particularly important to in-flight maintenance, but which were not covered in this study. Some of these could have fallen within the scope of this effort, but were excluded because of limitations on budget.

Sensors and Actuators - Unlike processing electronics, sensors and actuators exhibit a diversity of package types, electro-mechanical devices and special features. This diversity makes generalization difficult, but a maintenance philosophy concerning these items must be developed concurrently with that of processing electronics. Many of the criteria developed in the present study will doubtlessly be applicable, but a sizable cross section of the sensors and actuators should be studied in this respect, with new criteria being generated as necessary.

The checkout and calibration of these devices also presents a challenge that should be worked along with in-flight maintenance.

Mechanics and Hydraulics - Presently an in-flight maintenance study is underway which concerns propulsion systems, but if none are in progress on mechanics and hydraulics, they should be initiated. The criteria developed should be correlated with those developed for electronics to insure a unified approach to IFM.

Display and Controls - While greatly different than sensors and actuators, the displays and controls likewise exhibit a great diversity and often uniqueness. A comprehensive study of in-flight maintainability of these items is needed.

Checkout and Malfunction Isolation - Our checkout and malfunction isolation philosophy was drawn from a set of requirements that admittedly do not cover the totality of electronics to be found in a spacecraft. Further requirements from such areas as sensors, actuators, displays, controls and processing electronics other than those studied are necessary in order to develop the proper configuration for a checkout and malfunction isolation system. It is our recommendation that the checkout system be structured to meet requirements, not vice-versa.

Interprogram Commonality - It seems feasible to design and package electronic modules so that they would be interchangeable between dissimilar vehicles, for instance the space base and space shuttle. While we found a disappointingly low module commonality in the systems we studied, it is nonetheless probable that different programs will indeed need some similar functional electronics, and the degree of commonality might well be high enough to warrant a coordination effort.

IFM Experiment - It is our opinion that an experiment is needed to test criteria, packaging and checkout philosophy for in-flight maintenance. Needless to say, the hardware and conditions should approximate those expected on the space station.

X. ABBREVIATIONS AND ACRONYMS

ACE	-	Automatic Checkout Equipment
A/D	-	Analog/Digital
AM	-	Amplitude Modulated
ASCP	-	Attitude Set Control Panel
BIC	-	Built in Checkout
BIT	-	Built in Test
BITE	-	Built in Test Equipment
BIST	-	Built in System Test
C&W or		
CAW	-	Cuation and Warning
CAGE	-	Computerized Aerospace Ground Equipment
CDR	-	Commander
CDU	-	Coupling Data Unit
CFM	-	Cubic Ft./Min.
CM	-	Command Module
CMC	-	Command Module Computer
CMG	-	Control Moment Gyro
CMP	-	Command Module Pilot
CSC	-	Cosecant
CSM	-	Command/Service Module
CTE	-	Central Timing Equipment
D&C	-	Displays and Controls
D/A	-	Digital/Analog
DRG	-	Digital Ranging Generator
DSKY	-	Display Keyboard
DSE	-	Data Storage Equipment
ECA	-	Electronic Control Assembly
EDA	-	Electronic Display Assembly
EEC	-	Enable Error Counter
EMI	-	Electromagnetic Interference
EVA	-	Extra Vehicular Activity
FDAI	-	Flight Director/Attitude Indicator
FL	-	Filter
FLAW	-	Functional Loop/Alternate Ways
FM	-	Frequency Modulated
FMEA	-	Failure Mode Effects Analysis
G & C	-	Guidance and Control
G & N	-	Guidance and Navigation
GA	-	Gyro Assembly
GDC	-	Gyro Display Coupler
GP/FPI	-	Gimbal Position/Fuel Pressure Indicator
HG	-	High Gain
IF	-	Intermediate Frequency

ABBREVIATIONS AND ACRONYMS - continued

IFM	-	In-Flight Maintenance
IMU	-	Inertial Measurement Unit
ISS	-	Inertial Subsystem
KW	-	Kilowatt
λ (Lambda)	-	Failure Rate
LM	-	Lunar Module
LMP	-	Lunar Module Pilot
LRU	-	Lowest Replaceable Unit
LSI	-	Large Scale Integration
MC	-	Megacycle
MDA	-	Motor Drive Amplifier
MDIV	-	Malfunction Defection, Isolation and Verifi- cation
MOD	-	Modulator
MPD	-	Multi Parameter Display
MSA	-	Main Summing Amplifier
MSI	-	Medium Scale Integration
MXR	-	Mixer
OCS	-	On-Board Checkout System
OSS	-	Optical Subsystem
PAM	-	Pulse Amplitude Modulated
PC	-	Printed Circuit
PCM	-	Pulse Code Modulated
PIPA	-	Pulse Integrated Pendulous Accelerometer
PEA	-	PIPA Electronics Assembly
PGNCS	-	Primary Guidance, Navigation and Control System
PM	-	Phase Modulated
PMP	-	Pre-Modulation Processor
PPS	-	Pulses per Second
PS	-	Power Supply
PSA	-	Power Servo Amp.
PSIA	-	Pounds/Square Inch Ambient
PTT	-	Push to Talk
PVR	-	Precision Voltage Reference
R/R	-	Remove/Replace
RC	-	Rotation Control
RCS	-	Reaction Control System
RCV	-	Receiver
RF	-	Radio Frequency
RFP	-	Request for Proposal
RJEC	-	Reaction Jet On/Off Control
RRT	-	Rendezvous Radar Transponder
SCS	-	Stabilization and Control System

ABBREVIATIONS AND ACRONYMS - continued

SCE	-	Signal Conditioning Equipment
SCT	-	Scanning Telescope
SM	-	Service Module
SPS	-	Service Propulsion System
SS	-	Space Station
SXT	-	Sextant
TC	-	Translation Control
TVC	-	Thrust Vector Control
TVSA	-	Thrust Vector Servo Assembly
UDL	-	Up Data Link
USBE	-	Unified S-Band Equipment
VAC	-	Volts A.C.
VCO	-	Voltage Controlled Oscillator
VDC	-	Volts D.C.
VHF	-	Very High Frequency
XFRMR	-	Transformer
XMIT	-	Transmitter

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